As you are now the owner of this document which should have come to you for free, please consider making a donation of £1 or more for the upkeep of the (Radar) website which holds this document. I give my time for free, but it costs me around £300 a year to bring this document to you. You can donate here https://blunham.com/Radar, thank you.

Do not upload this copyright pdf document to any other website. Breaching copyright may result in a criminal conviction and large payment for Royalties.

This document was generated by me, Colin Hinson, from a document held at R.A.F. Henlow Signals Museum which is believed to be out of copyright or Crown Copyright. It is presented here (for free) under the Open Government Licence (O.G.L.) if under Crown Copyright and this version of the document is my copyright (along with the Crown Copyright) in much the same way as a photograph would be. It should be noted that most of the pages are identifiable as having been processed by me. If you believe the original document to be under copyright, please contact me.

The document should have been downloaded from my website https://blunham.com/Radar, or any mirror site named on that site. If you downloaded it from elsewhere, please let me know (particularly if you were charged for it). You can contact me via my Genuki email page: https://www.genuki.org.uk/big/eng/YKS/various?recipient=colin

You may not copy the file for onward transmission of the data nor attempt to make monetary gain by the use of these files. If you want someone else to have a copy of the file, point them at the website (https://blunham.com/Radar). **Please do not point them at the file itself as it may move or the site may be updated.**

I put a lot of time into producing these files which is why you are met with this page when you open the file.

In order to generate this file, I need to scan the pages, split the double pages and remove any edge marks such as punch holes, clean up the pages, set the relevant pages to be all the same size and alignment. I then run Omnipage (OCR) to generate the searchable text and then generate the pdf file.

Hopefully after all that, I end up with a presentable file. If you find missing pages, pages in the wrong order, anything else wrong with the file or simply want to make a comment, please drop me a line (see above).

If you find the file(s) of use to you, you might like to make a donation for the upkeep of the website – see https://blunham.com/Radar for a link to do so.

Colin Hinson

In the village of Blunham, Bedfordshire, UK.

AP 3456D

ROYAL AIR FORCE

MANUAL

FLYING

VOLUME D

AIRCRAFT INSTRUMENTS

AND INSTRUMENT SYSTEMS

By Command of the Defence Council

I. T. Durnett

MINISTRY OF DEFENCE
APRIL 1969

ROYAL AIR FORCE

MANUAL

FLYING

VOLUME D

AIRCRAFT INSTRUMENTS

AND INSTRUMENT SYSTEMS

By Command of the Defence Council

I.T. Dunnett

MINISTRY OF DEFENCE
APRIL 1969

NOTE TO READERS

1. The Royal Air Force Manual—Flying (AP 3456) is a publication sponsored by the Director of Training (Flying) (RAF), Ministry of Defence. It is a general reference on flying consisting of ten volumes. The extent and depth of coverage of the subject matter is aimed at the post-OCU aircrew member, with provision for more advanced and detailed information in the form of annexes to the main chapters.

2. The AP 3456 series comprises:

AP 3456, Volume A The Principles of Flight
Volume B Aircraft Design Features and Systems
Volume C Radio and Radar
Volume D Aircraft Instruments and Instrument Systems
Volume E The Aircrew, Airmanship and Aircraft Performance
Volume F Aircraft Piloting
Volume G Air Navigation
Volume H Aircraft Weapons Employment
Volume J Survival
Volume K Mathematics, Physics and Electronics

To assist in the location of material in the above volumes, the Part and Section headings for each volume are given immediately following the contents list for this volume.

3. The subject matter of this publication may be affected by Defence Council Instructions or other orders. If possible Amendment Lists are issued to correct this publication accordingly, but it is not always practicable to do so. When a Defence Council Instruction or order contradicts any portion of this publication, the Instruction or order is to be taken as the overriding authority.

4. Suggestions for changes concerned with the coverage of subject matter should be forwarded through normal channels to the following address:

Ministry of Defence, DDTF(RAF), Adastral House, Theobalds Road, WC1X 8RU

Any suggestions should preferably be accompanied by a draft covering the proposed new material, and any relevant references should be quoted, or an indication given of the source of new material.

5. Readers noting any errors in the text are requested to notify the editor direct at the following address:

Editor AP 3456, Royal Air Force College Cranwell, Sleaford, Lincs, NG34 8HB

AIRCRAFT INSTRUMENTS AND INSTRUMENT SYSTEMS

CONTENTS

PART	SECT	CHAP	TITLE
1			**AIR DATA INSTRUMENTS**
	1		**Temperature**
		1	Principle of Temperature Measurement
		2	Thermometers
	2		**Height**
		1	Introduction to Barometric Height Measurement
		2	Simple, Sensitive and Cabin Altimeters
		3	Servo-Assisted Altimeters
	3		**Speed and Distance**
		1	*Not Used*
		2	Vertical Speed Indicators
		3	Air Speed Indicators
		4	Machmeters
		5	Air Mileage Units
		6	Air Mileage Indicator, Mark 1
		7	Air Data Systems
		8	True Air Speed Unit, Mark 1
		9	Combined Speed Indicators
2			**HEADING, ATTITUDE AND ALIGNMENT INSTRUMENTS**
	1		**Direct Indicating Compass Systems**
		1	Principles of Direct Indicating Compass Systems
		2	E2 Series Compasses
		3	Pilot-Type Compasses
	2		**Gyroscopic References**

PART	SECT	CHAP	TITLE
		1	Introduction to Gyroscopes
		2	Master Reference Gyro, Mark 1
		3	CL11 Directional Gyro
		4	Sperry Twin Gyro Platform (MRG Mk 2)
		5	Turn and Slip Indicators
		6	The Direction Indicator, Mark 1
		7	Direct Reading Attitude Indicators
	3		**Remote Indicating Compasses**
		1	Introduction to Gyro-Magnetic Compasses
		2	Mk 4 and Mk 5 Series Gyro-Magnetic Compasses
		3	Military Flight System, Mark 1 Series—Compass Element
		4	Mk 7 Series Gyro-Magnetic Compasses
		5	Spare
		6	Polar Path Compass
		7	Spare
		8	C2G and C2J (GM8A) Gyro-Magnetic Compass Systems
		9	*Not Used*
		10	C12 Gyro-Magnetic Compass System
	4		**Flight Instrument Display Systems**
		1	The Zero Reader, ZL2
		2	The Military Flight System Director Horizon
		3	The Integrated Flight Instrument System
		4	The Head-Up Display
	5		**Alignment Instruments**
		1	Medium Landing Compass

PART	SECT	CHAP	TITLE
		2	Pattern 2 Prismatic Compass
		3	Watts Datum Compass, Marks 1 and 2
		4	MC-1M Magnetic Compass Calibrator
		5	The Precise Heading Test Set
	6		**Compass Correction and Calibration**
		1	Magnetic Compass Deviations
		2	Compass Swinging Procedures
		3	The Analysis of the Compass Swing
		4	The Magnetic Survey of Compass Bases
3			**DR NAVIGATION SYSTEMS**
	1		**Control Systems**
		1	Remote Indication and Control
		2	Servomechanisms
	2		**Computing Principles and Circuits**
		1	Analogue Computers
		2	Digital Computers
	3		**Position Computing and Display Systems**
		1	Air Position Indicator, Mark 1
		2	Wind Finding Attachment, Mark 1B
		3	Ground Position Indicator, Mark 1B
		4	Ground Position Indicator, Mark 4 Series
		5	Type 9476 Computers
		6	Type 947 and 9478 Display Units
		7	The Ferranti Navigation Display and Computer
		8	Ground Position Indicator, Mark 7
		9	The Tactical Air Navigation System

PART	SECT	CHAP	TITLE
	4		**Inertial Navigation Systems**
		1	Principles of Inertial Navigation
		2	INS Errors and Mixed Systems
		3	The Ferranti Inertial Platform
		4	The Elliott E3 Heading Reference System
		5	The Elliott E3R Heading Reference System
4			**ASTRONOMICAL INSTRUMENTS**
	1		**Sextants**
		1	Bubble Sextants, Mark 9AM, 9BM
		2	Periscopic Sextant, Mark 2
		3	Periscopic Sextants, Marks 2A, 2B
		4	The Kollsman Periscopic Sextant
	2		**Astro Compasses**
		1	Astro Compass, Mark 2A
		2	Spare
		3	Spare
5			**MISCELLANEOUS AIRCRAFT INSTRUMENTS**
	1		**Engine and Ancillary Instruments**
		1	Tachometer and Position Indicators
		2	Pressure Gauges
		3	Fuel Contents Gauges and Flowmeters
		4	Warning Systems and Devices
		5	Accelerometers and Fatigue Meters
		6	Torque Meters

PART	SECT	CHAP	TITLE
	2		**Optical Instruments**
		1	Drift Recorder, Mark 2
6			**GLOSSARY OF TERMS**

ABRIDGED CONTENTS LIST FOR AP 3456 SERIES

Vol	Part	Sect	Title
A			**THE PRINCIPLES OF FLIGHT**
	1		**The Principles of Flight — Fixed Wing Aircraft**
		1	Basic Principles of Flight
		2	Design and Construction Features
		3	High Speed Flight
		4	Propellers
	2		**The Principles of Flight — Helicopters**
		1	Basic Principles of Helicopter Flight
		2	Design and Construction Features
B			**AIRCRAFT DESIGN FEATURES AND SYSTEMS**
	1		**Propulsion**
		1	Propulsion Theory
		2	Piston Engines
		3	Gas Turbines
		4	Miscellaneous Propulsion Systems
		5	Gas Turbine Engine Derivatives
		6	Aviation Fuels
	2		**Aircraft Ancillary System**
		1	Principles
		2	Ancillary Systems
C			**RADIO AND RADAR**
	1		**Communications**
		1	Very High Frequency and Ultra High Frequency Equipment
		2	High Frequency Equipment
		3	Intercommunication Installations
		4	Secondary Surveillance Radar
		5	Communication Procedures
	3		**Airborne Height, Direction and Range Measurement**
		1	Airborne Height Measurement
		2	Airborne Direction Measurement
		3	Direction Transmitters
		4	Airborne Range and Bearing Measurement
	3		**Hyperbolic Navigation Systems**
		1	Principles of Hyperbolic Systems
		2	Hyperbolic Systems and Airborne Receivers

ABRIDGED CONTENTS — continued

Vol	Part	Sect	Title
	4		**Doppler Navigations Systems**
		1	Doppler Principles
		2	Doppler Equipments
	5		**Airborne Radars and Reconnaissance Systems**
		1	Principles of Airborne Radars and Reconnaissance Systems
		2	Airborne Radars and Reconnaissance Systems Equipments
	6		**Approach and Landing Aids**
		1	Precision Approach Aids
		2	Landing Aids
	7		**Glossary of Terms**
D			**AIRCRAFT INSTRUMENT AND INSTRUMENT SYSTEMS**
	1		**Air Data Instruments**
		1	Temperature
		2	Height
		3	Speed and Distance
	2		**Heading, Attitude and Alignment Instrument**
		1	Direct Indicating Compasses
		2	Gyroscopic References
		3	Remote Indicating Compasses
		4	Flight Display Systems
		5	Alignment Instruments
		6	Compass Correction and Calibration
	3		**DR Navigation Systems**
		1	Control Systems
		2	Computing Principles and Circuits
		3	Position Computing and Display Systems
		4	Inertial Navigation Systems
	4		**Astronomical Instruments**
		1	Sextants
		2	Astro Compasses
	5		**Miscellaneous Aircraft Instrument**
		1	Engine and Ancillary Instruments
		2	Optical Instruments
	6		**Glossary of Terms**

ABRIDGED CONTENTS – continued

Vol	Part	Sect	Title
E			**THE AIRCREW, AIRMANSHIP AND AIRCRAFT PERFORMANCE**
	1		**Aircrew Environment**
		1	The Aircrew Member
		2	Oxygen Equipment and Flying Clothing
	2		**Airmanship**
		1	Aircraft Servicing and Ground Handling
		2	Emergencies
		3	Escape from Aircraft in Flight
	3		**Aircraft Performance**
		1	Piston Engine Aircraft
		2	Jet Engine Aircraft
		3	Turboprop Aircraft
		4	Helicopters
		5	Scheduled Performance
		6	Weight and Balance
F			**AIRCRAFT PILOTING**
	1		**Basic and Advanced Flying**
		1	Basic Flying
		2	Advanced Flying
	2		**Helicopter Flying**
		1	Basic Flying Techniques
		2	Special Helicopter Techniques
G			**AIR NAVIGATION**
	1		**Basic Principles**
		1	Distance, Direction and Position on the Earth
		2	The Triangle of Velocities
		3	Position Lines and Fixes
		4	Aids to Plotting and Calculation
	2		**Maps and Charts**
		1	Principles of Map Projection
		2	Types of Projection
		3	Geodesy and Survey
	3		**Astro Navigation**
		1	The Celestial Sphere
		2	Sextant Observations
		3	The Sun, Moon and Time

ABRIDGED CONTENTS — continued

Vol	Part	Sect	Title
	4		**Flight Planning**
		1	Aeronautical Documents
		2	Flight Planning Procedures
	5		**Navigation Techniques**
		1	Basic Techniques
		2	Applied Navigation
		3	Special Techniques
		4	Tactical Navigation
	6		**Glossary of Terms**
H			**AIRCRAFT WEAPONS EMPLOYMENT**
	1		**Weapon and Target Assessment**
		1	Introduction to Aircraft Weapons Employment
		2	Effects of Aircraft Weapons
		3	Initial Consideration of Targets
		4	Vulnerability of Specific Targets
		5	Force Requirements
	2		**Guided Missiles**
		1	Introduction to Guided Missiles
		2	Missile Propulsion
		3	Control Systems
		4	Guidance
		5	Air to Surface Missiles
		6	Air to Air Missiles
	3		**Ballistic and Sighting**
		1	Ballistics
		2	Sights and Sighting
		3	Strike Sights
		4	Blind Bombing
	4		**Electronic Warfare**
		1	Electronic Warfare
	5		**Weapons**
		1	Aircraft Bombs
		2	Aircraft Guns
		3	Aircraft Flares
		4	Aircraft Rocket Systems

ABRIDGED CONTENTS – continued

Vol	Part	Sect	Title
J			**SURVIVAL**
	1		**Survival Techniques**
		1	General Aspects of Survival
		2	Environmental Survival
		3	Combat Survival
		4	Conduct After Capture
	2		**Survival Organisation and Equipment**
		1	Search and Rescue
		2	Survival Equipment
		3	Survival Training
K			**MATHEMATICS, PHYSICS AND ELECTRONICS**
	1		**Mathematics**
		1	Calculation Aids
		2	Algebra
		3	Plane and Spherical Trigonometry
		4	Calculus
		5	Vector Analysis
		6	Statistics
	2		**Physics**
		1	Heat
		2	Light
		3	Sound
		4	Mechanics
		5	Magnetism and Electricity
	3		**Electronics**
		1	Basic Principles
		2	Communications and Radar

ABRIDGED CONTENTS LIST FOR AP 3456 SERIES

VOL	PART	SECT	TITLE
A			**THE PRINCIPLES OF FLIGHT**
	1		**The Principles of Flight—Fixed Wing Aircraft**
		1	Basic Principles of Flight
		2	Design and Construction Features
		3	High Speed Flight
		4	Propellers
	2		**The Principles of Flight—Helicopters**
		1	Basic Principles of Helicopter Flight
		2	Design and Construction Features
B			**AIRCRAFT DESIGN FEATURES AND SYSTEMS**
	1		**Propulsion**
		1	Propulsion Theory
		2	Piston Engines
		3	Gas Turbines
		4	Miscellaneous Propulsion Systems
		5	Gas Turbine Engine Derivatives
		6	Aviation Fuels
	2		**Aircraft Ancillary Systems**
		1	Principles
		2	Ancillary Systems
C			**RADIO AND RADAR**
	2		**Communications**
		1	Very High Frequency and Ultra High Frequency Equipment
		2	High Frequency Equipment
		3	Intercommunication Installations
		4	Secondary Surveillance Radar

VOL	PART	SECT	TITLE
		5	Communications Procedures
	2		**Airborne Height, Direction and Range Measurement**
		1	Airborne Height Measurement
		2	Airborne Direction Measurement
		3	Directional Transmitters
		4	Airborne Range and Bearing Measurement
	3		**Hyperbolic Navigation Systems**
		1	Principles of Hyperbolic Systems
		2	Hyperbolic Systems and Airborne Receivers
	4		**Doppler Navigation Systems**
		1	Doppler Principles
		2	Doppler Equipments
	5		**Airborne Radars and Reconnaissance Systems**
		1	Principles of Airborne Radars and Reconnaissance Systems
		2	Airborne Radars and Reconnaissance Systems Equipment
	6		**Approach and Landing Aids**
		1	Precision Approach Aids
		2	Landing Aids
	7		Glossary of Terms
D			**AIRCRAFT INSTRUMENTS AND INSTRUMENT SYSTEMS**
	1		**Air Data Instruments**
		1	Temperature
		2	Height
		3	Speed and Distance
	2		**Heading, Attitude and Alignment Instruments**
		1	Direct Indicating Compasses

VOL	PART	SECT	TITLE
		2	Gyroscopic References
		3	Remote Indicating Compasses
		4	Flight Instrument Display Systems
		5	Alignment Instruments
		6	Compass Correction and Calibration
	3		**DR Navigation Systems**
		1	Control Systems
		2	Computing Principles and Circuits
		3	Position Computing and Display Systems
		4	Inertial Navigation Systems
	4		**Astronomical Instruments**
		1	Sextants
		2	Astro Compasses
	5		**Miscellaneous Aircraft Instruments**
		1	Engine and Ancillary Instruments
		2	Optical Instruments
	6		**Glossary of Terms**
E			**THE AIRCREW, AIRMANSHIP AND AIRCRAFT PERFORMANCE**
	1		**The Aircrew Environment**
		1	The Aircrew Member
		2	Oxygen Equipment and Flying Clothing
	2		**Airmanship**
		1	Aircraft Servicing and Ground Handling
		2	Emergencies
	3		**Aircraft Performance**

VOL	PART	SECT	TITLE
		1	Piston Engine Aircraft
		2	Jet Engine Aircraft
		3	Turboprop Aircraft
		4	Helicopters
		5	Scheduled Performance
		6	Weight and Balance
F			**AIRCRAFT PILOTING**
	1		**Basic and Advanced Flying**
		1	Basic Flying
		2	Advanced Flying
	2		**Helicopter Flying**
		1	Basic Flying Techniques
		2	Special Helicopter Techniques
G			**AIR NAVIGATION**
	1		**Basic Principles**
		1	Distance, Direction and Position on the Earth
		2	The Triangle of Velocities
		3	Position Lines and Fixes
		4	Aids to Plotting and Calculation
	2		**Maps and Charts**
		1	Principles of Map Projection
		2	Types of Projection
		3	Geodesy and Survey
	3		**Astro Navigation**
		1	The Celestial Sphere
		2	Sextant Observations

VOL	PART	SECT	TITLE
		3	The Sun, Moon and Time
	4		**Flight Planning**
		1	Aeronautical Documents
		2	Flight Planning Procedures
	5		**Navigation Techniques**
		1	Basic Techniques
		2	Applied Navigation
		3	Special Techniques
		4	Tactical Navigation
	6		**Glossary of Terms**
H			**AIRCRAFT WEAPONS EMPLOYMENT**
	1		**Weapon and Target Assessment**
		1	Introduction to Aircraft Weapons Employment
		2	Effects of Aircraft Weapons
		3	Initial Considerations of Targets
		4	Vulnerability of Specific Targets
		5	Force Requirements
	2		**Guided Missiles**
		1	Introduction to Guided Missiles
		2	Missile Propulsion
		3	Control Systems
		4	Guidance
	3		**Ballistic and Sighting**
		1	Ballistics
		2	Sights and Sighting
		3	Strike Sights

VOL	PART	SECT	TITLE
		4	Blind Bombing
	4		**Electronic Warfare**
		1	Electronic Warfare
J			**SURVIVAL**
	1		**Survival Techniques**
		1	Environment Techniques
		2	Medical Aspects of Survival
		3	Survival in Enemy Territory
		4	Conduct After Capture
	2		**Survival Equipment**
		1	Aircraft Escape Equipment
		2	Survival Flying Equipment
	3		**Search and Rescue**
		1	Search Organization
		2	Rescue Equipment and Procedure
	4		**Survival Training**
		1	Survival Training
K			**MATHEMATICS, PHYSICS AND ELECTRONICS**
	1		**Mathematics**
		1	Calculation Aids
		2	Algebra
		3	Plane and Spherical Trigonometry
		4	Calculus
		5	Vector Analysis
		6	Statistics
	2		**Physics**

VOL	PART	SECT	TITLE
		1	Heat
		2	Light
		3	Sound
		4	Mechanics
		5	Magnetism and Electricity
	3		**Electronics**
		1	Basic Principles
		2	Communications and Radar

PART 1

AIR DATA INSTRUMENTS

Section

1 **Temperature**

2 **Height**

3 **Speed and Distance**

PART 1

SECTION 1

TEMPERATURE

Chapter

1 Principles of Temperature Measurement

2 Thermometers

Annex-Heating Errors in Outside Air Temperature Thermometers

CHAPTER 1

PRINCIPLES OF TEMPERATURE MEASUREMENT

CONTENTS

	Paras
Introduction	1–2
NON-ELECTRICAL THERMOMETER PRINCIPLES	
General	3
Principles	4
ELECTRICAL THERMOMETER PRINCIPLES	
General	5
Principles	6
SENSING ELEMENTS OF ELECTRICAL THERMOMETERS	
Temperature Sensitive Resistance Bulbs	7–10
Thermocouples	11–18
INDICATORS FOR ELECTRICAL THERMOMETERS	
Ratiometer Type Indicator—S 63 Series	19–22
Millivolt Meter Type Indicator	23–24
Balanced Bridge Thermometer Indicator	25–27

Introduction

1. Thermometers installed in aircraft provide information on:

a. The outside air temperature, to enable true airspeed and height to be computed from indicated values.

b. The operating temperatures of various engine components, lubricants and exhaust gases, thus enabling the engines to be operated most efficiently, and giving timely warning of conditions likely to lead to damage or failure.

c. The temperature of various compartments within the aircraft.

2. Thermometers employ sensing elements whose physical or electrical properties change with temperature. They can be broadly divided into two categories—electrical and non-electrical. In this chapter the principles used in both types are briefly examined, and the more common types of sensing element and indicator used in electrical thermometers are described in some detail. In Chapter 2, specific examples of each type of outside air temperature and engine temperature thermometer are described. Full constructional details of thermometers in current use are contained in AP 1275B and code referenced publications in the AP 112G series.

NON-ELECTRICAL THERMOMETER PRINCIPLES

General

3. Non-electrical thermometers can be considered as single units, even though some of them include a remote indication facility. They are of fairly simple construction, the main features of each type being described in the early part of Chapter 2.

Principles

4. Non-electrical thermometers used in aircraft depend on one of the following physical characteristics:

a. The expansion of mercury when heated.
b. The change in vapour pressure of a pure volatile liquid with change of temperature.
c. The differential expansions of dissimilar metals, exploited in the use of bimetallic sensing elements.

ELECTRICAL THERMOMETER PRINCIPLES

General

5. It is convenient to consider electrical thermometers as comprising two components, the temperature sensitive element and the indicator. Various types of each component are described in this chapter, the complete thermometers being exemplified by those described in Chapter 2.

Principles

6. Aircraft electrical thermometers depend on one of the following electrical characteristics:

a. The variation with temperature of the electrical resistivity of certain metals.
b. The generation of an emf when heat is applied to one junction of a thermocouple.

SENSING ELEMENTS OF ELECTRICAL THERMOMETERS

Temperature Sensitive Resistance Bulbs

7. Resistance bulbs used with electrical thermometer systems have various shapes and sizes, but consist essentially of a resistance in the form of a wire coil, contained in a sealed steel tube, and connected to the electrical circuit of the indicator unit. The coil may be of nickel or platinum.

8. **Nickel Law Bulbs—Type S 84.** The operation of a nickel resistance depends upon the relation:

$R_T = R_o(1 + \alpha\, t + \beta\, t^2)$, where
R_T = Resistance at t°C,
R_o = Resistance at 0°C, and α and β are constants, α being the constant for the temperature coefficient of resistance of nickel.

9. **Platinum Law Bulbs—S 110 Series.** These have largely superseded the nickel law bulbs. Where platinum is used, the space in the bulb is filled with hydrogen gas, to give an improved response time (*ie* more rapid heat exchange between the medium and the sensitive element). The resistance relationship for platinum is:

$R_T = R_o\ (1 + \alpha t - \beta t^2)$, where α and β are the constants for platinum.

10. **Mounting.** For engine temperature measurement the bulb is provided with a bulkhead or plate mounting. When used for outside air temperature measurement the element may be coiled horizontally and mounted on a base plate, in thermal contact with but electrically insulated from the plate. The plate is installed on the underside of the airframe. Probe mountings used in the Total Head Thermometer and the Rosemount Outside Air Temperature Probe are described in Chapter 2.

Thermocouples

11. The main use of thermocouples in aircraft is to measure the temperature of hot engine gases. Usually several thermocouple

probes connected in parallel are positioned in the gas stream to obtain an average temperature reading.

12. The use of the thermocouple to measure temperature depends on the fact that if two strips of dissimilar metal are joined together at both ends, and heat is applied to one joint (the hot junction), a small emf is generated, the amount of which depends upon the metals used (often nickel-chromium and nickel-aluminium alloys), and the difference in temperature between the hot and cold junctions.

13. The thermocouple is enclosed inside a metal guard tube, holes in which allow the gases to flow across one junction. An indicator, which is basically a millivoltmeter calibrated to read in degrees Centigrade, is connected into the circuit. The indicator is compensated for variations in cold junction temperature.

14. **Stagnation Type**. The stagnation type of thermocouple is named after its housing, which is designed to bring the air almost to rest. It is suitable for use when exhaust gas velocities are high, and is used in gas turbine engines. The metal elements are enclosed in a ceramic insulator, except for the hot junction which projects into the gas stream. The insulator and elements are enclosed in a metal sheath, which has inlet and outlet holes of different diameters at its closed end to allow the gases to circulate around the junction. The holes are staggered as shown in Fig 1 to reduce the gas velocity.

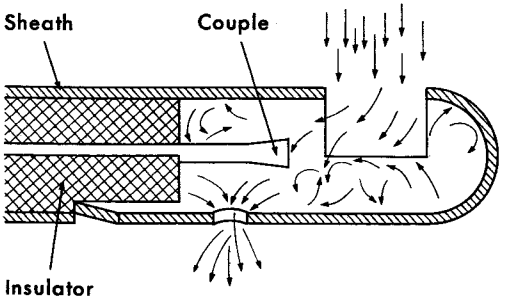


Fig 1 Stagnation Type of Thermocouple

15. **Rapid Response Type**. This is of similar construction to the stagnation type, but the inlet and outlet holes are diametrically opposed, as shown in Fig 2. It is suitable for use with lower exhaust gas velocities, and is therefore used in propeller turbine engines.

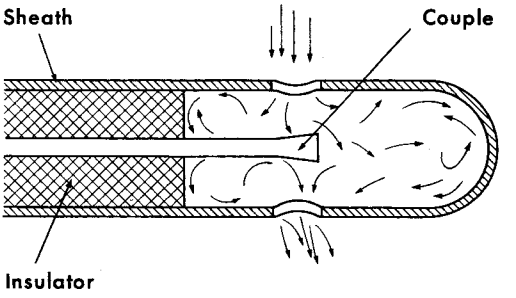


Fig 2 Rapid Response Type of Thermocouple

16. **Engine Cylinder Type**. In this type of thermocouple the metal elements are usually of copper and constantan, and the hot junction is simply passed through a sealed joint in the cylinder head and secured to the underside.

17. **Multiple Element Types.** Each thermocouple probe may be of single, double or triple element construction. A single element circuit provides temperature indication only; a double element provides an additional circuit to give a temperature signal to the maximum gas temperature control system, which feeds the information to the fuel control unit; and a triple element may be used to provide a circuit for a warning system, such as an exhaust gas analyser.

18. **Cold Junction Temperature Compensation**. Changes in the cold junction temperature may be compensated for by:

a. Use of a bi-metallic spiral acting on the hairspring controlling the indicator pointers.

b. Electrical compensation, using a Wheatstone bridge type resistance thermometer network. The voltage supplied to this type of compensator is in turn regulated by a voltage compensator, also using a Wheatstone bridge network.

INDICATORS FOR ELECTRICAL THERMOMETERS

Ratiometer Type Indicator—S 63 Series

19. The Type S 63 series indicator operates as a ratiometer, which measures the ratio of currents flowing in each of two moving coils. It is used in conjunction with temperature sensitive resistance bulbs to indicate temperature.

20. The basic circuit and the disposition of the coils are shown in Figs 3 and 4. The two coils are wound in opposition on a common former to which is attached a pointer. The former is free to rotate in the field of a permanent magnet about a stationary soft iron core. One of the coils on the former carries a steady reference current, while the other coil carries a current which varies with the resistance, and hence the temperature, of the bulb. The magnetic field in the air gap in which the moving coil rotates is not uniform owing to the shape of the core, and the ratio of the currents in the two coils will vary with the temperature of the bulb. The coil assembly will therefore rotate until the coils are in that part of the field where the torques produced in each coil are equal and opposite. The former will then cease to rotate and the pointer will indicate the temperature of the bulb.

21. The accuracy of the ratiometer is virtually unaffected by aircraft electrical supply variation, since a change of voltage does not alter the ratio of the currents in the two coils. The off scale position may be at either end of the scale, depending on the dial presentation.

22. A typical presentation is shown in Fig 5. Further details of construction and variations in type are contained in AP 112G-0504-16.

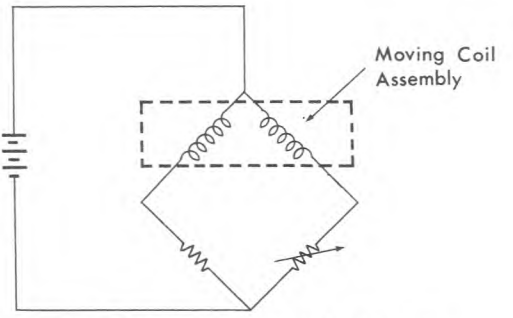
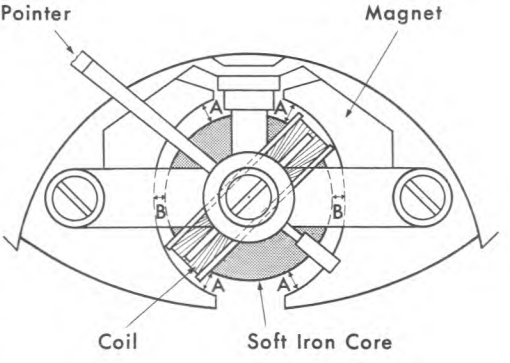


Fig 3 Ratiometer Indicator—Basic Circuit



Note.
Field air gap at points marked A greater than at points marked B

Fig 4 Ratiometer Indicator—Disposition of Coils

Fig 5 A Typical Indicator, Type S63

Millivoltmeter Type Indicator

23. Some of the Type S 149 temperature indicators and the S 64 indicator are of the voltmeter type, used in conjunction with a thermocouple. Individual indicators are described in AP 112G-0508-1.

24. In a typical voltmeter type of indicator the millivolt input from a thermocouple is applied to a servopotentiometer, which provides a high output to operate the indicator. The indicator is a normal voltmeter where the movement of a moving coil in relation to the field of the permanent magnet is proportional to the voltage applied to the coil.

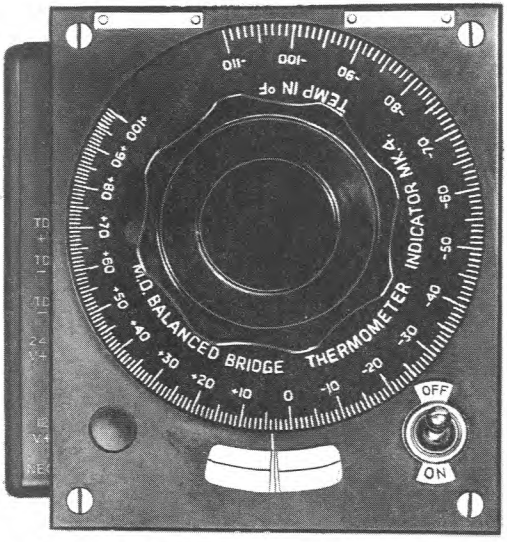


Fig 6 The Balanced Bridge Thermometer Indicator Mk 4

Balanced Bridge Thermometer Indicator

25. The balanced bridge thermometer indicator Mk 4 is fitted to some aircraft and used, in conjunction with a platinum resistance bulb, to measure outside air temperature. The indicator is illustrated at Fig 6.

26. The indiciator is essentially a Wheatstone bridge, one arm being the resistance of a thermometer bulb, and the adjustable arm consisting of a manually operated variable resistance of the circular slide wire type. The bridge is balanced when the galvanometer at the bottom of the indicator gives a zero reading; the position of the slider is then related to the resistance and hence the temperature of the thermometer bulb. This temperature is then read off the circular scale positioned by the slider control. The indicator is used with a resistance bulb Type A, B or C.

27. The following simple procedure is used to determine the outside air temperature:

a. Switch the ON-OFF switch to ON.
b. Rotate the control knob until the galvanometer pointer indicates zero deflection.
c. Read the temperature on the circular scale against the datum.
d. Switch to OFF.

CHAPTER 2

THERMOMETERS

CONTENTS

	Paras
Introduction	1
NON-ELECTRICAL THERMOMETERS	
Mercury Type	2–4
Vapour Pressure Type	5-7
Bi-metallic Element Type	8–9
REMOTE INDICATING ELECTRICAL THERMOMETERS	
General	10
Resistance Bulb Type	11–12
Thermocouple Type	13–14
SPECIAL APPLICATIONS	
Total Head Thermometer	15–18
Rosemount Outside Air Temperature Probe	19–23
THERMOMETER ERRORS	
General Errors	24
Outside Air Temperature Thermometer Errors	25–28
Annex—Heating Errors in Outside Air Temperature Thermometers	

Introduction

1. In Chapter 1 the basic principles of temperature sensing elements and indicators were described. In this chapter typical thermometers using each of the temperature sensing techniques listed at paras 4 and 6 of Chapter 1 are described in general terms. Detailed descriptions of these and other specific types of thermometer are contained in AP 1275B and the AP 112G series of publications. Thermometer errors are described in the Annex to this Chapter.

NON-ELECTRICAL THERMOMETERS

Mercury Type

2. The mercury thermometer, of which the Mk 1A oil temperature thermometer is an example, consists of a steel bulb connected by capillary tubing to a bourdon tube pressure gauge. An increase in temperature results in the expansion of the mercury, a flow in the capillary, and a distension of the bourdon tube to which the pointer is attached.

3. To minimize errors due to variations in the temperature of components other than the bulb, the volumes of the capillary tube and the bourdon tube are kept small relative to the volume of the bulb. To compensate for changes of temperature at the indicator, a bi-metallic strip, which reacts in opposition to the motion of the spirals of the bourdon tube, is incorporated between the free end of the bourdon tube and the centrally mounted pointer spindle. If the locations of the thermometer and its indicator require such a length of capillary tube that its volume is significant in relation to that of the bulb, small steel chambers containing rods of Invar are spaced along the capillary tube. These are designed such that the change in volume of the capillary tube and the chambers is equal to that of the mercury in those components for the same temperature change; variations in temperature of the capillary tubing do not then affect the reading of the indicator.

4. **Uses.** This type of thermometer may be used to indicate engine lubricating oil temperature, or outside air temperature. In the latter case the bulb is protected by a shield from the sun's radiation.

Vapour Pressure Type

5. The vapour pressure type of thermometer, of which the transmitting radiator thermometer Mk 8H is typical, makes use of the fact that the vapour pressure of a pure volatile liquid is dependent only on the temperature of the liquid. The instrument consists of a brass or copper bulb, half filled (at normal ground temperature) with a liquid such as ether, the remaining space containing, in this case, only ether vapour. A copper capillary tube dips into the ether, and connects at the indicator to a bourdon pressure gauge tube and pointer. The capillary and bourdon tube are filled with a liquid ether. Any particular temperature of the bulb produces a definite pressure of the ether vapour, which is transmitted through the liquid to register as the corresponding temperature indication at the pointer. As with the mercury thermometer, the volume of the capillary tube and bourdon tube is kept small compared with that of the bulb.

6. The movement of the bourdon tube in the pressure gauge for a given temperature change will depend on the difference between the internal and external pressures. Thus for a constant bulb temperature, an aircraft climb would register as an increase of temperature. The error is not large; a standard instrument with the bulb at 80°C would read 85°C at 20,000 feet.

7. **Uses.** The vapour pressure type of thermometer may be used for indicating the coolant temperature of liquid cooled engines.

Bi-metallic Element Type

8. In this type of thermometer, typified by the direct reading air thermometer Mk 2, a helical bi-metallic element housed in a metal tube coils and uncoils in response to variations in air temperature. The bi-metallic element consists of two strips of dissimilar metals welded together and formed into a helix, one end of which is anchored to the tube, while the other is free to rotate. To the free end is attached a spindle, and a pointer which moves over the graduated temperature scale.

9. **Uses.** This type of thermometer is used for direct measurement of outside air temperature. The tube containing the bi-metallic helix is enclosed in an outer sun shield provided with vent holes to allow free air circulation.

REMOTE INDICATING ELECTRICAL THERMOMETERS

General

10. The temperature sensing element of electrical transmitting thermometers is either a bulb containing a coil of wire which has a high temperature coefficient of resistance, or a thermocouple.

Resistance Bulb Type

11. In these thermometers an indicator employing the ratiometer principle is used to indicate the temperature derived from the resistance of a resistance bulb.

12. **Uses.** Indicators can be paired with various bulbs to produce complete thermometers for a variety of applications. They are used as air temperature thermometers, and also to measure oil, coolant, and induction charge temperatures. The electrical transmitting thermometer Mk 2 is of this type.

Thermocouple Type

13. An indicator, which is in effect a sensitive millivoltmeter, measures the small emf produced by a thermocouple.

14. **Uses.** This type of thermometer is used to measure the exhaust gas temperature (jet pipe temperature) of turbine or propeller turbine engines, or the cylinder head temperature of piston engines. It is usually named according to its function, *eg* the Engine Cylinder Thermometer, and the Exhaust Gas Thermometer Type B.

SPECIAL APPLICATIONS

Total Head Thermometer

15. The total head thermometer, illustrated in Fig 1, employs a nickel wire mounted in a pitot head shaped housing as the temperature sensitive element. It is used in the true airspeed unit, where it is connected as the variable arm of a resistance bridge monitoring the ambient temperature of the free air system.

16. The design of the housing is such that the slowing down of the air produces a rise in temperature close to the theoretical rise (*see* the Annex); the true airspeed unit, which is calibrated in terms of indicated temperature, can therefore accept the signal from the total head thermometer as an unmodified input.

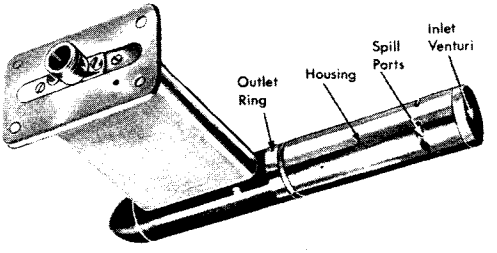


Fig 1 Total Head Thermometer

17. Most of the air entering the venturi—that which has been in contact with the venturi tube wall—is spilled through ports in the housing, but a small amount—the centre flow—is passed over the tube containing the element at a very reduced speed, and discharged through holes in the outlet ring. The holes ensure sufficient circulation of air over the element to reduce the thermal lag to acceptable limits.

18. The temperature range for this particular application of the total head thermometer is restricted to $-40°C$ to $+60°C$, with an accuracy of $\pm 1°C$.

Rosemount Outside Air Temperature Probe

19. The Rosemount air temperature probe, used with a ratiometer indicator, gives accurate total air temperature during flight in icing conditions.

20. The probe (Figs 2 and 3) consists of a centre body, mounted to the aircraft skin, containing a hermetically sealed platinum resistance element, and incorporating an air scoop and a de-icing element.

21. The de-icing heater, embedded in the material of the probe, is a tube containing an axial wire heating element, and operates continuously throughout flight. The heating is not thermostatically controlled but is self-compensating in that as temperature rises the resistance rises and so reduces the power consumption. The radiation shield protects the sensing element from the heating effects of the de-icing heater.

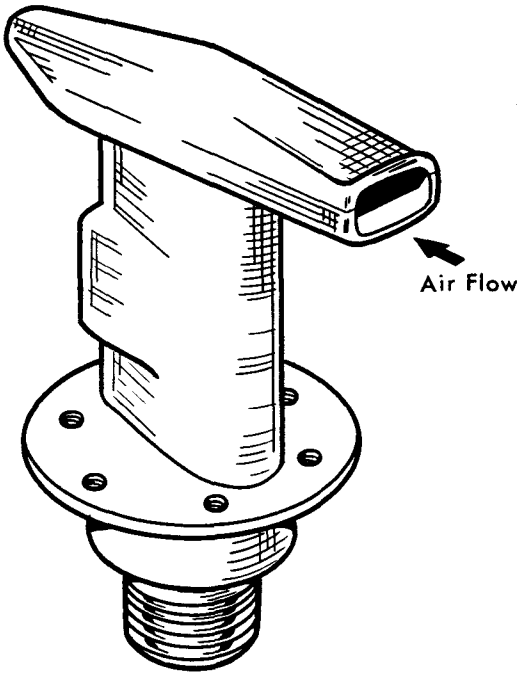


Fig 2 Outside View of the Rosemount Probe

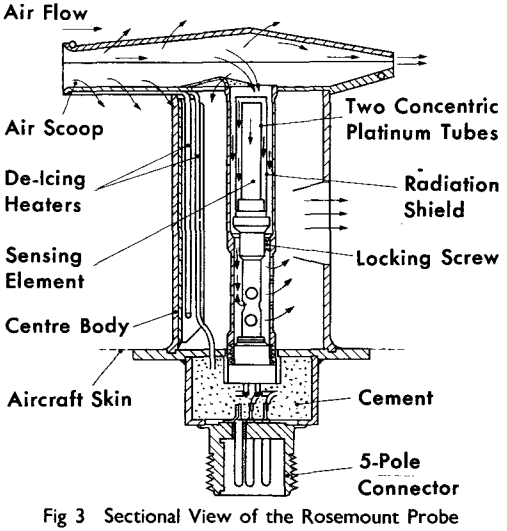


Fig 3 Sectional View of the Rosemount Probe

22. **Operation.** In flight, the air pressure within the probe is higher than that outside, thus boundary layer air is drawn off via bleed holes, as shown in Fig 3. In addition, the flow within the probe separates, part of the flow turning through a right angle before passing around the sensor. This produces particle separation which prevents the water droplets coming into contact with the sensor, and also prevents the sensor from damage by sand particles *etc*. This design permits the use of a delicate sensitive resistance element with fast response, and allows the use of de-icing heat to the exterior of the probe with negligible effect on the total temperature sensed.

23. **Limitations.** The Rosemount probe will operate accurately only with a satisfactory airflow over the probe. This condition is met under all conditions of flight, but when there is little or no airflow over the probe the probe body will be heated sufficiently to cause the element to sense temperatures in excess of ambient. The indicator will therefore over-read and, in cases where the heater is on for extended periods in zero airflow, the pointer will move past full-scale. However, the indicator movement will not be damaged by these conditions. The over-reading will persist for some time after commencement of taxying, until the airflow stabilizes the temperature of the probe. **Warning.** Operation of the probe heater on the ground, with no airflow over the probe, will result in the probe body temperature reaching a maximum of approximately 300°C.

THERMOMETER ERRORS

General Errors

24. The accuracy of indicated readings of remote indicating thermometers will depend largely on the accurate performance of the sensing elements, and the accuracy of the indicators and compensating devices. Appropriate chapters of AP 1275A and AP 112G quote calibration accuracy figures for

individual indicators; the thermometers are usually required to be accurate to within a small percentage of the full scale reading.

Outside Air Temperate Thermometer Errors

25. Aircraft thermometers used for measuring outside air temperature are subject to three types of error: instrument error, error due to environmental effects such as ice accretion or solar radiation on the thermometer probe, and errors due to kinetic or frictional heating.

26. **Instrument Error.** Instrument error is caused by imperfections in manufacture or operation of the instrument. The errors are usually small, and may often by allowed for by calibrating the instrument and fitting a correction card to the aircraft.

27. **Environmental Errors.** Solar heating effects can be reduced by mounting the sensitive element beneath the wing or fuselage (if a flat plate element), or by fitting it in a sun shield through which air is allowed to pass freely (if a probe mounted element). Protection of the element from ice accretion effects can be achieved by incorporating a heater, from which the sensing element is shielded, as in the Rosemount outside air temperature probe. Residual errors due to environmental effects cannot be calculated, and no corrections can be made for them.

28. **Heating Errors.** If the element is mounted in a probe projecting into the airflow, it will register the temperature of air which has been brought to rest. Its kinetic energy will have been converted into heat, and its temperature will not therefore be representative of that of the ambient air. If mounted on a flat plate flush with the fuselage, the element will be subject to the effects of frictional heating. Errors due to adiabatic or frictional heating can be calculated and allowed for by use of correction formulae: Either of two formulae may be used. These are:

$$T_1 = T_2 - k\left(\frac{V_T}{100}\right)^2 \quad \ldots\ldots\ldots\ldots \text{(1) or}$$

$$T_1 = \frac{T_2}{1 + 0.2\ k\ M^2} \quad \ldots\ldots\ldots\ldots\ldots\ldots \text{(2),}$$

where T_1 = Correct outside air temperature
T_2 = Indicated outside air temperature
V_T = TAS in mph
M = Mach number
k = Recovery factor of the temperature bulb.

In formula (1) temperatures are in °C, in formula (2) in °A 'k' is determined by flight testing, and its value is to be found in the operating instructions for the aircraft. The use of formula (1) with TAS in mph may be inconvenient, but a fair approximation may often be achieved by omitting 'k' and using TAS in knots. By coincidence, the error of not using 'k' cancels the error of using knots in place of mph for some aircraft. A full treatment of heating errors and the derivation of the formulae is given at the Annex.

HEATING ERRORS IN OUTSIDE AIR TEMPERATURE THERMOMETERS

CONTENTS

	Paras
Introduction	1
Stagnation Point Probes	2–5
Flate Plate Sensor	6
The Determination of 'k' Factor	7–8

Introduction

1. Heating errors in outside air thermometers are dependent upon the type of temperature sensing element mounting. The types are:

a. A probe protruding from the skin of the aircraft into the airflow. This is known as a stagnation point probe.

b. A flat plate let into the skin of the aircraft.

Stagnation Point Probes

2. Heating of a stagnation point probe is caused by the kinetic energy of the airflow being released as heat when the air is brought to rest by the probe. Indicated temperature is therefore in excess of true temperature. Assuming pressure changes are adiabatic the rise in temperature at the probe may be calculated from Bernoulli's equation for compressible flow.

3. If P, ϱ, V, T, and E represent pressure, density, velocity, temperature (°C) and internal energy in a streamline flow, and suffixes 1 and 2 represent free stream and stagnation point conditions, then:

$$\frac{P_1}{\varrho_1} + \tfrac{1}{2}\ V_1^2 + E_1 = \frac{P_2}{\varrho_2} + \tfrac{1}{2}\ V_2^2 + E_2 \quad \ldots\ldots (1)$$

But by the gas law, where R_a is the gas constant for air,

$$\frac{P_1}{\varrho_1} = R_a\ T_1, \text{ and } \frac{P_2}{\varrho_2} = R_a\ T_2.$$

In the free stream, $V_1 = V_T(TAS)$.

At a stagnation point $V_2 = 0$.

$\therefore$ from equation (1),

$$\tfrac{1}{2}\ V_1^2 = (E_2 - E_1) + R_a(T_2 - T_1) \quad \ldots\ldots (2)$$

If there is a change in temperature there must be a change in internal energy, as V_1 and R_a are constant for any set of conditions. Assuming the change in internal energy becomes heat, then:

$$(E_2 - E_1) = JC_v(T_2 - T_1) \quad \ldots\ldots (3),$$

where J = mechanical equivalent of heat, C_v = Specific heat of air at constant volume.

Combining (2) and (3),

$$\tfrac{1}{2}\ V_T^2 = (T_2 - T_1)(R_a + JC_v) \quad \ldots\ldots (4)$$

From thermodynamics:

$$R_a = J(C_p - C_v) \quad \ldots\ldots (5)$$

where C_p = specific heat of air at constant pressure.

Substituting (5) in (4)

$$\tfrac{1}{2}\ V_T^2 = J\ C_p(T_2 - T_1)$$

$$\therefore\ T_2 - T_1 = \frac{V_T^2}{2JC_p} \quad \ldots\ldots (6)$$

$\frac{1}{2JC_p}$ is a constant, say 'n',

$\therefore\ T_2 - T_1 = n\ V_T^2$.

When temperature is in °C and V_T is in mph, $n = 1.003 \times 10^{-4}$.

$$\therefore\ T_2 - T_1 = \frac{1.003}{10^4}\ V_T^2$$

$$\simeq \left(\frac{V_T{}^2}{100}\right) \quad \ldots\ldots (7)$$

4. The temperature rise can also be expressed in terms of Mach number (M).

$$V_T^2 = C^2\ M^2 \quad \ldots\ldots (8)$$

But also $C^2 = \frac{B,}{\varrho}$ where B is the adiabatic volume elasticity of air; and $B = \gamma s$,

where $\gamma = \frac{C_{p,}}{C_v}$ and s = free stream pressure.

$$\therefore C^2 = \frac{\gamma s.}{\varrho}$$

By the gas law $\frac{s}{\varrho} = R_a T$,

$$\therefore C_2 = \gamma R_a T \quad \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots (9)$$

Substituting (9) in (8),

$$V_T^2 = \gamma R_a T M^2.$$

Substituting in (6),

$$T_2 - T_1 = \frac{\gamma R_1 T_1 M^2,}{2 JC_p}$$ where temperatures are in °A.

Substituting for R_a from equation (5),

$$T_2 - T_1 = \frac{\gamma J(C_p - C_v) T_1 M^2}{2 JC_p}$$

$$= \frac{(\gamma - 1)}{2} T_1 M^2, \text{ since } \frac{C_p}{C_v} = \gamma.$$

$$\therefore \frac{T_2}{T_1} = 1 + \frac{(\gamma - 1)}{2} M^2.$$

γ for air = 1.4,

$$\therefore \frac{T_2}{T_1} = 1 + 0{\cdot}2\ M^2.$$

5. Both the above equations assume a full value of adiabatic heat rise. This is not realized in practice since no energy exchange is perfect. The theoretical value is reduced by a factor 'k' known as the recovery factor of the thermometer probe. The recovery factor is the fraction of the kinetic energy of the air stream which the sensing element recovers in reducing the velocity of the stream. 'k' is typically around 0.8. The full correction formulae are:

$$T_1 = T_2 - k \left(\frac{V_T}{100}\right)^2$$; where temperatures are in °C or °A, and

$$T_1 = \frac{T_2}{1 + 0.2\ kM^2}$$, where temperatures are in °A.

Flat Plate Sensor

6. The flat plate sensor is unaffected by adiabatic heating as it does not protrude in the airflow. However the passage of air past a flat plate does heat it due to frictional effects. By coincidence the heat rise approximates to that generated at a stagnation point probe due to adiabatic heating. For this reason the same correction formulae are used for the flat plate and the stagnation point sensors.

The Determination of 'k' Factor

7. 'k' factor is determined empirically for each aircraft type from flight test data. The value for a particular aircraft type is stated in the aircraft operating data.

8. 'k' factor is assumed to be constant. However variation does occur due to changes of heat transfer from the sensing element. Large variations (up to 50%) can occur due to direct solar heating, icing or flight through cloud or rain.

Printed in the UK for HMSO
by Hobbs the Printers of Southampton
(1562/86) Dd8908616 8/86 G436

PART 1
SECTION 2
HEIGHT

Chapter

1 **Introduction to Barometric Height Measurement**

2 **Simple, Sensitive and Cabin Altimeters**

3 **Servo-Assisted Altimeters**

CHAPTER 1

INTRODUCTION TO BAROMETRIC HEIGHT MEASUREMENT

CONTENTS

	Paras
Introduction	1
THE ATMOSPHERE	
General	2–6
Standard Atmospheres	7–13
PRINCIPLES OF HEIGHT MEASUREMENT	
Basic Principles	14–15
Simple Altimeter	16–17
Sensitive Altimeter	18–20
Servo-Assisted Altimeter	21–22
ERRORS IN PRESSURE ALTIMETERS	
Types of Error	23–25
Instrument Error	26
Pressure Error	27–30
Time Lag	31
Hysteresis Error	32
Barometric Error	33–36
Temperature Error	37–39
Blockages	40–41

Introduction

1. Pressure altimeters are instruments which indicate aircraft height above a selected pressure datum. They operate on the principle that air pressure decreases with height. They are in fact aneroid barometers graduated to indicate height instead of pressure.

THE ATMOSPHERE

General

2. The atmosphere is a relatively thin layer of gases surrounding the Earth, becoming more diffuse with increasing distance from the Earth's surface. Water vapour is present in variable amounts, particularly near the Earth's surface.

3. The atmosphere can be divided into a number of layers each with a tendency to a particular temperature distribution. These layers are shown in Fig 1. The figures are arbitrary and refer to certain standard atmospheres which will be defined later.

4. The lower layer, the troposphere, extends to a height known as the tropopause,

and the outstanding characteristic of this layer is the fairly regular decrease of temperature with height. The tropopause tends to become lower towards the Earth's poles and higher towards the equator. The region above the tropopause is known as the stratosphere, extending up to the stratopause. In this region the temperature is assumed to remain more or less constant. Above the stratosphere there is the chemosphere, a region in which the temperature tends to increase with height.

5. **Pressure Lapse Rate.** As height increases, pressure decreases, since the weight of air above decreases. The decrease in pressure is not, however, proportional to the increase in height because the density of the air varies with height, as does the value of *g*, although to a lesser extent. It is possible to deduce an expression for the pressure lapse rate at a constant temperature and thus establish a relationship between pressure and height. A practical approximation for the lower levels of the atmosphere is that a decrease in pressure of one millibar is equal to an increase in height of 30 ft.

6. **Temperature Lapse Rate.** Temperature does not remain constant but varies with height in a complex manner. The temperature lapse rate depends on the humidity of the air, and is itself a function of height. This variation greatly affects the relationship between pressure and height. To calibrate an altimeter to indicate barometric height it is necessary to make some assumptions as to the temperature structure of the atmosphere. The relationship can be expressed in mathematical form for each of the various layers of the atmosphere and the instrument can then be calibrated accordingly.

Standard Atmospheres

7. **Definition.** A standard atmosphere is an arbitrary statement of conditions which is accepted as a basis for comparison of aircraft performance and calibration of aircraft flight instruments. Because of the extreme variability of conditions in the atmosphere, the standard can only represent the average conditions over a limited area of the globe. Most standards so far adopted are related primarily to the mean atmospheric conditions in temperate latitudes of the northern hemisphere.

8. **Isothermal Atmosphere.** The early standard atmosphere assumed the air to be dry and remain at a constant 10°C, hence the name isothermal. It only gave a rough approximation of altitude in the lower levels of the atmosphere and is now out of use.

9. **ICAN Standard Atmosphere.** In 1924 the International Committee on Air Navigation (ICAN) proposed an international standard atmosphere which was widely adopted. The assumed characteristics of the atmosphere were as follows:

a. The air is dry and its chemical composition is the same at all altitudes.

b. The value of *g* is constant at 980.62 cm/sec^2.

c. The temperature and pressure at mean sea level are 15°C and 1,013.2 millibars.

d. The temperature lapse rate is 1.98°C per 1,000 ft up to a height of 36,000 ft above which the temperature is assumed to remain constant at – 56.5°C.

On the basis of these assumptions, the ICAN formulated equations relating to the variation of pressure with height in the troposphere and in the stratosphere.

10. **ICAO Standard Atmosphere.** Between 1950 and 1952 the International Civil Aviation Organisation (ICAO) proposed and adopted another standard atmosphere. The assumptions made were the same as those of the ICAN atmosphere except that mean sea level pressure was taken as 1,013.25 mb and the value of *g* as 980.665 cm/sec^2. Equations were then formulated for determining height from barometric pressure which were valid up to 65,800 ft. The differences between calibration values for the ICAN and the ICAO atmospheres are insignificant be-

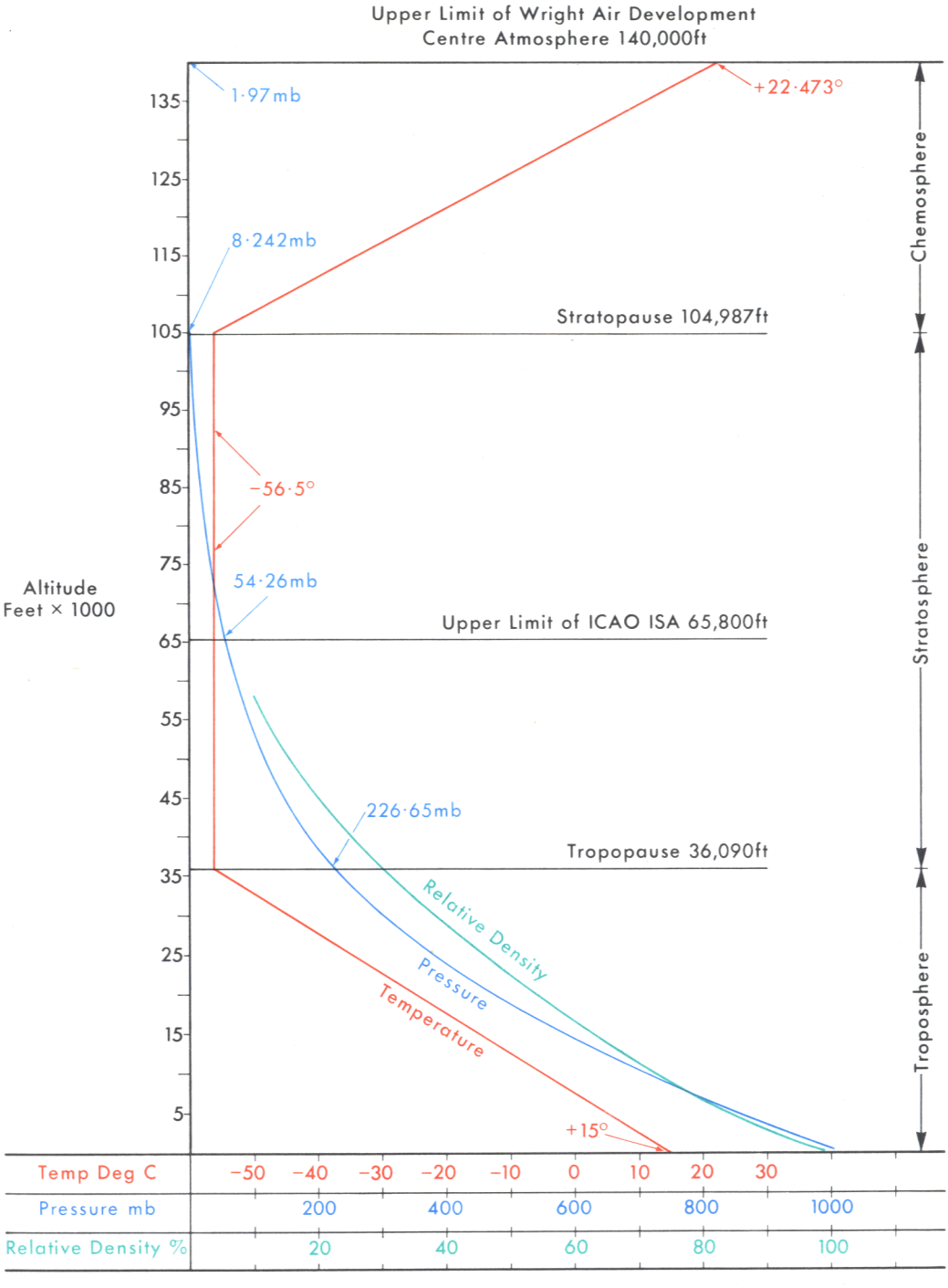


Fig 1 Structure of the Atmosphere

low 63,000 ft. The ICAO standard atmosphere is taken as the ISA (International Standard Atmosphere) and is referred to as ISA in the Flight Information Handbook.

11. **WADC Standard Atmosphere.** The advance in aircraft performance and the introduction of missiles showed the need for a standard atmosphere whose range extended beyond the 65,800 ft limit of the ICAO atmosphere. The Wright Air Development Centre (WADC) atmosphere was introduced to supplement the ICAO atmosphere to enable instruments to be calibrated up to 140,000 ft and the asumptions made are approximately the same as those of the ICAO atmosphere, with the exception that from the tropopause upwards the variation of temperature is purely arbitrary. Equations based on the WADC atmosphere have been deduced for the troposphere, the stratosphere and the chemosphere, up to 140,000 ft.

12. **ARDC Standard Atmosphere.** In order to give a further extension of the range of pressure altimeters the Air Research and Development Command (ARDC) standard atmosphere was introduced in 1953. Once again the assumed values are very similar to those of ICAO but slightly different equations have been evolved which make it possible for instruments to be calibrated up to 1,850,870 ft. Since both the WADC and the ARDC standard atmosphere use the ICAO pressure laws as a basis, the tables showing variation of pressure and temperature with altitude at lower altitudes are very similar. It is only at altitudes above 80,000 ft that the figures begin to diverge.

13. **Civil Atmospheres.** The WADC and ARDC atmospheres are accepted by the RAF, USAF, CDF and effectively by all NATO Air Forces. Civil aviation has its own standard atmosphere, ICAO 64. This atmosphere provides for heights up to 2,320,000 ft and is similar to WADC and ARDC up to 80,000 ft where significant differences begin to appear.

PRINCIPLES OF HEIGHT MEASUREMENT

14. A pressure altimeter consists basically of a thin corrugated metal capsule which is evacuated and sealed. It is prevented from collapsing completely by means of a leaf spring, or in some cases, by its own rigidity. Equilibrium is maintained between the pressure of the atmosphere on the faces of the capsule and the tension of the spring. Changes in atmospheric pressure alter this state of equilibrium and the capsule will expand for a decrease in pressure, or contract for an increase in pressure, until a new balance is obtained. The movement of the capsule face is transmitted to a pointer moving over a scale graduated according to one of the standard atmospheres.

15. Changes of pressure are measured relative to a datum, usually that of mean sea level. To allow for variations in actual mean sea level pressure from that assumed by the standard atmosphere, a method of altering the datum pressure is required (para 34).

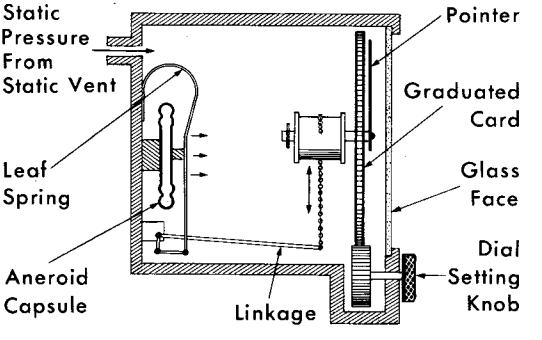


Fig 2 Simple Altimeter – Schematic

Simple Altimeter

16. Fig 2 shows a simple altimeter. The simple altimeter consists of a single capsule mounted in an airtight case, the case being fed with static pressure from the aircrafts static tube or vent. As the aircraft climbs, the pressure in the case falls, allowing the

spring to pull the capsule faces apart. Conversely a decrease in height compresses the capsule faces. This linear movement is magnified and transmitted to a pointer moving over a card graduated in feet in accordance with one of the standard atmospheres.

17. **Datum Setting.** The altimeter, normally calibrated according to the ICAN law, will indicate zero at a pressure of 1013.2 mb and a temperature of 15°C, when the lubber line, visible through a hole in the face of the dial, is aligned with the index marks on each side of the hole. Adjustment of this datum is made by moving the pointer itself by means of a small knob on the side of the dial. If the mechanism is set so that airfield elevation is indicated by the pointer before take-off, it will thereafter indicate altitude above mean sea level, provided the prevailing sea level pressure does not change. If the pointer is set to zero before take-off, it will read height above the airfield, again provided the surface pressure at the airfield remains the same.

Sensitive Altimeter

18. The sensitive altimeter is designed for more accurate height measurement than the simple altimeter; the principle of operation is similar.

19. **Construction.** The sensitive altimeter usually utilizes two or more evacuated capsules to give greater sensitivity for small changes in pressure. Multiple pointers, operated by an arrangement of gear wheels, are provided to enable larger scales to be used. One pointer rotates every 1,000 ft, a second every 10,000 ft and a third every 100,000 ft. A bi-metallic strip inserted between the capsule and the shaft which transmits capsule movement to the gearing provides temperature compensation in the instrument. This temperature compensation should not be confused with the error caused by flying from Warm to Cold air and *vice versa.*

20. **Datum Setting.** A sensitive altimeter has a millibar scale, which makes it possible to set the actual mean sea level pressure as the datum from which height is determined. Movement of the pressure setting knob rotates the millibar scale and also the pointers of the instrument to give the correct indication of altitude relative to the mean sea level pressure set. If the knob is turned so the pointers indicate airfield elevation before take-off, then the millibar scale will read actual sea level pressure. Airfield level pressure may be set on the millibar scale, in which case the altimeter will read zero on the ground and in flight will read height above the airfield. The datum pressure may be reset during flight to allow for changes in pressure on the ground.

Servo-Assisted Altimeter

21. The chief limitation of the directly operated capsule altimeter is its increasing inaccuracy with increasing height above altitudes of approximately 60,000 ft. At these altitudes the changes in pressure for a given height change are very much smaller than at ground level. For example, the pressure difference from sea level to 5,000 ft is 170.18 mb (ICAO) which gives a considerable capsule deflection. The pressure difference between 60,000 ft and 65,000 ft is only 15.32 mb which will give a very much smaller deflection. The effect of any frictional forces in the mechanism is very much magnified at high altitudes.

22. The servo-assisted altimeter was designed to relieve the capsule of the work required to drive the mechanical linkage. Changes of barometric pressure are detected by the deflection of an evacuated capsule, but instead of mechanical amplification, the capsule assembly is made part of a servo system. In this way the system is very sensitive to small changes in pressure and the servo provides the power to drive the indicating mechanism. The result is greater accuracy at all altitudes and a very much increased motive power in the system.

ERRORS IN PRESSURE ALTIMETERS

Types of Error

23. Altimeter errors are considered under two categories. Instrument and installation errors and errors caused by non-standard atmospheric conditions.

24. The errors inherent in the instrument and installation are as follows:

a. Instrument error.
b. Pressure error.
c. Time lag.
d. Hysteresis error.

25. The errors due to variations in the atmospheric conditions are as follows:

a. Barometric error.
b. Temperature error.

Instrument Error

26. Since capsule movements are of necessity greatly magnified it is impossible to avoid completely the effect of small irregularities in the mechanism. Certain instrument tolerances have to be accepted, and errors generally increase with height. Any residual error may be applied as a correction, but it is usually insignificant.

Pressure Error

27. Pressure error arises because the true external static pressure is not accurately transmitted to the instrument. A false static pressure can be created in the vicinity of the pressure head due to the air flow over it.

28. Pressure error is negligible at low altitudes and speeds, especially when a static vent in the fuselage is used instead of slots in the pressure head. The error becomes significant at high speeds.

29. For aircraft travelling at very high speeds it has been found necessary to design a pressure head to supply both pitot and static pressure and mount it at the foremost part of the aircraft. As the speed of the aircraft rises, a bow wave approaches the pressure head causing an increasing error in the recorded static pressure. At a speed close to Mach 1 the bow wave passes over the static slots. When this occurs the error falls rapidly to a small value which depends on the shape of the pressure head.

30. Correction for pressure error takes the form of a correction to be applied to the indicated height and must be determined for each aircraft type by calibration. Air data computers are designed to compensate for pressure error.

Time Lag

31. Since the response of the capsule and linkage is not instantaneous, the altimeter needle lags whenever height is increased or decreased rapidly causing an under-read on climbs and an over-read on descents. The latter error could be dangerous and should be allowed for in rapid descents. The amount of lag is proportional to the rate of descent. Time lag is virtually eliminated in the servo-assisted altimeter.

Hysteresis Error

32. A capsule under stress exhibits an imperfect elastic response. The capsule will have a different deflection for a given pressure change according to whether height is increasing or decreasing. This effect varies with particular conditions and is difficult to predict. It is most noticeable after sharp climbs and descents.

Barometric Error

33. Barometric error occurs when the actual sea level barometric pressure differs from that assumed by the standard atmosphere in accordance with which the altimeter has been calibrated.

34. Barometric error is simply corrected by making allowance for the difference in pressure. In the simple altimeter this is done by setting the pointer to the airfield elevation before take-off and in the more sensi-

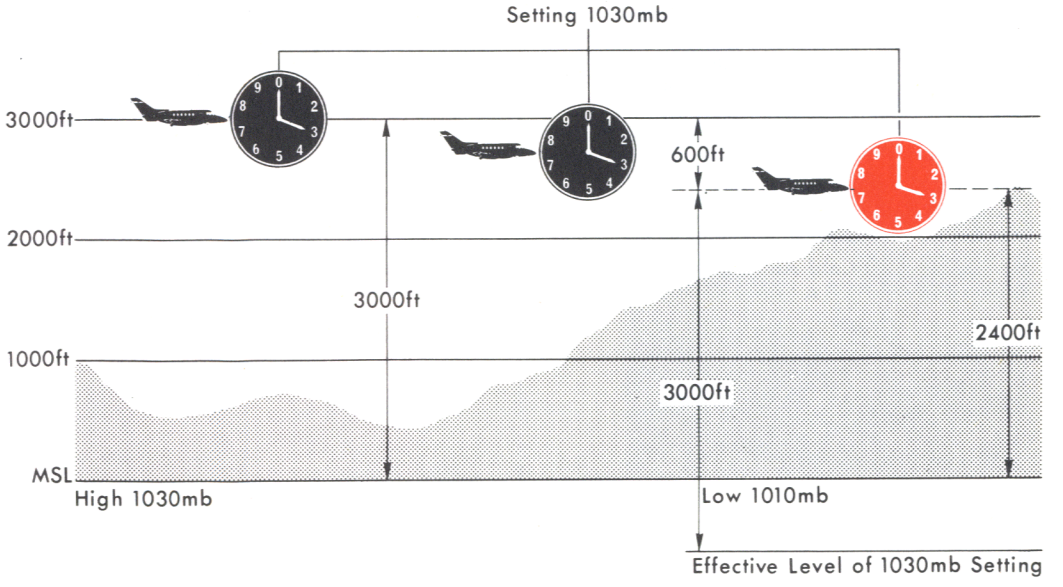


Fig 3 Effect of Barometric Error

tive types by setting the actual sea level pressure on the millibar scale. The millibar scale also permits changes of datum pressure to be made in flight when an aircraft flies over an area where the mean sea level pressure differs from that set. There is no method of doing this with the simple altimeter.

35. The effect on the altimeter in an aircraft flying from an area of high pressure to an area of low pressure is shown in Fig 3. In this case the aircraft flies from an area where the MSL pressure is 1030 mb to one where the MSL pressure is 1010 mb but retains a setting of 1030 mb on the altimeter. It can be seen that the datum is effectively lowered so that the altimeter reads high. Similarly in an aircraft flying from low pressure without a change of datum, the altimeter would read low. In summary, from HIGH to LOW the altimeter reads HIGH, and conversely from LOW to HIGH the altimeter reads LOW.

36. **Orographic.** Differences from the standard atmosphere pressure may occur when air is forced to rise/descend over hills and mountains. For known reasons low pressure tends to occur in the lee of mountains, with high pressure on the windward side. In addition, vertical movement of air created by hills and mountains can produce a change in temperature from ISA calculations inducing further errors in altimeter readings. Generally, when temperatures are less than ISA an aircraft will be lower than the altimeter reading.

Temperature Error

37. Temperature error, like barometric error, arises whenever the atmospheric conditions differ from those assumed by the standard atmosphere used to calibrate the instrument. The ICAO standard atmosphere assumes a temperature lapse rate of 1.98°C per 1,000 ft up to 36,090 ft, and then a constant temperature of – 56.5°C. If the actual temperature lapse rate differs from the assumed one, as it very often does, then the indicated height will be incorrect.

38. The pressure at a certain level over a column of cold air is less than the pressure at

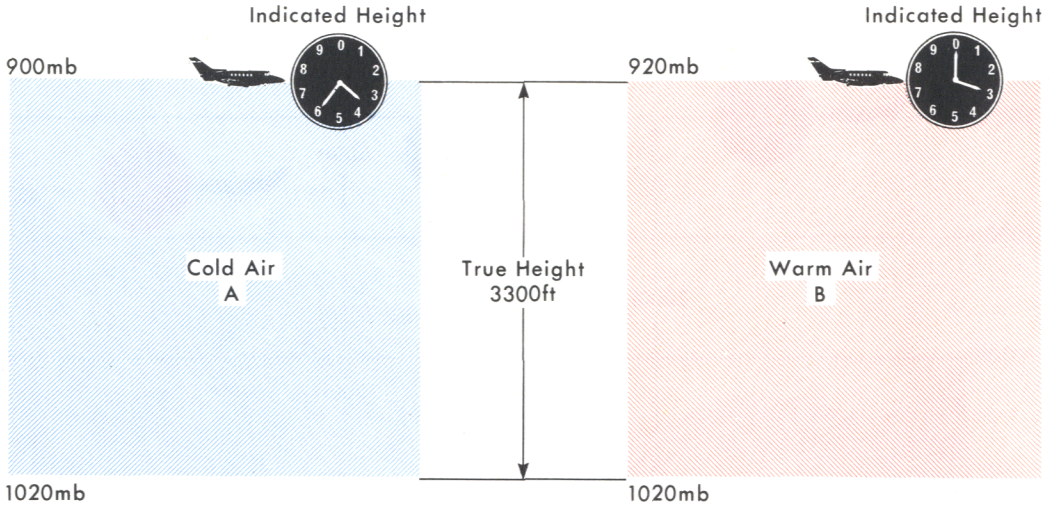


Fig 4 Effect of Temperature Error

the same level over a column of warm air assuming the same surface pressures, because cold air is more dense than warm air. Therefore at a given level and with equal surface pressures, an altimeter will read high when the temperature structure is colder than the assumed one, and low when the temperature structure is warmer than the assumed one. This is illustrated in Fig 4. The error is approximately 4 ft/1000 ft for each 1 deg C of difference from ISA.

39. The correct height may be obtained from the indicated height by the use of a specially designed computer such as the DR Computer Mk 4. At high altitudes differences of 1,000 ft and over between absolute and indicated heights are common. A temperature correction table for Obstacle Approach Clearance is provided in the Flight Information Handbook.

Blockages

40. Should the static tube or vent become blocked in any way, the pressure within the instrument case will remain constant and the altimeter will continue to register the height indicated when the blockage occurred. Some aircraft have an emergency source of supply, usually an internal vent. The static pressure inside an aircraft differs to some extent from that of the external atmosphere, and is influenced by the opening of windows, ventilators, *etc,* so that a different correction for pressure error is necessary. This is normally given in the Aircrew Manual for the type of aircraft.

41 Where there is no alternative source of static pressure the altimeter may be brought into operation again by the careful breaking of the face of one of the other pressure instruments of lesser importance, *ie* the vertical speed indicator or the machmeter. This will allow internal static pressure to the altimeter. Readings will be subject to large errors.

Note: If this method of obtaining an alternative source of static pressure is used in aircraft with pressurized cabins it is imperative that the pressurization system is switched off.

Printed in UK for HMSO, Dd 8907781 11/85 NB33998 **3347**

CHAPTER 2

SIMPLE, SENSITIVE AND CABIN ALTIMETERS

CONTENTS

SIMPLE ALTIMETERS

	Paras
Mk 16 and Mk 17 Altimeters	1–3

SENSITIVE ALTIMETERS

Mk 19, Mk 20 and Mk 23 Altimeters	4–8
Mk 26 and Mk 27 Altimeters	9–12
Mk 28 Altimeters	13–17
Mk 31 Altimeter	18–24

CABIN ALTIMETERS

General	25
Description	26
Accuracy	27

DATA SUMMARY

Data Table	28

SIMPLE ALTIMETERS

Mk 16 and Mk 17 Altimeters

1. **Description.** The Mk 16 and Mk 17 Altimeters are simple altimeters which operate as described in Chapter 1. They have a single pointer moving over a scale calibrated to the ICAN standard atmosphere. The Mk 17 differs from the Mk 16 in that it has a double capsule and a slightly different linkage system. Fig 1 shows the Mk 17 Altimeter face.

2. **Datum Pressure Setting.** The datum setting lubber line can be seen through a hole in the upper left quadrant of the instrument face. With the line set against the datum marks, by means of the setting knob at the bottom of the dial, the instrument will indicate height above mean sea level for a standard mean sea level pressure of 1013·2 mb. Allowance can be made for actual

Fig 1 Mk 17 Altimeter

pressure by setting airfield elevation before take-off.

3. **Accuracy.** The Mk 16 and Mk 17 Altimeters are accurate to ±100 ft at 0 ft and ±1,000 ft at 35,000 ft.

SENSITIVE ALTIMETERS

Mk 19, Mk 20 and Mk 23 Altimeters

4. **Description.** The Mk 19, Mk 20 and Mk 23 Altimeters are of the sensitive type, as described in Chap 1, designed to register small changes in altitude. They have the three pointer type of display so that height can be read to within 25 ft. They are calibrated to the ICAN standard atmosphere.

Fig 2 Mk 19B Altimeter

5. **Datum Pressure Setting.** The barometric pressure setting mechanism operates a set of veeder counters, displayed in a window in the lower half of the dial. The desired setting is made using the knurled knob at the bottom of the instrument.

6. **Mk 19 Series Altimeters**

a. *Mks 19A and 19B Altimeters.* The Mk 19B incorporates a low altitude warning sector which appears in a window just below the centre of the dial when the height is 16,000 ft or lower. The sector flag has stripes in black and white and gives the pilot a clear warning of low altitude in rapid descents. The Mk 19B is shown in Fig 2. The earlier type Mk 19A is the same as the Mk 19B but does not have the low altitude sector flag.

b. *Mk 19F Altimeter.* The Mk 19F utilizes a scroll type of indication in place of the 10,000 ft pointer. The device consists of a disc situated behind the dial and driven by the pointer gearing. A sector of the disc is painted white and the remainder black, the dial having a spiral type aperture from zero to 60,000 ft. The leading edge of the white sector corresponds to the 10,000 ft pointer on the other types of the Mk 19 series. As the indicated altitude increases the curved slot fills with white and vice versa.

c. *Vibrator Assembly.* In some installations a vibrator is used to assist the rate of response, especially in the upper altitude ranges of the capsule operated Mk 19 series when the inherent vibration of the aircraft is very small. The vibrator is clamped to the rear of the altimeter by means of a screw-tightened metal strap. The vibrator unit consists of a body housing a magnet, coil, armature and diaphragm. Between the diaphragm and the altimeter case is a rubber pad. Application of 115 V, 400 Hz single phase voltage to the coil causes a solenoid action of the armature proportional to the frequency of the applied current. The resulting vibrations are transmitted to the mechanism of the altimeter; the diaphragm and rubber pad being held in close contact with the rear of the altimeter case by the metal strap.

The Mk 19 series altimeters are accurate to ±70 ft at 0 ft and ±900 ft at 60,000 ft respectively.

7. **Mk 20 Series Altimeters.** The Mk 20 altimeter is fundamentally the same as the Mk 19A, with no low altitude warning sector. There are slight constructional differences and different working ranges. The Mk 20A has an upper limit of 35,000 ft but later versions go up to 50,000 ft. The altimeter is accurate to ±400 ft at 30,000 ft and ±750 ft at 50,000 ft.

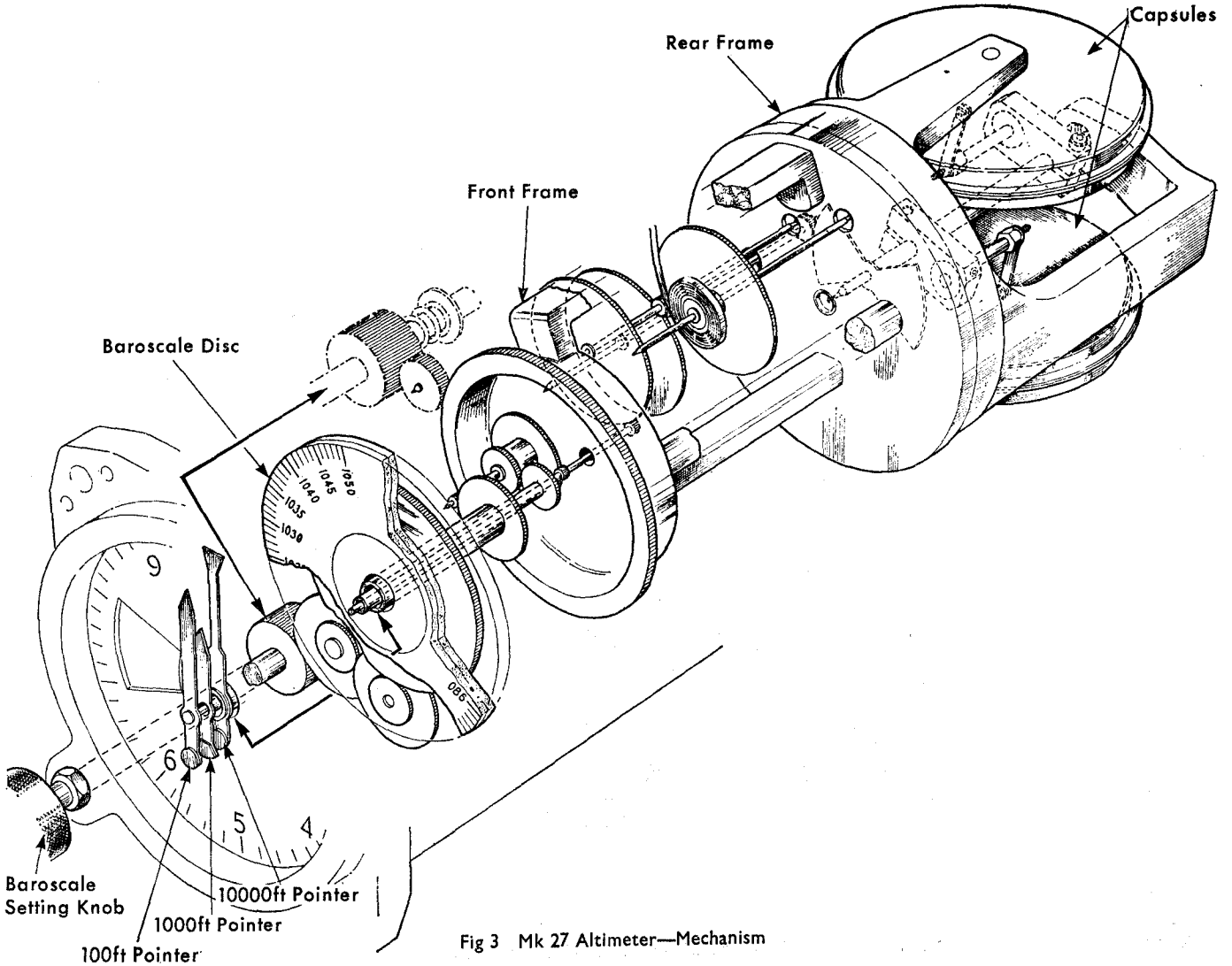


Fig 3 Mk 27 Altimeter—Mechanism

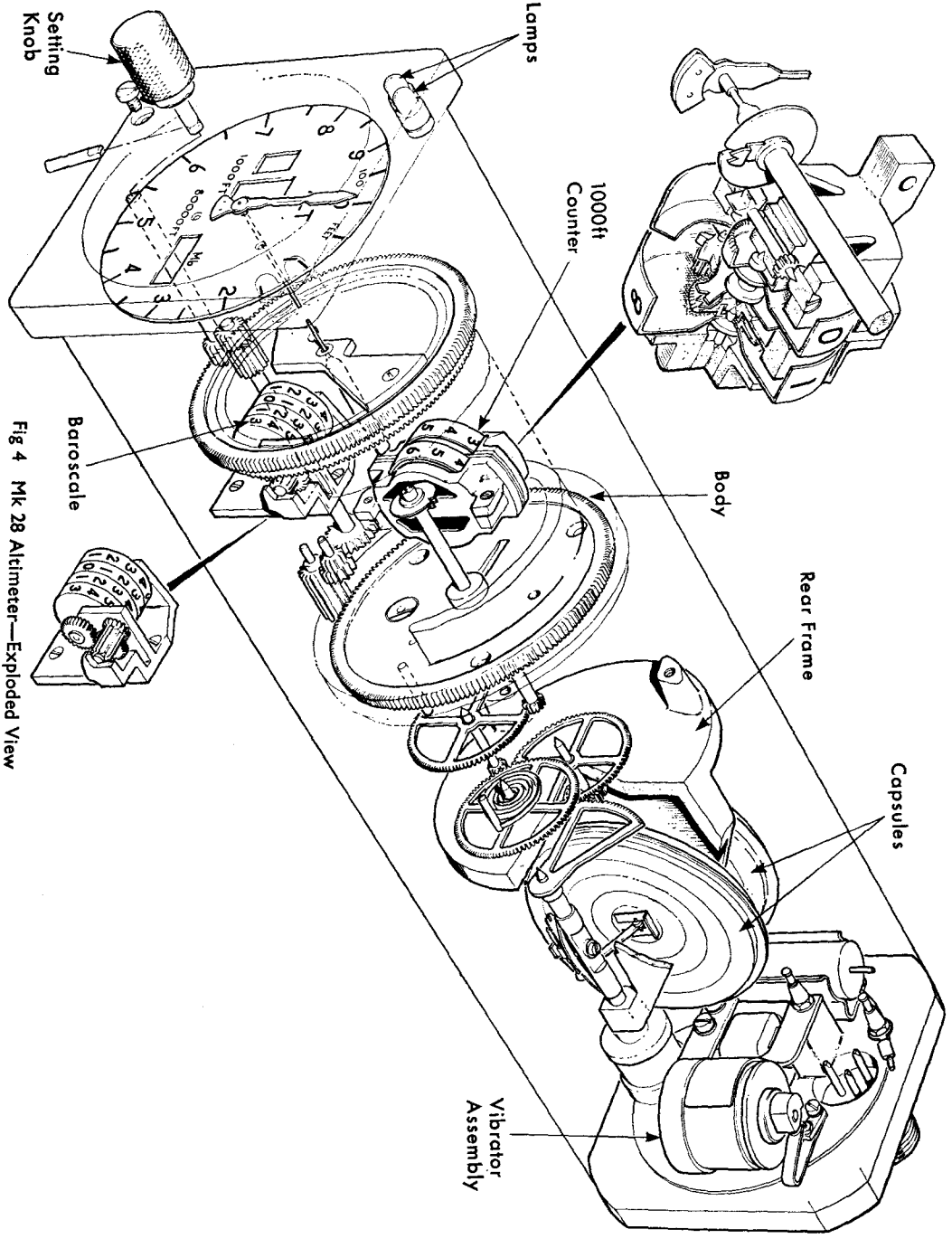


Fig 4 Mk 28 Altimeter—Exploded View

8. **The Mk 23 Series Altimeters.** The Mk 23 Altimeter has a range of 0 to 80,000 ft. It is similar to the Mk 19B shown in Fig 2. The low altitude warning sector appears at 16,000 ft and completely fills the window at 10,000 ft. This instrument is accurate to ±70 ft at 0 ft, ±600 ft at 40,000 ft and ±1,500 ft at 80,000 ft.

Mk 26 and Mk 27 Altimeters

9. The Mk 26 and Mk 27 altimeters have identical mechanisms, however the Mk 27 has a differently shaped front to the case and incorporates integral lighting. They are both capsule operated, direct reading, three pointer instruments with a range of −2,000 ft to 60,000 ft. They are calibrated to the ICAO standard atmosphere.

10. **Description.** The altimeters are housed in a 2¼ inch diameter sealed case with a static pressure connector at the rear. Two capsules are connected directly, with suitable gearing, to the three pointer display, *see* Fig 3.

11. **Datum Setting.** The baroscale setting, which is set by the knob in the lower left corner of the instrument, is indicated by a disc graduated from 950 to 1,050 millibars. This is located in the 9 o'clock position on the dial face.

12. **Accuracy.** These instruments are accurate to ±50 ft at 0 ft, ±175 ft at 10,000 ft, ±600 ft at 40,000 ft and ±2,000 ft at 60,000 ft.

Mk 28 Altimeters

13. The Mk 28 Series are sensitive altimeters having a single pointer and a two-digit counter presentation. The instruments of this series have similar mechanism but vary in instrument lighting and dial markings. They operate over the range −1,000 ft to 80,000 ft and are calibrated to the WADC standard atmosphere.

14. **Operation.** The altimeter is housed in a 3-inch square case which is sealed except for the static pressure connection at the rear. Two capsules drive, through suitable gears, the pointer which makes one revolution per 1,000 ft. Thousands and tens of thousands of feet are indicated on a two-digit counter. The counter drive mechanism, *see* Fig 4, consists of an arm and a disc each having a "V" shaped slot which engages a counter drive pinion. Because the counter is operated by the pneumatic forces of the capsules only, it is not possible to make the digit change an instantaneous operation. The arm mechanism is arranged to change the counter by one digit whilst the pointer moves from 9 to 0 on the dial, the disc holds the counters steady at other times.

Note: Because the action of digit changing imposes an additional load on the capsule mechanism, there is normally some hesitation of the pointer during the changeover period. The hesitation becomes more noticeable at high altitudes due to the decreased driving force of the capsules. The hesitation is to be expected and is not an unserviceability.

15. **Datum Setting.** The millibar counter movement is controlled by the rotation of the setting knob located at the lower left corner of the instrument, and the reading set is displayed on counters through a window in the lower right quadrant of the dial.

16. **Vibrator Assembly.** This series of altimeter is fitted with an integral vibrator assembly. It has been found that an induced vibration in the mechanism reduces the initial opposition to motion of the moving parts, particularly the digit counter, and reduces the frictional lag in the system. This altimeter is more sensitive and more accurate than the other non-servo-assisted altimeters because of this feature.

17. **Accuracy.** This instrument is accurate to ±30 ft at 0 ft, ±100 ft at 10,000 ft, ±260 ft at 40,000 ft and ±600 ft at 60,000 ft.

Mk 31 Altimeter

18. The Mk 31 Altimeter is a sensitive capsule operated instrument displaying pressure altitude by means of a two digit counter, a drum and pointer. A coded electrical output of altitude is provided by two digitizers. The encoded output is fed to a code converter which converts the digitized code into ICAO Gillham code for use by an IFF transponder. The instrument operates over the range —1,000 ft to +45,000 ft. An internal view of the altimeter is shown at Fig 5.

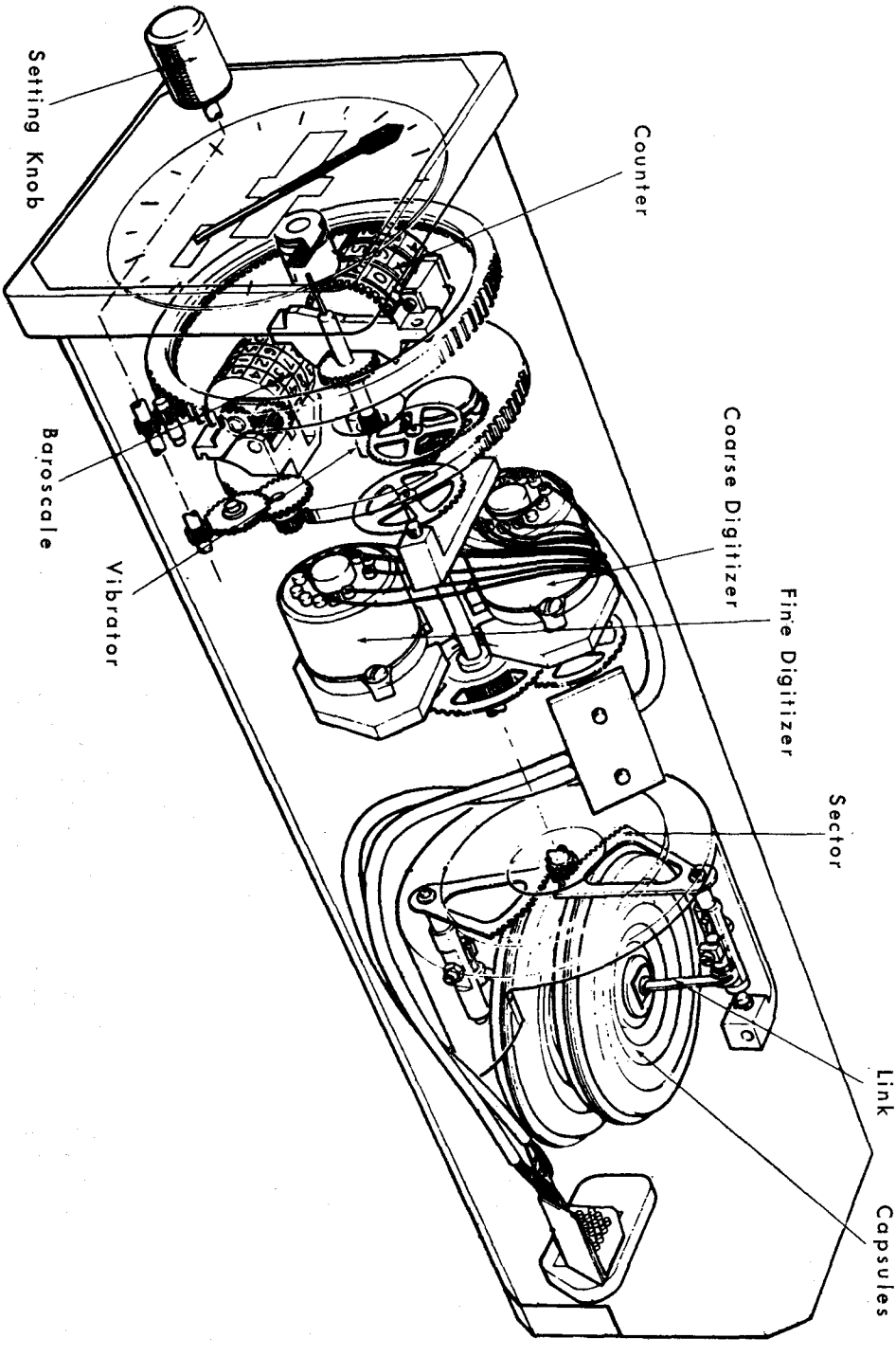


Fig 5 Mk 31 Altimeter—Internal View

19. **Operation.** The altimeter is housed in a 3¼ inch square case which is sealed except for the static pressure connector at the rear. The mechanism comprises a conventional two capsule operated unit connected through gearing to the pointer and drum counter assembly. The gearing also engages two low torque digitizers. The gearing drives the pointer and also the drum so that both indicate hundreds of feet. The drum in turn is coupled to a two digit counter which registers in thousands and tens of thousands of feet. Counter changes, at every thousand feet, take place when the pointer traverses the intervals between 900 and 1,000 ft. The tens of thousands feet counter is marked with diagonal black and white hatching in place of the conventional zero figure to bring attention to altitudes below 10,000 ft.

20. **Datum Setting.** The millibar counter movement is adjusted by a barometric setting knob located at the lower left corner of the instrument and the reading set is displayed on counters through a window in the lower right quadrant of the dial. Rotation of the setting knob causes the pointer to rotate and also alters the millibar counter setting. The digitized height signal is not affected however, as this is always referenced to a datum pressure of 1013·25 mb.

21. **Digitizer Assembly.** Two digitizers, one coarse and one fine, are mounted on the mechanism body, a pinion on each being engaged by common gearing from the main shaft. Each digitizer rotor consists of magnetic laminations set in a rotor shaft mounted in jewelled bearings. Each digitizer receives a pulse train from an oscillator within the externally mounted code converter. The pulses are applied to the stator of each digitizer where they are modified according to the position of the magnetic laminations within the rotor relative to the stator. Thus the angular position taken up by each rotor and shaft relative to its stator determines the digital signal produced by the assembly. The reply pulses are fed to the code converter where any ambiguity between the coarse and fine digitizers is resolved by means of a secondary interrogation pulse derived from the fine digitizer reply pulse. The total reply pulse is modified from a pure binary pattern to ICAO Gillham code for use by an IFF transponder.

22. **Code Converter.** The code converter changes the induction digitized altitude information received from the altimeter into ICAO Gillham code. The unit consists of an electronics assembly mounted within a 3¼-inch square case. A local oscillator within the converter provides an interrogating pulse train to the two digitizers and the pulse groups received from the digitizers are changed into a continuous parallel binary output in accordance with the Gillham code. This binary output is stored in quad latch circuits which form a nine bit parallel store. The store is updated once during every pulse cycle (at a prf of 20-100 Hz). The least significant bit represents a one hundred foot band of altitude. The binary state of the store is used by an IFF transponder for automatic altitude reporting.

23. **Vibrator Assembly.** The altimeter is fitted with an integral vibrator assembly mounted on the mechanism frame. A 28 V DC supply is fed to a coil which drives the spring/iron mass vibrator. This supply also energizes a warning flag solenoid and the oscillator in the code converter. In the event of a power supply failure, the solenoid will be de-energized to allow the warning flag to appear in an aperture in the dial, at the same time the store in the code converter will revert to a recognized fail safe pattern of ALL ZEROS.

24. **Accuracy.** This altimeter is accurate to ±35 ft at 0 ft, ±100 ft at 20,000 ft and ±230 ft at 40,000 ft.

CABIN ALTIMETERS

General

25. There are numerous types of cabin altimeters designed for use in pressurized aircraft. They indicate cabin pressure, in terms of altitude, so that the crew may regulate the oxygen supply accordingly. These altimeters are generally of the simple type, having one pointer moving over a scale graduated in tens of thousands of feet (or in thousands of feet in some cases). They are calibrated according to the standard atmospheres given in Chap 1.

Description

26. A sectional view of a typical example of a cabin altimeter is shown in Fig 5. It consists of a multiple stack of capsules driving, through gears, a single pointer. The case is vented to cabin pressure. A change in the cabin pressure causes the capsules to expand or contract in the same way as normal altimeters, thus turning the pointer.

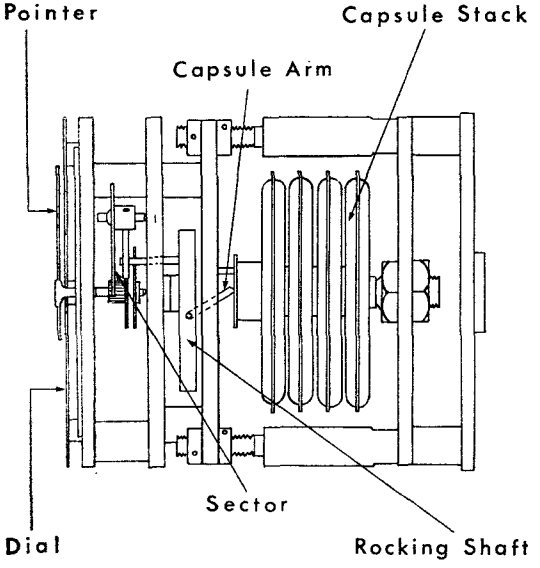


Fig 6 Sectional View of a Cabin Altimeter

Accuracy

27. Cabin altimeters are normally simple instruments. They usually do not have error compensating devices although some do have compensation for temperature. Generally all suffer, in some form or other, from the errors listed in Chap 1, paras 23 to 28, *ie* instrument and installation errors or errors due to variation in atmospheric conditions from the standard atmosphere to which they are calibrated. Taking into account these errors and the fact that cabin pressure, in terms of altitude, will not normally fall below the equivalent of 30,000 ft, the instrument should be accurate to better than $\pm$500 ft throughout this range.

DATA SUMMARY

Data Table

28. A summary of the pertinent data of the altimeters included in this chapter is given in Table 1.

Type	Calibration	Operating Range	Accuracy
Mk 16	ICAN	0–35,000 ft	± 100 ft at 0 ft ± 1,000 ft at 35,000 ft
Mk 17	ICAN	0–35,000 ft	± 100 ft at 0 ft ± 1,000 ft at 35,000 ft
Mk 19	ICAN	0–60,000 ft	± 70 ft at 0 ft ± 900 ft at 60,000 ft
Mk 20	ICAN	0–35,000 ft	± 70 ft at 0 ft ± 400 ft at 30,000 ft ± 750 ft at 50,000 ft
Mk 23	ICAN	0–80,000 ft	± 70 ft at 0 ft ± 600 ft at 40,000 ft ± 1,500 ft at 80,000 ft
Mk 26	ICAO	−2,000 to +60,000 ft	± 50 ft at 0 ft ± 175 ft at 10,000 ft ± 600 ft at 40,000 ft ± 2,000 ft at 60,000 ft
Mk 27	ICAO	−2,000 to +60,000 ft	± 50 ft at 0 ft ± 175 ft at 10,000 ft ± 600 ft at 40,000 ft ± 2,000 ft at 60,000 ft
Mk 28	WADC	−1,000 to +80,000 ft	± 30 ft at 0 ft ± 100 ft at 10,000 ft ± 260 ft at 40,000 ft ± 600 ft at 60,000 ft
Mk 31	ICAO	−1,000 to +45,000 ft	± 35 ft at 0 ft ± 100 ft at 20,000 ft ± 230 ft at 40,000 ft

CHAPTER 3

SERVO-ASSISTED ALTIMETERS

CONTENTS

	Paras
Introduction	1
MK 22 ALTIMETER	
Description	2
Components	3
Altimeter Unit	4–7
Amplifier Unit	8
Principle of Operation	9–14
Accuracy	15
MK 22J ALTIMETER	
Introduction	16
Altitude Encoder	17–18
Pressure Error Correction	19
Baroscale Setting	20
Datum Adjustments	21–22
MK 29 ALTIMETERS	
Description	23–25
Servo Mode of Operation	26–28
Standby Mode of Operation	29–32
Datum Pressure Setting	33
Accuracy	34
MK 30A ALTIMETER	
Introduction	35–37
Description	38–39
Operation	40–47
Accuracy	48
DATA SUMMARY	
Data Table	49

Introduction

1. Servo-assisted altimeters use the same basic principles as the simple and sensitive altimeters described so far, *ie* pressure changes are measured by the expansion or contraction of evacuated capsules. In servo-assisted altimeters the linkages have been developed to be more sensitive and to provide adequate transmission power.

MK 22 ALTIMETER

Description

2. The Mk 22 Altimeter is of the servo-assisted type, designed to give accurate measurement of height from −1,000 ft up to 100,000 ft. The altimeter is calibrated according to the ICAO/WADC standard atmospheres.

Components

3. The equipment comprises two main units, the altimeter itself and an amplifier. Later versions of both units have been produced but differ only in minor constructional details from the basic units which are described in this chapter. The altimeter is illustrated in Fig 1.

Fig 1 Mk 22 Altimeter

Altimeter Unit

4. The altimeter unit, shown in Fig 1, contains the capsule assembly, the servo system, the height indicating mechanism and the datum pressure setting mechanism. The dial on the face of the instrument gives both digital and pointer indications of height. The millibar counters show through a window in the lower half of the dial

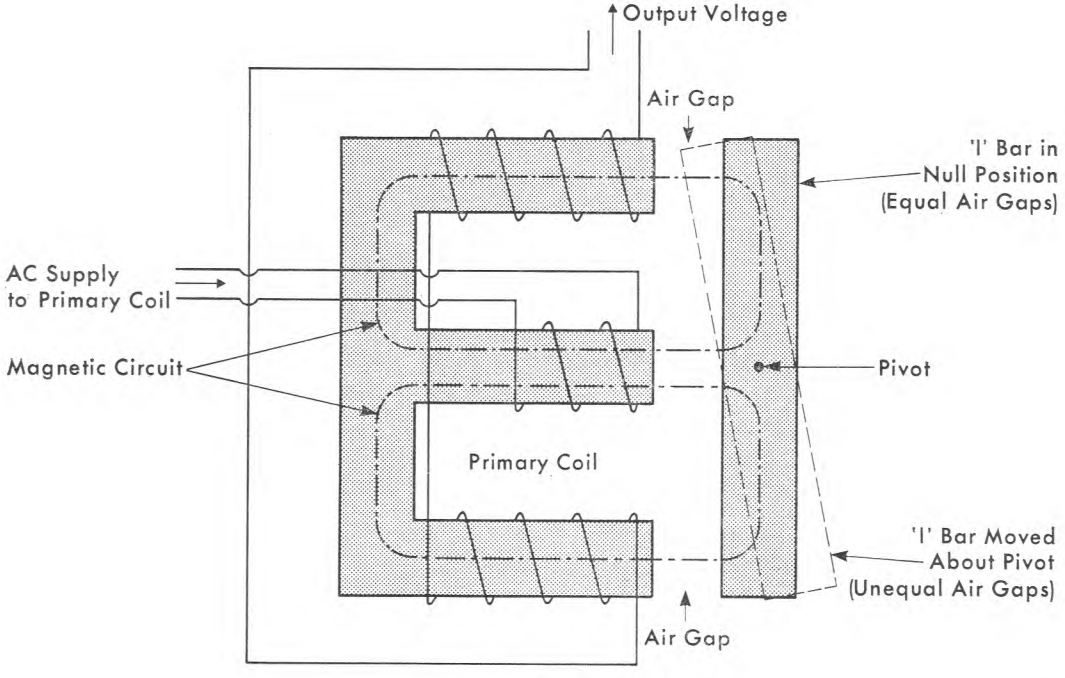


Fig 2 E and I Inductive Pick-Off

and the OFF flag through a window in the upper half.

5. **Height Indication.** Height is indicated by a veeder counter system and a single pointer. The counters are visible through four windows, the last of which contains both tens and units of feet. The first two figures, tens of thousands and thousands, are large and are coloured white; the last three are smaller and are coloured green. Height is shown to the nearest 50 ft. The pointer moves round the dial, which is graduated in 50 ft divisions, from 0—1,000 ft. The combined system indicates height up to 100,000 ft.

6. **Millibar Scale.** The millibar scale is also in veeder counter form. Pressure setting is achieved by rotation of the knob at the bottom of the dial.

7. **OFF Flag.** The OFF flag appears when the power supply fails or if the servo mechanism runs off at either end of the height range.

Amplifier Unit

8. The amplifier unit contains the power supply transformers and the signal amplifying circuits. It also provides the reference voltages for the inductive pick-off and AC motor. The power supply for the equipment is 115 V, single phase AC at 400 Hz. The power supply switch is normally located in some convenient position on the pilot's instrument panel.

Principle of Operation

9. The E and I pick-off is an electromechanical device used to obtain a null signal condition as part of a servo loop. The arrangement is shown in Fig 2.

10. An AC supply is fed to the centre limb of the E bar. This causes equal alternating fluxes to be produced in the outer limbs, providing the air gaps between the extremities of these limbs and the I bar are equal. Equal voltages are then

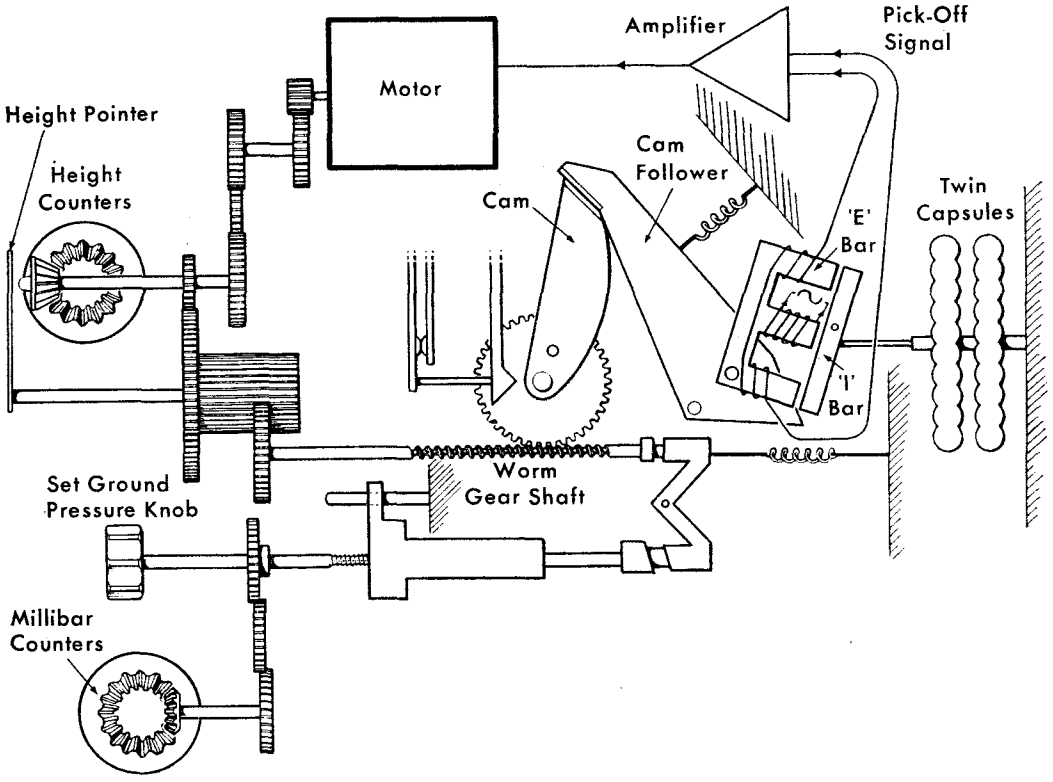


Fig 3 Mk 22 Altimeter, Schematic

induced in the coils wound on the outer limbs. The secondary coils are connected so that the voltages are 180° out of phase, and so when equal and opposite voltages are produced, there is no resultant voltage output.

11. When the I bar is moved about the pivot and the E bar remains stationary, the two air gaps are unequal and the reluctance of each circuit changes. The voltage induced in one coil increases while the other decreases. There will therefore be a resultant voltage output, the phase of which will depend on the direction of movement of the I bar.

12. In the Mk 22 Altimeter the movement of the aneroid capsules is transmitted through a linkage to the I bar of an E and I inductive pick-off as shown in Fig 3. The amplitude of the AC voltage output from the secondary depends on the amount of deflection of the I bar, which is a function of the pressure change. The polarity of the output signal will depend on whether the capsules expand or contract.

13. The output signal is amplified and is used to drive the motor. The speed of rotation of the motor will depend on the amplitude of the signal and the direction of rotation will be determined by the phase of the signal. The motor drives the gear train which rotates the height veeder counters and the pointer. The motor also drives, through gearing, a cam which imparts an angular movement to a cam follower. The E bar of the inductive pick-off is attached to this follower. The sense of the movement is such that the E bar is driven until it reaches a position where the air gaps are again equal thus completing the servo loop. This system is very sensitive to small changes in pressure and provides good torque to drive the indicating mechanism.

14. **Datum Pressure Setting.** The pressure setting knob is linked to the cam as shown in Fig 3. Rotation of the knob causes the worm shaft to slide forwards or backwards and thus rotate the cam slightly. The angular movement of the cam alters the relationship between the E and the I bars; there is then an output produced which causes the counters to rotate and drives the inductive pick-off back to its neutral position. Thus the datum from which height is measured is effectively altered. The value of the datum pressure is recorded on the millibar scale.

Accuracy

15. The instrument accuracy of the system is ± 1 millibar over the whole range. The indicated accuracy is ± 30 ft at 0 ft, ± 100 ft at 40,000 ft, ± 300 ft at 60,000 ft and $\pm 4{,}000$ ft at 100,000 ft. Rates of climb and descent of 10,000 ft per minute can be accurately followed without appreciable time lag.

MK 22J ALTIMETER

Introduction

16. The Mk 22J Altimeter is basically similar to the Mk 22 described above, but it has some extra facilities and some mechanical modifications which are given below. A schematic diagram of the Mk 22J Altimeter is given in Fig 4.

Altitude Encoder

17. The altimeter has an altitude encoder (digitizer) to give a coded height output which, when transmitted via a remote transponder, enables the height sensed by the capsules to be monitored on the ground.

18. The solenoid activating the warning flag, when power supply is interrupted, also operates a relay to switch off the digitizer output.

Pressure Error Correction

19. A facility is provided to enable a pressure error correction, supplied from a remote pressure transducer, to be applied to the altimeter output.

Baroscale Setting

20. The baroscale setting mechanism is modified to become independent of the servo-loop and is interconnected with the height display by a differential gear mechanism.

Datum Adjustments

21. The position of the capsules may be altered by an adjustment screw for the initial setting of the instrument datum.

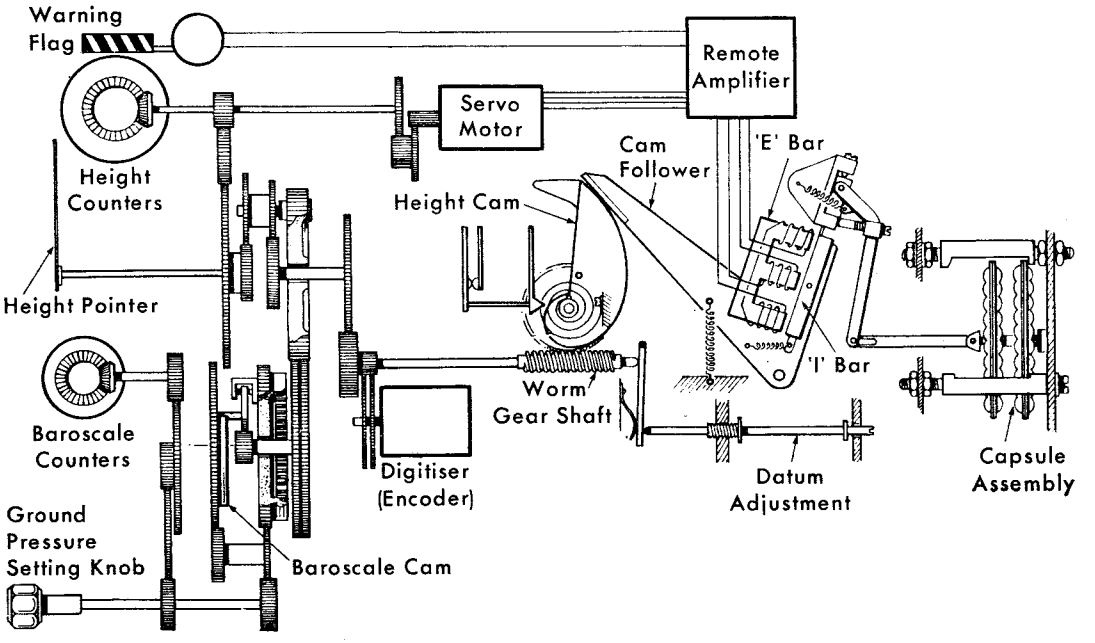


Fig 4 Mk 22J Altimeter, Schematic

22. A second adjustment screw, identified "zero adjust", carries out the datum adjustment previously done by the baroscale setting mechanism. This screw causes the worm drive to the cam to slide backwards or forwards so altering the relationship between the E and I bars.

MK 29 ALTIMETERS

Description

23. The Mk 29 Altimeter series are servo-assisted altimeters with automatic reversion to normal capsule operation in the event of power or other failure. All the instruments are basically similar, with changes in the lighting and connections to suit the aircraft to which they are fitted. In the normal mode of operation the instrument is supplied with a pressure corrected altitude from a remote pressure sensor, Air Data Computer or Mk 30 Altimeter, so that a more accurate indication of altitude is given. In the event of an electrical failure or on demand, these instruments will operate as uncorrected precision pressure altimeters. The Mk 29 Altimeter is designed to operate from $-1{,}000$ ft to $\pm 80{,}000$ ft and is calibrated to the ICAO 1964 standard atmosphere.

24. Indication of altitude is provided by a pointer, a drum and a two digit counter. The pointer and drum show hundreds of feet whilst the counter shows thousands and tens of thousands of feet. The display also shows the selected barometric pressure and a flag to indicate the mode of operation. To bring attention to altitudes below 10,000 ft, the left hand counter is marked with diagonal white stripes instead of the figure "0", and for altitudes above 80,000 ft, the left hand counter is marked with diagonal red stripes instead of the figure "9". Two dummy zeros are placed to the right of the counter and drum to complete the altitude display. The sealed case is filled with static pressure via a connector at the rear.

25. The mode of operation is selected by a RESET/STBY knob on the front of the instrument. The knob is spring loaded to the central position.

Servo Mode of Operation

26. The altimeter is engaged in its normal operating mode, the servo mode, by momentarily setting the mode selection switch to RESET and letting it return to the central position. This supplies power to the amplifier and motor circuits and the STBY flag is removed from display.

27. A synchro transmitter (CTB), with a very low inertia rotor, normally referred to as a synchrotel, is mounted on the altimeter drive shaft, *see* Fig 5. The synchrotel gives an error signal related to the difference between the synchrotel input signal and the rotor position. The synchrotel input signal is the corrected altitude signal from an Air Data Computer or a Mk 30 Altimeter. The rotor position is determined by the capsule displacement. The amplified error signal from the synchrotel rotor is fed to the motor which nulls the synchrotel rotor error and drives the pointer.

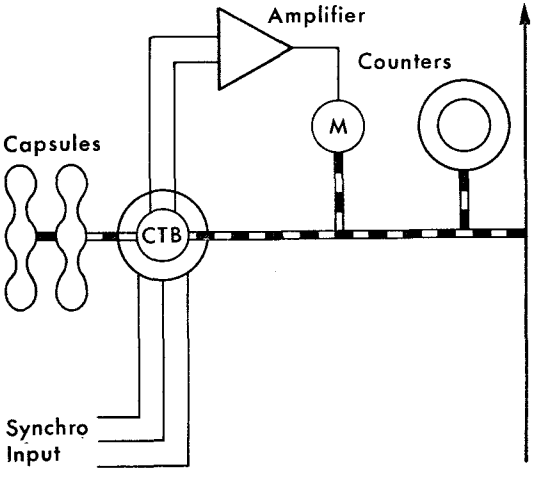


Fig 5 Mk 29 Altimeter, Schematic

28. **Error Detection.** A fail safe detection circuit is incorporated so that the system automatically reverts to the standby mode of operation in the event of one of the following faults:

a. Primary power failure.
b. Servo amplifier failure.
c. Servo motor failure.
d. Failure within the detection circuits.
e. A difference greater than 4,500 ft between the standby and servo altitudes.

In any of these circumstances the STBY flag will drop into view, the vibrator will come into operation, and standby operation will continue until the RESET switch is made.

Standby Mode of Operation

29. The standby mode of operation can be selected by setting the switch on the front of the instrument to STBY and then letting it return to the central position or (*see* para 28) it may be automatically selected by the error detection circuits. When the standby mode is selected, it is indicated by a flag bearing the legend STBY which appears in the top left quadrant of the instrument face.

30. A change in static pressure causes the two capsules to expand or contract. The capsules are connected by a suitable linkage and gear system to a shaft which drives the altimeter pointer. The pointer is geared to the drum so that both show the same height indication in hundreds of feet. The drum, in turn, is geared to the two digit counters which register thousands and tens of thousands of feet.

31. A small vibrator, operating only in the standby mode, helps overcome the inertia and friction lag of the system and thus gives better sensitivity

32. During the standby mode, the synchrotel and motor constitute a small additional load on the capsules, but because of their small size, low inertia and friction, the effect on the altimeter performance is negligible. A semi-exploded view of the altimeter is given in Fig 6.

Datum Pressure Setting

33. The datum setting is controlled by the rotation of the setting knob in the bottom left of the instrument face. The knob when turned engages a small gear to the millibar counters, a larger gear to the capsule mechanism frame and a bevel gear and worm drive to the body of the synchrotel. Rotation of the setting knob causes simultaneous rotation of the millibar counters, the capsule mechanism, the pointer and the synchrotel.

Accuracy

34. These instruments are accurate to $\pm$30 ft at 0 ft, $\pm$80 ft at 10,000 ft, $\pm$230 ft at 40,000 ft and $\pm$800 ft at 60,000 ft when operated in the standby

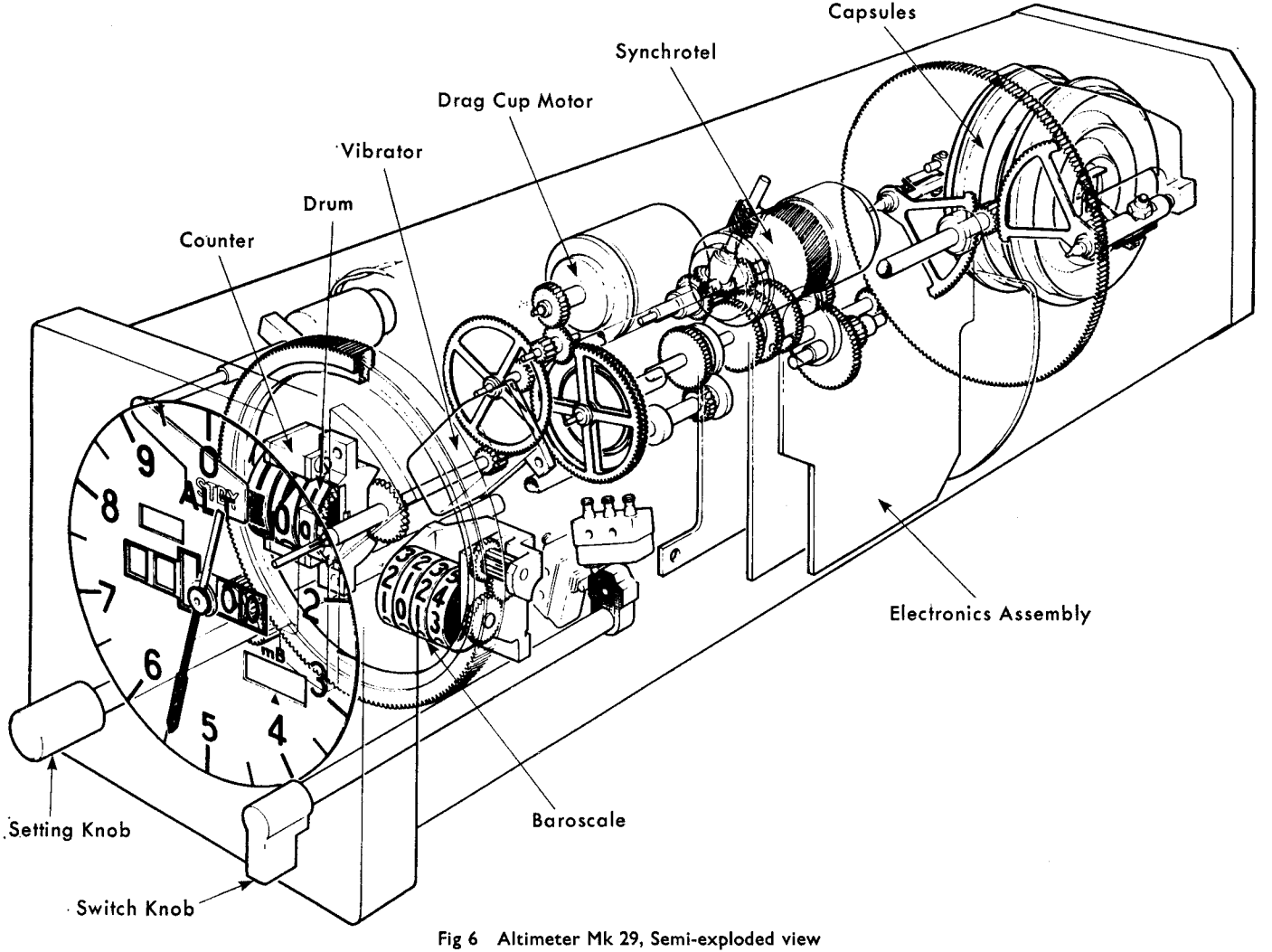


Fig 6 Altimeter Mk 29, Semi-exploded view

mode. In the servo mode the reading will be within ±~~25~~ ft of the height from the instrument supplying the altitude data.

MK 30A ALTIMETER

Introduction

35. The Mk 30A Altimeter is a pressure servoed encoding altimeter displaying pressure corrected altitude over a calibrated range of −900 ft to +60,500 ft. The instrument provides electrical outputs to a Mk 29 series altimeter and to a pressure error correction unit (PECU) and also an encoded height output for altitude reporting. The instrument is housed in a 3¼ inch square flanged case. A typical system is shown in Fig 7.

36. Indication of altitude is given by a counter-counter-drum-pointer presentation which is fluorized. In the event of a servo malfunction or power failure, a red and black diagonal striped flag drops into view to obscure the counter and drum display. The tens of thousands of feet counter is marked with diagonal black and white stripes to bring attention to altitudes below 10,000 ft. At negative altitudes the counter is marked with red and white stripes.

37. In the event of a malfunction within the associated PECU, a pressure error warning flag, marked PE, drops into view at the top of the display. A barometric setting knob is fitted at the lower left front of the instrument to allow the setting of the selected barometric pressure. The selected pressure is displayed on a four-digit millibar counter in the lower right quadrant of the dial.

Description

38. The altimeter is housed in a sealed case which is connected to the static system of the aircraft via a static port. At the rear of the instrument is a capsule with its link attached to a rocking shaft and sector. The sector is coupled through gearing to a synchrotel rotor and a coarse sector disc. A servo motor is geared to the synchrotel stator, to the rotor of the altitude encoder and a synchro transmitter, and also to a twin gang potentiometer.

39. The altitude encoder is of the brush type in which brushes pass over conductive segments to produce an eleven bit digital output in ICAO Gillham code which is used as a direct output to an IFF transponder for automatic altitude

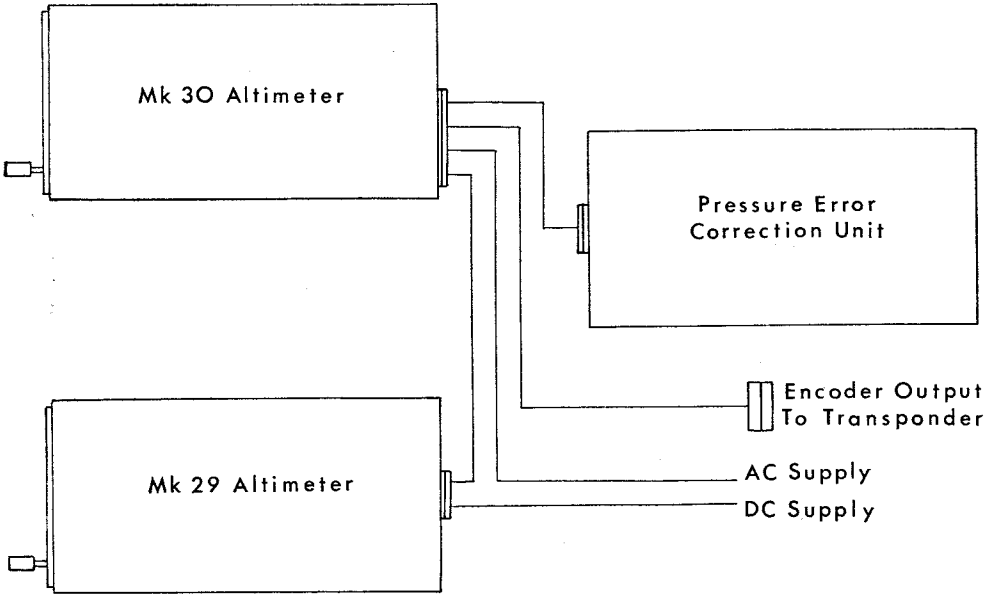


Fig 7 Mk 30A Altimeter System

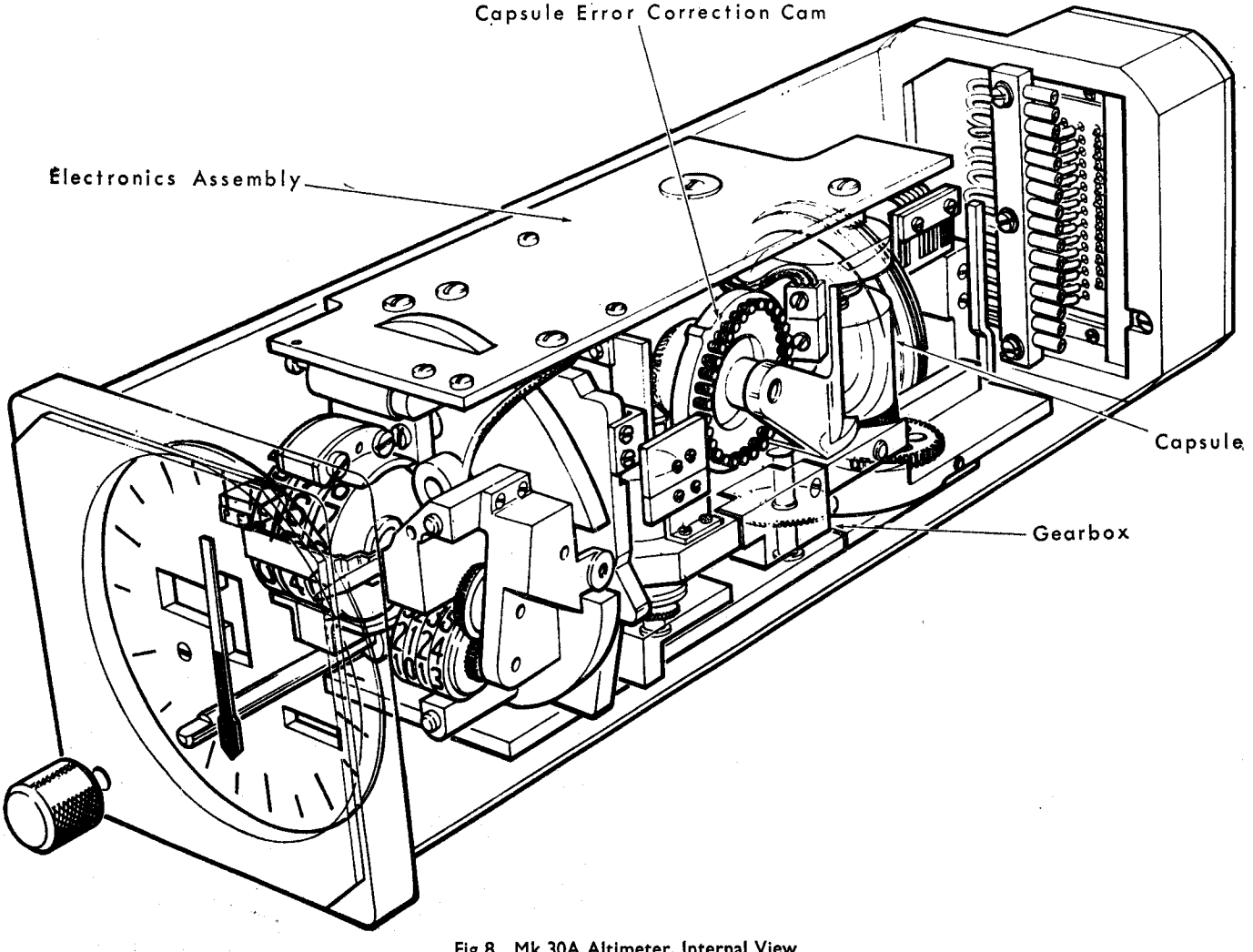


Fig 8 Mk 30A Altimeter, Internal View

reporting. The gearing also drives an adjustable cam which is used to compensate for capsule error. A limit switch disconnects the electrical supply if the instrument indicates an altitude beyond the range of $-3{,}000$ ft or $\pm 80{,}000$ ft. An internal view of the altimeter is shown at Fig 8.

Operation

40. A change of static pressure causes a deflection of the capsule which is transmitted through the mechanism to drive the rotor of the synchrotel and the coarse sector alignment disc. The electrical output from the synchrotel stator is fed to the control phase of the servo motor whose output shaft, through gearing, drives the display pointer and the stator of the synchrotel. The synchrotel is thus driven to the null position, completing the servo loop.

41. The pointer and drum, being directly coupled, both indicate hundreds of feet. The drum is coupled to the two digit counters which indicate thousands and tens of thousands of feet.

42. **Capsule Error Correction Cam.** Driven by the servo motor, the capsule error correction cam has a roller running on its periphery. The roller is connected to the rotor of a differential synchro, the output of which is connected to the synchrotel. The profile of the cam is calibrated to match the errors of the capsule and thus the synchro output adds a correction term into the servo loop.

43. **Coarse Sector Alignment Disc.** This disc is usually maintained at mid-travel and produces no output. If, however, the error between capsule displacement and synchrotel output is such that the synchrotel error approaches 180° displacement (*eg* due to a temporary interruption of power supply at altitude), a switch within the disc is made and a fixed signal is fed to the servo motor to override the error and hence bring the altimeter gear train to the correct datum.

44. **Datum Pressure Setting.** As stated in para 37, a conventional barometric setting knob allows datum pressure to be set on a four-digit counter on the face of the instrument. However, the setting knob has no effect on the altitude encoder, the synchro transmitter and the potentiometer outputs which are always referenced to a datum pressure of 1013·25 mb.

45. **Potentiometer Outputs.** The twin gang potentiometer is driven, via the gear train, by the servo motor output shaft. Each gang produces a voltage output related to height:

a. One gang produces a voltage which is a function of height expressed as ΔH. This is fed to the PECU to be used as a correction term.

b. The other gang produces a voltage inversely proportional to static pressure expressed as $1/S$. This is also fed to the PECU as a further correction term.

46. **PECU.** The PECU supplies to the differential synchro in the altimeter a synchro signal corrected by the voltages from the potentiometer. This synchro signal constitutes the pressure error correction factor and it is added to the correction factor from the capsule error correction cam at the differential synchro to give a total correction to the servo loop. The altimeter thus displays altitude corrected for pressure errors, and also provides a pressure error corrected synchro output to the Mk 29 series altimeter.

47. **Failure Monitoring.** Two electronic monitors, one in the altimeter servo motor circuit and one in the PECU, control relay contacts which operate, in case of failure in either unit, as follows:

a. In case of failure of the servo motor amplifier circuit or if the synchrotel null voltage rises above a level equivalent to 600 ft for between 3 and 7 seconds, the warning flag solenoid is de-energized and the warning flag drops over the counter and drum display. At the same time the 115 V supply to the synchro transmitter, the Mk 29 Altimeter and the PECU is disconnected.

b. In case of failure in the PECU, the PE warning flag solenoid is de-energized and the PE failure warning flag drops into view to indicate a PE malfunction. At the same time the PECU correction signal is removed. However, the synchro output to the Mk 29

Altimeter remains but is uncorrected for PEC.

c. Because the relay contacts operated by the two monitors are connected in series in the altitude encoder common line, the encoder only gives an output when both the altimeter and the PECU are operating correctly. If a failure is indicated by either monitor, the encoder output is disconnected.

Accuracy

48. This instrument is accurate to ±20 ft at 0 ft, ±60 ft at 30,000 ft and ±120 ft at 60,000 ft.

DATA SUMMARY

Data Table

49. The following table summarizes the pertinent data of the altimeters included in this chapter.

Type	Calibration	Operating Range	Accuracy
Mk 22	ICAO/WADC	−1,000 to +100,000 ft	± 30 ft at 0 ft ± 100 ft at 40,000 ft ± 300 ft at 60,000 ft ± 4,000 ft at 100,000 ft
Mk 29	ICAO 1964	−1,000 to +80,000 ft	± 30 ft at 0 ft ± 80 ft at 10,000 ft ± 230 ft at 40,000 ft ± 800 ft at 60,000 ft
Mk 30A	ICAO 1964	−900 to + 60,500 ft (calibrated)	± 25 ft at 0 ft ± 60 ft at 30,000 ft ± 120 ft at 60,000 ft

PART 1

SECTION 3

SPEED AND DISTANCE

Chapter

1 *Not issued*

2 **Vertical Speed Indicators**

3 **Air Speed Indicators**
Annex: **Calibration of Air Speed Indicators**

4 **Machmeters**
Annex: **Theoretical Basis of Mach Number Measurement**

5 **Air Mileage Units**

6 **Air Mileage Indicator, Mark 1**

7 **Air Data Systems**

8 **True Air Speed Unit, Mark 1**

9 **Combined Speed Indicators**

CHAPTER 2

VERTICAL SPEED INDICATORS

CONTENTS

	Paras
Introduction	1
Principle	2
Construction of the VSI, Mk 3	3–6
Operation of the VSI, Mk 3	7–9
The Instrument Face	10
Other Types and Marks of VSI	11
Errors	12–14
Pilot's Serviceability Checks	15–16

Introduction

1. A vertical speed indicator (VSI), also known as a rate of climb and descent indicator (RCDI), is a sensitive differential pressure gauge. It records the rate of change of atmospheric pressure in terms of rate of climb or descent when an aircraft departs from level flight.

Principle

2. The principle employed is that of measuring the difference of pressure between two chambers, one within the other. The pressure of the atmosphere is communicated directly to the inner chamber, and through calibrated chokes or constrictions to the outer chamber. If the atmospheric pressure changes, as when climbing or descending, the lag rate between the outer and inner chambers is a measure of the rate of climb or descent of the aircraft.

Construction of the VSI, Mk 3

3. There are many types of VSI in service use. The main constructional features of the Kelvin and Hughes Type, Mk 3 Series, are described in the following paragraphs. Details in which some of the other types differ are mentioned at para 11.

4. The mechanism consists essentially of a sensitive capsule, a metering unit, and a suitable magnifying linkage (*see* Fig 1). The capsule is connected by a link to the rocking shaft and sector which meshes with a pinion carrying the pointer. The pointer zero setting arrangement is such that rotation of the adjusting screw at the front of the case moves the capsule support towards or away from the remainder of the mechanism.

5. The mechanism is contained in an air-tight case, which forms the outer chamber of the instrument. Air pressure from the aircraft's static system is fed to the inside of the capsule via the pressure connector and a capillary tube. The pressure in this system is permitted to escape into the case via the metering unit, which comprises an orifice and a capillary tube.

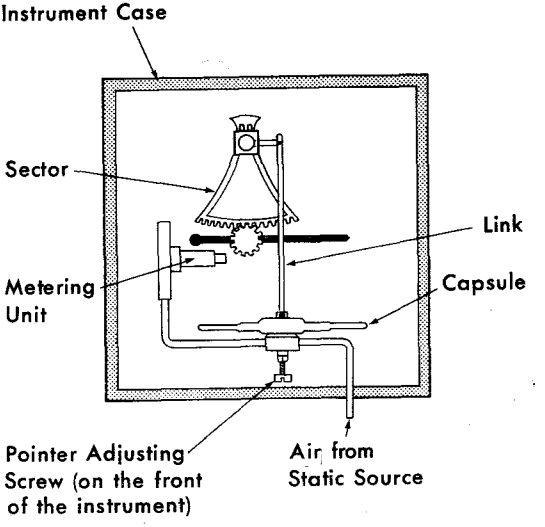


Fig. 1 General Construction of a VSI, Mk 3

6. The restraining action of two springs (not shown in Fig 1) bearing on a stem connected to the capsule centre piece can be varied, thus providing a calibration facility.

Operation of the VSI, Mk 3

7. The instrument indicates the pressure dif-

ference between the inside and outside of the capsule, that is, the pressure difference across the metering unit. The metering unit is therefore required to give a definite pressure difference for any given rate of climb or descent, while compensating for variations in temperature and pressure of the atmosphere with changes of altitude. The compensation is achieved by incorporating in the metering unit both an orifice and a capillary, whose sizes are chosen so that the indicator readings remain correct over a wide range of temperature and altitude conditions. Their effects are as follows:

a. *Pressure (Altitude) Compensation.* The pressure difference across an orifice for a given rate of climb decreases with increasing altitude, and a purely orifice-type of metering unit, which indicated the rate of climb correctly at sea level, would indicate too small a rate at altitude. On the other hand, the pressure difference across a purely capillary-type of metering unit at a constant rate of climb increases with increasing altitude (at constant temperature), and such a unit would indicate too big a rate of climb at altitude. The reason for this difference in behaviour is that the flow of air through a capillary is laminar, while that through an orifice is turbulent. A combination of the two types of metering unit can be found which gives satisfactory altitude compensation over a limited pressure range at a given temperature.

b. *Temperature Compensation.* The pressure difference across the capillary depends on the viscosity of the air, which is proportional to the absolute temperature and, therefore, decreases with decreasing temperature. The pressure difference across the orifice varies inversely as the temperature of the air, and will therefore increase with the decrease in temperature. This effect is opposite to the effect of temperature change on the capillary, so that compensation can be obtained by a combination of the two types.

8. The instrument is connected to the static side of the aircraft's pitot/static system, and measures the rate of change of static pressure. When the pressure varies due to changing altitude, the pressure change in the case lags behind that in the capsule. This lag is caused by the metering unit, which restricts the flow of air into and out of the case, whereas the flow of air to the inside of the capsule is unrestricted. The resulting differential pressure deflects the capsule. This movement is magnified and transmitted to the pointer via the link, rocking shaft and sector. The rocking shaft transposes the linear movement of the capsule into rotary movement.

9. In level flight, the pressure inside the capsule and the case are the same, and the pointer remains at the horizontal zero position. When the aircraft climbs, the static pressure decreases, and the capsule collapses slightly, causing the pointer to indicate a rate of climb. The fall in pressure in the case lags behind that in the capsule until level flight is resumed and the pressures equalize. In a descent, the increase in pressure in the case lags behind the increase in static pressure in the capsule, and the capsule is expanded, a rate of descent being indicated.

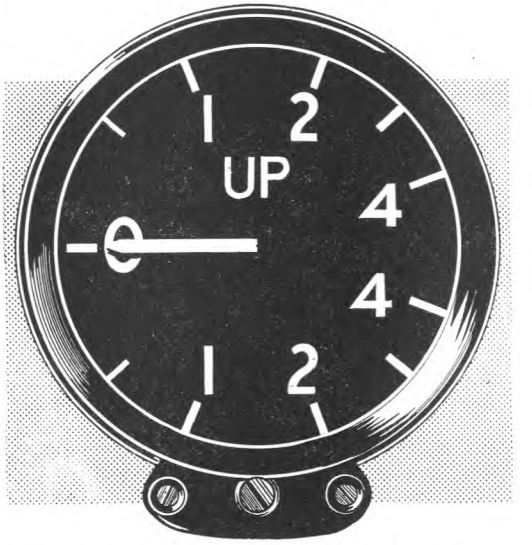


Fig 2 VSI, Mk 3(P)—Instrument Face

The Instrument Face

10. A single pointer indicates the rate of climb or descent by registering against a background dial (*see* Fig 2). The zero position is horizontal pointing to 9 o'clock. Climb is indicated by clockwise, and descent by anti-clockwise pointer rotation. The dials are graduated for rates of up to 4,000 feet per minute, there being a gap between the end graduations on the climb and descent scales in the 3 o'clock position of the dial, and the pointer movement being restricted to approximately 170 degrees from zero, in each direction, to avoid ambiguity.

Other Types and Marks of VSI

11. The coded air publications 112G Series contain details of the many types of VSI in current use. They employ the same basic principles as the Mark 3, but differ in construction, the main points of difference being as follows:

a. *Pressure Sensitive Element.* In some instruments a diaphragm separating two chambers replaces the capsule.

b. *Dial Graduation.* Some instruments are graduated to maximum rates of climb or descent of 2,000 feet per minute only. The scale may be linear or logarithmic, and sub-division of the scales varies with different instruments.

c. *Metering Unit.* Some instruments employ a leak assembly comprising porous diffuser tubes, and the Mk 1 Series uses a series of jewel holes to achieve a pressure differential between the two sides of a diaphragm. Methods of temperature compensation vary accordingly.

Errors

12. **Lag.** When an aircraft is suddenly put into a steady climb or descent a delay of a few seconds occurs before the pointer settles at the appropriate rate of climb or descent, due to the time required for the pressure difference to develop.

13. **Pressure Error.** If the static head or vent (and thus the ASI) is subject to a large pressure error, the VSI may wrongly indicate a rate of climb or descent whenever a considerable change of airspeed occurs, especially during take-off.

14. **Static Line Blockage.** Blockage of the static line by ice or any other obstruction renders the VSI completely unserviceable, the pointer remaining at zero whatever the vertical speed.

Pilot's Serviceability Checks

15. **On the Ground.** Before flight, pilots should ensure that the pointer reads zero or that the index error, if any, is within the permissible limits. These are:

a. Plus or minus 200 feet per minute when the local air temperature is within the range —20°C to +50°C, or

b. Plus or minus 300 feet per minute when the local air temperature is outside the above range.

16. **In the Air.** Except as in para 13, the instrument should read zero during level flight. The accuracy of the instrument indications may be checked using a stop-watch during steady climb or descent.

CHAPTER 3

AIR SPEED INDICATORS

CONTENTS

	Paras
Introduction	1
Principle	2–6
Construction	7–9
Calibration	10
ASI Errors	11–16
Blocked or Leaking Pressure Systems	17–19
Annex: Calibration of Air Speed Indicators	

Introduction

1. A knowledge of the speed at which an aircraft is travelling through the air *ie* the airspeed, is essential both to the pilot for the safe and efficient handling of the aircraft and to the navigator as a basic input to the navigation calculations. The instrument which displays this information is the air speed indicator (ASI).

Principle

2. An aircraft, stationary on the ground, is subject to normal atmospheric or static pressure which acts equally on all parts of the aircraft structure. In flight the aircraft experiences an additional pressure on its leading surfaces due to a build up of the air through which the aircraft is travelling. This additional pressure due to the aircraft's forward motion is known as dynamic pressure and is dependent upon the forward speed of the aircraft and the density of the air according to the following formula:

$$p_t = \tfrac{1}{2}\rho V^2 + p$$

where p_t = the pitot pressure, (also known as total head pressure or stagnation pressure)

p = the static pressure

ρ = the air density

V = the velocity of the aircraft.

Rearranging the formula, the difference between the pitot and the static pressures is equal to $\frac{1}{2}\rho V^2$ (the dynamic pressure). The air speed indicator measures this pressure difference and provides a display indication graduated in units of speed.

3. Fig 1 illustrates the principle, in its most simple form, on which all air speed indicators function. The ASI is a sensitive differential pressure gauge operated by pressures picked up by a pressure head, which is mounted in a suitable position on the airframe. The simplest pressure head consists of an open ended tube, the pitot tube, aligned with the direction of flight, and a second tube, the static tube, which is closed and streamlined at the forward end but which has a series of small holes drilled radially along its length.

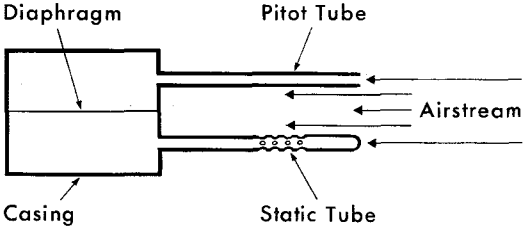


Fig 1 Principle of Air Speed Indicator

4. When moved through the air, the pitot tube will pick up pitot pressure made up of static pressure and dynamic pressure. The pitot pressure is led through a pipe-line to one side of a sealed chamber, divided by a thin flexible diaphragm. The static tube is unaffected by dynamic pressure as its end is closed, however, the small holes will pick up local static pressure. The static pressure is led through a second pipeline to the other side of the diaphragm.

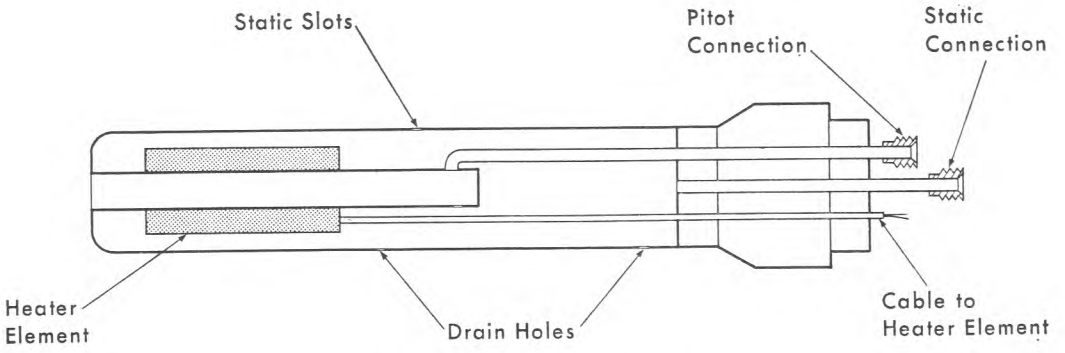


Fig 2 A Typical Pressure Head

5. The diaphragm is subjected to the two opposing pressures. However, the static pressure component of the pitot pressure is balanced by the static pressure on the other side of the diaphragm so that any diaphragm movement is determined solely by the dynamic, or pitot excess, pressure. Movement of the diaphragm is transmitted through a mechanical linkage to a pointer on the face of the ASI where the pitot excess pressure $(p_t - p)$ is indicated in terms of speed.

6. In practice it is more usual for the pitot tube and the static tube to be combined into a single pressure head with the pitot tube built inside the static tube. A heater is placed between the pitot and static tubes to prevent ice forming and causing a blockage. Drain holes in the head allow moisture to escape and various traps are used to prevent dirt and water from entering the open ended tube. A typical pressure head is shown in Fig 2.

Construction

7. Most air speed indicators in current use have a capsule instead of a diaphragm, however, the principle of operation is exactly the same. The capsule, acting as the pressure sensitive element is mounted in an airtight case. Pitot pressure is fed into the capsule and static pressure is fed to the interior of the case, which thus contains the lower pressure. A pressure difference will cause the capsule to open out, the movement being proportional to pressure. A link, quadrant and pinion can be used to transfer this movement to a pointer and dial calibrated in knots.

8. As stated in para 2, the pitot excess pressure varies with the square of the speed and a linear pressure/deflection characteristic in the capsule produces an uneven speed/deflection characteristic of the pointer mechanism, giving unequal pointer movements for equal speed changes. To produce a linear scale between the capsule and pointer it is necessary to control the characteristic of the capsule and/or the mechanism. Control of the capsule is difficult due, among other reasons, to the magnification factor of the mechanism. It is more usual to control the mechanism to produce a linear scale shape by changing the lever length as the pointer advances. Depending on the manufacturer of the ASI, detailed points of construction will vary, however, the basic principle holds good for all. A typical simple ASI is shown in Fig 3.

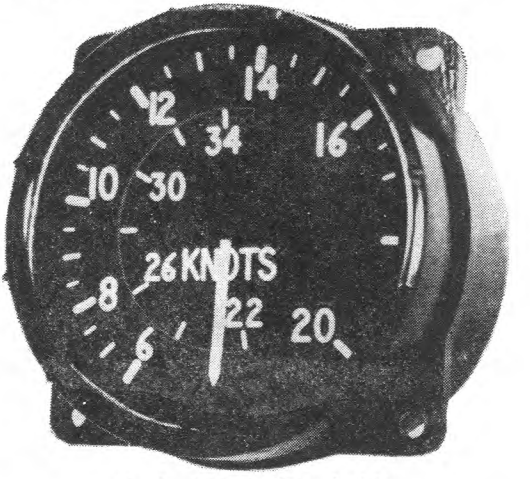


Fig 3 A Typical Simple ASI

9. **The Sensitive Air Speed Indicator.** The sensitive ASI is identical in principle to the simple ASI, operates from the normal pitot/static system, but provides a more sensitive indication. This is achieved by an increase in the gear train from the capsule, so that two pointers may be moved over an evenly calibrated dial. Because of this increase in the gear train, more power is required to operate the gears and this is provided by a stack of two capsules. This capsule assembly has a linear pressure/deflection characteristic which is more closely controlled than the single capsule used in the simple ASI. A typical sensitive ASI is shown in Fig 4.

Fig 4 A Two Pointer Sensitive ASI

Calibration

10. Since dynamic pressure varies with air speed and air density, and since air density varies with temperature and pressure, standard datum values have to be used in the calibration of air speed indicators. The values used are the sea level values of the standard ICAO atmosphere. The formula given in para 2 is only an approximation and one of two formulae is used for calibration of a particular ASI depending on the speed range of the instrument. More details of calibration and the formulae used are given in the Annex to this chapter.

ASI Errors

11. The ASI pointer registers the amount of capsule movement due to dynamic pressure, however, the dial is calibrated according to the formulae mentioned above which assume constant air density (standard sea level density) and no instrument defects. Any departure from these conditions or disturbance in the pitot or static pressures being applied to the instrument will result in a difference between the indicated and the true air speed and thus an error in the display. There are four sources of error:

a. Instrument error.

b. Pressure error.

c. Compressibility error.

d. Density error.

12. **Instrument Error.** Instrument error is caused by manufacturing tolerances in the construction of the instrument. The error is determined during calibration and any necessary correction is combined with that for pressure error (*see* para 13).

13. **Pressure Error.** Pressure error results from disturbances in the static pressure around the aircraft due to movement through the air. Size of the error depends upon:

a. The position of the pressure head, pitot head or static vent.

b. The angle of attack of the aircraft.

c. The speed of the aircraft.

Most of the error results from variations in the local static pressure caused by the airflow over the pressure head. In lower speed aircraft the static head is often divorced from the pitot tube and positioned where the truest indication of static pressure is obtained *eg* on the fuselage midway between nose and tail. In such a case the static pipeline terminates at a hole in a flat brass plate known as the static vent. It it usual to have two static vents, one either side of the aircraft to balance out the effects of sideslip which produces an increase of pressure on one side of the aircraft and a corresponding decrease in pressure on the other side. The use of static vents eliminates almost all the error caused by the pressure head. Any remaining error is determined by flight trials. Unfortunately the use of a static vent becomes less

acceptable for high performance aircraft since, at Mach numbers exceeding 0·8, the flow of air around the static vent may be unpredictable. In such cases a high speed pitot-static head is used and, as before, pressure error is determined by flight trials. The correction for pressure error (PEC) is tabulated in the Aircrew Manual for the aircraft type and is also combined with that for instrument error (IEC) and recorded on a correction card mounted adjacent to each ASI. The card correction (*ie* Inst-Error + PEC) should be applied to the indicated airspeed (IAS) to obtain rectified air speed (RAS).

14. **Compressibility Error.** The calibration formulae contain a factor which is a function of the compressibility of the air. At higher speeds this factor becomes significant. However the calibration formulae use standard mean sea level values and an error is introduced at any altitude where the actual values differ from those used in calibration. At altitude, the less dense air is more easily compressed than the denser air at sea level, resulting in a greater dynamic pressure which causes the ASI to over-read. In addition compressibility increases with increase of speed, therefore compressibility error varies both with speed and altitude. Compressibility error and its correction can be calculated by using the circular slide rule of the DR Computer Mk 4A or the Height and True Air Speed Computer Mk 4 (*see* AP 3456G Part 1, Sect 4, Chaps 3 and 5). Application of the compressibility error correction (CEC) to RAS produces equivalent air speed (EAS).

15. **Density Error.** As has already been explained, dynamic pressure varies with air speed and the density of the air. Standard mean sea level air density is used for calibration purposes. Thus, for any other condition of air density, the ASI will be in error. As altitude increases, density decreases and IAS, and thus EAS, will become progressively lower than true air speed (TAS). The necessary correction can be calculated from the formula:

$$EAS = TAS \sqrt{\frac{\rho}{\rho_o}}$$

where: ρ = the air density at the height of the aircraft

ρ_o = the air density at mean sea level.

In practice, the density error correction (DEC) is obtained from a graph or by the use of a circular slide rule such as the DR Computer Mk 4A or the Height and True Air Speed Computer Mk 4 (*see* AP 3456G Part 1 Sect 4 Chaps 3 and 5).

16. **Summary.** The relationship between the various air speeds and the associated errors can be summarized as follows:

$$RAS = IAS + PEC + IEC$$
$$EAS = RAS + CEC$$
$$TAS = EAS + DEC$$

Blocked or Leaking Pressure Systems

17. **Blockages**

a. *Pitot.* If the pitot tube is blocked *eg* by ice, the ASI will not react to changes of air speed in level flight. However, the capsule will act as a barometer producing an indication of increase in speed if the aircraft climbs or a decrease in speed if the aircraft dives.

b. *Static.* If the static tube is blocked, the ASI will over-read at lower altitudes and under-read at higher altitudes than that at which the blockage occurred.

18. **Leaks.**

a. *Pitot.* A leak in the pitot tube causes the ASI to under-read.

b. *Static.* A leak in the static tube, where the outside pressure is lower than static *ie* some unpressurized aircraft, will cause the ASI to over-read. Where the outside air is higher than static *ie* in a pressurized cabin the ASI will under-read.

19. **Effects.** Whilst any under-reading of the ASI is undesirable it is not necessarily dangerous. However, over-reading of the ASI is dangerous since a stall will occur at a higher indicated air speed than that specified for the aircraft.

CALIBRATION OF AIR SPEED INDICATORS

CONTENTS

	Paras
Theoretical Basis of Air Speed Measurement	1–2
Calibration of Air Speed Indicators	3–6
Numerical Value of Formula	7

Theoretical Basis of Air Speed Measurement

1. Fig 1 illustrates schematically an aerofoil moving forward with a speed of V. Air speed is determined by measuring the static pressure of the undisturbed air p and the total, stagnation or pitot pressure p_t *ie* the pressure at a point on a streamline which has been brought to rest on the aerofoil.

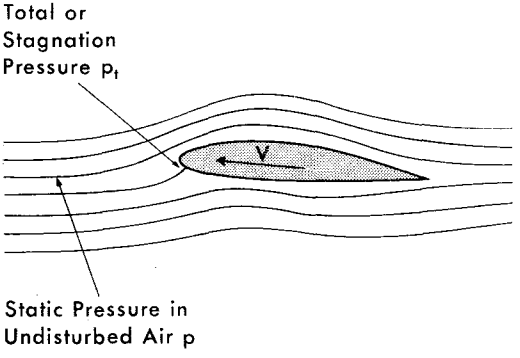


Fig 1 Pressures used for Air Speed Measurement

2. Ignoring the effect of position energy, the total energy for unit volume of fluid of density ρ at a static pressure p and moving with a velocity V equals $\frac{1}{2}\rho V^2 + p$. Thus to evaluate V, the total energy, the static pressure and the density must be measured. By reducing the velocity of the air to zero, the pressure energy equals the total energy, *ie* pressure energy equals $\frac{1}{2}\rho V^2 + p$. This relationship can be expressed in terms of pressure thus:

$$p_t = \tfrac{1}{2}\rho V^2 + p$$

$$\text{or} \quad p_t - p = \tfrac{1}{2}\rho V^2 \qquad \ldots.(1)$$

For the purposes of this formula it has been assumed that air is a "perfect fluid", which is substantially correct at low subsonic speeds (<0·4M). However, when considering the compressibility effect of the air in the pitot tube at higher speeds, the relationship between the true air speed V, the static and the pitot pressures is given by:

$$p_t = p\left(1 + \tfrac{1}{2}\cdot\frac{\gamma-1}{\gamma}\cdot\frac{\rho V^2}{p}\right)^{\gamma/(\gamma-1)} \qquad \ldots.(2)$$

where γ = ratio of specific heats of air at constant pressure and constant volume.

Expanding equation (2) gives:

$$p_t = p\left(1 + \tfrac{1}{2}\cdot\frac{\rho V^2}{p} + \tfrac{1}{8}\frac{\rho^2 V^4}{\gamma p^2} + \frac{2-\gamma}{48}\cdot\frac{\rho^3 V^6}{\gamma^2 p^3} + \ldots\right) \qquad \ldots.(3)$$

But velocity of sound in air,

$$a = \sqrt{\frac{\gamma p}{\rho}} \qquad \ldots.(4)$$

Hence, substituting in (3) the formula becomes:

$$p_t - p = \tfrac{1}{2}\rho V^2\left\{1 + \tfrac{1}{4}\left(\frac{V}{a}\right)^2 + \frac{2-\gamma}{24}\left(\frac{V}{a}\right)^4 + \ldots\right\} \qquad \ldots.(5)$$

Calibration of Air Speed Indicators

3. Using only the first two terms of the expansion (5), an approximate relationship between the true air speed V, the speed of sound a, the air density ρ and Δp (the difference in the two measured pressures p_t and p) is obtained. This is used in the calibration of British air speed indicators; by assuming standard ICAO mean sea level values for density ρ_0 and speed of sound a_0, and by defining the rectified air speed V_r as the value obtained from the formula in terms of the pressure differences Δ_p using the standard values. The air speed calibration formula is therefore:

$$\Delta p = \tfrac{1}{2}\rho_o V_r^2\left(1+\tfrac{1}{4}\cdot\frac{V_r^2}{a_o^2}\right) \quad \ldots.(6)$$

4. At low speeds, the second term in equation (6), the compressibility factor, may be neglected and the true air speed and equivalent air speed V_i (which now equals V_r, the RAS) are then related by the expansion quoted in equation (1) above *ie*:

$$\tfrac{1}{2}\rho_o V_i^2 = \tfrac{1}{2}\rho V^2, \text{ or } V_i = V\sqrt{\frac{\rho}{\rho_o}}$$

5. American practice is to calibrate air speed indicators using the exact formula in equation (2), *ie* with γ as a constant and assuming standard values p_0 for p and ρ_0 for ρ. The relationship between V_i and Δp is given by:

$$V_i = \left[\frac{2\gamma p_o}{(\gamma-1)\rho_o}\left\{\left(\frac{p_o+\Delta p}{p_o}\right)^{(\gamma-1)/\gamma}-1\right\}\right]^{\frac{1}{2}}$$

or $\ldots.(7)$

$$V_i = \left[\frac{2a_o^2}{(\gamma-1)}\left\{\left(\frac{\Delta p}{p_o}+1\right)^{(\gamma-1)/\gamma}-1\right\}\right]^{\frac{1}{2}} \quad \ldots.(8)$$

which becomes, when $\gamma = 1{\cdot}4$ is substituted:

$$V_i = \left[5a_o^2\left\{\left(\frac{\Delta p}{p_o}+1\right)^{\frac{2}{7}}-1\right\}\right]^{\frac{1}{2}} \quad \ldots.(9)$$

or

$$= a_o\left[5\left\{\left(\frac{\Delta p}{p_o}+1\right)^{\frac{2}{7}}-1\right\}\right]^{\frac{1}{2}} \quad \ldots.(10)$$

Significant differences do not arise between the two formulae until sonic speeds are approached. The error in the simplified formula (6) is 4 kt at an IAS of 600 kt.

6. The formulae quoted above, *ie* equations (6) and (10) are suitable for use at air speeds up to the speed of sound but should not be used for supersonic speeds. However the basic method of measurement using modified formulae can be extended to supersonic speeds.

Numerical Value of Formula

7. For calibration purposes, the differential pressure Δp is expressed in millimeters of water at a temperature of 15°C. Assuming the standard mean sea level conditions ($\rho_0 = 1{\cdot}226$ Kg/m^3 and $a_0 = 660{\cdot}6$ kt), the numerical form of equation (6) is:

$$\Delta p \text{ (mm water at 15°C)} = 0{\cdot}01658\ V_r^2\ (1 + 0{\cdot}57\times 10^{-6}V_r^2)$$

where V_r is in knots.

This relationship is shown in Fig 2 which also illustrates the simpler relationship, assuming an incompressible fluid, $\Delta p = \frac{1}{2}\rho_0 V_r^2$. The difference between the two curves amounts to 2 kt at an RAS of 200 kt and 16 kt (Indicated) at an RAS of 400 kt.

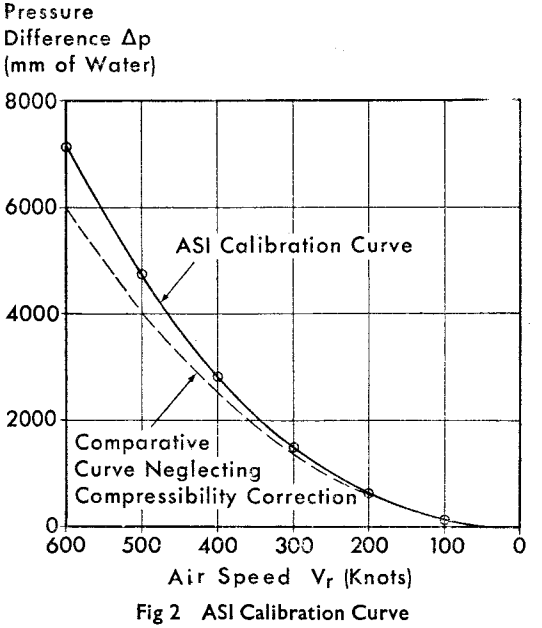


Fig 2 ASI Calibration Curve

CHAPTER 4

MACHMETERS

CONTENTS

	Paras
Introduction	1–2
Principle	3
Construction	4–8
Calibration	9
Errors	10–12

Annex: Theoretical Basis of Mach Number Measurement

Introduction

1. **Mach Number.** As an aircraft's speed approaches the speed of sound, the airflow around the aerofoils exhibits a marked change, characterized by the occurrence of shock waves. These will occur locally, depending on the aircraft design, at some speed below the speed of sound and will increase in effect and extent as the speed is further increased. They can cause loss of aerodynamic lift, changes in aerodynamic stability erratic control loads, loss of control effectiveness and buffetting. The onset of these shock waves and their subsequent effects occur, for a given aircraft type, when the true air speed is a certain proportion of the local speed of sound. For convenience, the ratio of true air speed to the local speed of sound is considered as a single entity. It is called Mach number and is usually expressed thus:

$$\text{Mach number (M)} = V/a$$

where: V = True air speed
a = Local speed of sound

The Mach numbers corresponding to subsonic, transonic and supersonic speeds are 0–0·75M, 0·75–1·2M and 1·2M to infinity respectively. The Mach number at which the shock waves initially occur is called Critical Mach Number.

2. **Machmeter.** Because of the effect of the shock waves on stability and control of the aircraft, it is important that the pilot knows his speed in terms of Mach number. This is achieved by an instrument called a Machmeter which gives a direct display of Mach number and usually has an adjustable index which can be set to the Critical Mach Number of the aircraft in which it is installed.

Principle

3. As explained in para 1, the local Mach number varies with the true air speed and the speed of sound. True air speed is a function of pitot excess pressure, *ie* the difference between pitot and static pressure, and density. The local speed of sound is a function of static pressure and density. As the density factor is common to both functions, Mach number can be expressed as:

$$M = \frac{V}{a} = f\left(\frac{p_t - p}{p}\right)$$

where: V = True air speed
a = Local speed of sound
p_t = Pitot pressure
p = Static pressure

The machmeter is designed to measure the ratio of pitot excess and static pressures and the dial is calibrated to show the corresponding Mach number. A fuller description of the relationship between Mach number, pitot excess pressure, static pressure and the true air speed with the associated numerical values is given in the Annex to this chapter.

Construction

4. A typical machmeter is shown in Fig 1. It consists essentially of a sealed case containing two capsule assemblies and the necessary mechanical linkages. The interior of the case is connected to the static pressure pipeline. The interior of one capsule unit, the air speed capsule, is connected to the pitot pressure pipeline. The second capsule unit, the altitude capsule, is sealed and evacuated to respond to static pressure changes.

Fig 1 A Typical Machmeter

5. The air speed capsule measures the pressure difference between pitot and static pressure and therefore expands or contracts as air speed increases or decreases. The movement of the capsule is transferred by the air speed link to the main shaft, causing it to rotate and move a pivoted arm (the ratio arm) in the direction A–B (*see* Fig 2).

6. The altitude capsule responds to changes of static pressure, expanding or contracting with variation of altitude. The movement of the capsule is transferred to the ratio arm, via a spring and pin, causing it to move in the direction C–D. The pin is pointed at both ends and rests in cups on the altitude capsule and ratio arm; the spring providing the tension necessary to retain the pin in position.

7. The position of the ratio arm depends, therefore, upon both pitot excess and static pressures. Movement of the ratio arm controls the ranging arm which, through linkage and gearing, turns the pointer thus displaying Mach number corresponding to the ratio of pitot excess pressure to static pressure. An increase of altitude and/or air speed results in a display of higher Mach number.

8. Critical Mach Number is indicated by a specially shaped lubber mark located over the dial of the machmeter. It is adjustable so that the critical Mach number for the particular type of aircraft in which the machmeter is installed may

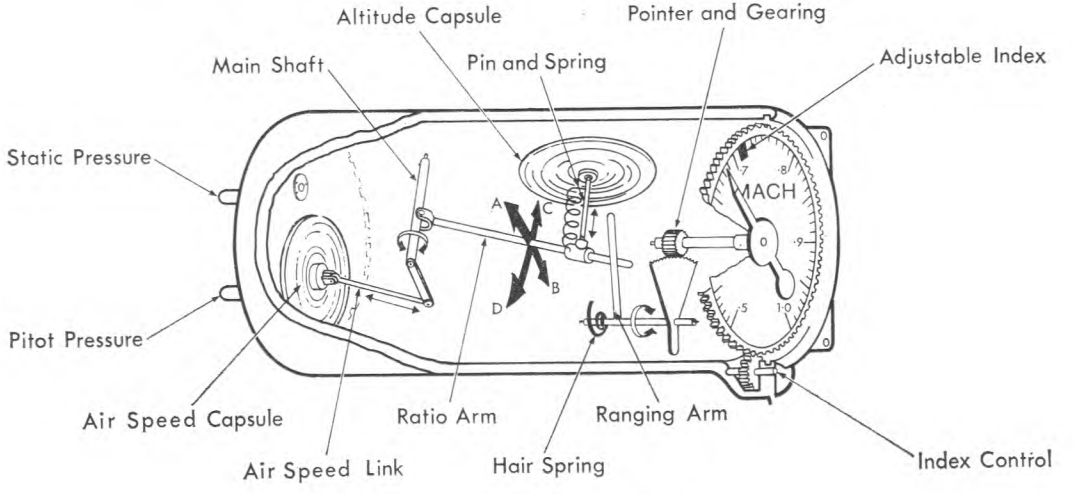


Fig 2 Principle of Operation of a Machmeter

be preset. Presetting can be carried out by an adjusting screw on the front of the instrument.

Calibration

9. Machmeters are calibrated to formulae relating Mach number to the atmospheric conditions of pressure and density. The laws involved between the pressures and the speeds take a different form at subsonic and supersonic speeds. At supersonic speeds a shock wave forms ahead of the pitot-static head and a different process is involved from that at subsonic speeds. The two flight regimes have to be considered separately, although the subsonic and supersonic relationships must reduce to the same values at $M = 1$. The formulae used for calibration of Machmeters are given in the Annex to this chapter.

Errors in Machmeters

10. As Mach number is effectively a function of the ratio of pitot excess pressure to static pressure, only those errors in the measurement of this ratio will affect the machmeter. There are only two such errors; instrument error and pressure error. Variations in air density and temperature from the standard mean sea level values have no effect.

11. **Instrument Error.** Like all instruments, machmeters are subject to tolerances in manufacture which produce errors that vary from instrument to instrument. These are, however, small and are, typically, of the order of $\pm$ 0·01M over a range of 0·5 to 1·0M.

12. **Pressure Error.** The machmeter operates from the same pressure source as the air speed indicator and is therefore subject to the same pressure errors as discussed in Chap 3. However, the effect of pressure error is relatively greater on the machmeter as the ratio of pitot excess pressure (p_t–p) to static pressure (p) is being measured rather than just the pitot excess pressure (p_t–p) in the case of the ASI.

THEORETICAL BASIS OF MACH NUMBER MEASUREMENT

CONTENTS

	Paras
Introduction	1
Subsonic Speeds	2
Supersonic Speeds	3–5
Application of Formulae	6

Introduction

1. The laws involved between pressures, *ie* pitot and static pressure, and speeds, *ie* true air speed and local speed of sound, take a different form at subsonic and supersonic speeds. At subsonic speeds the processes involved are isentropic (*ie* adiabatic and reversible), but at supersonic speeds a shockwave forms ahead of the pitot-static head and a nonisentropic (*ie* adiabatic but irreversible) process is involved. These two flight regimes have to be considered separately but as stated in the main chapter, both the subsonic and supersonic relationships must equate to the same value at M = 1.

Subsonic Speeds

2. Bernouilli's Law states that, if a compressible airflow is brought to rest isentropically, the total or pitot pressure (p_t) is related to the static pressure (p) and the true air speed (V) by the equation:

$$\frac{p_t}{p}=\left(1+\tfrac{1}{2}\cdot\frac{\gamma-1}{\gamma}\cdot\frac{\rho V^2}{p}\right)^{\gamma/(\gamma-1)} \quad \ldots.(1)$$

where: ρ = Air density
γ = Ratio of specific heats at constant pressure and constant volume.

But the speed of sound in air is given by:

$$a \text{ (speed of sound)} = \sqrt{\left(\frac{\gamma p}{\rho}\right)}$$

or

$$a^2=\frac{\gamma p}{\rho}$$

Substituting in equation (1):

$$\frac{p_t}{p}=\left(1+\frac{\gamma-1}{2}\cdot\frac{V^2}{a^2}\right)^{\gamma/(\gamma-1)} \quad \ldots.(2)$$

But V/a = Mach number (M), therefore:

$$\frac{p_t}{p}=\left(1+\frac{\gamma-1}{2}\cdot M^2\right)^{\gamma/(\gamma-1)} \quad \ldots.(3)$$

Equation (3) can be rewritten for M as:

$$M^2=\frac{2}{\gamma-1}\left\{\left(\frac{p_t}{p}\right)^{(\gamma-1)/\gamma}-1\right\} \quad \ldots.(4)$$

If

$$\left(\frac{p_t-p}{p}+1\right)$$

is substituted for

$$\left(\frac{p_t}{p}\right)$$

equation (4) becomes:

$$M^2=\frac{2}{\gamma-1}\left\{\left(\frac{p_t-p}{p}+1\right)^{(\gamma-1)/\gamma}-1\right\} \quad \ldots.(5)$$

Which becomes, when $\gamma = 1{\cdot}4$ is substituted:

$$M^2=5\left\{\left(\frac{p_t-p}{p}+1\right)^{\frac{2}{7}}-1\right\} \quad \ldots.(6)$$

This is the calibration law used for machmeters at subsonic speeds. It can be seen that the value of M is dependent on (p_t-p) and p.

Supersonic Speeds

3. At supersonic speeds the reading of the machmeter is still dependent on the ratio between the pitot excess pressure and static pressure. It is, however, necessary to allow for the presence of a normal shockwave ahead of the pitot tube; the flow process is no longer an isentropic one. The pitot tube no longer picks up the total free stream pressure but the total pressure behind the shockwave.

4. The ratio of the total pressure behind a normal shockwave (p_{t2}) to the static pressure (p) is given by the Rayleigh Supersonic Pitot Formula:

$$\frac{p_{t2}}{p} = \frac{\left(\frac{\gamma+1}{2}M^2\right)^{\gamma/(\gamma-1)}}{\left\{1+\frac{2\gamma}{\gamma+1}(M^2-1)\right\}^{1/(\gamma-1)}} \quad \ldots.(7)$$

This replaces equation (2) which is only valid for an isentropic process. The pitot tube will pick up p_{t2} which is sometimes called the Impact Pressure. At speeds below $M = 1$, the impact pressure is identical to the total pressure p_t.

5. Equation (7) becomes, when $\gamma = 1{\cdot}4$ is substituted:

$$\frac{p_{t2}}{p} = \left(\frac{6}{5}M^2\right)^{3{\cdot}5} \cdot \left(\frac{6}{7M^2-1}\right)^{2{\cdot}5} \quad \ldots.(8)$$

Which by resolution becomes:

$$\frac{p_{t2}}{p} = \frac{166{\cdot}92M^2}{\left(7-\frac{1}{M^2}\right)^{2{\cdot}5}} \quad \ldots.(9)$$

If

$$\left(\frac{p_{t2}-p}{p}+1\right)$$

is substituted for

$$\frac{p_{t2}}{p},$$

equation (9) becomes:

$$\frac{p_{t2}-p}{p} = \frac{166{\cdot}92M^2}{\left(7-\frac{1}{M^2}\right)^{2{\cdot}5}} - 1 \quad \ldots.(10)$$

This is the calibration law for machmeters at supersonic speeds. Once again the Mach number is dependent on $(p_{t2}-p)$ and p.

Application of Formulae

6. For values of

$$\left(\frac{p_t-p}{p}\right) < 0{\cdot}893,$$

the subsonic formula is used. For higher values of

$$\left(\frac{p_t-p}{p}\right),$$

the supersonic formula is used.

CHAPTER 5

AIR MILEAGE UNITS

CONTENTS

	Paras
Introduction	1
Principle	2–5
AIR MILEAGE UNIT, MARK 1	
Introduction	6
Description	7–14
Operation	15–17
Accuracy	18
AIR MILEAGE UNIT, MARK 2	
Introduction	19
Description	20–24
Operation	25
Accuracy	26
AIR MILEAGE UNIT, MARK 4	
Introduction	27
Description	28–37
Operation	38–39
Accuracy	40
AIR MILEAGE UNIT, MARKS 4A AND 4B	
Description	41–42

Introduction

1. One requirement of certain types of automatic DR systems is an input of true air speed to a high accuracy. The Air Mileage Unit (AMU) is designed to provide a continuous measure of true air speed in the form of a shaft rotation rate proportional to TAS. This rotation of the AMU shaft may be used, for example in the Air Position Indicator, where it is combined with true heading in a resolving mechanism enabling a fully automatic air plot to be maintained. The shaft rotation may also be used to provide air distance flown to an Air Mileage Indicator (AMI).

Principle

2. If a centrifugal fan, fitted in an airtight case, is driven by an electric motor, the pressure produced at the periphery of the fan will depend upon the pressure and density of the air at the centre of the fan, the radius and the angular velocity of the fan. Similarly, the pressure developed in the pitot tube of an aircraft depends upon the pressure and density of the air outside the aircraft and the aircraft's true air speed.

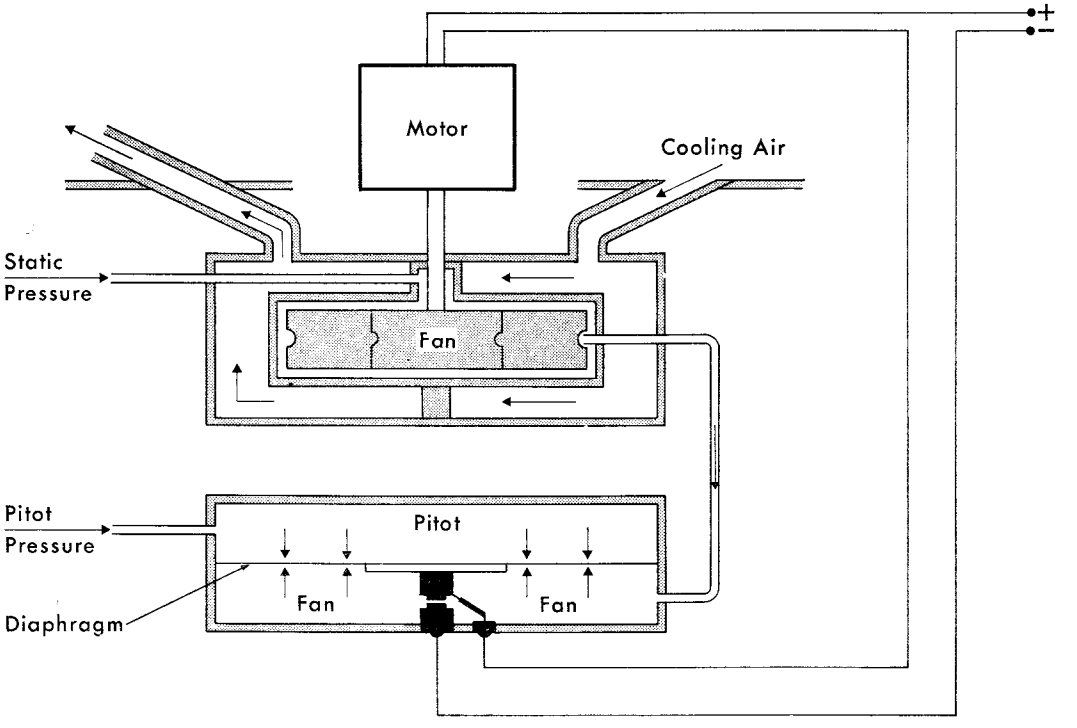


Fig 1 Principle of Air Mileage Unit

3. The speed of rotation of the fan can be controlled by balancing the pressure at the periphery of the fan against the pressure developed in the pitot tube. If the air at the centre of the fan is of the same pressure and density as the air outside the aircraft, the pressure balance will result in the fan, and the electric motor, rotating at a speed proportional to true air speed.

4. Fig 1 shows how this basic principle could be utilized in its most simple form. Static pressure is fed to the centre of the fan and the density at the fan centre is maintained equal to the atmospheric density by a flow of cooling air around the fan chamber to maintain it at atmospheric temperature. The fan pressure is applied to a diaphragm whose movement is opposed by the application of pitot pressure on the other side. This diaphragm operates a contact controlling the fan drive motor. Whenever the pitot pressure exceeds the fan pressure, the contact is made and the motor speeds up until the fan pressure exceeds the pitot pressure, when the contact is broken. With this arrangement, the fan drive motor will hunt about a mean speed proportional to true air speed

5. Unfortunately this simple arrangement has disadvantages which preclude its use in this form. However the principle of the simple AMU forms the basis of all practical air mileage units, and additional features are incorporated to overcome the disadvantages.

AIR MILEAGE UNIT, MARK 1

Introduction

6. The AMU, Mk 1 is designed to provide an output of TAS, over a range of 100 to 300 knots, by means of a shaft rotating at a speed proportional to TAS. This shaft output can be used to operate the mechanism of the air position indicator (API), the air mileage indicator (AMI) or both.

Description

7. The principle of operation of the AMU, Mk 1 is as described for the simple AMU in paras 2 to 4. This description is therefore limited to the additional features which are incorporated in the AMU, Mk 1.

8. **Diaphragm Areas.** The fan produces a pressure which is small in comparison with the aircraft's pitot pressure, since for a fan to produce a pressure equal to pitot pressure it would need a large radius, a high speed of rotation or a combination of both. A composite diaphragm arrangement is introduced where the higher pitot pressure acting on a diaphragm of small area is balanced by a lower fan pressure acting upon a diaphragm of larger area. Fig 2 shows how this ratio (9·6 : 1) of diaphragm areas is achieved; static pressure being introduced solely to balance the static pressure components of both pitot and fan pressures.

Pressures: Pitot Excess + Static = Pitot
Fan Excess + Static = Fan

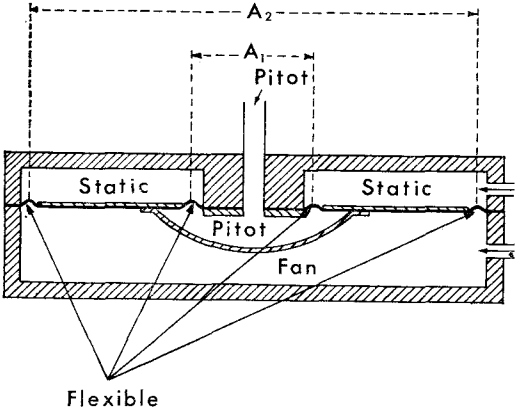


Fig 2 Diaphragm Areas

9. **Low Air Speed Cut-Out.** In the diaphragm chamber an additional diaphragm is fitted, together with a second pair of contacts. This diaphragm shown in Fig 3 is spring loaded to keep the contacts open when the unit is not operating. When the pitot pressure exceeds the spring pressure, which occurs with pitot pressures corresponding to air speeds of 65 to 75 knots, the contacts are closed. These contacts are in series with the motor which can therefore only operate when the contacts are closed. To facilitate ground testing, the low speed diaphragm operates a push rod fitted to the pitot dome which holds the main diaphragm contacts closed when the unit is not operating. In flight, the low speed spring is compressed by pitot pressure, and the low speed diaphragm is clear of the top of the push rod which then plays no part in the operation of the instrument.

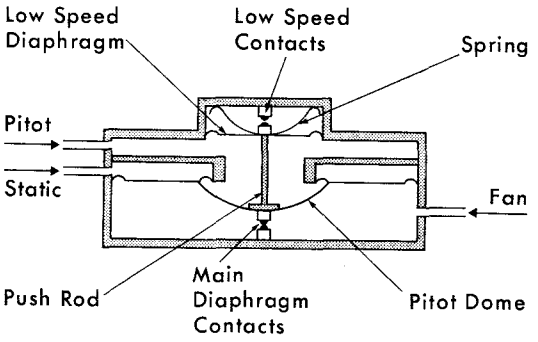


Fig 3 AMU, Mk 1—Diaphragm Assembly

10. **Valve Relay.** To avoid arcing and thus sooting, of the main diaphragm contacts, they are fed with a low operating current from a valve relay. This relay operates contacts in the motor circuit. The circuit is arranged so that when power is applied to the unit, the relay operated contacts are in the same position, open or closed, as the main diaphragm contacts.

11. **Gear Train.** In compliance with the AMU, Mk 1 operating range of 100 to 300 knots TAS, maximum motor speed is 6,000 rpm giving 1,200 revolutions per nautical mile. The output shaft is driven by the motor shaft through a 50 : 1 reduction gear, producing 24 revolutions for each nautical mile of distance flown.

12. **Control Panel.** Control of the AMU is affected either by controls fitted to the API or, if no API is fitted to the aircraft, by means of the Switch Panel, Mk 1 used in conjunction with the AMI (*see* Chap 6). The AMU controls are:

a. An ON/OFF switch.

b. An indicator lamp in the motor circuit which "winks" regularly when the AMU is

functioning correctly.

c. A GROUND TEST push button which by-passes the low speed cut-out switch.

The indicator lamp is not fitted on the Switch Panel, Mk 1.

13. **Calibration and Adjustment.** The simple AMU is based on the principle that there is a linear relationship between fan speed and true air speed, shown by the straight line in Fig. 4. However, compressibility of the air in the pitot tube will be greater than that in the fan, resulting in an increase in fan speed. Heating of the air drawn into the fan chamber cannot be entirely eliminated and therefore the density of the fan air is not the same as the outside air. Once again this will result in an increase in fan speed to balance the two pressures. The total effect of these errors is shown by the blue curve in Fig 4. Two adjustments are provided to allow for correction of the fan speed. These adjustments are:

a. *High Speed Adjustment.* The working position of the diaphragm and hence the ratio of diaphragm areas is altered by means of the adjustable lower main diaphragm contact. This adjustment can therefore be used to reduce the speed of the fan, and is used to produce 5000 rpm at 250 knots under standard sea level conditions, as shown by the red curve in Fig 4. Fan speed below 250 knots will now be too slow.

b. *Low Speed Adjustment.* The weight of the diaphragm is supported by a balance weight on a lever over a knife-edged fulcrum. A spring above the balance weight gives a downward force of $\frac{1}{2}$ lb on the lever. The spring can be set to apply bias to the diaphragm, since at any point on the lever, other than directly over the knife-edged fulcrum, the spring provides a couple on the lever altering the effect of the balance weight. In this case the spring is adjusted to apply a downward force to the diaphragm, causing the fan to increase speed to balance the additional effect of this force. The bias is set to correct the fan speed at a TAS of 100 knots under standard sea level conditions (*see* green curve in Fig 4). This adjustment has negligible effect on the high speed adjustment. Details of both the high speed and the low speed adjustments are shown in Fig 5.

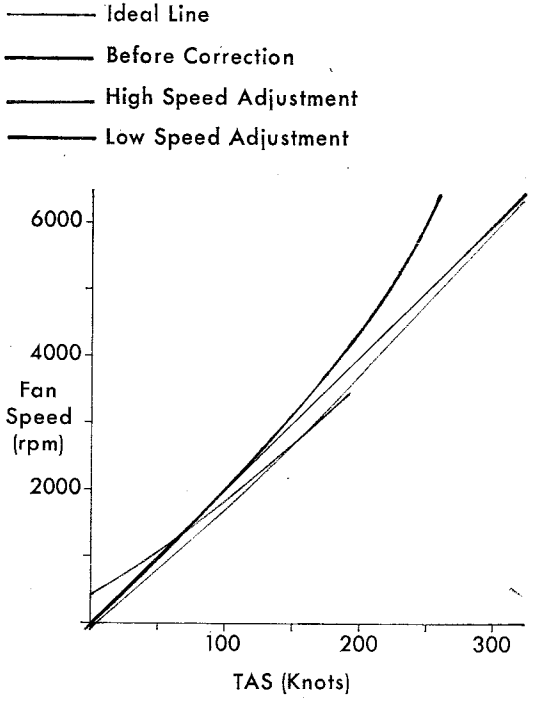


Fig 4 AMU, Mk 1—Calibration Curves

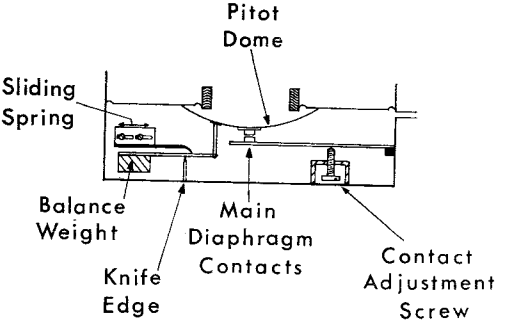


Fig 5 AMU, Mk 1—Calibration Adjustments

14. **Transmission.** Rotation of the output shaft is normally transmitted through a flexible drive to the API. In addition, a transmitter fitted to the output shaft produces electrical pulses to operate the AMI. Where the distance between the AMU and the API is greater than the maxi-

mum flexible drive length of 5 feet 6 inches, an electrical transmission system is used. An M-type transmitter connected to the output shaft electrically drives a repeater motor connected to the API by a short length of flexible drive. The M-type transmitter also incorporates a contactor to operate the AMI.

Operation

15. **Ground Test.** Operation of the GROUND TEST button by-passes the low speed contacts (*see* Fig 6) With the MAIN switch on and 30 seconds having been allowed for valve warm-up, the relay contacts close (the main diaphragm contacts being held closed by the push rod) and the motor circuit is energized. The motor builds up speed until the fan pressure exceeds the pressure of the low speed spring. At this point the main diaphragm and relay contacts open, the motor circuit is completed through the high resistance path of the parallel lamp and resistance, and the motor slows down until the fan pressure falls to a lower value than spring pressure. The cycle is then repeated resulting in the lamp "winking" and the API and/or AMI recording air distance. If the test button is operated before the valves have warmed up, the lamp will be permanently lit until the relay contacts close when it will start to "wink" as described above.

16. **In Flight.** Pitot pressure acting on the low speed diaphragm closes the low speed contacts as soon as the air speed reaches 65 to 75 knots. Providing the MAIN switch is on and the relay valves are warmed-up, the AMU starts to operate. Pitot pressure holds the main diaphragm contacts, and thus the relay contacts, closed and the motor builds up speed until fan pressure exceeds pitot pressure. The contacts then open, the motor slows down until pitot pressure exceeds fan pressure, the contacts again close and the cycle restarts. Regular "winking" of the lamp indicates that the relay contacts and thus the motor, fan, diaphragm contacts and valve relay are operating correctly (*see* Fig 7).

17. **Fault Diagnosis and Rectification.** The most likely faults which develop in the AMU are

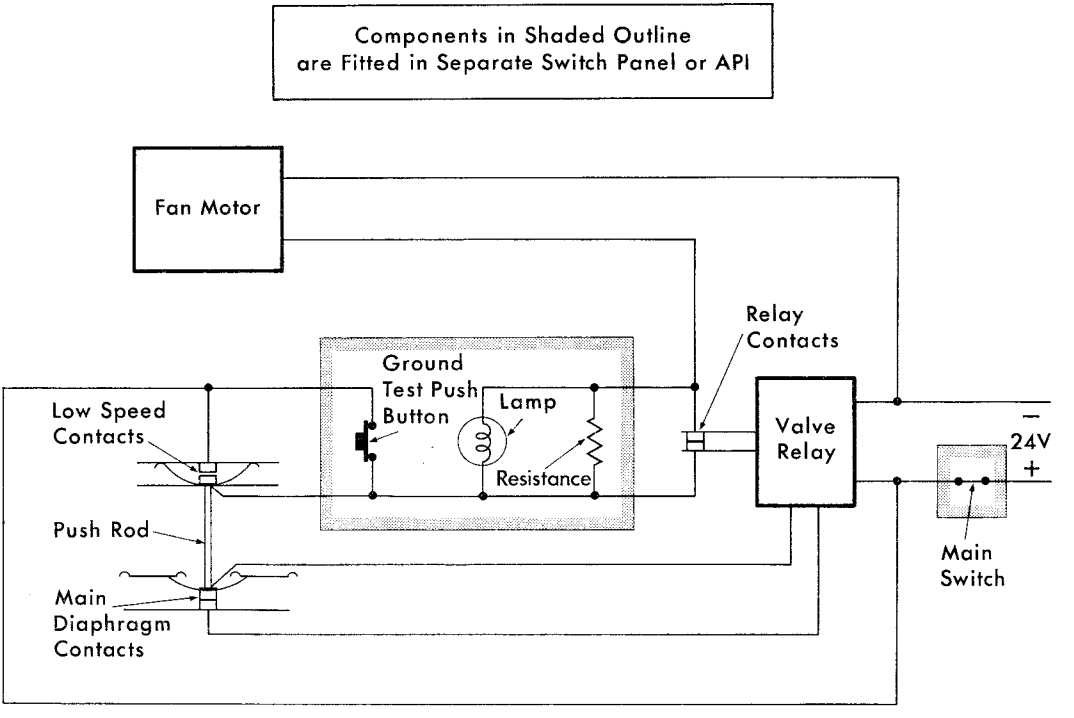


Fig 6 AMU, Mk I—Electrical Circuit

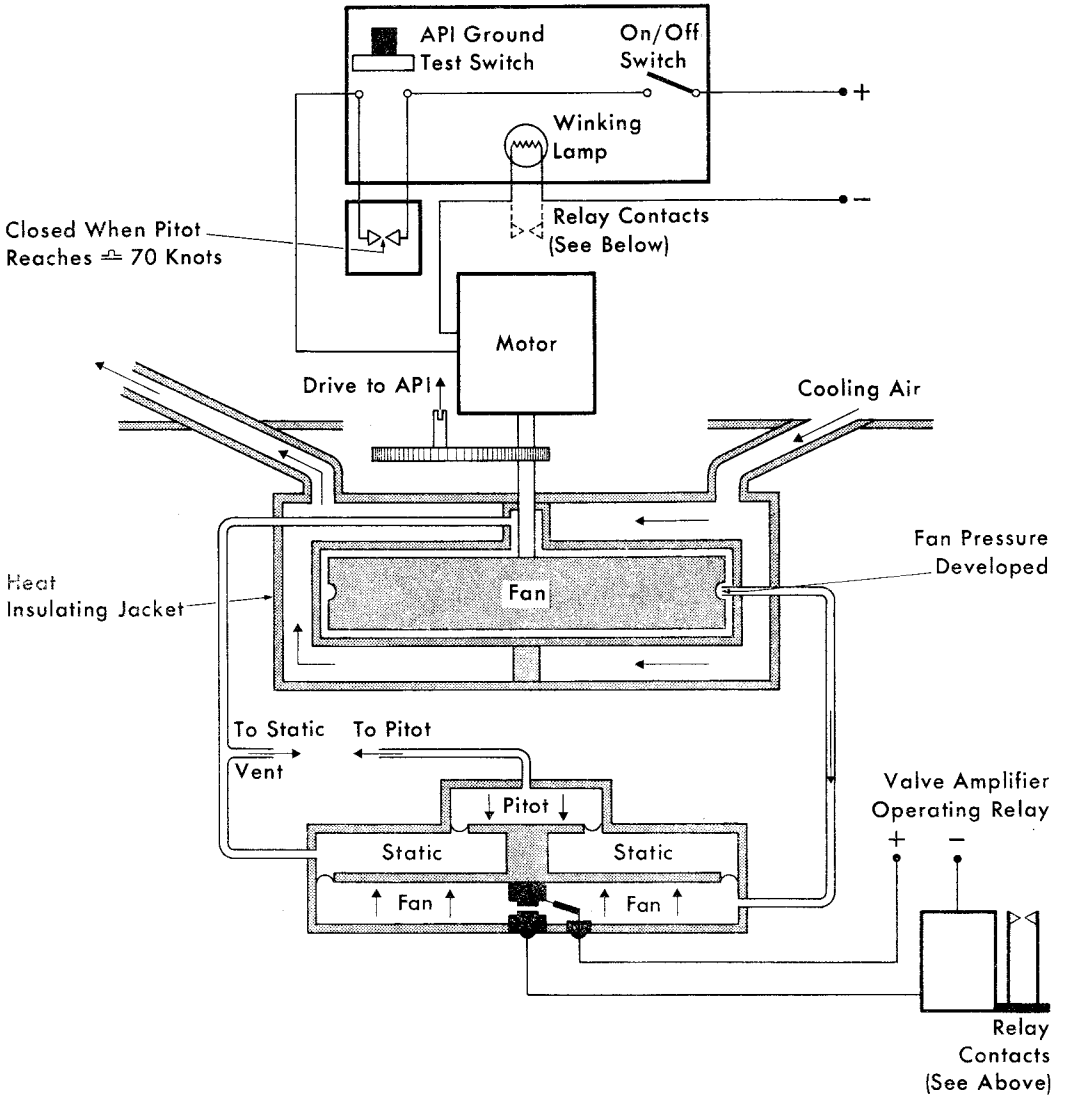


Fig 7 AMU, Mk I—Simplified Schematic

indicated by either the lamp being on or off permanently:

a. *Lamp On Permanently.* This indicates a fault in the AMU. Switch off.

b. *Lamp Off Permanently.* This indicates either a fused lamp or a fault in the AMU. Check output of air distance by manual plotting methods. If reasonably accurate continue to use with caution. If output is obviously in error, switch off the AMU.

Accuracy

18. Accuracy over the operating range of 100–300 knots is of the order of ± 2%. Like all instruments that use pitot/static pressures, the AMU is subject to pressure error (*see* Chap 3).

AIR MILEAGE UNIT, MARK 2

Introduction

19. The Air Mileage Unit, Mk 2, is similar in design to the AMU, Mk 1. The operating principle is the same, but there are additional features which extend the speed range to 400 knots. Errors due to compressibility and heating are kept to a minimum.

Description

20. The description of the AMU, Mk 1 given above applies, with certain modifications, to the AMU, Mk 2. This description is therefore limited to those modifications and additions peculiar to the AMU, Mk 2.

21. **Fan.** The corrections for compressibility and adiabatic heating described in para 13 are only satisfactory up to an air speed of 300 knots. The AMU, Mk 2 is therefore fitted with a variable bladed fan which gives a greater increase in fan pressure at higher speeds to compensate for these errors. The fan (*see* Fig 8) is fitted with phosphor bronze strips, on leaf springs, at the blade tips. Under the action of centrifugal force these strips bend outwards and increase the effective radius of the fan, and fan pressure is increased with the increased radius as well as with increase in speed. Stops are fitted to the backing plate to prevent the tips fouling the fan scoop if overspeeding should occur. The high speed and low speeds adjustments described in para 13 are retained for correction below 300 knots.

22. **Fan Cooling System.** To provide a greater flow of cooling air in order to maintain the fan chamber air temperature close to that of the outside air at altitude, the inner heat insulating jacket of the AMU, Mk 1 has been removed. The slight loss of heat insulation is more than balanced by the increased flow of cooling air.

23. **Diaphragm Areas.** The top speed of the fan motor is the same as the AMU, Mk 1, *ie* 6,000 rpm, but, in the case of the AMU, Mk 2, is equivalent to 400 knots. Therefore at a given TAS, the fan speed of the Mk 2 is three-quarters that of the Mk 1. Likewise the Mk 2 fan develops less pressure than the Mk 1 at the same air speed. This results in an increase in the ratio of pitot pressure to fan pressure which requires a similar change in the diaphragm area ratio. The diaphragms of the AMU, Mks 1 and 2 are externally the same, but by machining the diaphragm castings and altering the dimensions of the diaphragm annular backing plate, the Mk 2 ratio is changed to 21·6 : 1.

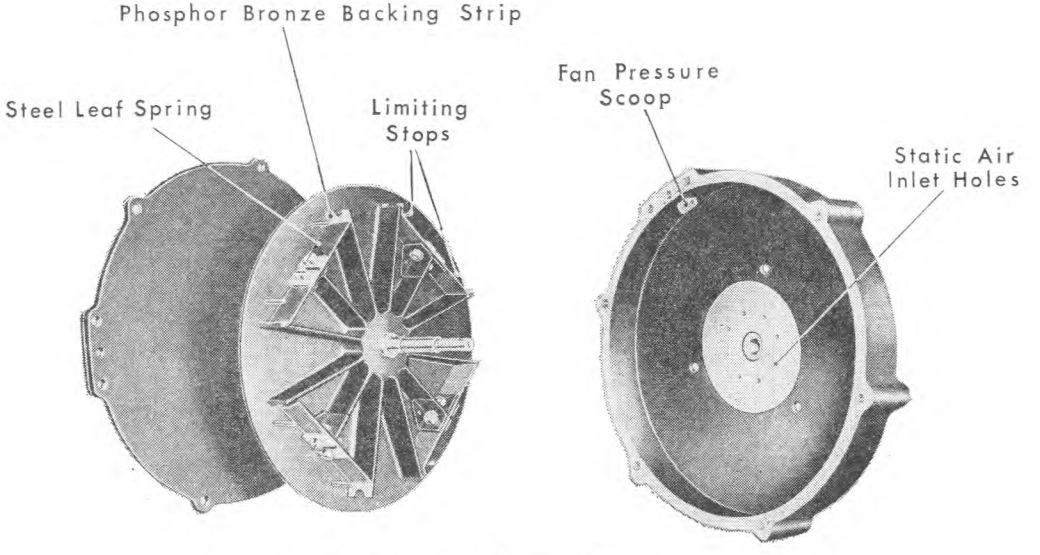


Fig 8 AMU, Mk 2—Details of Fan Assembly

24. **Gear Train.** To maintain the output of 24 revs per nautical mile, the motor drives the output shaft through a 37·5 : 1 reduction gear, *ie* 6,000 rpm is equivalent to 400 knots or 900 rev of the motor per nautical mile.

Operation

25. Operation of the AMU, Mk 2 is exactly the same as the AMU, Mk 1 (*see* paras 15–17). AMU controls are fitted to the API (*see* Fig 9) or the Switch Panel, Mk 1.

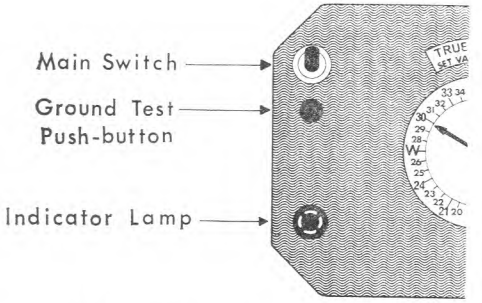


Fig 9 AMU Controls Fitted to API

Accuracy

26. Accuracy over the operating range of 100–400 knots is of the order of ±2%. The instrument is also subject to pressure error.

AIR MILEAGE UNIT, MARK 4

Introduction

27. The Air Mileage Unit, Mk 4 is a development of the earlier models and provides an electrical output from a transmitter driven by a shaft rotating at a speed proportional to TAS. Operating range is 150–550 knots. An additional electrical transmission supplies a 28 volt DC pulse for every 0·1 nm of air distance flown to be used by an AMI.

Description

28. The AMU, Mk 4 operates on the same basic principle as the simple AMU, but in order to correct for the effects of compressibility and adiabatic heating at high speeds, pitot pressure is fed to the centre of the fan instead of static pressure. Thus the fan air density increases with air speed and the fan runs more slowly to balance the two pressures.

29. **Diaphragm Arrangement.** Because of the change from static to pitot pressure in the fan, a new diaphragm arrangement is required, this is shown in Fig. 10. Since fan pressure is greater than pitot pressure, it must be applied over a smaller area than pitot pressure; static pressure is again introduced to balance the static component of both the fan and pitot pressures.

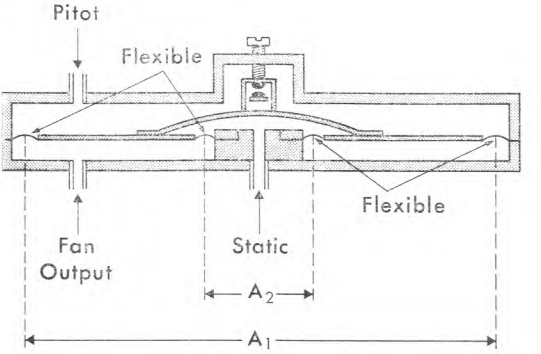
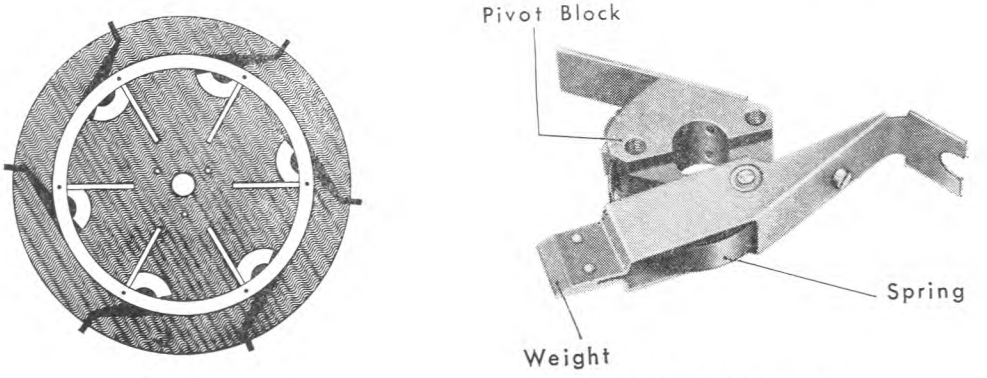


Fig 10 AMU, Mk 4—Diaphragm Assembly

30. **Fan.** Feeding pitot pressure to the fan makes too great a compensation for compressibility and heating and the fan runs too slowly at high air speeds. The fan is therefore designed so that its effective radius decreases as its speed increases. As shown in Fig 11, the blade tips are mounted on pivots and spring loaded to the extended position. Small weights are fitted on the inner ends of the arms carrying the blade tips and under the action of centrifugal force, the weights overcome spring tension and move out, progressively retracting the blade tips. The fan therefore increases its speed to balance the pressures.

31. **Fan Cooling.** The fan is mounted in the base of the instrument and the AMU, Mk 4 is mounted in the aircraft so that the base is flush with the aircraft skin and exposed to the outside airstream. In addition a cooling girdle is fitted to circulate air around the exterior of the fan chamber.

32. **Motor Control System.** The main diaphragm contacts control a valve relay which operates two



a. Fan Assembly

b. Fan Blade Assembly

Fig 11 AMU, Mk 4—Details of Fan Assembly

sets of contacts, one set short-circuiting a resistance and lamp in parallel as in the AMU Mks 1 and 2, and the other set operating a reversing relay controlling a servo-motor. The servo-motor drives the wipers of a bank of four resistors in series with the fan motor adjusting the resistance to a value corresponding to the required motor speed. This system reduces the hunt of the fan motor to a minimum.

33. **Low Speed Cut-Out.** The low speed diaphragm in the AMU, Mk 4 is divorced from the main diaphragm assembly and consists of a separate chamber divided by a spring-loaded diaphragm, the spring normally keeping the contacts apart. Pitot and static pressure are fed to opposite sides of the diaphragm and when pitot pressure reaches an equivalent of 60-90 knots, the contacts are closed.

34. **Ground Test Solenoid.** The main diaphragm contacts cannot be held closed by the low speed diaphragm as it is divorced from the main assembly. Instead, the ground test button, as well as short-circuiting the low speed contacts, causes a solenoid to be energized. This solenoid is linked to the main diaphragm balance weight, and when energized, lifts the balance weight which in turn causes the main diaphragm contacts to close.

35. **Gear Ratios.** The maximum motor speed is 8,250 rpm equivalent to a TAS of 550 knots *ie* 900 rev per nautical mile. The shaft rotation drives an 18·75 : 1 reduction gear which supplies the API transmitter shaft with 48 rev per nautical mile. In addition a further reduction gear of 90 : 1 supplies the AMI contactor shaft with 10 rev per nautical mile.

36. **Transmission.** The API transmitter is an M-type transmitter producing 24 steps of 15 degrees for each revolution and 48 revolutions per nautical mile. This transmission is received in the API by a repeater motor which, through a 2 : 1 reduction gear, provides a mechanical drive of 24 rev per nautical mile. Transmission to the AMI is by 28 volt DC pulses generated by a pair of cam-operated contacts at a rate of one pulse per 0·1 nm of air distance flown.

37. **Control Panel.** The AMU, Mk 4 is controlled from a separate panel shown in Fig. 12. The controls mounted on the front are as follows:

a. *ON/OFF Switch.* This switch allows power to be supplied to the AMU and should be switched on prior to take off. This ensures that the valves are warmed up.

b. *ELECTrical TRANSmission Switch.* This controls the electrical transmissions to the API and the AMI.

c. *GROUND TEST Button.* This button initiates the ground test procedure (*see* para 38).

d. *Lamp Shutter Switch.* This controls a three-position shutter over the lamp allowing bright or dim viewing or complete obscurity. A fourth position allows access to the lamp for bulb replacement.

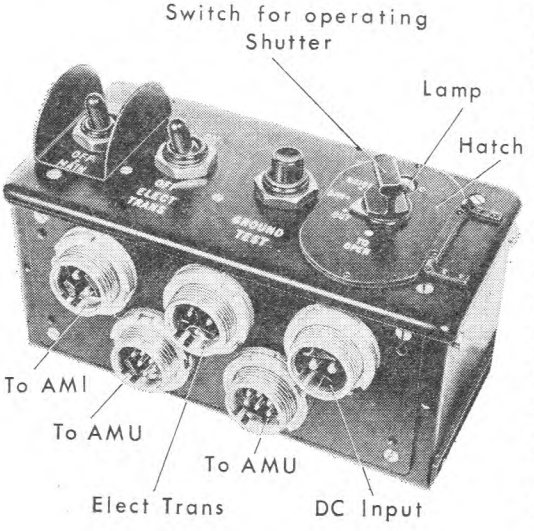


Fig 12 AMU, Mk 4—Control Panel

Operation

38. **Ground Test.** Switch on the MAIN switch and allow at least 30 seconds for the valves to warm up. When the GROUND TEST button is pressed, the low speed contacts are short-circuited and the ground test solenoid is energised closing the main diaphragm contacts. The relay contacts close, the servo-motor reduces the resistance in the motor circuit and the fan motor accelerates until fan pressure builds and opens up the main diaphragm contacts. The relay contacts then open, the servo-motor reverses increasing the resistance in the motor circuit thus slowing down the fan motor until the main diaphragm contacts again close. The cycle is then repeated. During ground test the AMU may run at any speed corresponding to a TAS range of 70–270 knots. The lamp is likely to "wink" more rapidly than in flight. To check the transmission to the API or the AMI, the ELECTrical TRANSmission switch must be on.

39. **In Flight.** Switch on the MAIN and ELECTrical TRANSmission switches and allow at least 30 seconds for the valves to warm up. When airspeed reaches 60–90 knots, pitot pressure closes the low speed and the main diaphragm contacts. The relay contacts close, the servo-motor reduces the resistance in the motor circuit and the fan motor accelerates until fan pressure has built up sufficiently to open the main diaphragm contacts. The relay contacts then open, the reversing relay controls the servo-motor to increase the resistance and the motor slows down until the main diaphragm contacts close again. The cycle is continually repeated with the fan motor hunting about a mean speed corresponding to the true air speed. The lamp will "wink" showing tha tthe AMU is operating correctly. The GROUND TEST button should not be pressed during flight. After flight the MAIN and ELECTrical TRANSmission switches should be placed to the "OFF" position.

Accuracy

40. The operating range of the AMU Mk 4 is 150–550 knots. Over the range 300–550 knots accuracy is of the order of $\pm 2\frac{1}{2}\%$; outside this range accuracy is reduced. The AMU Mk 4 is similarly subject to pressure error as the earlier marks of instrument.

AIR MILEAGE UNIT, MARKS 4A AND 4B

Description

41. **AMU, Mk 4A.** The AMU, Mk 4A is similar to the Mk 4 instrument but an alternative valve and an additional relay are introduced to provide improved control of the power supply to the fan motor.

42. **AMU, Mk 4B.** The AMU, Mk 4B is similar to the Mk 4A instrument but an additional high speed contactor is introduced to provide a series of 28 volt DC pulses at a rate proportional to true air speed for use by the Navigation and Bombing System, Mk 1.

CHAPTER 6

AIR MILEAGE INDICATOR, MARK 1

CONTENTS

	Paras
Introduction	1
Principle	2
Description	3–5
Operation	6–8
Accuracy	9

Introduction

1. The Air Mileage Indicator, Mk 1 is designed to provide a continuous indication of air distance flown and is driven by the Air Mileage Unit (Mks 1, 2 or 4) or the True Air Speed Unit, Mk 1. It is primarily intended for use by the air engineer but may be of assistance to the navigator in maintaining a manual air plot when an API is not fitted or is unserviceable. This chapter should be read in conjunction with Chaps 5 and 8.

Principle

2. The air mileage unit (AMU) or the true air speed unit (TAU) provide, either directly or via a flexible driven electrical transmitter, 28 volt DC pulses at a rate of one pulse every 0·1 nautical mile of air distance flown. These pulses are used to drive a set of veeder type counters recording air mileage in steps of 0·1 nautical mile.

Description

3. The Air Mileage Indicator (AMI) installation consists of:

a. The indicator unit

b. Control switches.

4. **Indicator Unit.** The indicator unit (*see* Fig 1) consists of a set of veeder type counters which record the DC pulses from the contactor mechanism in the AMU, the electrical transmitter or the TAU. The counters are operated by a solenoid which is triggered each time a pulse is received from the AMU/TAU; each operation of the solenoid increases the counter reading by 0·1 nm. The counters can record up to 999·9 nm before automatically turning over to zero. A resetting knob on the left side of the indicator allows the counters to be reset to zero at any time. The figures of the hundreds, tens and units counters are coloured white and those of the tenths counter are coloured green.

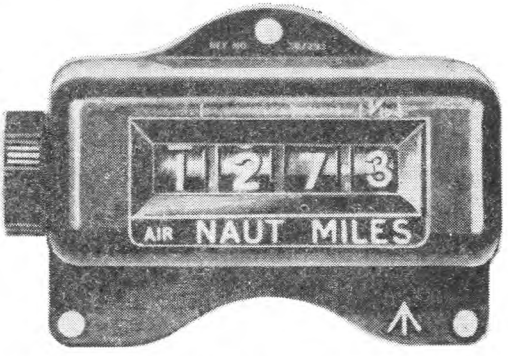


Fig 1 Air Mileage Indicator, Mk 1

5. **Control Switches.** The control switches for the the AMI are the same as for the AMU or TAU, but their position in the aircraft varies with the type of instrument being used to drive the installation. There are three possible alternatives:

a. *AMU, Mks* 1 *or* 2 *With No API Fitted.* In this installation a Switch Panel, Mk 1 (*see* Fig 2) is used. The panel carries the MAIN ON/OFF switch and the GROUND TEST button for the AMU.

b. *AMU, Mks* 1 *or* 2 *With API Fitted.* In

this installation the MAIN ON/OFF switch and the GROUND TEST button are part of the API.

c. *AMU, Mk 4 or TAU, Mk 1.* In this installation the controls are on the separate control panel of the AMU or the TAU (*see* Chaps 5 and 8). These control panels also have an ELECT TRANS switch which controls the transmission to the AMI and/or API.

Fig 2 Switch Panel, Mk 1

Operation

6. **Ground Test.** The ground test procedure is similar for all installations and the AMU/TAU controls *ie* the MAIN ON/OFF switch and the GROUND TEST button, are used. Test procedure is as follows:

a. Put MAIN switch ON and, in the case of the AMU, wait 30 seconds for the valves to warm up.

b. If the AMU, Mk 4 or TAU, Mk 1 control panel is being used, put the ELECT TRANS switch ON.

c. Press the GROUND TEST button. The AMU/TAU should commence operating and the AMI indicator should start recording air mileage.

d. Release the GROUND TEST button and reset the counters to zero.

e. Place the MAIN switch (and the ELECT TRANS switch if used) to OFF.

7. **In Flight.** Before take-off, check that the indicator counters are set to zero and put the switches on the appropriate panel to the ON position (*see* para 5). The AMI will automatically start to record when the AMU/TAU start operating during take-off. To reset the counters to zero, turn the resetting knob anti-clockwise. It is not necessary, during resetting, to switch off the AMU/TAU.

8. **Faults.** The AMI is a very simple instrument and is, therefore, very reliable. Any failure to record air distance is more likely to be due to a fault in the AMU, the TAU or the transmission rather than in the AMI itself.

Accuracy

9. The solenoid in the AMI and hence the counters, are operated directly by the pulses produced by the AMU/TAU. The AMI thus directly reflects the accuracy of the AMU/TAU being used. For this accuracy, refer to Chap 5 or 8.

CHAPTER 7

AIR DATA SYSTEMS

CONTENTS

	Paras
Introduction	1–3
Probes	4–6
Transducers	7–12
Air Data Computer	13–15
Summary	16

Introduction

1. There are various forms of air data systems used in modern aircraft but all are similar in principle and provide similar outputs depending on the particular requirement. This chapter describes a typical system rather than a specific aircraft application; for details of a particular aircraft installation, reference should be made to the relevant Aircrew Manual and/or the Air Technical Publication.

2. The air data system (ADS) is built around an air data computer (ADC) which forms an essential part of a modern flight/navigation/weapons system. The ADS measures basic air data inputs of pitot pressure, static pressure, air temperature and where necessary angle of attack, and computes outputs of flight parameters to the requirements of the aircraft's systems and displays.

3. A comprehensive ADS consists of:

a. Pitot, static and temperature probes to measure the basic air data.

b. Local incidence vane for angle of attack computation.

c. Transducers to convert the basic air data inputs into electrical or electro-mechanical signals.

d. Air data computer to process the data and provide the required outputs to the aircraft's systems and displays.

e. Power supplies to provide specific stabilised power for the ADS units.

Probes

4. **Pitot/Static.** Pitot and Static pressures are taken from the aircraft's pitot/static head or the pitot head and static vent (*see* Pt 1, Sec 2, Chap 1 and Pt 1, Sec 3, Chap 3).

5. **Temperature.** The outside air temperature probes are described in Chap 2.

6. **Angle of Attack.** Angle of attack is the angle, in the normally vertical plane of symmetry of the aircraft, at which the free-stream airflow meets an arbitrary longitudinal datum line on the aircraft. It is generally measured by a small pivoted vane whose axis of rotation is nominally horizontal and athwartships. The vane is usually mounted on the side of the fuselage near the nose or on a probe forward of the wing or nose.

Transducers

7. **General.** The transducers, which convert pressures, temperatures and angles to voltages or digital pulses, are the most vital elements of the air data system and are the limiting factor in their accuracy. All ADS measure static pressure (ps) and temperature (Tm). All outputs may be computed if a third measures pitot pressure (pt) or dynamic pressure (pt-ps). In some systems both

are measured for redundancy. The choice of other inputs varies among designs.

8. **Pressure Transducers.** A good pressure transducer must be sensitive to small pressure changes and must produce a usuable, repeatable, and accurate output with low hystersis; it must be insensitive to vibration, acceleration, corrosion, humidity, various kinds of radiation and changes in ambient temperature.

9. **Displacement Pressure Transducers.** The displacement transducer uses the expansion of an elastic diaphragm to actuate an electrical pick off that produces a direct analogue or digital output. The diaphragm may have a vacuum on one side to measure the absolute static pressure or absolute pitot pressure fed to the other side; alternatively, it may have static pressure on one side and pitot pressure on the other to measure dynamic pressure (pt-ps). Diaphragm motion may itself actuate the transducer as in the strain-gauge transducer illustrated in Fig 1 or it can serve as an input to a servo that drives the transducer, similar in principle to the Mk 22 altimeter mechanism (*see* Sect 2, Chap 3). A typical direct digital transducer is shown in Fig 2 where the output is the frequency of a vibrating wire which changes in proportion to the displacement of the diaphragm.

10. **Force-Balance Pressure Transducers.** This transducer balances the force on the diaphragm against that of a calibrated standard, such as a spring. When a pressure change occurs, a null detecting pick off detects a small motion of the diaphragm and drives a servo to restore the diaphragm to null. The nulling force measures the pressure difference across the diaphragm. This system reduces the stringent requirements of elasticity of the diaphragm but increases them on the restoring servo, whether

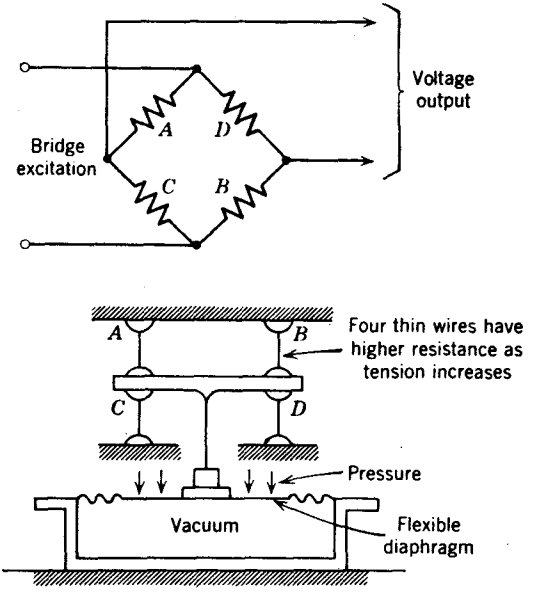


Fig 1 Strain-gauge Absolute Pressure Transducer

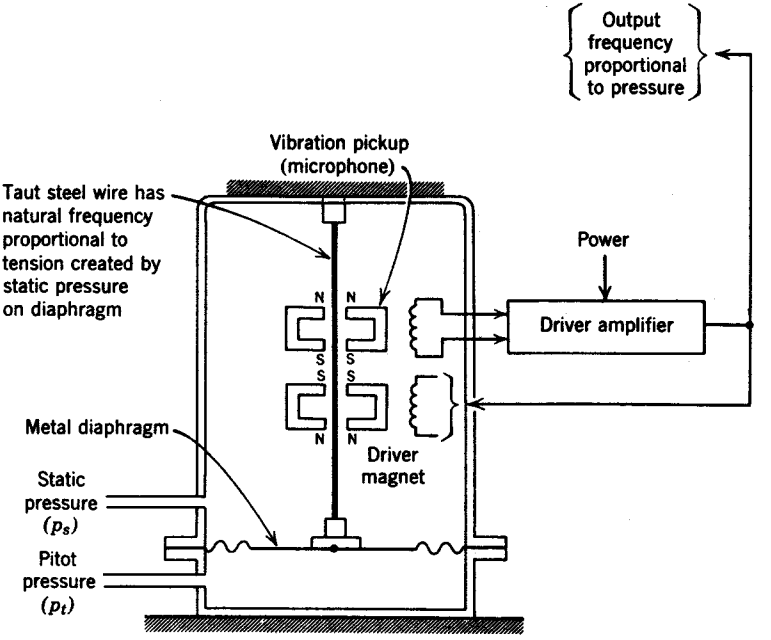
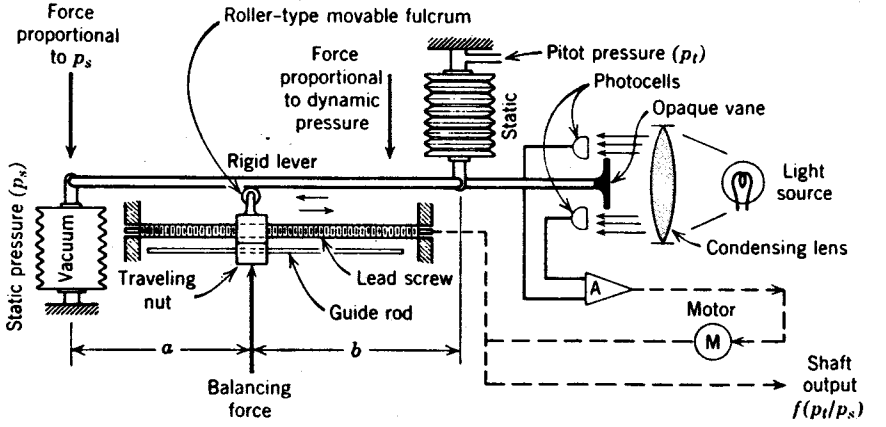


Fig 2. Vibrating-wire Differential Pressure Transducer



Servo runs to balance light on two photocells

When balanced,
$p_s\, a = (p_t - p_s)\, b$

Fig 3 Force-balance Mach Transducer

mechanical or electrical. A form of this transducer is used in the True Air Speed Unit described in Chapter 8. Simple analog computations can be performed within a transducer as illustrated in Fig 3 where a force balance mechanical transducer uses a photocell detector to read motion of a lever and a servoed pivot to null the lever motion. Mach No can be measured directly by balancing the lever with ps at one end and (pt-ps) at the other.

11. **Temperature Transducers.** These transducers are contained within the probes and are described in Sect 1, Chap 2.

12. **Anuglar Transducers.** The angle of attack vane movement is measured by the displacement method and transmitted via a potentiometer, synchro or digital encoder to the ADC.

The Air Data Computer

13. The ADC processes the inputs to calculate the corrections required to the raw input data and compute the outputs from the corrected data. The pressure and temperature correction and computation algorithms used are given in Pt 1, Sections 1, 2 and 3 of this Volume. The angle of attack correction is determined empirically as a function of the measured angle and Mach number.

14. **ADC Outputs.** An ADC in a complex military aircraft may supply 50 or more different outputs, many being different forms of the same computed parameter; for example, pressure altitude may be a potentiometer voltage, a synchro output, and a digital code. For the convenience of other sub-systems functions of the basic parameters such as square roots, logarithms, and special aerodynamic functions may be calculated. Fig 4 illustrates a possible configuration of an ADC for a high performance aircraft. The outputs may be fed direct to displays and individual sub-systems or to a main computer which controls the distribution of the information.

15. **Serviceability Tests.** ADS usually have a ground test facility whereby calibrated signals are manually switched into the system in place of the transducer outputs to give known outputs to the displays. In flight, power supplies, amplifiers and other vital channels are continuously monitored to display failure warning indicators. Digital ADCs include a built in test function in the computation programme.

Summary

16. An Air Data System avoids the bulk and complexity of piping to the large number of pressure instruments required to cover all the applications in a modern aircraft. It provides, through its Air Data Computer, the comprehensive data essential to efficient operation of modern aircraft.

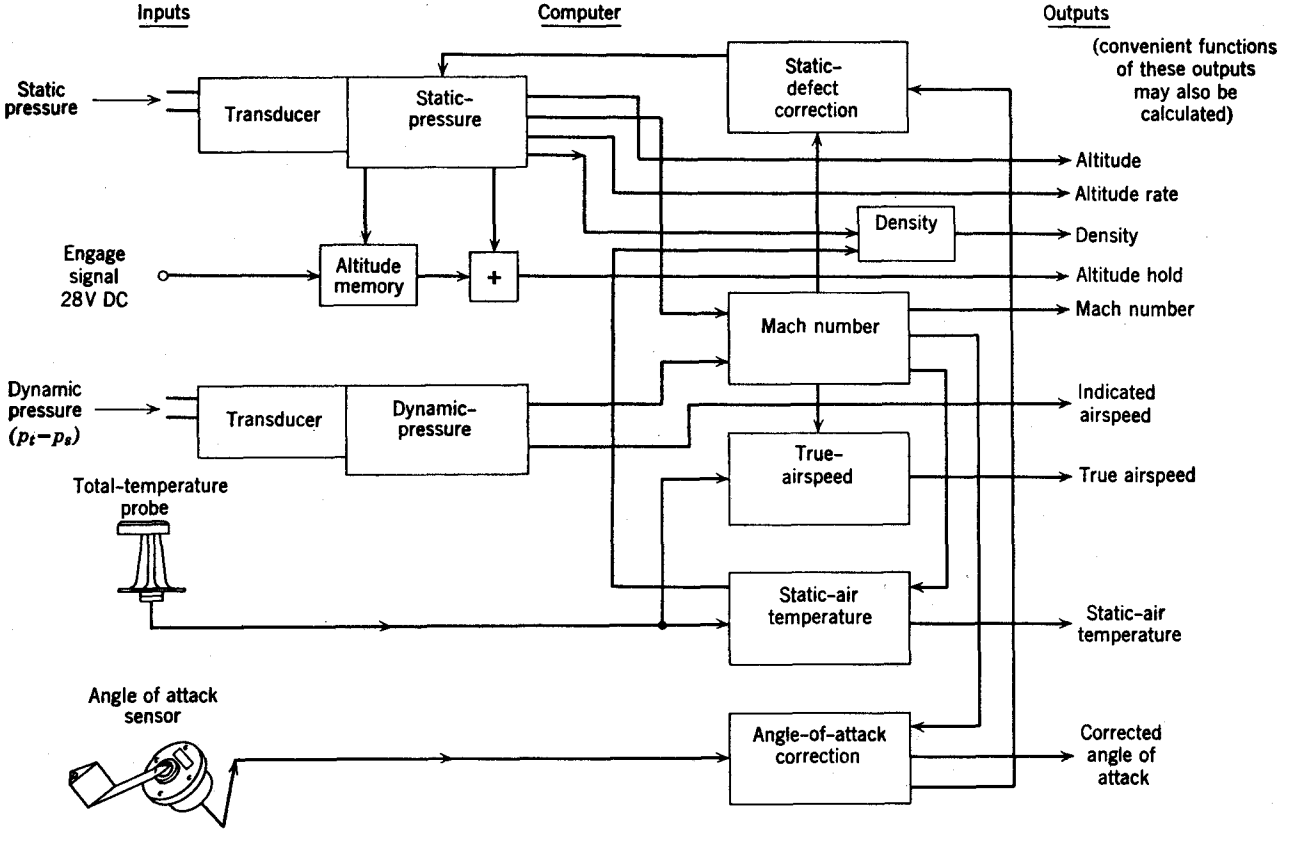


Fig 4 Air Data Computer

Printed in UK for HMSO, Dd No 8417159 7/85 NB30957 3347

CHAPTER 8

TRUE AIR SPEED UNIT, MARK 1

CONTENTS

	Paras
Introduction	1–4
Sequence of Computation	5–6
Pressure Transducers	7–11
Computation of log(P–S) and log(S)	12–15
Computation of log{(P–S)/S}	16
Conversion of log{(P–S)/S} to log f(M)	17
Pressure Error Correction	18
Computation of Mach Number	19
Computation of True Air Speed	20–22
Computation of Air Miles Flown	23
Description	24–27
Operation—Ground Test	28–30
Operation in Flight	31–33
Accuracy	34

Introduction

1. The True Air Speed Unit, Mk 1 (TAU) is an electro-mechanical device which provides accurate output signals proportional to the true air speed of an aircraft and the air distance flown. A pressure error corrected Mach number output is also available.

2. Optimum accuracy is obtained by compensating the mechanism for pitot-static probe pressure errors and by precise measurement of the air-stream temperature. Outputs are provided as follows:

a. A *DC* voltage, proportional to true air speed over the range 150–600 knots, is fed to the navigation and bombing system through a variable air speed unit.

b. A 28 v DC pulse voltage at 0·1 nm intervals operates the air mileage indicator.

c. Mach number output, over the range 0·3M to 1·0M, is by a synchro control transmitter.

3. The unit operates in conjunction with a true air speed unit control panel and must be connected to a nickel resistance thermometer probe through a special lead. The TAU, Mk 1 may be generally considered as a replacement for the AMU, Mk 4 series.

Theory of Computation

4. True air speed may be computed according to the formula:

$$TAS = aM \qquad \ldots.(1)$$

where a = Speed of sound in air
M = Mach number

The speed of sound in air cannot be measured directly but is proportional to the square root of the absolute air temperature:

$$a = C\sqrt{T} \qquad \ldots.(2)$$

where T = Absolute free air temperature
C = A constant

Combining equations (1) and (2):

$$TAS = CM\sqrt{T} \qquad \ldots.(3)$$

T cannot be measured directly, but can be found from the adiabatic temperature formula:

$$T = T_i/(1+0{\cdot}2KM^2) \qquad \ldots.(4)$$

where Ti = Absolute indicated stagnation temperature
K = A constant (the recovery factor of the bulb)

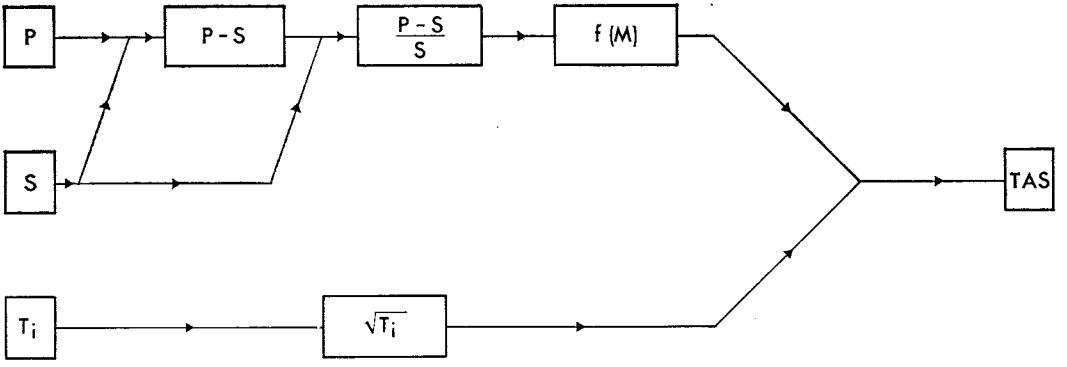


Fig 1 Basic Sequence of Computation

Combining equations (3) and (4):

$$TAS = C\sqrt{\frac{M^2 T_i}{1+0{\cdot}2M^2K}} \quad \ldots.(5)$$

This can be expressed as:

$$TAS = C \cdot f(M) \cdot \sqrt{T_i} \quad \ldots.(6)$$

Where

$$f(M) = \sqrt{\frac{M^2}{1+0{\cdot}2M^2K}} \quad \ldots.(7)$$

Equation (6) is used as the basis of computation.

Sequence of Computation

5. The physical quantities measured by the TAU, Mk 1 are pitot pressure (P), static pressure (S) and stagnation air temperature (T_i). Mach number can be expressed as a function of the ratio (P—S)/S, (*see* Annex to Chap 4), so that:

$$f(M) \text{ becomes } f\left\{f\left(\frac{P-S}{S}\right)\right\}$$

Since for any one value of (P—S)/S there is only one value of f(M), a combination of P and S is used to produce this term. T_i provides the term $\sqrt{T_i}$ and the basic combination of the terms to give true air speed is shown in Fig 1.

6. The process of computation is simplified by the use of logarithmic values so that addition and subtraction may be used instead of multiplication and division. The sequence of this simplified solution is shown in Fig 2.

Pressure Transducers

7. Uncorrected static and pitot—static pressures are measured by two transducers, which operate on the force balance principle, providing rotary

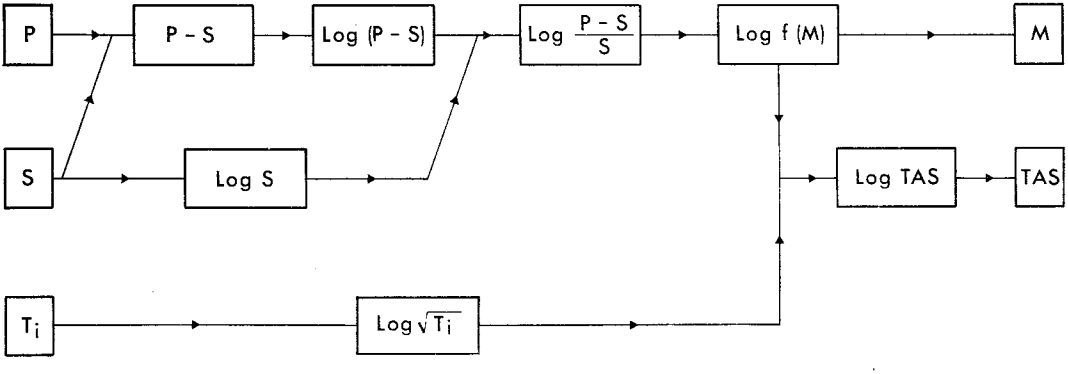


Fig 2 Sequence of Simplified Solution

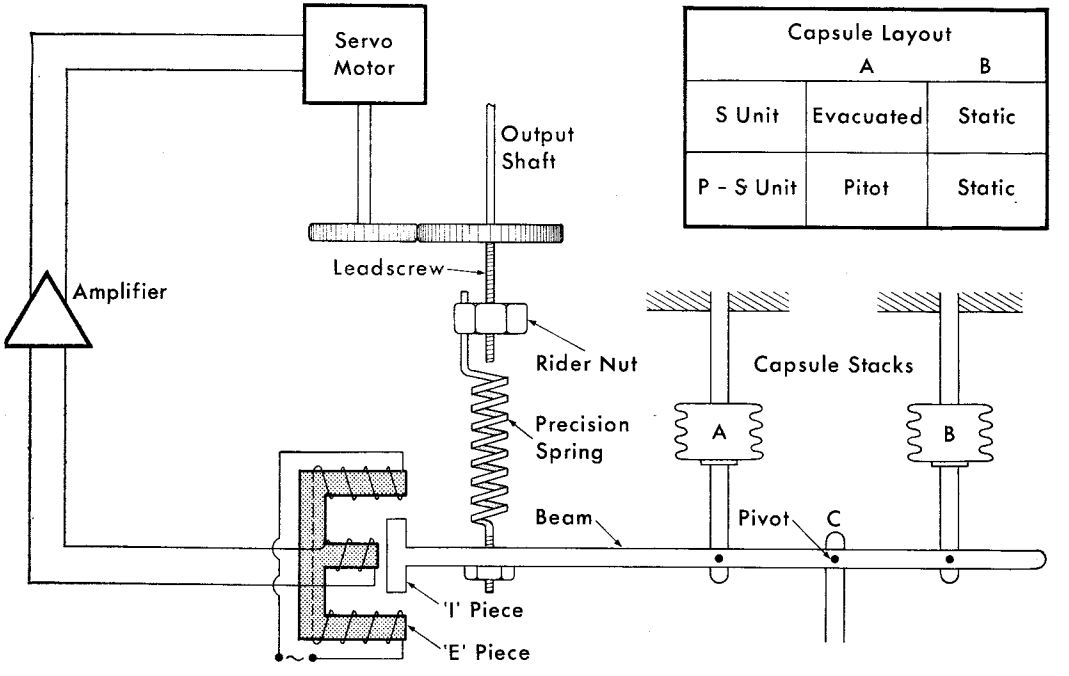


Fig 3 Force Balance Pressure Transducer—Schematic

outputs proportional to the applied pressures. Each transducer employs two capsule stacks as shown in Fig 3. In the (P—S) transducer, pressure is applied to the interior of each capsule (pitot to one and static to the other), whilst in the (S) transducer one capsule is evacuated and sealed, static pressure being applied to the interior of the other.

8. **Description.** The capsule stacks (A and B) are connected to a beam on either side of the pivot. A pressure applied to the capsules causes a deflection of the beam. Fixed to the end of the beam is an I piece forming part of an E and I pick-off, which senses the deflection. The E piece which is firmly fixed to the main casting, has two identical windings on the outer limbs and one on the centre limb. The coils on the outer limbs are energised by an AC supply and the coil on the centre limb comprises the secondary.

9. **Operation.** When the I piece moves relative to the E piece, a change of reluctance occurs inducing a voltage in the coil on the centre limb. The phase and magnitude of this voltage is dependent upon the direction and amount of movement of the beam. The voltage is amplified and used to energize the control winding of a servo motor which is geared to drive a lead screw and output shaft. The lead screw carries a nut which is coupled to the beam by a precision spring. When the motor shaft rotates, the force exerted upon the beam by the spring is changed until the beam, capsules and transformer pick-off are restored to a position of equilibrium. The angle of rotation of the output shaft is a precise measurement of the capsule force and therefore the applied pressure.

10. **Limit Switches.** To prevent damage to, or jamming of, the lead screw nut movement, two pairs of contacts are fitted, one at each end of the nut movement. As the nut nears the end of its travel, it will close one or other of the contacts causing reversal of the servo motor. The lead screw reverses and the nut will move away from, and release, the contacts. In this way the mechanism will hunt at the extremities of its range without damage or jamming. Mechanical stops prevent derangement of the transducer by rapid changes of pressure.

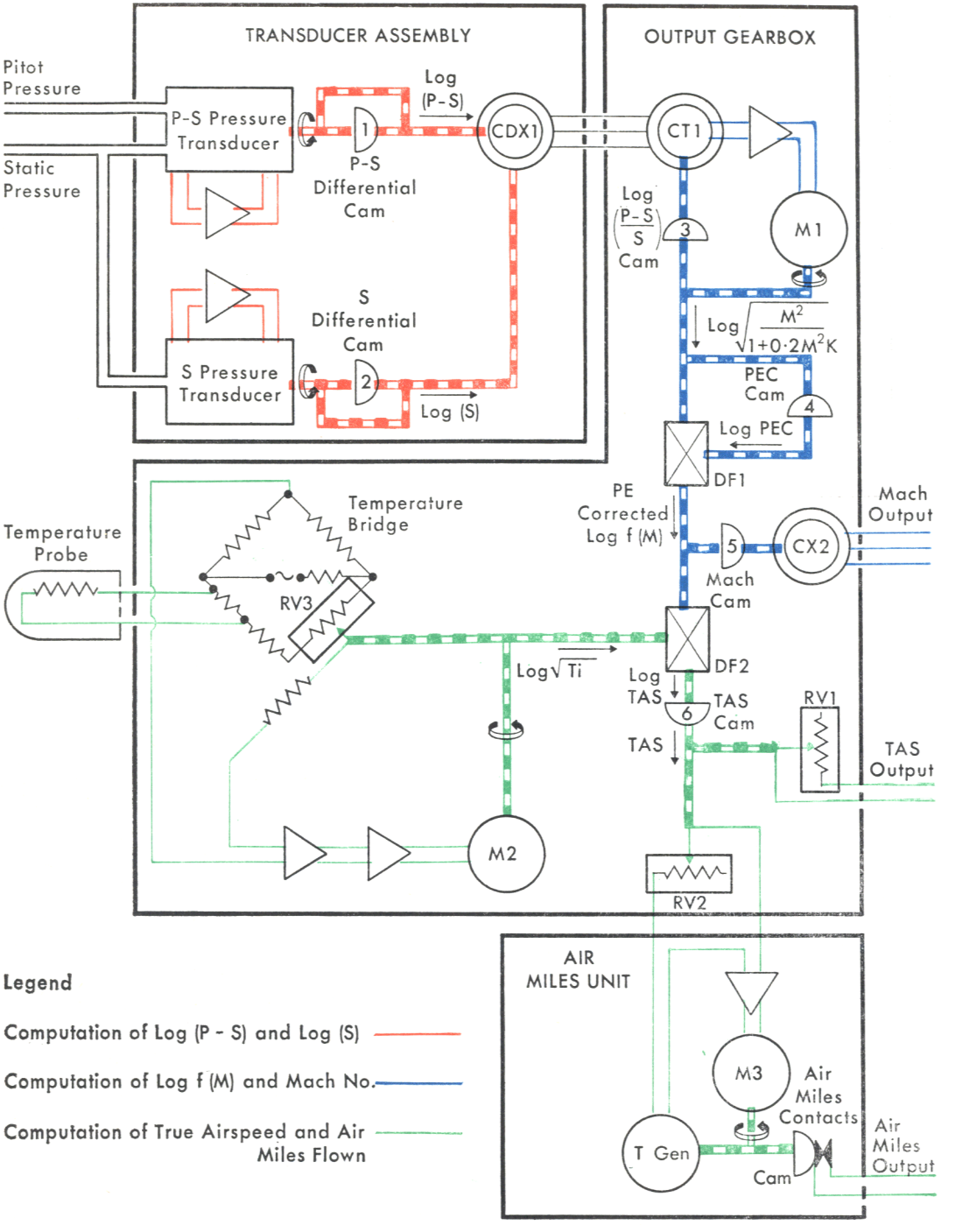


Fig 4 True Air Speed Unit—Schematic

11. **Temperature Compensation.** The accuracy of the transducer is very much dependent upon the accuracy of the precision spring. The transducer is temperature compensated by a bimetal strip attached to the main frame and coupled to the beam with a spring. Changes in ambient temperature deflect the bimetal strip which applies a corrective force to the beam, via the spring, to modify the output.

Computation of log(P—S) and log(S)

12. The mechanical output from the transducers, *ie* output shaft angles proportional to the applied (P—S) and (S) pressures, are converted to log (P—S) and log(S) by two differential cams (1 and 2) contained in the log gearbox which is part of the transducer assembly. A block schematic diagram of the TAU and the flow of information through the unit can be followed by reference to Fig 4.

13. The principle of a simple cam and its follower is shown in Fig 5. The input shaft rotation turns the cam through the angle proportional to the input value. Spring loaded against the cam is the cam follower fixed to the output shaft. As the cam turns, the follower is moved through an angle depending upon the cam shape, and the output shaft rotates through the same angle; the design of cam and follower being such that for, say, an input of x, the output is log x.

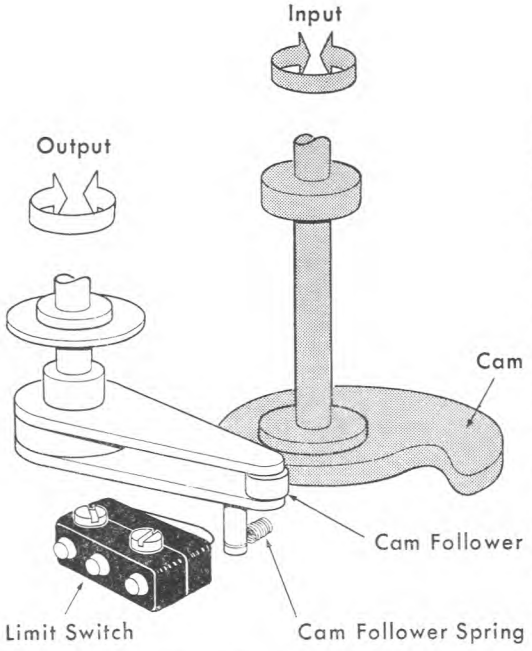


Fig 5 Simple Cam and Follower

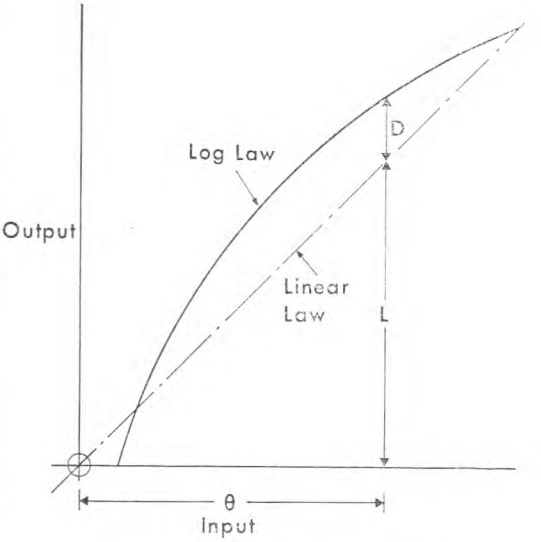


Fig 6 Differential Cam—Input/Output Relationship

14. The differential cams (1 and 2) are required to give an output in log form as shown by the curved line in Fig 6. To produce this output, a lift L is required for an angle of rotation and this would normally necessitate the use of a cam with a very steep slope. Since the linear output can be obtained by a simple epicyclic gear, the profile of the cam is cut to give the difference between the linear law and the log law. This arrangement limits the maximum lift of the cam to dimension D, making the cam less steep and providing greater accuracy for a cam of given diameter.

15. **The Differential Cam.** Each differential cam (*see* Fig 7) is fixed to a shaft X which is driven through a gear train from the (P—S) or (S) transducer. In Fig 7 the components driven directly by the shaft are unshaded. The planetary wheels P run in a bearing fixed in the cam and, when the cam rotates, they revolve around the shaft on a fixed radius. Wheels A and B are free to rotate on shaft X, wheel A being driven by the sector of the cam follower. If a circular cam was employed,

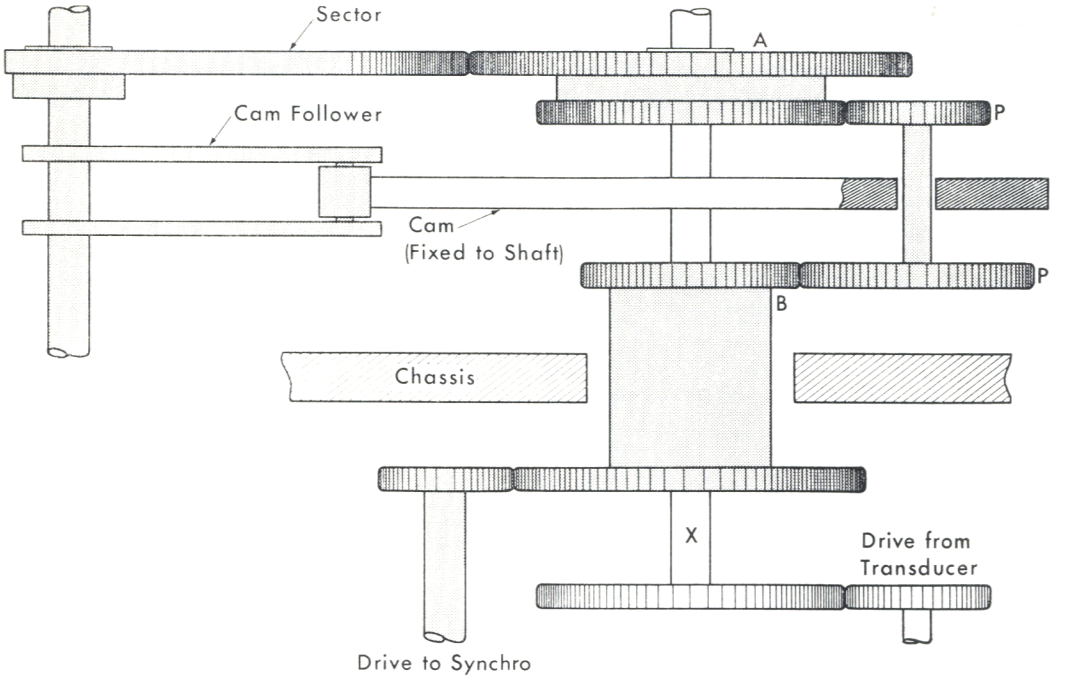


Fig 7 Differential Cam—Schematic

allowing rotation with no resulting follower movement, wheel A would not move. The planetary wheel P would roll around wheel A and, because one planetary wheel P is smaller than the other, wheel B would be driven. This output is fed to a differential synchro. If the cam is held stationary and the follower arm is moved, the sector turns wheel A. The planetary wheels P are driven by wheel A which, in turn, causes wheel B and the synchro drive gear to rotate. Any movement of the cam shaft is therefore modified by the cam follower, sector, wheel A and the planetary gears, according to the shape of the cam. The resulting movement is transmitted to a differential synchro. A diagram of one of the differential cams is shown at Fig 8.

Computation of log {(P—S)/S}

16. The cam outputs are geared to a synchro differential control transmitter (CDX 1), the log(S) output driving the stator and the log(P—S) output driving the rotor. The transmission from CDX 1 therefore corresponds to an angle proportional to $\{\log(P-S) - \log S\}$ or $\log\{(P-S)/S\}$.

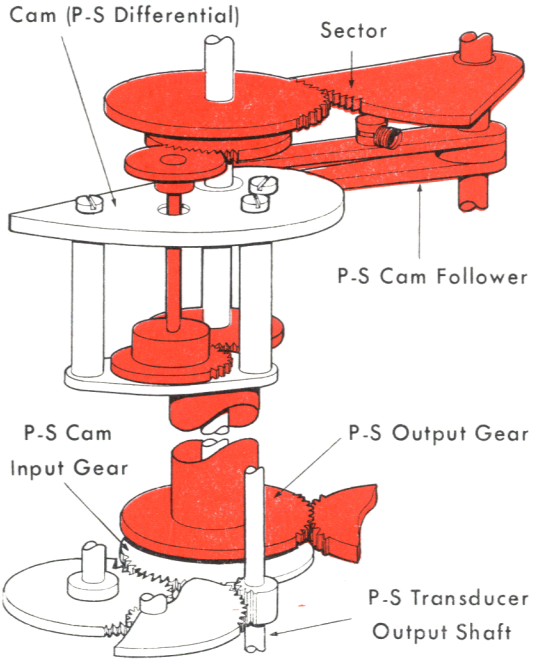


Fig 8 P—S Differential Cam

Conversion of log {(P—S)/S} to log f(M)

17. Signals from the differential synchro (CDX 1) are received by a control transformer (CT 1). The servo motor (M 1) of the associated servo system drives cam 3, which has its cam follower geared to the rotor shaft of the control transformer (CT 1). As the control transformer is driven to the null position, cam 3 modifies the servo motor output to:

$$\log \sqrt{\frac{M^2}{1+0{\cdot}2M^2K}} \text{ or } \log f(M)$$

Pressure Error Correction

18. The mechanical output from the servo motor, proportional to log f(M), is fed directly to one side of a mechanical differential (DF 1). The differential also receives another input from the same source, which is modified by cam 4. This is the pressure error correction (PEC) cam which applies a corrective mechanical signal to offset pressure errors in the pitot and static pressures fed to the TAU. The amount of correction applied is determined by the cam which is shaped to suit conditions applicable to a particular aircraft type. A circular cam, which makes no pressure error correction, is used for test purposes. This is supplied with the TAU by the manufacturer. Alternative cams and the aircraft to which they apply are shown in Table 1.

Type Letter	Identification Colour	Aircraft Type
A	Red	Vulcan B Mk 2 (all variants)
F	Metal	Victor B Mk 2
H	Blue	VC 10

Note: *Cam Type F (metal) is also used for test purposes*

Table 1 PEC Cam Details

Computation of Mach Number

19. By using suitable gearing, the output from the differential (DF 1) is divided into two identical rotations *ie* proportional to the PE corrected log f(M). One of these rotations is converted to Mach number by the Mach cam (cam 5) and its attendant follower which is geared to drive the rotor of the Mach output synchro transmitter (CX 2) which, in turn, provides the Mach output. The other rotation is fed directly to a second mechanical differential (DF 2).

Computation of True Air Speed

20. The second differential (DF 2) is used to introduce the temperature function which is proportional to log $\sqrt{T_i}$. The basic information to provide this function is derived from the nickel resistance thermometer probe which measures the airstream temperature.

21. **Computation of log $\sqrt{T_i}$.** The temperature probe is connected to one arm of a Wheatstone bridge (temperature bridge), the remaining arms of the bridge being formed by a motor driven potentiometer and two resistors of fixed value. A change in temperature causes a change in the resistance of the probe and therefore an out-of-balance condition in the bridge. The out-of-balance signal across the bridge is amplified and fed to the control winding of the temperature bridge servo motor (M 2). The servo motor output shaft is geared to drive the potentiometer in the direction required to restore a balanced condition in the bridge. By careful calibration of the temperature bridge, the motor can be made to rotate through an angle proportional or equal to $\log\sqrt{T_i}$. This is the temperature function introduced into the differential DF 2.

22. **True Air Speed Output.** The two mechanical inputs fed to the differential (DF 2) produce an output proportional to log TAS. The TAS cam (cam 6) and its attendant cam follower is included to reduce this output to true air speed. The cam follower, rotating as true air speed, is geared to drive a volts/knots potentiometer (RV 1) With the application of a suitable external power supply, this potentiometer gives a volts per knot output equivalent to true air speed.

Computation of Air Miles Flown

23. The true air speed output from cam 6 is also used to compute air miles flown. The voltage from the air miles potentiometer (RV 2), which varies proportionately with true air speed, is opposed by the voltage output from a tacho-generator. Any resultant difference voltage is amplified and fed to the control phase of the air

miles motor (M 3) which drives the generator. The motor therefore rotates at a speed proportional to true air speed. The motor shaft also drives a cam which operates the air miles contacts. When connected to an external power supply, the contacts provide a pulse voltage to operate the air mileage indicator. The pulses occur at 0·1 nm intervals.

Description

24. The true air speed unit system comprises two main units; the true air speed unit itself and the control panel.

25. **True Air Speed Unit.** The mechanism is housed in a drum-shaped container (*see* Fig 9) fitted with a mounting flange and a services panel incorporating the plug and socket connectors and the pitot and static pressure inlets. The unit consists essentially of three main sub-units:

a. *The Transducer Assembly.* This contains the (P—S) and (S) pressure transducers, the differential cams 1 and 2 and associated gearing, and the differential control transmitter.

b. *The Output Gearbox.* This contains the temperature bridge, the PEC cam, the Mach cam and TAS potentiometers.

c. *The Air Miles Unit.* This contains the motor, tacho-generator and contacts associated with the air miles flown output.

26. **The Control Panel.** There are two alternative control panels, both provide identical facilities and are similar in construction and appearance. The later model is provided with a transilluminated lighting system consisting of an engraved Perspex front panel and two miniature lamp holders positioned above and below the GROUND TEST button. Fig 10 shows the earlier

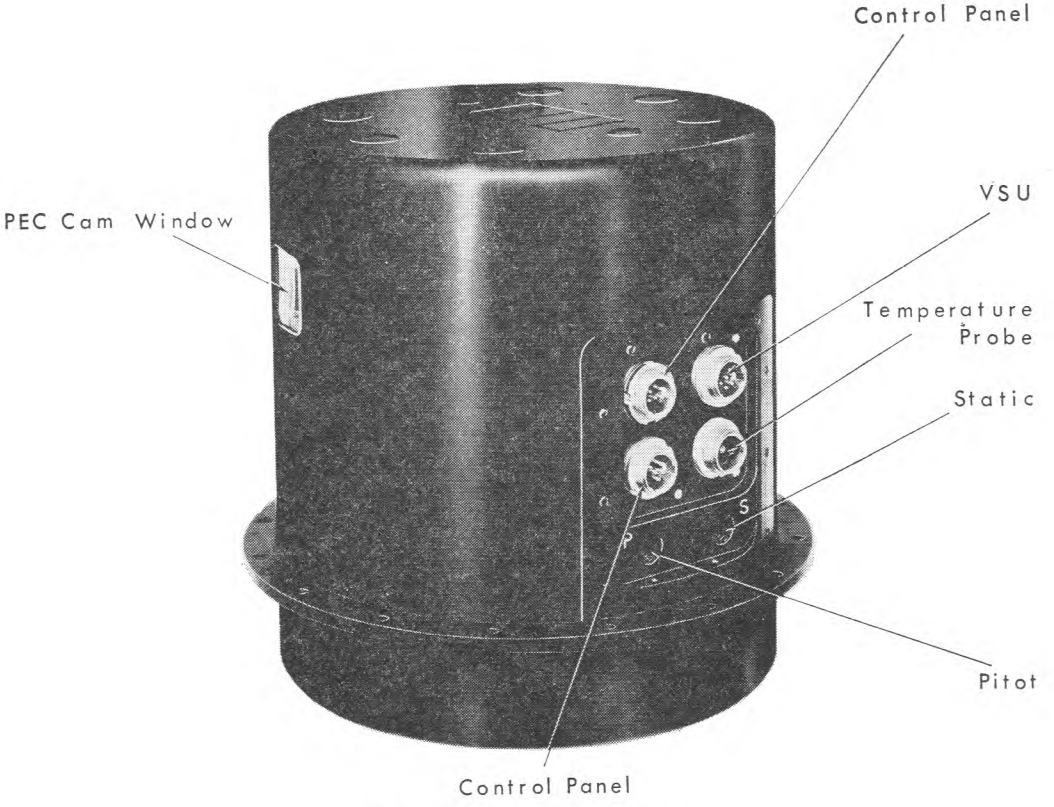


Fig 9 The True Air Speed Unit

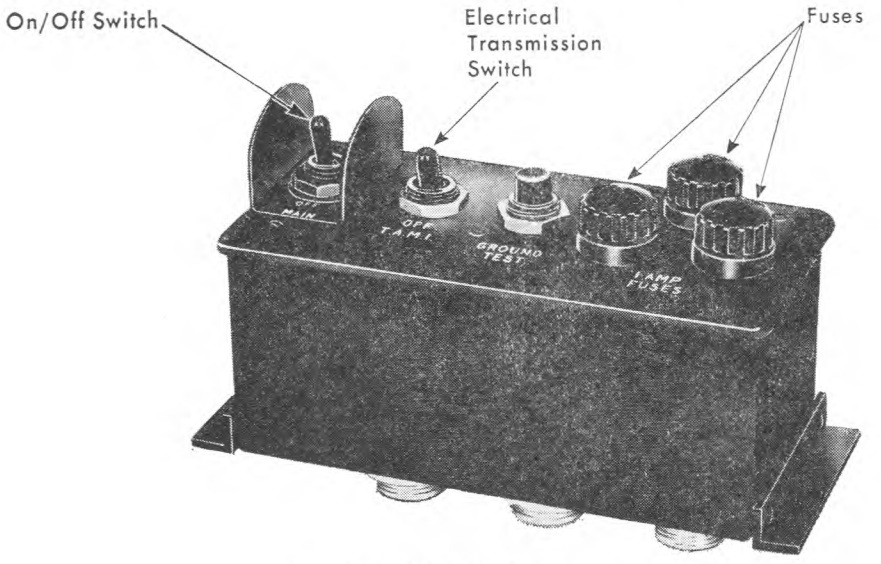


Fig 10 TAU Control Panel

model; it has a MAIN switch with guard plates, an ON/OFF switch controlling the AMI, a GROUND TEST button and three one amp fuses, two for the 115 v AC supply to the TAU and one for the 28 v DC supply to the AMI. It is also used as a junction box and feeds the main supplies to the TAU and the TAU outputs to the associated equipment.

27. **TAU Installation.** The outputs of Mach number, TAS and air miles flown are used as inputs to the aircraft avionics system. *See* Fig 11.

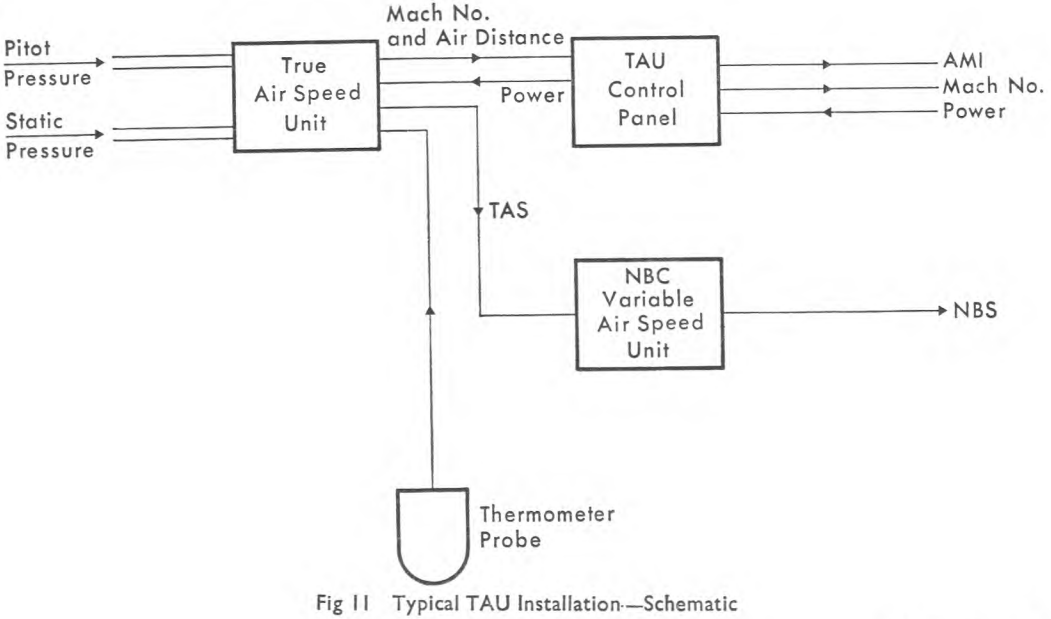


Fig 11 Typical TAU Installation—Schematic

Operation—Ground Test

28. A ground test circuit is incorporated in the TAU, Mk 1 to enable the functioning of the complete unit, with the exception of the pneumatics, to be checked in the aircraft without recourse to additional equipment. This is achieved by arranging for the unit to provide a predetermined output. When the GROUND TEST button is pressed, with the MAIN switch ON, a pre-arranged voltage is passed to the transducer amplifiers, overriding any normal signal, and the servo motors rotate to a set value. At this value, two cams, one on the (P—S) shaft and one on the (S) shaft, open contacts and break the test circuit. The servo motors stop and the predetermined values of (P—S) and (S) are fed through the unit. These values give a TAS output of 400 kt if the probe temperature is 0°C. If the probe temperature varies from 0°C, the TAS output will also vary. Table 2 gives the approximate speeds (within ± 8 kt) that should be obtained when the circular Type F test PEC cam is installed in the unit.

Probe Temperature °C	TAS Output–knots (± 8 kt)
—30	377
—20	385
—10	392
0	400
+10	407
+20	414
+30	421
+40	428
+50	435

Table 2 Ground Test—TAS Output Relative to Temperature (with circular PEC cam fitted)

29. When the test button is released, the normal signals from each E and I pick-off are fed to the amplifiers. Normal operation of the transducers will then return the displaced beams to their null position and the outputs will return to zero. The MAIN switch may then be put OFF.

30. If the GROUND TEST button should be accidentally pressed during flight, false signals will be fed to the transducer amplifiers and the outputs will not be accurate. No damage, however, should be caused as the transducers are protected by reversal switches, which operate in this case, and by the normal transducer limit switches.

Operation in Flight

31. The only actions necessary before take off are to place the MAIN switch and the AMI switch to the ON position. When air speed reaches 100 kt on the take off run, the TAU starts operating.

32. **Low Air Speed Cut-Out.** A double switch, operated by a cam on the (P—S) shaft, is set to operate at a (P—S) angle equivalent to 100 kt. This ensures that there are no instrument outputs below this air speed.

33. **Power Failure Switch.** The volts per knot potentiometer (RV 1) is fed from an external source so that, in the event of power failure to the TAU, it is still possible to have an air speed output. This output, however, could be incorrect and, to prevent any misleading indication, a power failure switch is fitted. This switch is operated by a servo motor energized by a supply from the TAU. When the supply is connected, the motor shaft rotates sufficiently to turn an eccentric cam and close the switch. In the event of power failure, a leaf spring returns the motor shaft and cam to their original positions and the switch is opened disconnecting the potentiometer from the output plug and reducing the TAS output voltage to zero.

Accuracy

34. The accuracies of the three outputs from the True Air Speed Unit, Mk 1 are given in Table 3.

Output	Range	Form of Output	Accuracy		
			Ambient Temperature	Altitude	
				0–40,000 ft	50,000 ft and above
TAS	150–600 kt	Potentiometer	20°C	± 6 kt	± 9 kt
			—40°C and +55°C	± 14kt	± 25 kt
Mach No	0·3–1·0M	Synchro Control Transmitter	20°C	±0·01M	±0·015M
			—40°C and +55°C	± 0·023M	± 0·04M
Air Miles	—	DC pulse at 0.1 nm intervals	± 2 knots additional to TAS actual error		

Table 3 Range and Accuracy of Outputs

CHAPTER 9

COMBINED SPEED INDICATORS

CONTENTS

	Paras
Introduction	1
Principle	2
Description	3–7
Presentation	8
Digital Mach/Air Speed Indicators	9–10
Range and Accuracy	11

Introduction

1. With the increased complexity of aircraft instrument panels in modern aircraft and the continual search for more room in an already restricted space, it is becoming the practice to combine two or more functions into one instrument. One area where this has been successfully carried out is with speed indicating instruments. A combined instrument showing both indicated air speed and Mach number is now fitted in some aircraft. The instrument can take one of three forms; a synchro driven tape strip display as described in Chap 7, a more simple capsule operated dial presentation or a capsule operated dial, synchro operated digital presentation.

Principle

2. The construction of the dial-type combined speed indicator is very similar to the machmeter and the same principles are employed. Reference should be made to Chap 4 where these principles are fully explained.

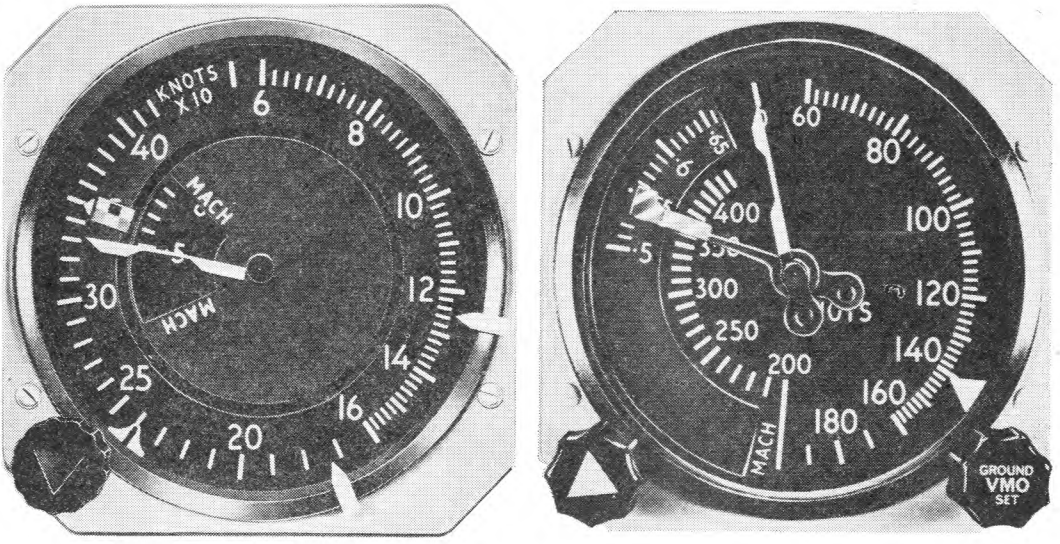
Description

3. The combined speed indicator (CSI) contains an air speed capsule and an altitude capsule. The air speed capsule directly drives, through a normal type linkage, a pointer which is read against a dial calibrated in IAS. The altitude capsule, expanding or contracting, reacts to static pressure and thus altitude. This movement, through a second linkage, modifies a parallel drive from the air speed capsule in a similar manner to the machmeter. This second drive is used to position against the air speed pointer, a rotateable disc graduated in Mach number. The Mach number disc rotates anti-clockwise as altitude increases whilst the pointer rotates clockwise with increasing IAS. Thus the pointer displays against the Mach number disc the correct Mach number for the particular air speed/altitude combination as well as the IAS against the fixed graduations on the dial.

4. Other functions are sometimes included in the CSI. These include:

a. A limit speed pointer.
b. Limit speed warning.
c. Outputs to control an auto-throttle system.

5. **Limit Speed Pointer.** Most aircraft performance data list a speed, expressed in Mach number and sometimes the equivalent IAS, which should not be exceeded under normal operating conditions or a speed which should not be exceeded under any conditions. Sometimes there is a somewhat lower speed, usually expressed in knots of IAS, which must not be exceeded at low level. For example, an aircraft may have a limiting Mach number of 0·9M equivalent at sea level (and ISA conditions) to 594 knots, at 10,000 ft to 509 knots, at 20,000 ft to 425 knots, at 30,000 ft to 347 knots, *etc.* However, at low level it may be restricted to 490 knots. It is possible, by means of a special linkage designed to suit the particular aircraft and connected to the altitude capsule, to display this information on the CSI. This is usually achieved by means of a distinctively coloured pointer; red or chequered. This limit



a Mach Aperture Inside IAS Scale

b. Mach Aperture Outside IAS Scale

Fig 1 Typical CSI Dial Presentations

(Reproduced by courtesy of Smiths Industries Ltd., Aviation Division)

speed pointer is set on the ground to the particular relevant limit speed, in this case 490 knots. As the aircraft climbs, an overriding stop maintains the pointer at this reading until a condition exists where 490 knots is equivalent to 0·9M. From then on the pointer moves anti-clockwise showing the IAS equivalent of 0·9M. During descent, the pointer will move clockwise until 490 knots is reached when the overriding stop again takes effect and the pointer remains at the maximum figure. At any time the pilot can assess his air speed in relation to his maximum permitted speed by the angle between the IAS pointer and the limit speed pointer.

6. **Limit Speed Warning.** In some CSIs a limit speed switch is incorporated which is closed when the IAS pointer reaches or exceeds the speed shown by the limit speed pointer. This switch operates either an audio or visual warning or both, to warn the pilot that he has exceeded his limit speed.

7. **Auto-throttle Control.** On aircraft where an auto-throttle system is installed, control of this facility may be achieved by a synchro system installed in the CSI. A moveable command pointer, manually set by a knob on the front of the instrument, positions the rotor of a synchro. The rotor of a second synchro is positioned by a low friction drive from the IAS pointer. When the IAS pointer reads the same as the command pointer, there is zero output from the pair of synchros. Any difference between the two pointers produces an error signal which is fed to the auto-throttle system adjusting the throttles so that the aircraft returns to the original selected speed.

Presentation

8. A single pointer is read against a fixed IAS dial calibrated in knots and a rotateable disc (the Mach disc) calibrated in Mach number. The Mach disc is set behind and viewed through

an aperture positioned either inside or outside the air speed scale. A second pointer, distinctively painted with diagonal lines or chequers, may be incorporated to show the limit speed at all altitudes. On some models, two manually positioned bezel mounted lubber marks are available to indicate any desired air speed for reference purposes. A single command lubber positioned manually by a knob on the front of the instrument, allowing the auto-throttle reference speed to be set, may also be incorporated. Typical presentations are shown in Fig 1a and b.

Digital Mach/Air Speed Indicators

9. A variation of the CSI is a model which shows IAS by a pointer indication and Mach number by a digital display. In this case the instrument contains two capsules (air speed and altitude) as explained above but these are used only to drive the air speed pointer and a limit speed pointer, if fitted. A synchro drive proportional to Mach number is received from the aircraft's air data computer and a servo loop drives a three counter digital display. Limit speed warning and auto-throttle control can be incorporated as described in paras 5 and 6.

10. **Presentation.** An air speed pointer is read against a fixed scale and a second pointer, distinctively marked, may be incorporated to show limit speed at all altitudes. A servo driven three drum counter provides a digital read out of Mach number to two or three places of decimals. A failure flag covers the counters in the event of of power failure or loss of the Mach number synchro signal from the air data computer. Moveable index lubber marks may be incorporated in the same manner as for the dial

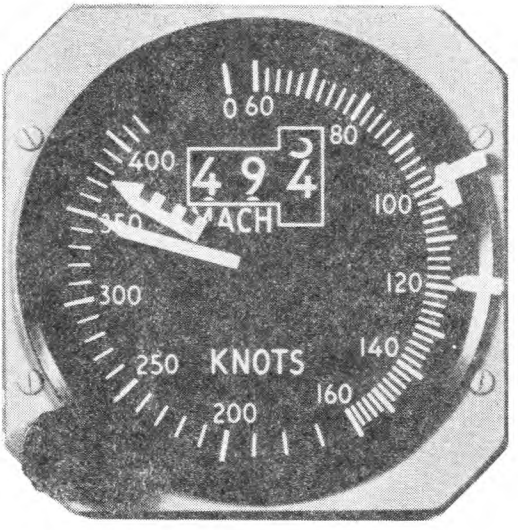


Fig 2 Typical Digital Mach/Air Speed Indicator

(Reproduced by courtesy of Smiths Industries Ltd., Aviation Division)

presentation CSI and control of an auto-throttle reference lubber mark by a knob on the front of the instrument may also be included. A typical digital Mach/air speed indicator is shown in Fig 2.

Range and Accuracy

11. The operating range of the CSI varies with the particular model but, typically, air speeds up to 800 knots and Mach number up to 2·5 can be covered. Typical instrument accuracies are $\pm$ 3 knots and $\pm$ 0·010M.

PART 2

HEADING, ATTITUDE AND ALIGNMENT INSTRUMENTS

Section

1 Direct Indicating Compasses

2 Gyroscopic References

3 Remote Indicating Compasses

4 Flight Instrument Display Systems

5 Alignment Instruments

6 Compass Correction and Calibration

PART 2
SECTION 1
DIRECT INDICATING COMPASSES

Chapter

1 **Principles of Direct Indicating Compass Systems**

2 **E2 Series Compasses**

3 **Pilot-Type Compasses**

CHAPTER 1

PRINCIPLES OF DIRECT INDICATING COMPASS SYSTEMS

CONTENTS

	Paras
Introduction	1
Properties	2
Horizontality	3–4
Sensitivity	5–6
Aperiodicity	7
ERRORS	
General	8
Turning and Acceleration Errors—Cause	9
Turning and Acceleration Errors—Effect	10–12
Summary	13
Minor Errors	14

Introduction

1. A direct indicating compass system (DICS) consists of a freely suspended magnet system which can align itself with the horizontal component of the Earth's magnetic field thus defining the direction of Magnetic North. By aligning a compass card with the north-seeking (red) end of the magnet system as shown in Fig 1, the aircraft's magnetic heading can be read off against a lubber line.

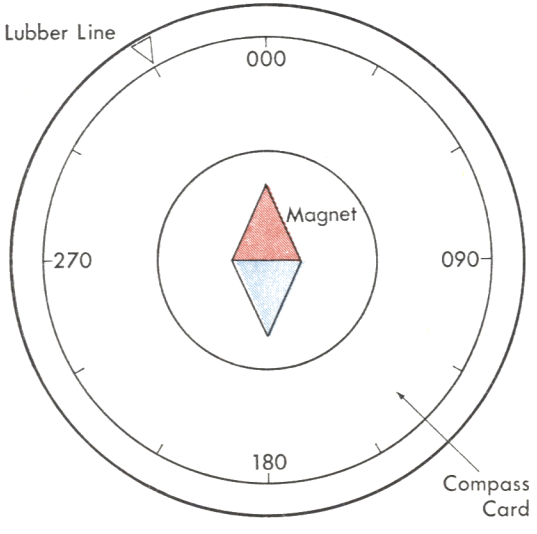


Fig 1 Basic DICS

Properties

2. A DICS must exhibit the following properties:

a. *Horizontality*. The magnet system must remain as near horizontal as possible.

b. *Sensitivity*. The magnet must be sensitive.

c. *Aperiodicity*. The magnet's behaviour must be aperiodic.

Horizontality

3. Freely suspended in the Earth's magnetic field, a magnet system will align itself with the direction of that field. At the magnetic equator the field direction is parallel to the Earth's surface; at all other places the magnet system is tilted in the direction of the total field (T), where T is the resultant of the horizontal and vertical fields.

4. If the magnet system were allowed to align itself with the T field it would be difficult to align the compass card accurately; moreover, the

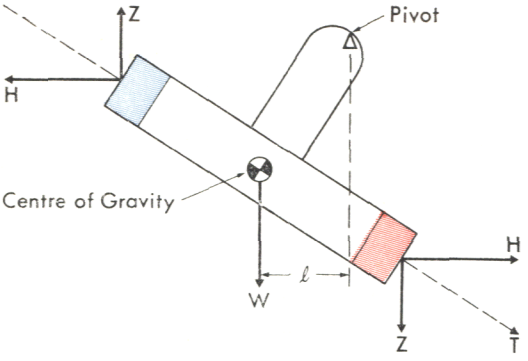


Fig 2 Pendulous Suspension

tendency to tilt would reduce the magnetic moment in the horizontal plane in which direction is measured. A pendulous suspension system is therefore used to overcome the magnet system's tendency to tilt. When the pendulously suspended magnet system tilts to align with T, the magnet system's centre of gravity is displaced from the vertical through the pivot (Fig 2). The magnet system's weight forms the couple W*l*, which acts to restore the magnet system to the horizontal. In UK latitudes the residual tilt in a well designed compass is approximately 2°.

Sensitivity

5. A DICS must be sensitive and able to indicate the local magnetic meridian quickly and accurately. Sensitivity may be increased by the following methods:

a. Increasing the magnetic moment of the magnet system.

b. Reducing the moment of inertia of the magnet system.

c. Reducing the friction at the suspension point.

6. A compromise is reached between the magnetic moment and the moment of inertia requirements by using a number of small, light, powerful magnets as the magnetic sensing element of the compass. Friction at the pivot is reduced by using jewelled bearings and also by suspending the magnet system in a fluid which reduces the weight acting on the pivot and lubricates the bearing.

Aperiodicity

7. The vibrations and oscillations experienced in flight by a suspended magnet system tend to cause undesirable periodic oscillations. Aperiodicity is achieved using a magnet system with a low moment of inertia and high magnetic moment, the same compromise applied for sensitivity. Damping filaments, which create drag forces and reduce the magnet system's tendency to oscillate, are also used.

ERRORS

General

8. In addition to the errors caused by external magnetic fields (described in Sect 6, Chap 1), DICS are subject to the errors covered in the following paragraphs.

Tuning and Acceleration Errors—Cause

9. If an aircraft fitted with a DICS is subjected to horizontal accelerations, the accelerating forces may cause errors in the indicated heading. The accelerations may be the result of speed changes or from the central acceleration experienced in a turn; both have similar effects on the compass system, the resultant errors being greatest when the accelerating force acts at right angles to the magnetic meridian with which the compass is aligned, *ie* when the aircraft changes speed on easterly or westerly headings, or turns through North or South. The errors are caused by the displacement of the magnet system's centre of gravity from the line through the pivot. This displacement results in the formation of couples which rotate the magnet system and produce heading errors.

Turning and Acceleration Errors—Effect

10. Consider an aircraft in the Northern Hemisphere increasing speed whilst heading West, or turning from North or South onto West. In both cases the accelerating force acts through the pivot which is the magnet system's point of attachment to the aircraft. The reaction force acts, not through the pivot, but through the magnet system's centre of gravity.

11. Looking down on the magnet system in Fig 3 it can be seen that a couple is produced

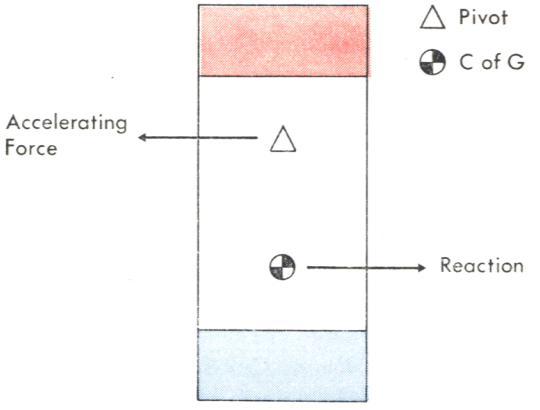


Fig 3 Accelerating Force Producing Couple

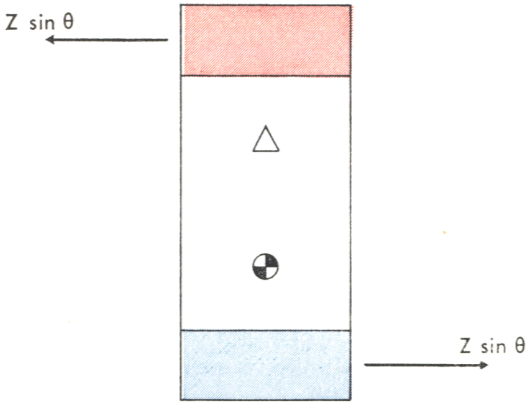


Fig 5 Couple Causing Turn

which turns the magnet system anti-clockwise. Considering the effect of these forces in the vertical plane together with the magnetic forces acting on the magnet, it can be seen from Fig 4 that the accelerating force and its reaction create a couple which tilts the magnet system out of the vertical. The vertical component of the Earth's magnetic field no longer acts through the pivot, but can be resolved into two orthogonal components. One component ($Z \cos \theta$) acts through the pivot, and the other ($Z \sin \theta$) at 90° to the pivot. θ is the angle of tilt. In Fig 4 it is shown that the component $Z \sin \theta$ tends to pull the blue end of the magnet to the right. An equal but opposite effect is created at the red end, and a magnetic couple is created which turns the magnet system anti-clockwise (Fig 5).

12. Two couples, one mechanical and one magnetic, turn the magnet system anti-clockwise. If the error is caused by an increase in speed, the effect is an apparent turn to North, *ie* the compass over-reads. If the error is caused by turning, the effect depends on the direction and rate of turn. In turns through North, the magnet system turns in the direction of turn and in all but the most violent manoeuvres, the indicated turn is slower than the actual turn, *ie* the compass under reads the turn indicating a turn of perhaps 20° for an actual turn through 45°. In turns through South, however, the magnet system turns in the opposite direction to the turn and the indicated turn is greater than the actual turn, *ie* the compass indicates a turn of perhaps 40° for an actual turn of 20°.

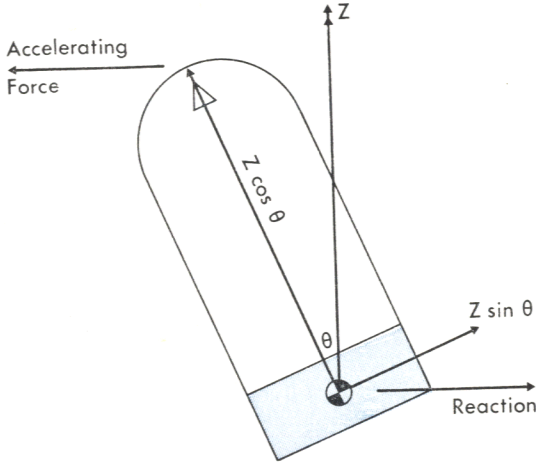


Fig 4 Acceleration Causing Tilt

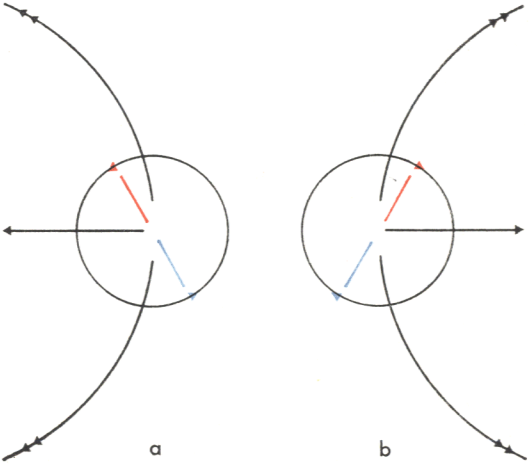


Fig 6 Effects of Turning and Acceleration Errors

Summary

13. The effects of turning and acceleration errors are illustrated in Fig 6 and summarized below:

a. *Northern Hemisphere*

(1) Acceleration on westerly headings and turns to the West cause the magnet system to rotate anti-clockwise.

(2) Acceleration on easterly headings and turns to the East cause the magnet system to rotate clockwise.

(3) Acceleration causes an apparent turn to the North.

(4) Turns through North cause the compass to under-indicate the turn.

(5) Turns through South cause the compass to over-indicate the turn.

b. *Southern Hemisphere.* The effects are reversed in the Southern Hemisphere.

Minor Errors

14. The following minor errors also occur:

a. *Scale Error.* Scale error is caused by errors in the calibration of the compass card or grid ring.

b. *Alignment Error.* Alignment error is caused by the incorrect mounting of the compass in the aircraft, or by a displaced lubber-line. The error is corrected by the compass swing.

c. *Centring Error.* Centring error occurs when the compass card is not centred on the magnet system pivot.

d. *Parallax Error.* When reading a DICS care must be taken to ensure that the eye is centred on the face of the compass. If the line of sight is offset parallax errors occur.

CHAPTER 2

E2 SERIES COMPASSES

CONTENTS

	Paras
Introduction	1
Compass, Type E2A	2–5
Compass, Type E2B	6
Compass, Type E2C	7
Serviceability Checks	8
Compass Swinging	9

Introduction

1. The magnetic compasses of the E2 series are miniature instruments, developed for use as emergency or standby compasses where lack of space precludes the installation of the larger standby compass of the P type. They are of the vertical card-type, the compass card being graduated every 10 degrees with figures every 30 degrees. The cardinal points are marked by the appropriate letter. The compasses are designed to give a bench accuracy of $\pm 2\frac{1}{2}°$ and an operational accuracy of $\pm 10°$. However, with good flight conditions and a stable heading indication, interpolation may be made to an extent comparable with bench accuracy.

Compass, Type E2A

2. The bowl of the compass is of a plastic material with the lubber line marked on the front inside of the bowl. The stem supporting the compass system is mounted on a bracket which is screwed into the base of the bowl. The compass bowl is completely filled with a silicone fluid, chosen because it has no detrimental effect on the material of the bowl and because its temperature/viscosity changes are small. A bellows at the rear of the bowl allows for change of the volume of the liquid due to variation in temperature.

3. The magnet system comprises a steel ring magnet to which is riveted a dome. The iridium-tipped pivot screws into the centre of the dome and rests in a sapphire cup secured to the vertical stem by the cupholder. The magnet system is therefore pendulously suspended. The compass card is attached to the magnet system by brackets riveted to the card and to the magnet. When the compass is inverted, the magnet system cannot become detached from the vertical stem since the clearance between the dome and the top of the bowl is insufficient to allow the pivot to leave the cup.

4. Two pairs of adjustable correctors are fitted to a metal plate which is secured to the top of the bowl. When the corrector magnets are in the neutral position, index lines engraved on small circular rotatable plates above the magnets are aligned with fixed lines engraved on the main plate. The main plate is also engraved with the letters B and C against the rotatable plates, denoting the coefficients for which the correction is being made. The operating heads for the correctors are situated at the top front of the bowl, one on either side of the filler plug. The letters B and C are engraved on the rim of the top plate above their respective corrector operating heads. The operating heads are turned by a small E2 compass corrector key.

5. Two radial slots in the compass mounting plate permit adjustment in azimuth to correct for coefficient A. A scale engraved on the rim of the top plate between the letters B and C enables the amount of movement to be assessed. The E2A compass is shown in Fig 1.

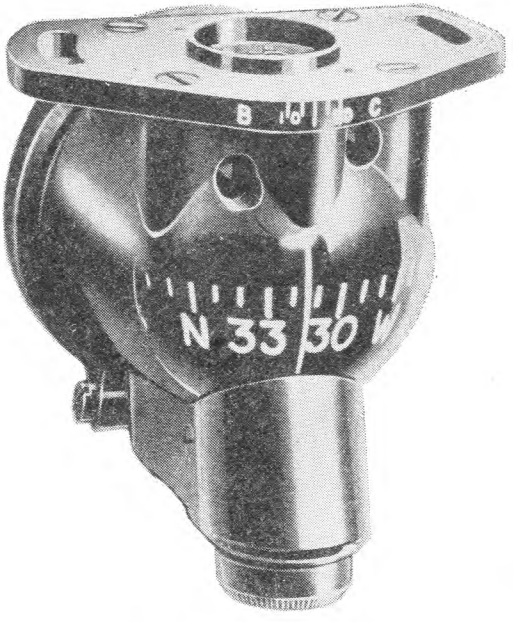


Fig 1 Compass, Type E2A

Fig 2 Compass, Type E2B and E2C

Compass, Type E2B

6. The E2B compass is a variant of the E2A compass, the difference being that the E2B compass incorporates direct red lighting of the compass bowl. This lighting is achieved by means of a miniature 28 v DC non-magnetic lamp and a red filter. The intensity of illumination of the lamp can be varied by means of a standard pattern dimmer switch. The E2B compass is used in aircraft where the cockpit lighting is red. The E2B compass is shown in Fig 2.

Compass, Type E2C

7. The E2C compass is almost identical to the E2B compass but has a clear instead of a red filter around the internal lamp. It thus has direct white lighting of the compass bowl instead of red. The E2C compass is used in aircraft where the cockpit lighting is white.

Serviceability Checks

8. The complete inspection of a compass of the E2 series is impracticable when the compass is in use, but the pilot should inspect the compass as follows:

a. The bowl must be completely filled with liquid and no bubbles should be present.

b. The bowl and liquid should be free from discolouration. It is possible that there will be slight discolouration, but the compass should not be declared unserviceable unless it is difficult to read the compass card under normal lighting conditions.

c. The bowl and liquid should not contain an excessive amount of sediment.

d. The compass bowl should not be cracked or damaged in any way.

Compass Swinging

9. The detailed instructions for swinging a compass of the E2 series are given in Sect 6.

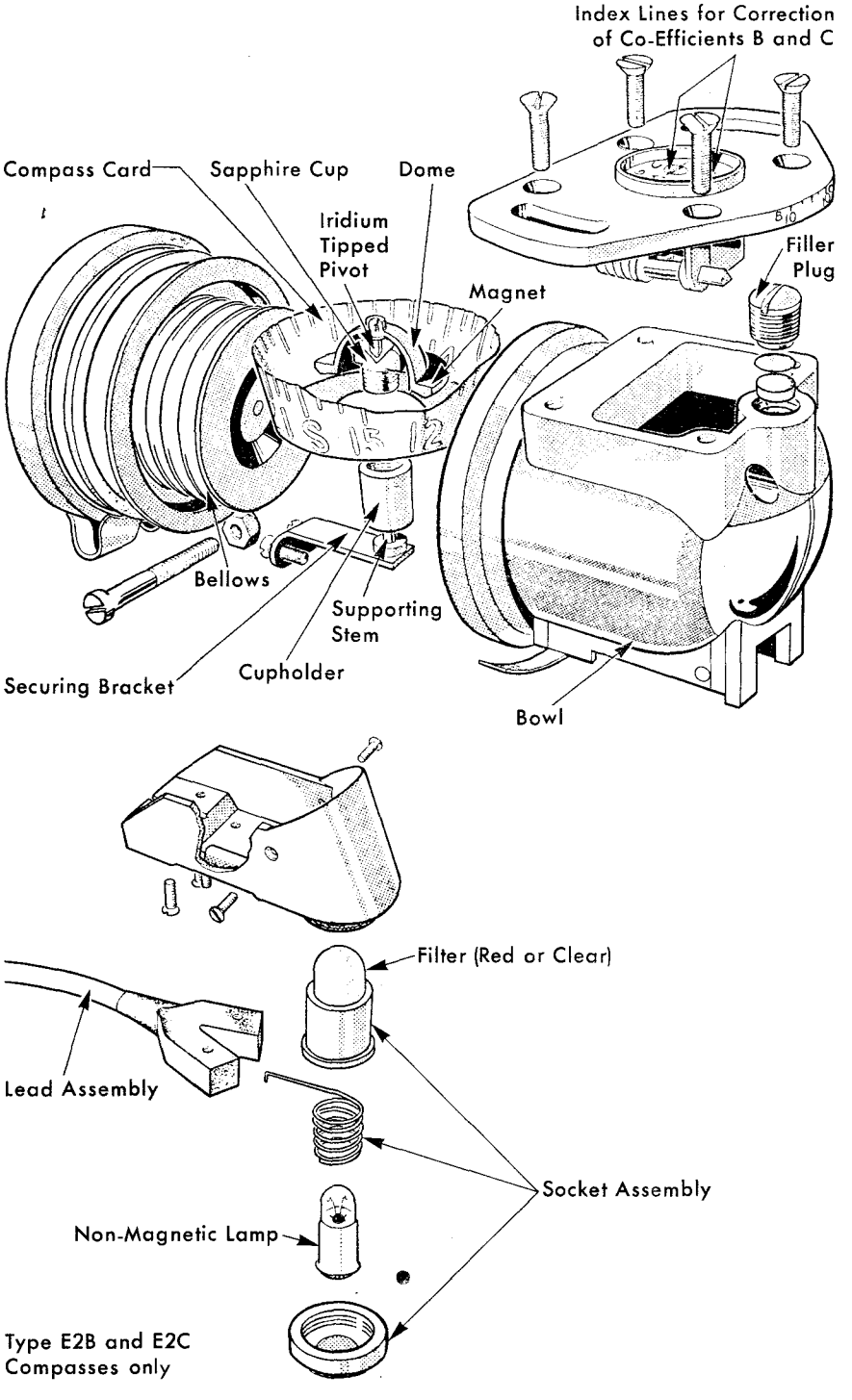


Fig 3 Compasses, E2 Series—Exploded View

CHAPTER 3

PILOT TYPE COMPASSES

CONTENTS

	Paras.
Introduction	1–2
COMPASSES	
Pilot's Magnetic Compass, Type P10	3–8
Pilot's Magnetic Compass, Type P11	9
Pilot's Magnetic Compass, Type P12	10–12
CORRECTOR DEVICES	
Adjustable Corrector No. 3	13–14
Adjustable Corrector No. 4	15
OPERATION	
Inspection	16–18
In-Flight Use	19-21
Compass Swinging	22

Introduction

1. Pilot-type magnetic compasses are provided for setting or reading the heading of an aircraft, and as standby compasses in aircraft fitted with remote indicating compasses. Those in use in the Service consist of an aperiodic magnet system suspended in a liquid-filled bowl which, by means of an intermediate floating ring, is permitted to move fore and aft, athwartships and vertically to minimize vibration but cannot turn in azimuth relative to the container in which it is suspended.

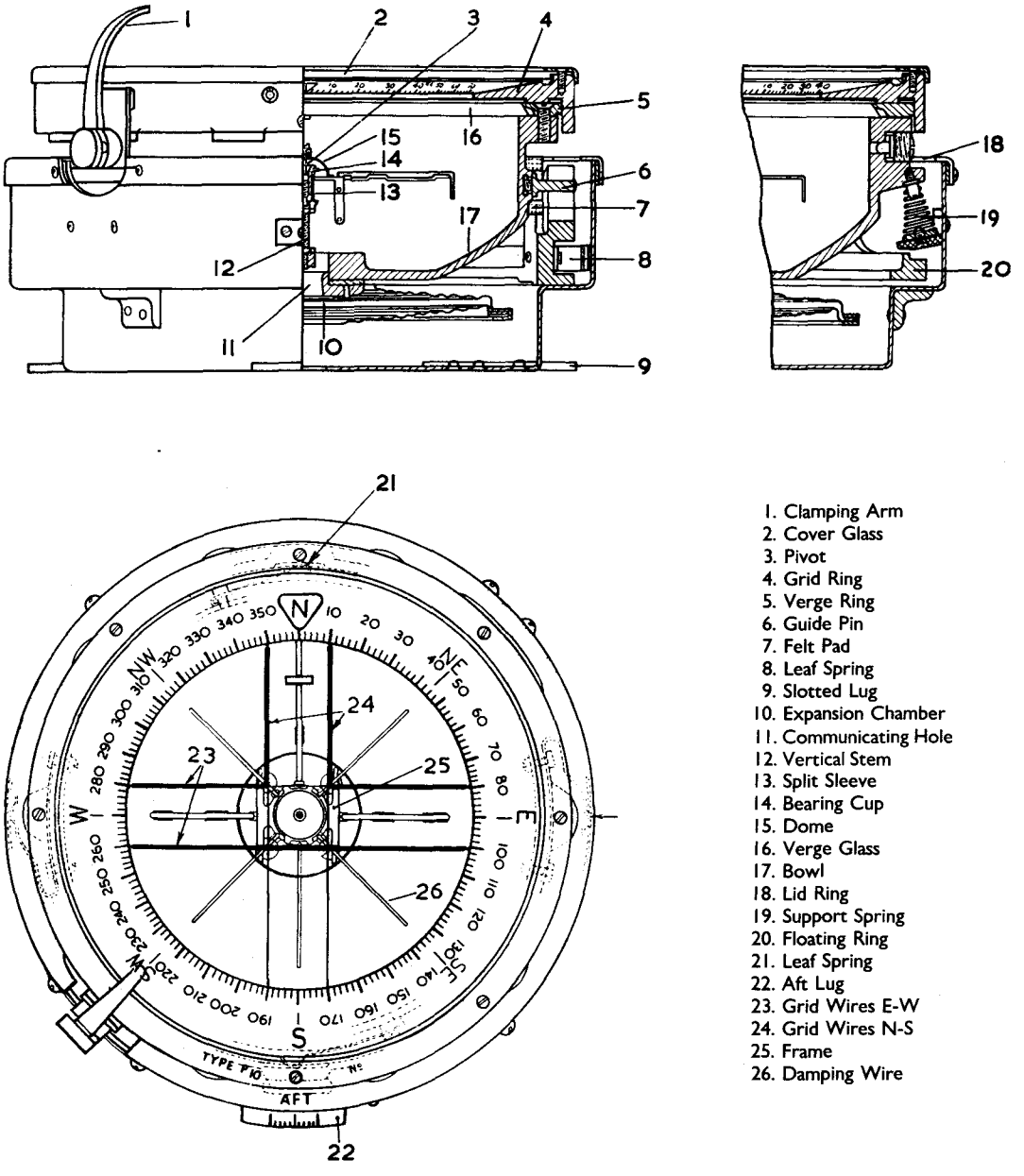
2. The information on magnetic compasses given in Chap 1 applies to pilot type compasses and should be read in conjunction with this chapter.

COMPASSES

Pilot's Magnetic Compass Type P10

3. The magnetic compass, type P10 has a black moulded bowl which is cylindrical in shape and filled with liquid. The bowl is sealed at the top by a verge glass and rubber gasket held in place by a verge ring. A hole in the bottom of the bowl connects with a bellows-type expansion chamber which accommodates changes in the volume of the liquid due to temperature changes.

4. A grid ring fits over the verge ring and is retained by screws which locate in a channel around the edge of the verge ring. By this means the grid ring may be turned on the verge ring but cannot be lifted off. The grid



1. Clamping Arm
2. Cover Glass
3. Pivot
4. Grid Ring
5. Verge Ring
6. Guide Pin
7. Felt Pad
8. Leaf Spring
9. Slotted Lug
10. Expansion Chamber
11. Communicating Hole
12. Vertical Stem
13. Split Sleeve
14. Bearing Cup
15. Dome
16. Verge Glass
17. Bowl
18. Lid Ring
19. Support Spring
20. Floating Ring
21. Leaf Spring
22. Aft Lug
23. Grid Wires E-W
24. Grid Wires N-S
25. Frame
26. Damping Wire

Fig 1 Pilot's Magnetic Compass, Type P10 – Sectional View

ring is graduated every two degrees and has figures every ten degrees. The cardinal and quadrantal points are marked with appropriate letters, those denoting the cardinal points being slightly larger, and the N having a red background. Four wires are stretched across the grid ring, two of them parallel to the E-W line and the other two parallel to the N-S line. These are known as grid wires and are used to align the grid ring with the magnet system. A cover glass is fitted to the grid ring to protect the scale and grid wires. The grid ring

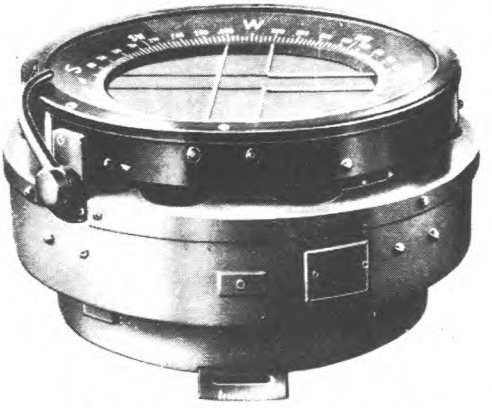


Fig 2 Pilot's Magnetic Compass Type P10

may be clamped to the bowl by operation of the clamping arm.

5. The magnet system, which is painted black, consists of four magnets attached to a frame which also carries damping wires. The North filament is distinguished by a cross. The system is pendulously suspended by a pivot resting in a sapphire bearing cup which is mounted on a vertical stem attached to a bridge in the bottom of the bowl. A split sleeve, secured by a nut, fits over the sapphire cup and its stem, to prevent the magnet system becoming detached during aerobatics.

6. To enable the compass to be used in darkness the lubber line, cardinal letters and certain essential grid wires and filaments are luminized.

7. The compass bowl is supported in a cylindrical container by four equally spaced support springs which isolate it from vibration. The floating ring, fitted with slots engaging with pins on the compass bowl and container, prevents movement of the bowl in azimuth but permits it in fore and aft, athwartships and vertical directions. Movement of the floating ring is damped by the use of metal leaf springs, two attached to the bowl and bearing on the floating ring and two attached to the floating ring and bearing on the container. The bowl is retained to the container by means of a lid ring, vertical movement being limited by lugs which come into contact with felt pads on the lid ring.

8. The compass is secured in the aircraft by bolts of non-magnetic material which pass through three slotted lugs. The slots allow a small degree of movement in azimuth so that correction for coefficient A may be made. The aft lug is graduated in degrees so that the amount of adjustment may be judged.

Pilot's Magnetic Compass, Type P11

9. The magnetic compass, type P11 is similar in design and appearance to the compass, type P10, but it is approximately two inches smaller in diameter. The compass bowl is of metal, and the grid ring and floating ring are of plastic material. The magnet system has two magnets only, and the vertical stem supporting the pivot is attached to a bridge running across a diameter of the bowl. Changes in the volume of the compass liquid due to temperature changes are accommodated by a metal bellows fixed in the bottom of the bowl.

Pilot's Magnetic Compass, Type P12

10. The magnetic compass, type P12 is designed as a roof-mounting compass, the heading being read through an adjustable mirror mounted below the compass. The compass can be fitted to the instrument panel of an aircraft but in this case the lamp housing must be omitted. In construction the compass, type P12, is similar to the compass, type P11, except that the vertical stem carrying the pivot is fitted to the centre of the verge glass.

11. A scale is engraved around the trim of the grid ring in addition to the scale on its face. This rim scale is read against a second lubber line, mounted on the side of the container, and illuminated by an electric lamp in a fitting attached to the side of the container.

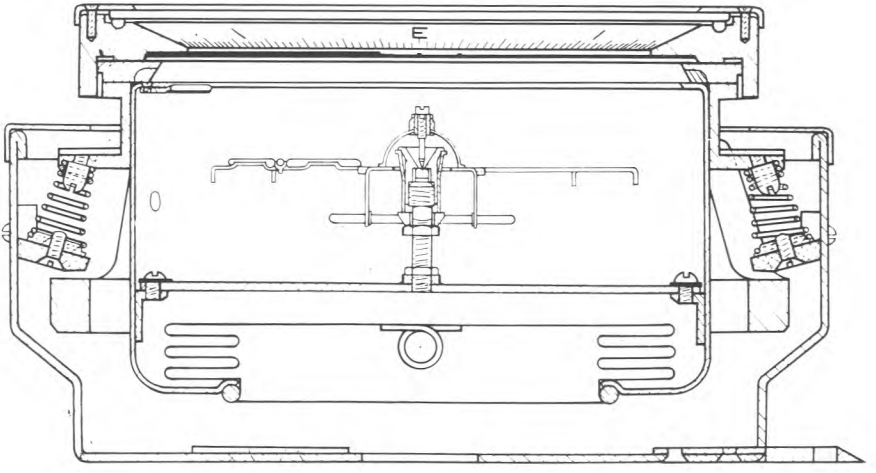


Fig 3 Pilot's Magnetic Compass, Type P11 – Sectional View

Fig 4 Pilot's Magnetic Compass, Type P12

12. An arm attached to the back of the container supports a plate glass mirror. The position of the arm can be adjusted and clamped in position by means of a wing nut and spring washer. The mirror is attached to the lower end of the adjustable arm by means of a universal ball joint which can be locked in position by operating a lever which projects beyond the edge of the mirror.

CORRECTOR DEVICES

Adjustable Corrector, No 3

13. The adjustable corrector, No. 3, is a micro-adjuster used to produce a local magnetic field to correct coefficients B and C of the compass deviation.

14. Two pairs of magnets are contained in a moulded cylindrical casing fitted with four mounting lugs and clearly marked to indicate the correct mounting position relative to the aircraft. Two operating heads, 180° apart, are provided for each pair of magnets in case one is inaccessible when the corrector is installed. The neutral position of the magnets is not marked; it must be found by turning the operating head to its two extreme

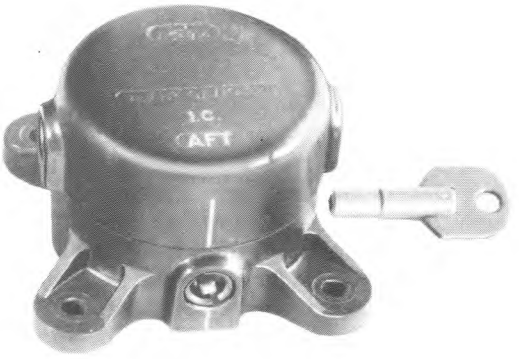


Fig 5 Adjustable Corrector, No 3 and Corrector Key

positions and then determining the mid-point. Coefficient B is adjusted by turning the fore-and-aft operating head, and co-efficient C by turning the athwartships one, with a corrector key.

Adjustable Corrector, No 4

15. The adjustable corrector, No 4 employs the same method as the corrector, No 3, for the correction of coefficients B and C. In addition, however, it embodies a pair of magnets at right angles to the two horizontal pairs to correct for coefficient R. These vertical corrector magnets can be adjusted only from one operating head situated in the base of the corrector.

OPERATION

Inspection

16. Compasses in use must be examined periodically by trained personnel. Pilots and navigators should examine their compasses and should be in a position to form an opinion as to serviceability, but they must obtain the advice of a specialist officer before condemning a compass. On the examination the following points must be checked but compasses may not be dismantled and force should not be applied.

a. The bowl must be completely filled with liquid and no bubbles should be present. The bowl and liquid must be free from discolouration.

b. The error due to pivot friction must not be more than two degrees.

c. The spring suspension of the bowl and all mechanical devices must be in good condition. The bowl should move freely in any direction and return to its original position without touching the container at any point.

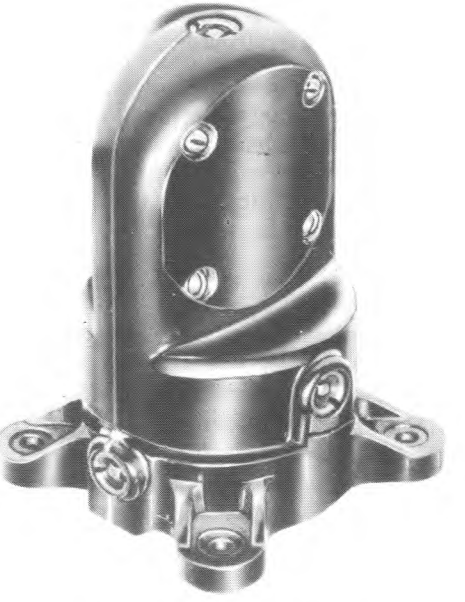


Fig 6 Adjustable Corrector, No 4

17. **Discolouration of the Liquid.** Discolouration of the compass spirit is caused by impurities in the liquid, in the bowl or in the joints, and normally only develops with age. It is probable that few compasses will be entirely free from discolouration, but those affected should not be condemned unless excessive pivot friction develops or it is difficult to see the luminous filaments of the magnet system under the normal lighting conditions. A sediment may form in the liquid and this will settle in the pivot cup and in the bottom of the bowl when the compass is stationary. A compass in which there is an

excessive amount of sediment should be considered unserviceable.

18. **Pivot Friction.** Excessive friction at the pivot of the magnet system may develop gradually or it may be caused by mechanical shock. It is often accompanied by discoloured compass liquid. To test the pivot friction proceed as follows:

a. Level the compass and note the indication.

b. By placing a small bar magnet above the magnet system, deflect the system about 10 degrees and hold it in this position for 30 seconds to allow the liquid to come to rest.

c. Remove the bar magnet, allow the magnet system to settle and note the indication.

d. Repeat b and c, deflecting the magnet system in the opposite direction.

e. Note the difference between the two positions in which the magnet system settled. This difference is a measure of the pivot friction.

In general, the pivot friction of a new compass measured in this way in a horizontal field of 0.18 gauss should not exceed 2 degrees. A compass may be considered serviceable if the difference in readings can be brought within the limit by lightly tapping the compass bowl. If the difference is permanent at more than 2 degrees the compass must be considered unserviceable. The test should be made on four headings 90° apart, and when the compass is installed in an aircraft this can conveniently be done when the compass is being swung. Since the frictional error is inversely proportional to the value of the Earth's horizontal magnetic field (H) and sensibly independent of temperature, the error can be calculated for different values of H. *Example:* What is the permissible error due to pivot friction

when H = 0.24?

Since it is 2° when H = 0.18 then

$$\frac{0.18}{0.24} = \frac{\text{New error}}{2}$$

$$\therefore \text{New permissible error} = \frac{0.18}{0.24} \times 2 = 1.5 \text{ degrees.}$$

In-Flight Use

19. To determine the heading of the aircraft, set the grid ring so that the E-W grid wires are parallel to the E-W luminized filaments of the magnet system, with the N pointer of the magnet system between the two N luminized grid wires. The heading is then shown on the scale and must be corrected by reference to the deviation card to obtain the magnetic heading of the aircraft.

20. To steer a magnetic heading, apply the deviation to the required heading to obtain the required compass reading. Set the grid ring so that the required compass heading is shown on the scale against the lubber line, and clamp it. Turn the aircraft until the E-W luminized filaments of the magnet system are parallel to the E-W grid wires with the N pointer of the magnet system between the two N luminized grid wires.

21. If possible, compass readings should only be made during steady flight, since errors are introduced by turning and acceleration. When the aircraft is tilted the grid ring may no longer be parallel to the magnet system, and care should be taken to avoid errors in setting and reading the grid ring. Readings should not be taken if the tilt is more than 20 degrees, as the magnet system is then in contact with the verge glass and is not free to rotate. The compass should always be read from directly above the scale to avoid parallax error. If all these precautions are taken, a P-type compass can indicate a heading with an accuracy of ± 2°.

Compass Swinging

22. Details for the swinging of P-type compasses are contained in Section 6.

PART 2

SECTION 2

GYROSCOPIC REFERENCES

Chapter

1 **Introduction to Gyroscopes**

2 **Master Reference Gyro, Mark 1**

3 **CL 11 Directional Gyro**

4 **Sperry Twin Gyro Platform (MRG Mk 2)**

5 **Turn and Slip Indicators**

6 **The Direction Indicator Mk 1**

7 ~~**Artificial Horizons and Horizon Gyro Units**~~
DIRECT READING ATTITUDE INDICATORS

CHAPTER 1

INTRODUCTION TO GYROSCOPES

CONTENTS

	Paras
Introduction	1
Classification of Gyroscopes	2–3
Definition of Terms	4
Laws of Gyrodynamics	5–23
THE RATE GYROSCOPE	
Principle of Operation	24–26
THE INTEGRATING GYROSCOPE	
Principle of Operation	27–32
THE DISPLACEMENT GYROSCOPE	
Definition	33
Wander	34–37
Earth Rotation	38–39
Transport Wander	40–41
Apparent Wander Table	42–43
Practical Corrections for Topple and Drift	44–47
Gimbal Lock	48
Gimbal Error	49
GYRO SUSPENSION METHODS	
Introduction	50–52

Introduction

1. A gyroscope consists of a symmetrical rotor spinning rapidly about its axis and free to rotate about one or more perpendicular axes. Freedom of movement about one axis is usually achieved by mounting the rotor in a gimbal, as in Fig 1 and complete freedom can be approached by using two gimbals, as illustrated in Fig 2. The physical laws which govern the behaviour of a gyroscope are identical to those which account for the behaviour of the Earth itself. The two principal properties of a gyro are rigidity in inertial space (inertial space being a fixed spatial datum) and precession. These properties are exploited in heading reference systems and inertial navigation systems, and some aircraft instruments which are described in this volume.

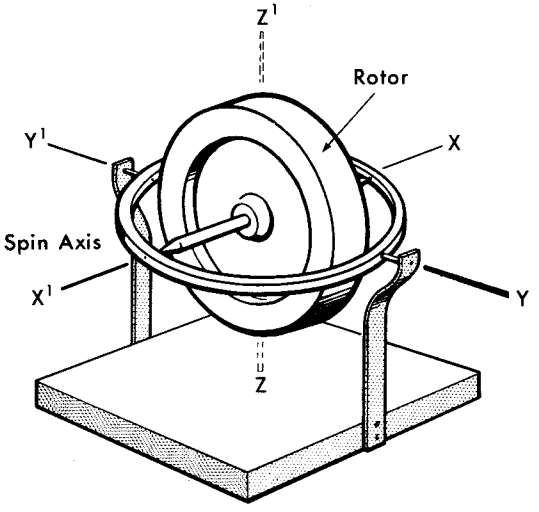


Fig 1 Gyroscope With One Degree of Freedom

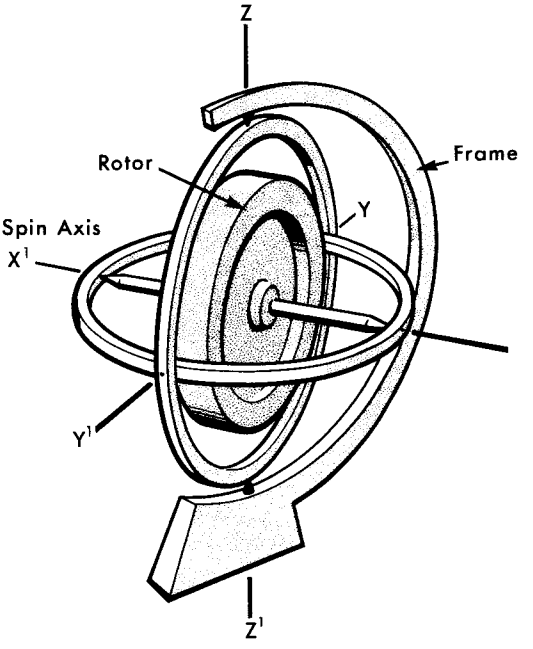


Fig 2 Gyroscope With Two Degrees of Freedom

Classification of Gyroscopes

2. Gyroscopes are classified in Table 1 in terms of the quantity they measure, namely:

a. Angular displacement from a known datum.

b. Rate of angular displacement of a vehicle.

c. The integral of an input with respect to time.

3. It should be realized however that the above classification is one of a number of ways in which gyroscopes can be classified. Referring to Table 1 it will be seen that a displacement gyroscope could be classified as a two degree of freedom gyro or a space gyro. Note also that the classification of Table 1 does not consider the spin axis of a gyroscope as a degree of freedom. In this chapter a degree of freedom is defined as the ability to measure movement about a chosen axis.

Definition of Terms

4. The following fundamental mechanical definitions provide the basis of the laws of gyrodynamics:

a. *Momentum.* Momentum is the product of mass and velocity. (mv).

b. *Angular Velocity.* Angular velocity is the instantaneous linear velocity at the periphery of a circle divided by the radius of the circle, $\frac{v}{r}$ or ω and is normally measured in radians per second.

c. *Angular Momentum.* If the mass (m) of a body is concentrated at a radius (r) from the axis of rotation, the angular momentum is the product of the instantaneous linear momentum (mv) and the radius. Angular momentum = mvr or $mr^2\omega$.

d. *Moment of Inertia.* The moment of inertia of a body is the summation of mr^2 for every particle of mass 'm' which it can

TYPE OF GYRO	USES IN GUIDANCE AND CONTROL	GYRO CHARACTER-ISTICS
Rate Gyroscope	1 Aircraft Instruments	1 Modified single degree of freedom gyro. 2 Illustrates the principle of secondary precession.
Rate Integrating Gyroscope	1 Inertial Navigation	1 Modified single deg. of freedom gyro. 2 Can also be a two degree of freedom gyro.
Displacement Gyroscope	1 Heading reference 2 Inertial Navigation Systems 3 Aircraft Instruments	1 Two degrees of Freedom. 2 Defines direction with respect to space, thus it is also called a space gyro, or free gyro.

Table I Classification of Gyros

be considered as comprising. In the case of a disc or wheel of uniform mass distribution throughout its radius r, the moment of inertia I about its axle is $\frac{mr^2}{2}$. For a cylinder of outside radius r_1 and inner radius r_2, $I = \frac{m(r_1^2 + r_2^2)}{2}$.

An alternative expression for Angular Momentum is the product of Moment of Inertia I, and Angular Velocity ω.

e. *Radius of Gyration.* The radius of gyration of a body is that distance from the axis of rotation at which all the mass of that body can be considered to act. It is normally denoted by k. Consequently, in calculations of moments of inertia of bodies of irregular shapes, $I = mk^2$, where m is the mass of the body.

f. *Gyro Axes.* In gyrodynamics it is convenient to refer to the axis about which the torque is applied as the input axis and that axis about which the precession takes place as the output axis. The third axis, the spin axis, is self evident.

Laws of Gyrodynamics

5. **Rigidity in Space.** If the rotor of a perfect displacement gyroscope is spinning at constannt angular velocity, and therefore constant angular momentum, no matter how the frame is turned no torque is transmitted to the rotor axis. The law of conservation of angular momentum states that the angular momentum of a body is unchanged unless a torque is applied to that body. It follows from this that the angular momentum of the rotor must remain constant in magnitude and direction. This is simply another way of saying that the rotor spin axis continues to point in the same direction in inertial space. This property of a gyro is defined in the First Law of Gyrodynamics.

6. **The First Law of Gyrodynamics.** If a rotating body is so mounted as to be completely free to move about any axis through the centre of mass, then its spin axis remains fixed in inertial space however much the frame may be displaced.

7. A space gyroscope loses its property of rigidity in space if the spin axis is subjected to random torques, some causes of which will be examined later.

8. **Precession.** Consider the free gyroscope in Fig 3, spinning with constant angular momentum about the XX^1 axis. If a small mass 'M' is placed on the inner gimbal ring it exerts a downward force 'F' so producing a torque 'T' about the YY^1 axis. By the laws of rotating bodies this torque should produce an angular acceleration about the YY^1 axis, but this is not the case, and

a. Initially the gyro spin axis will tilt through a small angle ($\varnothing$ in Fig 3), after which no further movement takes place about the YY^1 axis.

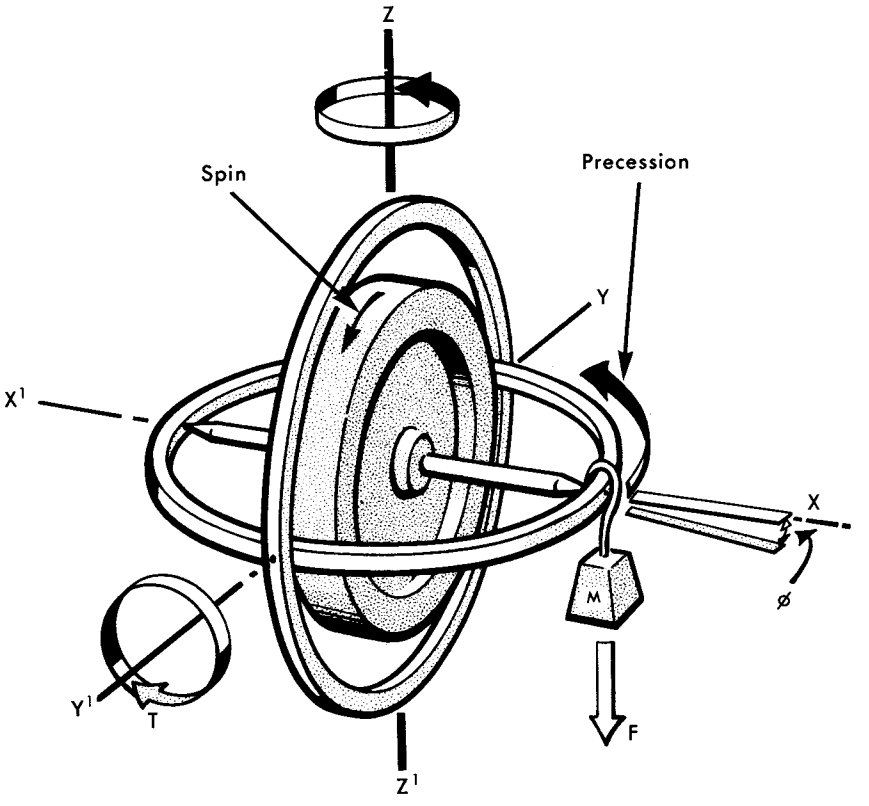


Fig 3 Precession

b. The spin axis then commences to turn at a constant angular velocity about the axis perpendicular to both XX^1 and YY^1, *ie* the ZZ^1 axis. This motion about the ZZ^1 axis is known as precession, and is the subject of the Second Law of Gyrodynamics.

9. **The Second Law of Gyrodynamics.** If a constant torque T is applied about an axis perpendicular to the spin axis of an unconstrained, symmetrial spinning body, then the spin axis will precess steadily about an axis mutually perpendicular to the spin axis and the torque axis. The angular velocity of precession Ω is given by:

$$\Omega = \frac{T}{I\omega}$$

10. The angle $\varnothing$ is proportional to T and is a measure of the work done. Its value is almost negligible and will not be discussed further.

11. Precession ceases as soon as the torque is withdrawn, but if the torque application is continued, precession will continue until the direction of spin is the same as the direction

of the applied torque. If however the direction of the torque applied about the inner gimbal axis moves as the rotor precesses, the direction of spin will never coincide with the direction of the applied torque.

12. **Direction of Precession.** Fig 4 shows a simple rule of thumb to determine the direction of precession:

a. Consider the torque as being due to a force acting at right angles to the plane of spin at a point on the rotor rim.

b. Carry this force around the rim through 90° in the direction of rotor spin.

c. The torque will apparently act through this point resulting the rotor precessing in the direction shown.

13. A full mathematical treatment of precession is beyond the scope of this chapter, but a brief explanation of Conservation of Angular Momentum and why precession occurs is given.

14. **Conservation of Angular Momentum.** In linear motion, if the mass is constant, changes in momentum caused by external forces will be indicated by changes in velocity. Similarly in rotary motion if the moment of inertia is constant, then the action of an external torque will be to change the angular velocity in speed or direction and in this way change the angular momentum. If, however, internal forces (as distinct from external torques) act to change the moment

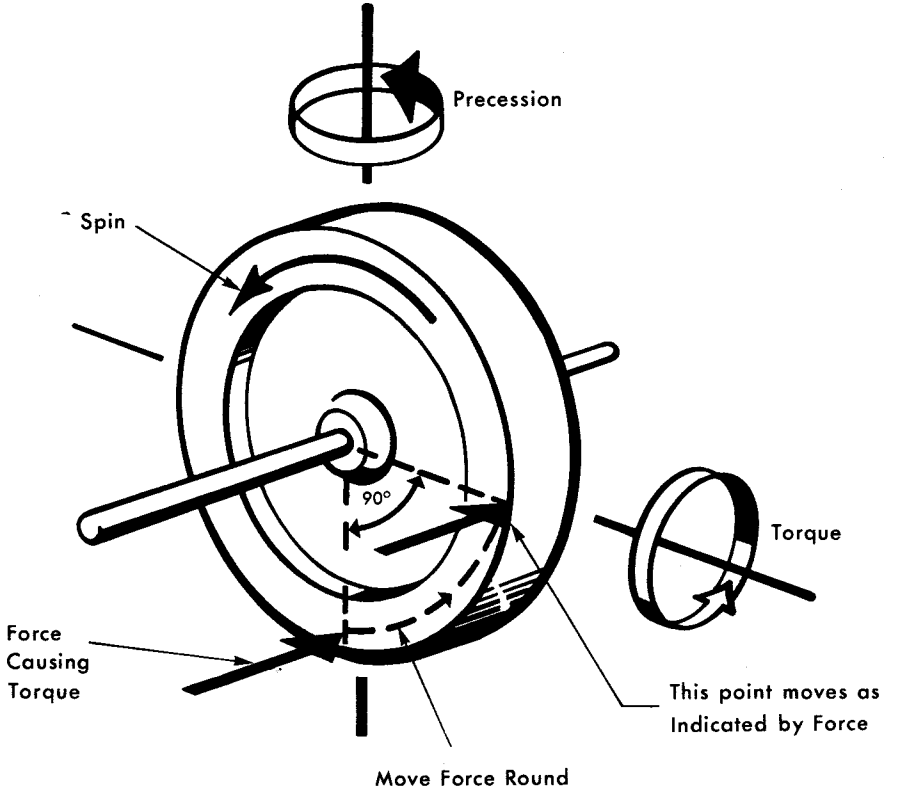


Fig 4 Determining of Precession

of inertia of a rotating system, then the angular momentum is unaffected. Angular momentum is the product of the moment of inertia and angular velocity, and if one is decreased so the other must increase to conserve angular momentum. This is the Principle of Conservation of Angular Momentum. Consider the ice-skater starting her pirouette with arms extended. If she now retracts her arms she will be transferring mass closer to the axis of the pirouette, so reducing the radius of gyration. If the angular momentum is to be maintained, then because of the reduction of moment of inertia the rate of her pirouette must increase, therefore:

a. If the radius of gyration of a rotating body is increased a force is considered to act in opposition to the rotation caused by the torque, decreasing the angular velocity.

b. If the radius of gyration is decreased a force is considered to act assisting the original rotation caused by the torque, so increasing the angular velocity.

15. **Cause of Precession.** Consider the gyroscope rotor in Fig 5a spinning about the XX^1 axis and free to move about the YY^1 and ZZ^1 axes. Let the quadrants (1, 2, 3 and 4) represent the position of the rotor in spin at one instant during the application of an external force to the spin axis, producing a torque about the YY^1 axis. This torque is tending to produce a rotation about the YY^1 axis while at the same instant the rotor spin is causing particles in quadrants 1 and 3 to recede from the YY^1 axis, increasing their moment of inertia about this axis, and particles in quadrants 2 and 4 to approach the YY^1 axis decreasing their moment of inertia about this axis. Particles 1, 2 and 4 tend to conserve angular momentum about YY^1, therefore:

a. If inertia (Y) increases, 1 and 3 exert forces opposing their movement about YY^1.

b. If inertia (Y) decreases, 2 and 4 exert forces assisting their movement about YY^1.

16. Hence 1 and 4 exert forces on the rotor downwards, whilst 2 and 3 exert forces upwards. These forces can be seen to form a couple about ZZ^1, Fig 5b, causing the rotor to precess in the direction shown in Fig 5c.

17. **Gyroscopic Resistance.** In demonstrating precession it was stated that after a small deflection about the torque axis movement about this axis ceased, despite the continued application of the external torque. This state of equilibrium means that the sum of all torques acting about this axis is zero. There must therefore be acting about this axis a resultant torque L which is equal and opposite to the external torque, as shown in Fig 6. This resistance is known as Gyroscopic Resistance and is created by internal couples in a precessing gyroscope.

18. **Formation of Gyroscopic Resistance.** Consider now the gyroscope in Fig 5c spinning about an axis XX^1 and precessing about the ZZ^1 axis under the influence of a torque T, about the YY^1 axis. The rotor quadrants represent an instant during the precession and spin. Using the argument of para 15, the particles in quadrants 1 and 3 are approaching the ZZ^1 axis and exerting forces acting in the direction of precession, while in quadrants 2 and 4 the particles are receding from the ZZ^1 axis and exerting forces in opposition to the precession. The resultant couple is therefore acting about the YY^1 axis in opposition to the external torque. This couple is the Gyroscopic Resistance. It has a value equal to the external torque thus preventing movement about the YY^1 axis.

19. Gyroscopic Resistance is always accompanied by precession, and it is of interest to note that if precession is prevented, gyroscopic torque cannot form and it as easy to move the spin axis when it is

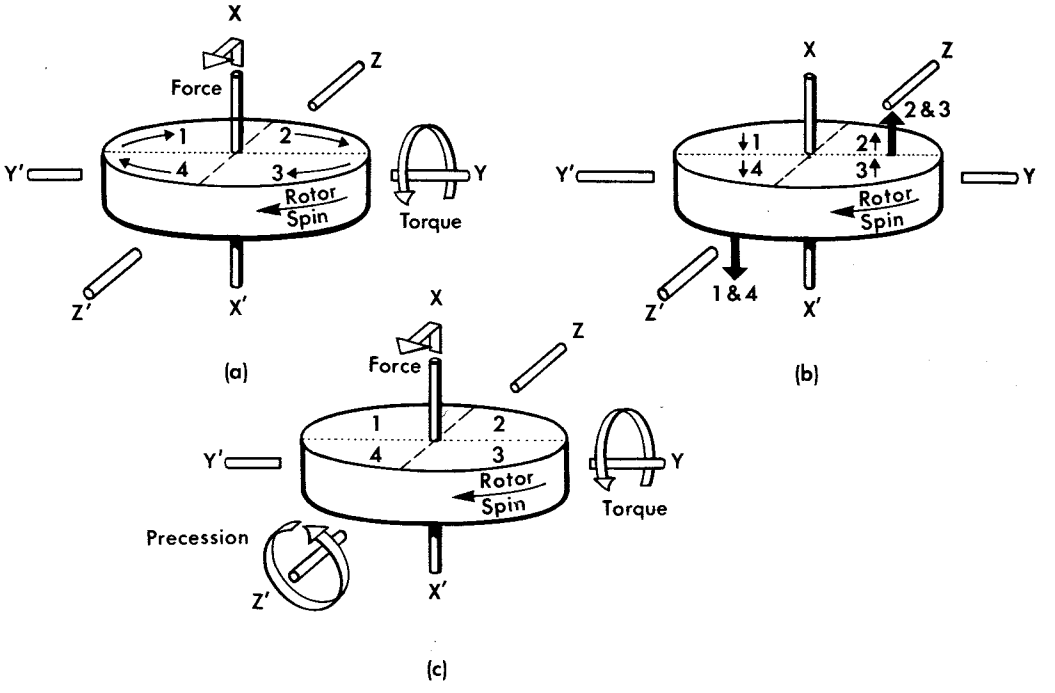


Fig 5 Instant of Spin and Precession

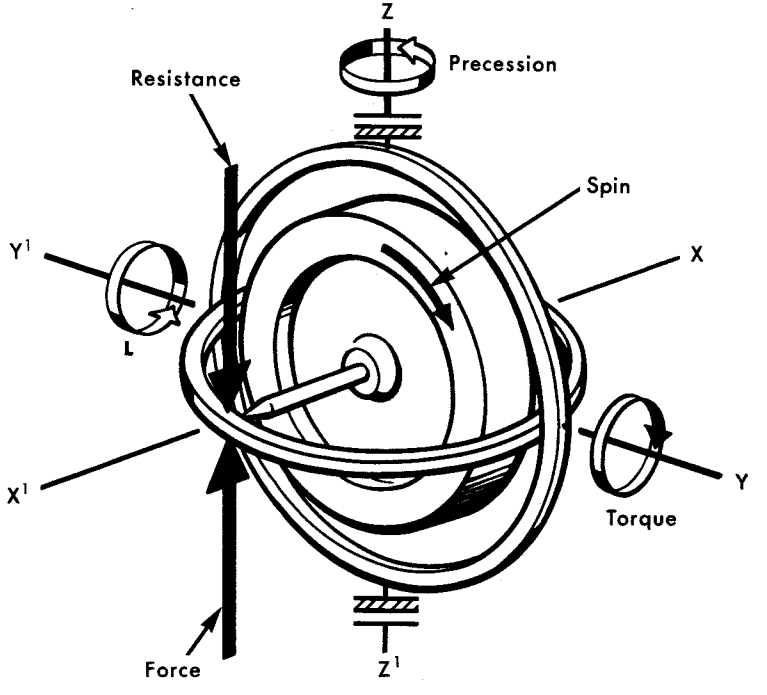


Fig 6 Gyro Resistance

spinning as when it is at rest. This can be demonstrated by applying a torque to the inner gimbal of a gyroscope with one degree of freedom. With the ZZ^1 axis locked the slightest touch on the inner gimbal will set the gimbal ring (and the rotor) moving. This behaviour is exploited in caging devices.

20. **Secondary Precession.** If a sudden torque is applied about one of the degrees of freedom of a perfect displacement gyroscope the following phenomena should be observed:

a. Nodding or nutation occurs. Here it is sufficient to note that nutation occurs only for a limited period of time and eventually will cease completely. Additionally nutation can only occur with a two degree of freedom gyro and to a large extent it can be damped out by gyro manufacturers.

b. A deflection takes place about the torque axis, (dip), which remains constant provided that the gyro is perfect and the applied torque is also constant.

c. The gyro precesses, or rotates, about the ZZ^1 axis.

21. If, however, an attempt is made to demonstrate this behaviour, it will be seen that the angle of dip will increase with time, apparently contradicting sub-para 20b.

22. To explain this discrepancy, consider Fig 7. If the gyro is precessing about the ZZ^1 axis, some resistance to this precession must take place due to the friction of the outer gimbal bearings. If this torque T is resolved using the rule of thumb given in para 12, it will be seen that the torque T causes the spin axis to dip through a larger angle. This

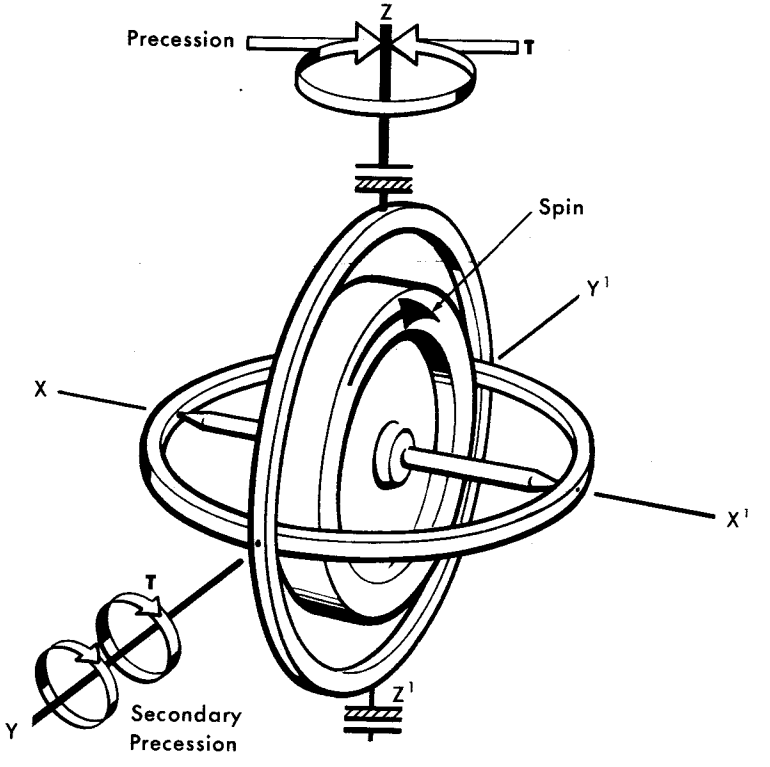


Fig 7 Precession Opposed by Secondary Precession

precession is known as secondary precession.

23. It should be noted that secondary precession can only take place when the gyro is already precessing, thus its name. Note also that secondary precession acts in the same direction as the originally applied torque.

THE RATE GYROSCOPE

24. Fig 8 shows a gyroscope with freedom about one axis YY^1. If the frame of the gyro is turned about an axis ZZ^1 at right angles to both YY^1 and XX^1, then the spin axis will precess about the YY^1 axis. The precession will continue until the direction of rotor spin is coincident with the direction of the turning about ZZ^1.

25. Suppose the freedom of this gyroscope about the gimbal axis is restrained by the springs connecting the gimbal ring to the frame as in Fig 9. If the gyroscope is now turned about the ZZ^1 axis, precession about the YY^1 axis is immediately opposed by a torque applied by the springs. It has been shown that any torque opposing precession produces a secondary precession in the same direction as the original torque (*see* para

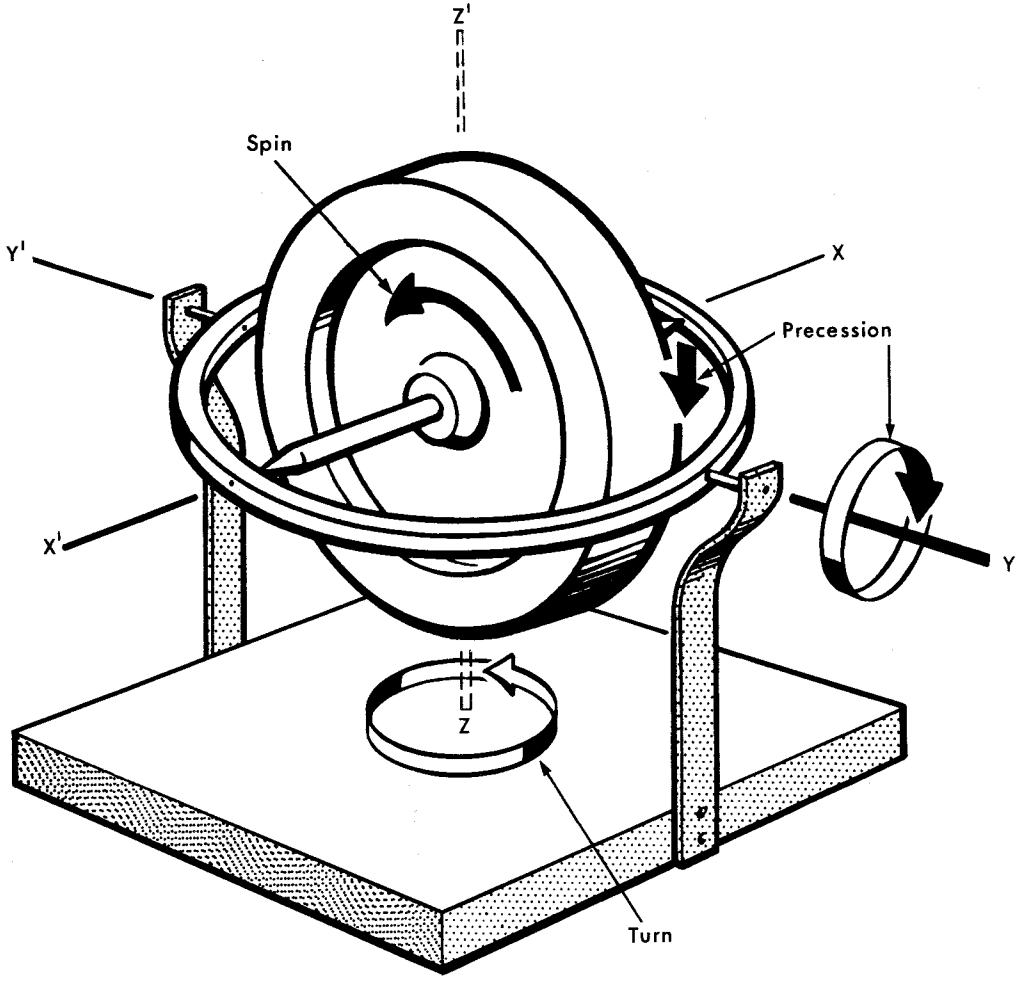


Fig 8 Gyro with One Degree of Freedom – Precession

23). If the turning of the frame is continued at a steady rate, the precession angle about the YY^1 axis will persist, distending one spring and compressing the other, thereby increasing the spring torque. Eventually the spring torque will reach a value where it is producing secondary precession about ZZ^1 equal to, and in the same direction as, the original turning. When this state is reached the gyroscope will be precessing at the same rate as it is being turned and no further torque will be applied by the turning. Any change in the rate of turning about the ZZ^1 axis will require a different spring torque to produce equilibrium, thus the deflection of the spin axis ($\varnothing$ in Fig 9) is a measure of the rate of turning. Such an arrangement is known as a Rate Gyroscope, and its function is to measure a rate of turn, as in the Rate of Turn Indicator.

26. The relationship between the deflection angle and rate of turn is derived as follows:

Spring Torque $\propto \varnothing$ or
Spring Torque = $K\varnothing$ (where K is a constant)
At equilibrium:
Rate of Secondary Precession = Rate of Turn,
ie $\frac{K\varnothing}{I\omega}$ = Rate of Turn
$\therefore \varnothing \propto$ = Rate of Turn $\times I\omega$

($I\omega$ is the angular momentum of the rotor and is therefore constant).

The angle of deflection can be measured by an arrangemment shown at Fig 10 and the scale calibrated accordingly.

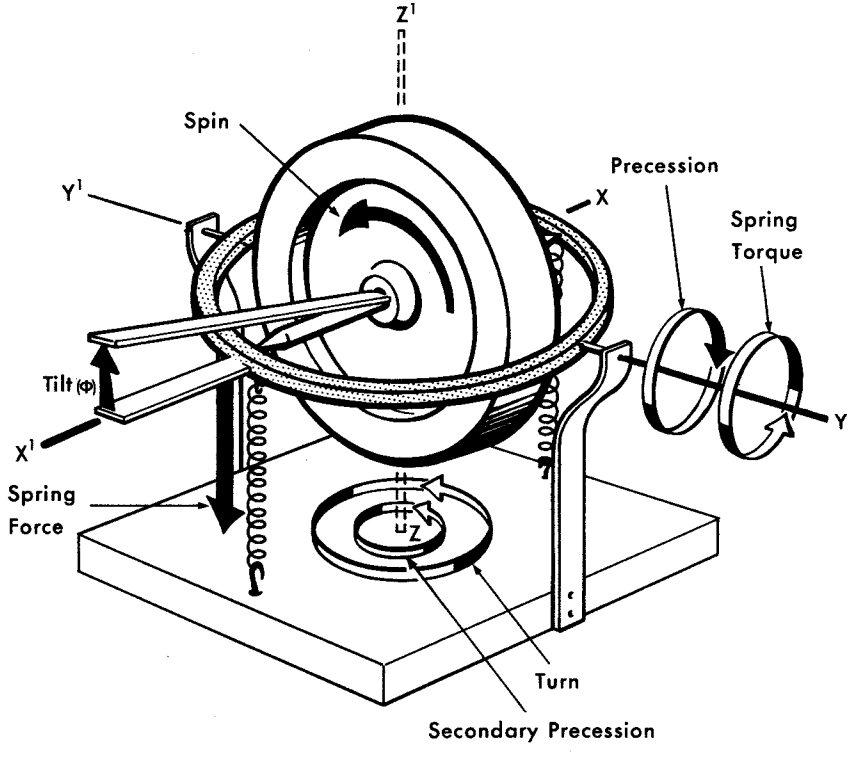


Fig 9 Rate Gyroscope

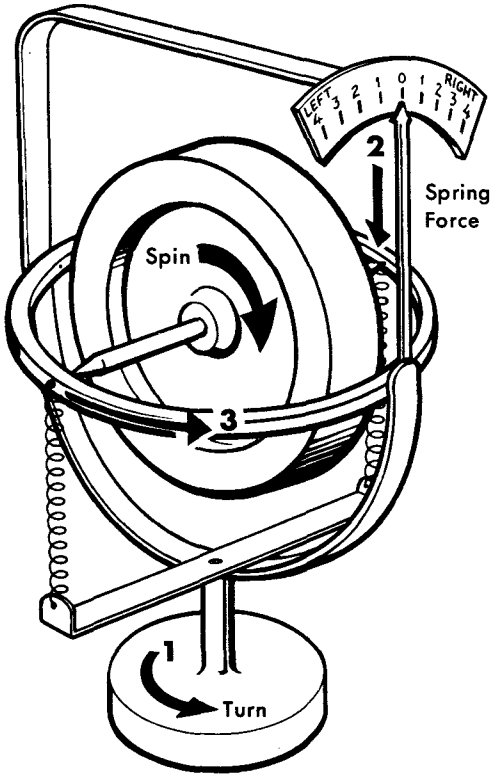


Fig 10 Rate of Turn Indicator

THE INTEGRATING GYROSCOPE

Principle of Operation

27. An integrating gyroscope is a single degree of freedom gyro using viscous restraint to damp the precessional rotation about the output axis. The integrating gyro is similar to the rate gyro except that the restraining springs are omitted and the only factor opposing gimbal rotation about the output axis is the viscosity of the fluid. Its main function is to detect turning about the input axis (YY^1 in Fig 11), by precessing about its output axis (ZZ^1 in Fig 11).

28. The integrating gyro was designed for use on inertial navigation stable platforms, where the requirement was for immediate and accurate detection of movement about three mutually perpendicular axes. Three integrating gyros are used, each performing its functions about one of the required axes. These functions could be carried out by displacement gyros, but the integrating gyro has certain advantages over the displacement type. These are:

a. A small input rate causes a large gimbal deflection (gimbal gain).

b. The gyro does not suffer from nutation.

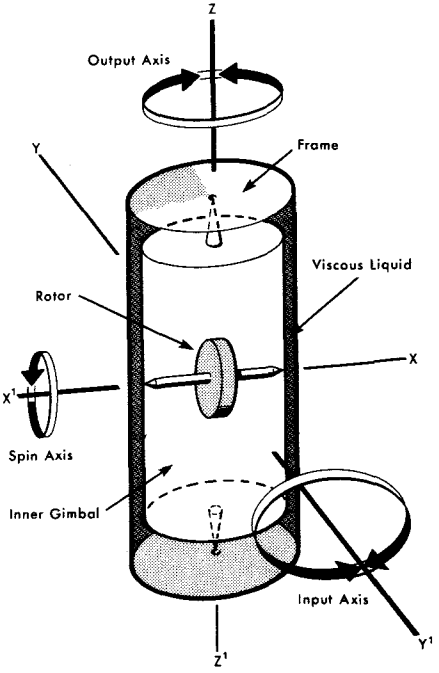


Fig 11 Simple Integrating Gyroscope

29. Fig 11 shows a simple integrating gyro. It is basically a can within which another can (the inner gimbal) is pivoted about its vertical (ZZ^1) axis. The outer can (frame) is filled with a viscous fluid which supports the weight of the inner gimbal so reducing bearing torques. The rotor is supported with its

spin (XX^1) axis across the inner gimbal. In a conventional non-floated gyro, ball bearings support the entire gimbal weight and define the output axis. In the floated integrating gyro the entire weight of the rotor and inner gimbal assembly is supported by the viscous liquid, thereby minimising frictional forces at the output (ZZ^1) axis pivot points. The gimbal output must however be defined and this is done by means of a pivot and jewel arrangement. By utilising this system for gimbal axis alignment, with fluid to provide support, the bearing friction is reduced to a very low figure.

30. The gyroscope action may now be considered. If the whole gyro in Fig 12 is turned at a steady rate about the input axis (YY^1), a torque is applied to the spin axis causing precession about the output axis (ZZ^1). The gimbal initially accelerates (precesses) to a turning rate such that the viscous restraint equals the applied torque. The gimbal then rotates at a steady rate about ZZ^1 proportional to the applied torque. The gyro output (an angle or voltage) is the summation of the amount of input turn derived from the rate and duration of turn and is therefore the integral of the rate input. (Note that the rate gyro discussed in paras 24 – 26 puts out a rate of turn only). The movement about the output axis may be made equal to, less than, or greater than

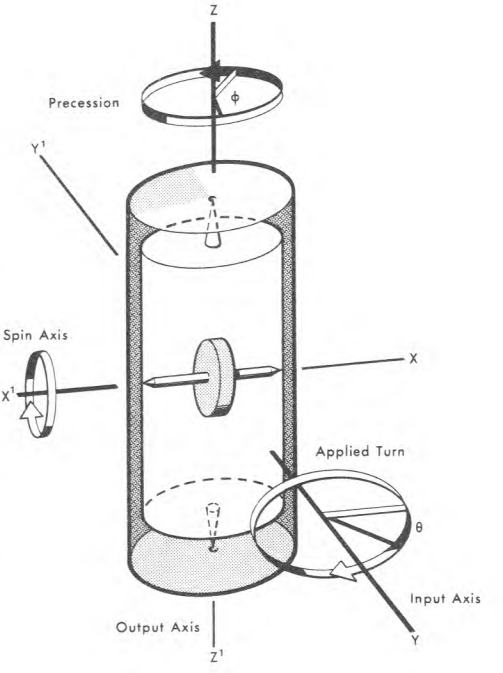


Fig 12 Function of Integrating Gyroscope

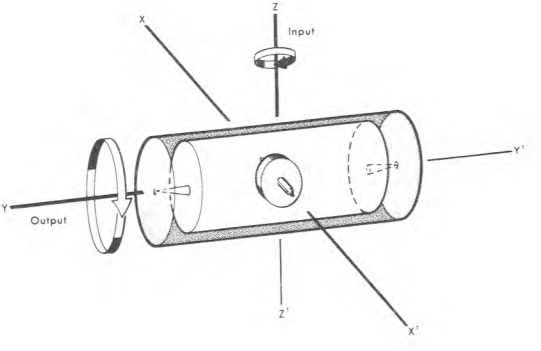


Fig 13 Azimuth Gyroscope

movements about the input axis by varying the viscosity of the damping fluid. By design the ratio between the output angle ($\varnothing$) and the input angle (θ) can be arranged to be of the order of 10 to 1. This increase in sensitivity is called gimbal gain.

31. A gyro mounted as in Fig 12 and sensing rotations about a horizontal input axis is known as a levelling gyro. Two levelling gyros are required to define a level plane. Most inertial platforms align the input axis of their levelling gyros with True North and East.

32. Motion around the third axis, the vertical axis, is measured by an azimuth gyro and the alignment of this is shown in Fig 13.

THE DISPLACEMENT GYROSCOPE

Definition

33. A displacement gyro is a two degree of freedom gyro. It can be modified for a particular task, but it always provides a fixed artificial datum about which angular displacement is measured.

Wander

34. Wander is defined as any movement of the spin axis away from the reference frame in which it is set.

35. **Causes of Wander.** Movement away from the required datum can be caused in two ways:

a. Imperfections in the gyro can cause the spin axis to physically move. These imperfections include such things as friction and unbalance. This type of wander is referred to as real wander since the spin axis is actually moving. Real wander is minimised by better engineering techniques.

b. A gyro defines direction with respect to inertial space, whilst the navigator requires earth directions. Two factors will cause a perfect gyro to lose its earth reference; earth rotation and vehicle movement or transport wander. It is therefore essential, in order to use a gyro to determine directions on earth, to correct it for apparent movement due to the fact that the earth rotates or that the gyro may be moving from one point on earth to another. This type of wander is called apparent wander.

36. **Drift and Topple.** It is more convenient to study wander by resolving it into two components:

a. *Drift,* which is defined as any movement of the spin axis in the horizontal plane around the vertical axis.

b. *Topple,* which is defined as any movement of the spin axis in the vertical plane around a horizontal axis.

37. Table 2 summarises the types of wander. From para 35 it should be apparent that the main concern when using a gyro must be to understand the effects of earth rotation and transport wander on a gyro.

Earth Rotation

38. In order to explain the effects of earth rotation on a gyro it is easier to consider a single degree of freedom gyro, since it has only one input and one output axis. The following explanation is based on a knowledge of rotational vector notation.

39. Consider a gyro input axis aligned with the earths spin axis; it would detect earth rate, Ωe, 15.04 degrees per hour. Consider the same gyro at a point A in Fig 14; it would be affected by earth rotation according to how its input axis was aligned, namely:

a. If its input axis were aligned with the earth's spin axis, it would still detect 15.04 degrees per hour.

b. *Azimuth Gyro.* If its input axis were aligned with the local vertical it would

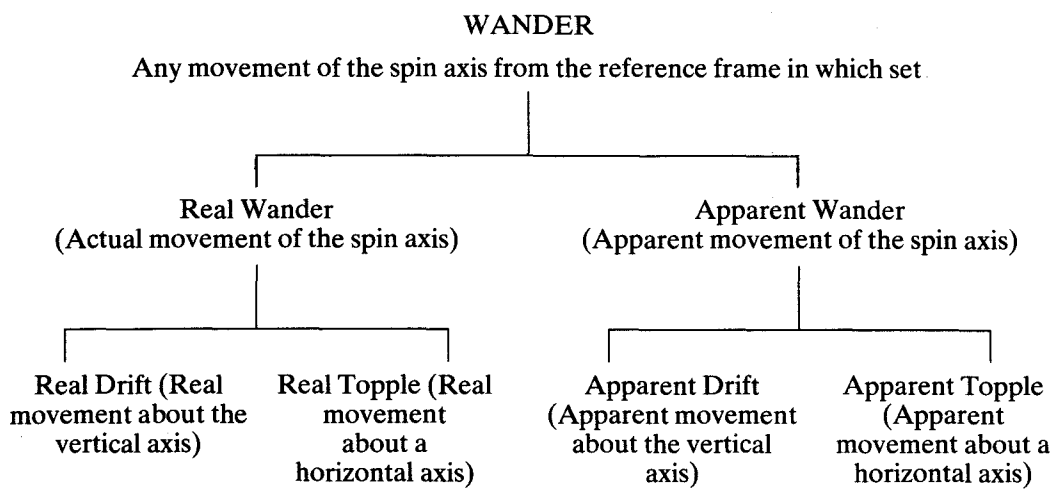


Table 2 Types of Wander

detect 15.04 × Sin Latitude (Sin λ) °/hr.
Note that by definition this is drift.

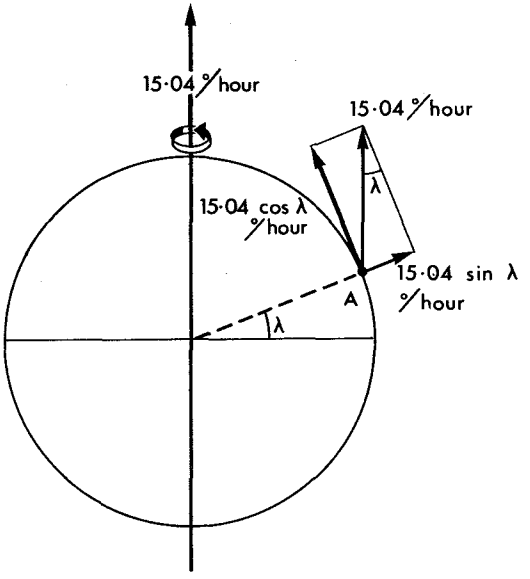


Fig 14 Components of Earth Rate

c. *Levelling Gyro.* If its input axis were aligned with local North, it would detect 15.04 × Cos Latitude (Cos λ) °/hr.
Note that by definition this is topple.

d. *Levelling Gyro.* Finally if the input axis were aligned with local East, that is, at right angles to the earth rotation vector, it would not detect any component of earth rotation.

Transport Wander

40. If an azimuth gyro spin axis is aligned with local North (ie the true meridian) at A in Fig 15 and the gyro is then transported to B, convergence of the meridians will make it appear that the gyro spin axis has drifted. This apparent drift is in addition to that caused by Earth rotation. The gyro has not in fact drifted; it is the direction of the True North which has changed. However, if the gyro is transported North-South, there is no change in the local meridian and therefore, no apparent drift. Similarly, as all meridians are parallel at the Equator, an East–West movement there produces no apparent drift.

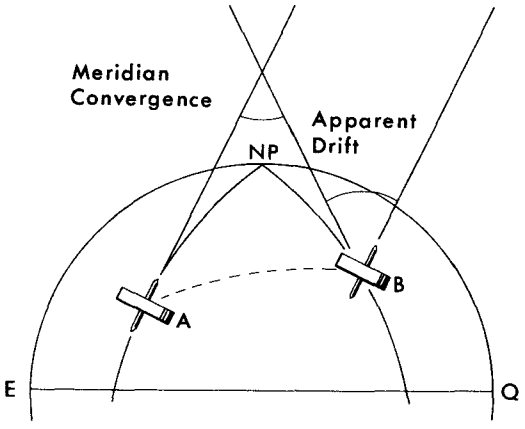


Fig 15 Apparent Drift

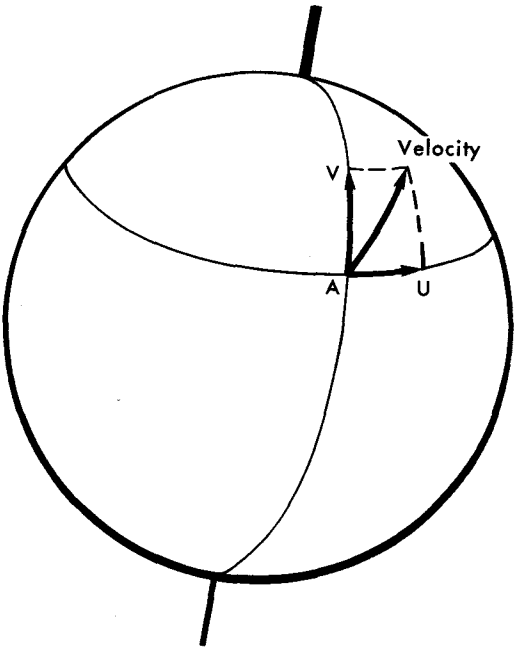


Fig 16 Transport Wander

Transport rate drift thus depends on the convergence of the meridians and the rate of crossing them; ie the East-West component of ground speed (U). The amount of convergence between two meridians is ch long × sin lat (see Annex A to AP 3456G Part 2 Sect 1 Ch 1 for proof). Any given value of U thus produces an increase in apparent gyro drift as latitude increases. The amount of drift due to transport rate may be found as follows:

Meridian conv. (C^o) = ch long × sin λ
Now ch long may be written as Eastings in nm divided by 60 (giving decimal quantities of degrees) and multiplied by secant λ

$$C^0 = \frac{\text{Eastings nm}}{60} \times \sec \lambda \times \sin \lambda$$

Now secant is the reciprocal of cosine, so

$$C^o = \frac{\text{Eastings nm}}{60} \times \frac{1}{\cos \lambda} \times \sin\lambda$$

and since

$$\frac{\sin}{\cos} = \tan,$$

$$\therefore C^o = \frac{\text{Eastings nm}}{60} \times \tan \lambda$$

Substituting U for Eastings:

$$C^o/\text{hr} = \frac{U \tan \lambda}{60} \ ^o/\text{hr}$$

or, as an approximation;

$$C^o/\text{hr} = \frac{U \tan \lambda}{R} \text{ Radians/hr}$$

where R = earth radius in nm.

41. Consider now two levelling gyros, whose input axes are North and East respectively, and whose output axes are vertical.

a. The East component of aircraft velocity in Fig 16 will be sensed by the North gyro as a torque of $\frac{U}{R}$ about its input axis. If the gyro is not corrected for this transport wander, it is said, by definition, to topple.

b. Similarly, due to the effect of aircraft velocity North, the East gyro will topple at the rate of $\frac{V}{R}$

Apparent Wander Table

42. All the equations derived in the study of earth rate and transport wander rate are summarized in Table 3. The units for earth rate can be degrees or radians, whilst for transport wander they are radians.

	Input Axis Alignment			
	Local North	Local East	Local Vertical	Correction Sign
Earth Rate	$\Omega e \cos \lambda$	Nil	$\Omega e \sin \lambda$	+
Transport Wander	$\frac{U}{R}$	$\frac{V}{R}$	$\frac{U}{R}$	+E —W
	Topple		Drift	

Ωe = Angular Velocity of the Earth
λ = Latitude. R = Earth's Radius
Earth Rate units degrees or radians per hour
Transport Wander units radians per hour

TABLE 3 Components of Drift and Topple—Earth Rate and Transport Wander Rate

43. **Correction Signs.** The correction signs of Table 3 apply only to the drift equations. As already stated these equations give corrections to apply to the gyro readings to obtain true directions. These correction signs will be reversed for the Southern Hemisphere.

Practical Corrections for Topple and Drift

44. If all the corrections of Table 3 were applied to three gyros with their input axes aligned to true North, true East and the local vertical, true directions would be defined continuously, and in effect the gyros would have been corrected for all apparent wander. However, these corrections make no allowance for the real drift of the gyro and consequently an error growth proportional to the magnitude of the real drift will exist. As a rough rule of thumb, an inertial platform employing gyros with real drift rates in the order of 0.01°/hr will have a system error growth of 1–2 nm/hr CEP.

45. Twin gyro platforms (TGPs), on the other hand, employ cheaper lower quality gyros whose drift rates are in the order of 0.1°/hr. If these real drift rates were not compensated for, system inaccuracies would be unacceptably large. For this reason, most TGP systems make use of the local gravity vector to define the level plane, thus compensating for both real and apparent drifts.

46. Specifically, gyro drifts may be corrected in the following ways:

a. *Topple.* Topple is normally corrected for in gyros by the use of either gravity switches (see Figs 17a, b and c), or by case levelling devices (see Figs 18a, b and c). These devices sense movement away from the vertical, and send appropriate signals to a torque motor until the vertical is re-established. The levelling accuracy of these methods is a approximately 1°.

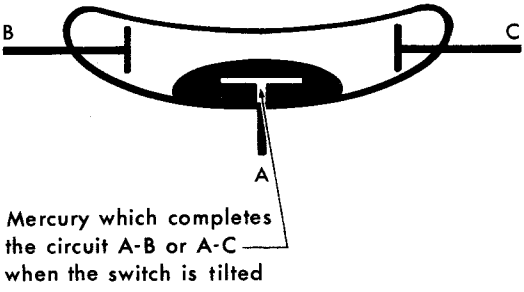


Fig 17a Gravity Sensitive Switch

b. *Drift.* Drift corrections can be achieved by:

(1). Calculating corrections using Table 3 and applying them to the gyro reading.

(2). Applying a fixed torque to the gyro so that it precesses at a rate equal to the earth rate for a particular latitude. Drift will be fully corrected for only at the selected latitude.

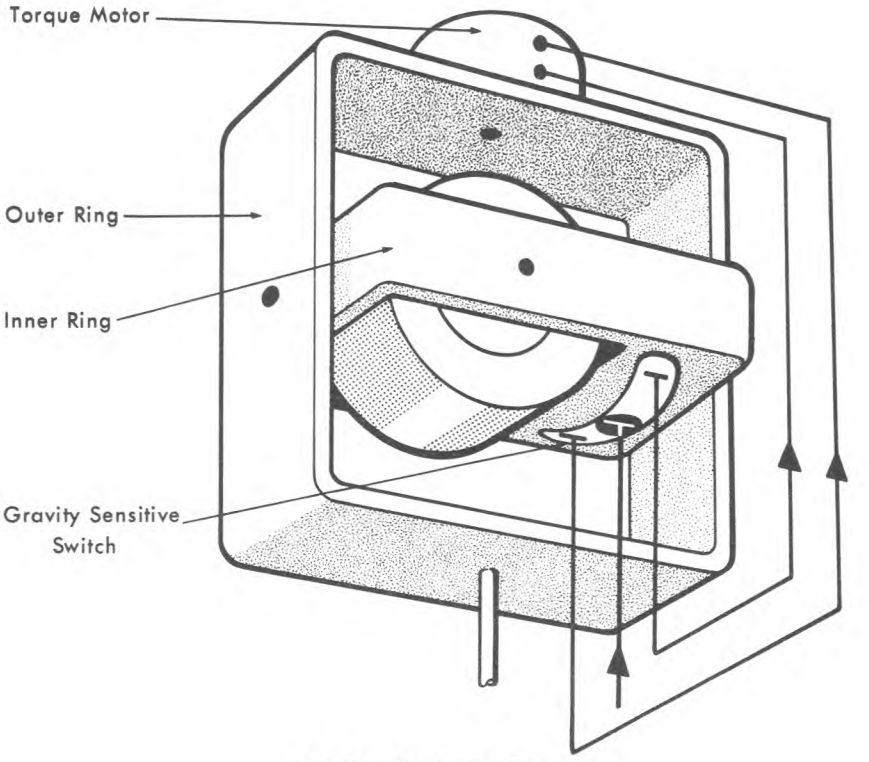


Fig 17b Gravity Levelling

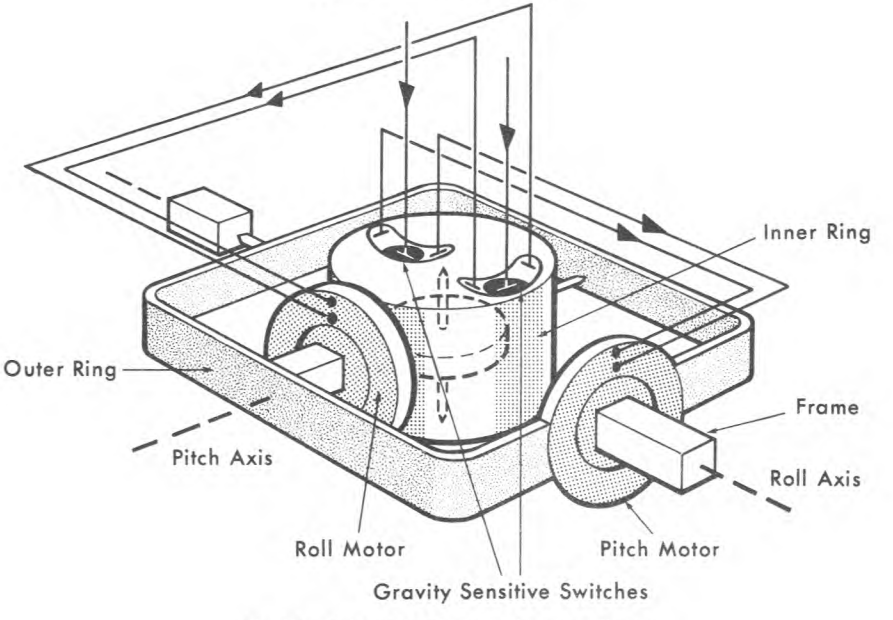


Fig 17c Schematic Electrical Erection System

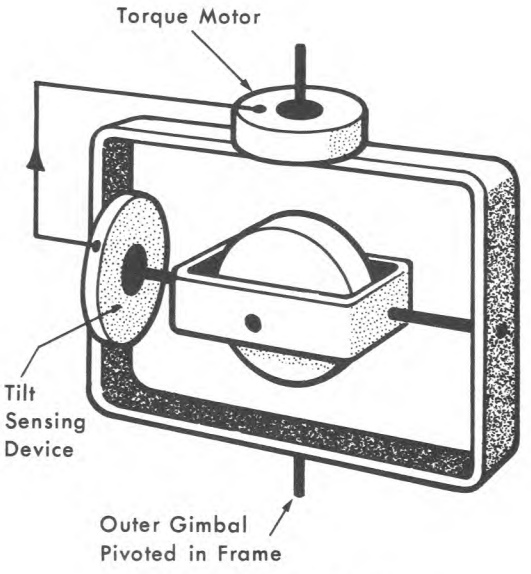


Fig 18a Electrical Case Levelling

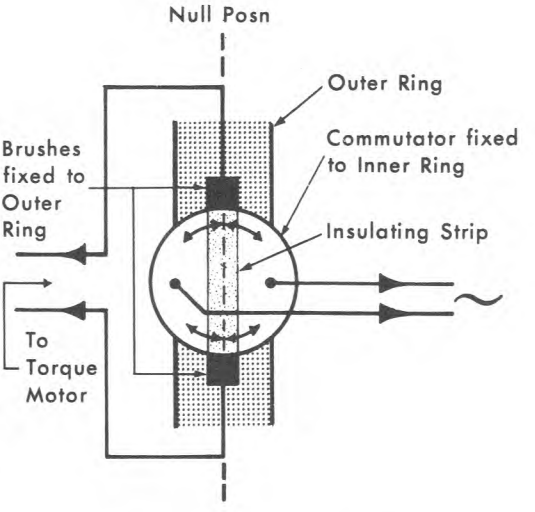


Fig 18c Synchro Sensing Device

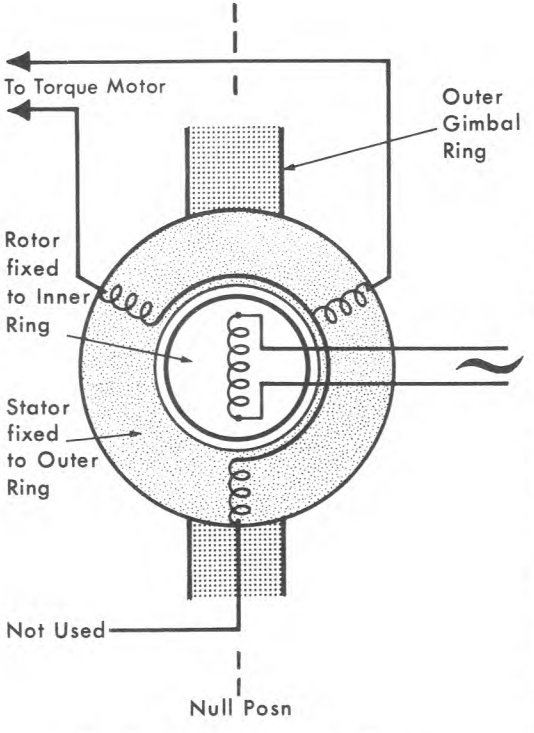


Fig 18b Commutator Sensing Device

(3.) Applying variable torques, using the same approach as in (2) above, but being able to vary the torque according to the latitude. These azimuth drift corrections make no allowance for real drift, which can only be limited by couupling the azimuth gyro to a flux valve.

47. To complete this study of the displacement gyro, it remains to mention a limitation and an error peculiar to this type of gyro, gimbal lock and gimbal error.

Gimbal Lock

48. Gimbal lock occurs when the gimbal orientation is such that the spin axis becomes coincident with an axis of freedom. Effectively the gyro has lost one of its degrees of freedom, and any attempted movement about the lost axis will result in real wander. This is often referred to as toppling, although drift is also present.

Gimbal Error

49. Gimbal error occurs when there is a misalignment between the aircrafts aerody-

namic axes and the navigation system axes. Most modern systems automatically compensate for these errors.

GYRO SUSPENSION METHODS

Introduction

50. The necessity for an accurate azimuth in modern navigation and attack systems and the development of inertial navigation have led to a requirement for low drift gyros. To achieve a low drift rate friction at the gimbal bearings must be reduced as much as possible.

51. The commonest method of suspending gyro wheels and gimbals is by means of prepacked ball bearings. Prepacking is necessary to ensure that no mass unbalance occurs, but it increases friction. The friction in the gyro wheel bearings is not critical, because the wheel is usually driven by a synchronous motor which ensures a constant rpm. However the friction in the gimbal bearings is important, because it will give rise to unwanted torques when attempting to move the gyro about the gimbal axes. Because of this a great deal of attention has been devoted to studying other methods of suspending gyro gimbals.

52. In the floated gyro the floated inner can takes the weight of the gyro off the gimbal bearings, thus reducing unwanted torques. Other methods of suspending gyros are described in Part 3, Sect 4, Chap 2, para 16.

Printed in England for Her Majesty's Stationery Office by Nelson Brothers, Printers, Chorley

CHAPTER 2

MASTER REFERENCE GYRO, MARK 1

CONTENTS

	Paras.
Introduction	1—2
General Description	3—8
Stabilization by Vertical Gyro	9—16
Operation of Azimuth Gyro	17—19
Performance and Accuracy	20—23

Introduction

1. The master reference gyro (M.R.G.) is the central unit of the flight instrument dynamic reference system fitted in several aircraft. It is capable of feeding practically continuous flight attitude and heading information to indicating instruments and other equipment requiring such information for their operation. The M.R.G. measures aircraft movement in the pitching, rolling and yawing planes, and indicates the movement by electrical outputs. Thus a single unit may replace the displacement gyroscopes used in direction indicators, artificial horizons, weapons, and radar systems.

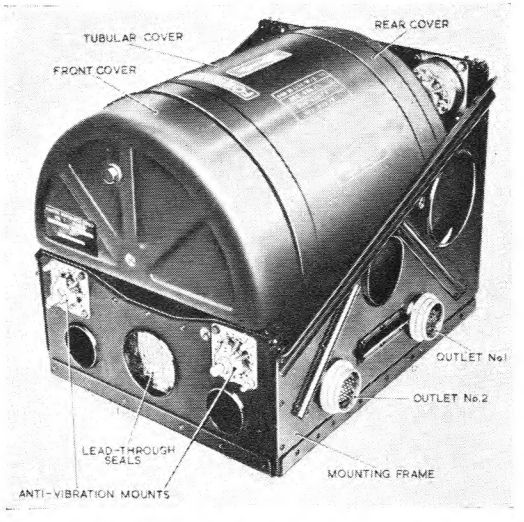


Fig. 1. M.R.G. Mk. 1, Type B, and Mounting Frame

2. The unit described in this chapter is the M.R.G. Mk. 1, Type B. Also in use are Types D and E which are almost identical in construction and operation to the Type B. The essential differences are in the types of electrical data transmission devices used. These differences are such as to render the various types of M.R.G. non-interchangeable within a system. The M.R.G. Mk. 2 is described in Chapter 4 of this Section.

General Description

3. The M.R.G. Mk. 1 consists of a sealed gyro unit and a mounting frame (Fig. 1). The sealed unit consists basically of a servo-controlled, gyro-stabilized platform, embodying an azimuth gyro and a vertical gyro. The latter maintains the platform in the horizontal plane, thus obviating gimballing errors in the azimuth gyro, and provides a datum from which angular movements of the aircraft in pitch and roll may be measured.

4. The vertical gyro spin axis is tied to the Earth's gravitational centre by the monitoring action of gravity seeking mercury switches mounted on the platform. The azimuth gyro spin axis is maintained relative to magnetic North by monitoring signals derived from a fluxgate type detector unit.

5. The vertical and azimuth gyro monitoring systems include pendulous devices which are subject to acceleration errors. Provision is therefore made to suppress the monitoring circuits when accelerations reach a predetermined value.

6. The platform assembly is pivoted fore-and-aft in a chassis. The chassis is enclosed in a hermetically sealed case which is filled with a mixture of helium and nitrogen, this arrangement offering protection against dampness and assisting in heat dissipation, as well as permitting operation in a rarefied atmosphere.

7. To ensure that the M.R.G. fore-and-aft axis is parallel to that of the airframe, the mounting frame is designed to fit into a mounting tray which is rigidly attached to the aircraft.

8. Electrical connecting leads are brought out from the sealed case via lead-through glass seals in the front cover, and terminate in the mounting frame. Essential services are duplicated as a safety measure. The necessary power supplies are nominally 115V, 400 c/s, 3—phase a.c. and 28V d.c.

Stabilization by Vertical Gyro

9. Fig. 2 shows the M.R.G. in schematic form. The assembly consists of an outer platform pivoted in the chassis about the X-X (fore-and-aft) axis, and an inner platform, consisting of two forked members which form one rigid-element. The inner platform is pivoted in ball-races inside the outer and is free to rotate about the Y-Y (athwartships) axis. The two platforms thus form a gimbal assembly with freedom about the roll and pitch axes, but fixed in azimuth. Fig. 3 shows a sectional view of the assembly.

10. When the aircraft banks, the rigidly attached chassis moves with it, but the inner and outer platforms remain in the horizontal plane under the influence of the vertical gyro. The angular displacement between the outer platform and the chassis is equal to the aircraft bank angle. Similarly, angular displacement between the inner and outer platforms is equal to the aircraft pitch angle.

11. **Pitching Control.** A potentiometer, VR3, is positioned at the outer gimbal ring bearing of the vertical gyro, with a movable pick-off fixed to the outer gimbal ring. When the aircraft pitches, the outer gimbal remains in the plane determined by the rotor, but the inner platform starts to move with the aircraft. Thus the resistor moves with the aircraft across the pick-off, and an indication of pitch movement is fed from the potentiometer to the servomotor SM1. The latter is thus energized in such a way as to turn the inner platform until the pick-off is in the centre position. At this point the servomotor is no longer energized, and the inner platform is slaved to the vertical gyro outer gimbal ring.

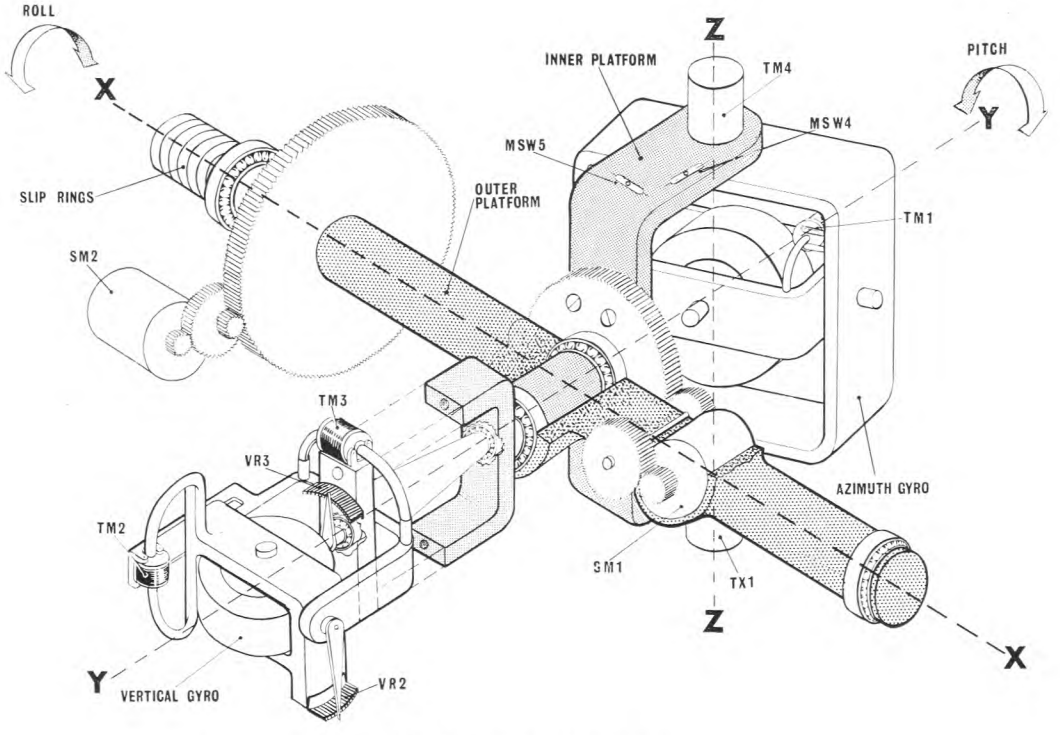


Fig. 2. M.R.G.—Schematic

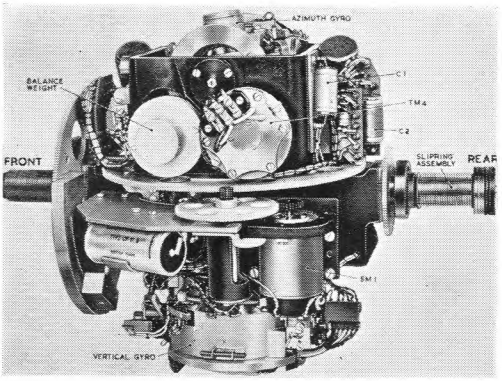


Fig. 3. Platform Assembly (Top View)

12. **Rolling Control.** A roll potentiometer, VR2, is positioned on the inner gimbal ring bearing. When the aircraft banks the inner gimbal ring remains in the plane of the spin axis, but the inner platform and outer gimbal ring start to roll with the outer platform. Thus the pick-off moves across the resistor, and an indication of roll movement is fed from the potentiometer to the servomotor SM2. The latter is thus energized in such a way as to turn the outer platform, and consequently the inner platform and outer gimbal ring, until the pick-off is in the centre position. At this point the servomotor is no longer energized and the outer platform is slaved to the vertical gyro inner ring.

13. **Data Transmission.** Pitch and roll information is transmitted from the servomotors, via bevel gears and synchros, to those positions in the aircraft where it is required.

14. **Erection in Pitch.** A pitch mercury switch, MSW 5, is mounted in the aircraft fore-and-aft axis on the inner platform. Any deviation of the platform from the horizontal, about the pitch axis, actuates the mercury switch which then energizes the control winding of the pitch torque motor TM2. The control winding is mounted on the outer gimbal ring of the vertical gyro, and the armature is mounted on the inner ring; the torque exerted on the inner gimbal ring when the torque motor is energized causes the outer ring to precess in such a direction as to restore the spin axis to the earth's vertical. The inner platform follows the outer ring, as described in para. 11, until the mercury switch reaches its null position, at which stage the process ceases.

15. **Erection in Roll.** Any deviation of the outer platform and therefore the inner platform from the horizontal about the roll axis, actuates the roll mercury switch MSW4 mounted athwartships on the inner platform. This energizes the control winding of the roll torque motor TM3 and causes precession of the inner gimbal ring. The outer platform follows the inner gimbal ring as described in para. 12, and the inner platform, carried by the outer platform about the bank axis, is restored to the horizontal; at this point the mercury switch reaches its null position and the process ceases.

16. **False Erection.** As stated in para. 5, provision is made to suppress the erection signal at predetermined error rates. However at lower rates false erection does occur and to keep these errors to a minimum, normal erection is limited to a comparatively low rate ($\simeq 3°$/min); the vertical gyro thus acts as a smoothing device for the information provided by the mercury switches. There is provision for selection of a fast erection rate (9° to 17° per minute when the gyro is running at full speed).

Operation of Azimuth Gyro

17. The azimuth gyro outer ring is free to take up any position in azimuth, and the spin axis is maintained at right-angles to the outer ring by an anti-topple motor, TM4, geared to the outer ring. The motor is energized by an anti-topple switch, the contact of which is mounted in the axis of the inner ring. The switch detects any tilt of the gyro spin axis greater than two degrees from the orthogonal, and causes the anti-topple motor to produce a torque on the outer ring which results in precession of the inner ring in such a direction as to keep the spin axis within two degrees of the orthogonal position. Since the outer ring is maintained in the vertical by the stabilized platform, gimballing errors in the azimuth gyro are eliminated.

18. Attached to the spindle of the outer ring at its lower end is the rotor of the azimuth synchro transmitter TX1. The rotor defines the position of the spin axis. The stator of TX1 is located in the inner platform, and defines the fore-and-aft axis of the aircraft. Any relative movement in azimuth between the spin axis of the gyro and the fore-and-aft axis of the aircraft is transmitted by TX1 to the heading indicator of the Gyro-magnetic Compass Mark 5—Fighter Type (G 5 F T).

19. **Azimuth Monitoring.** The output from the compass detector unit is compared with the compass heading indicator. If there is any misalignment an error signal energizes the control winding of the azimuth monitor torque motor TM1, mounted on the azimuth gyro outer ring. The armature of TM1 is mounted on the inner ring and the resultant torque between the control winding and armature causes precession of the outer ring in such a direction as to reduce the misalignment error. The precession rate of the outer ring is comparatively slow, and the azimuth gyro thus smooths the information from the detector unit.

Performance and Accuracy

20. The M.R.G. Mk. 1 should give an accuracy of $\pm 1\frac{1}{2}°$ in pitch and roll in straight unaccelerated flight, but this is degraded in accelerated and turning flight. This is particularly true of the bank error in slow turns at bank angles less than 10°. The free wander rate of both gyros is of the order of 15°/hr.

21. As previously stated the torque motors slaving the azimuth and vertical gyros are cut-out under certain conditions; these are:—

(*a*) When a sustained fore-and-aft acceleration or deceleration in excess of 0.083g is detected.

(*b*) When pitch angles exceed 20°.

(*c*) When bank angles exceed 10°.

Thus, when the aircraft executes any manoeuvre outside these limits, attitude and heading continue to be defined by free gyros which are subject to random drift errors. The total error introduced is equal to the wander rate x time.

22. The M.R.G. gives full attitude reference except when the pitch angle is within $\pm 2°$ of the vertical, in which configuration the platforms are in a state of gimbal lock. Once the pitch angle has reduced to less than $\pm 88°$, the platform is automatically re-orientated.

23. A thermal timing switch within the M.R.G. ensures that the equipment is operational within 20 ± 3 seconds from switch-on.

CHAPTER 3

C.L.11 DIRECTIONAL GYRO

CONTENTS

	Paras.
Introduction	1– 3
General Description	4–11
Rotorace Bearings...	12–14
Levelling System	15–16
Compensation for Apparent Drift	17–18
Advantages...	19

Introduction

1. The Sperry C.L.11, or Rotorace Gyro, is a directional gyroscope designed to provide a highly stable heading reference. The use of Rotorace (trade name) bearings on the inner gimbal axis results in the instrument having a random drift rate of about 1° per hour.

2. The gyro is normally used as an integral part of a gyro-magnetic compass system, where it is monitored by the magnetic element, but it may be used as a pure D.G. It is also adapted for use in more advanced attitude and heading reference systems.

3. This chapter deals only with the C.L.11 gyroscope. Its application to particular heading reference systems is treated in the chapters on those systems.

General Description

4. The C.L.11 gyro consists of two major assemblies, as shown in Fig. 1:—

(*a*) A sealed case assembly containing the gyro, the gimbal mechanism and the synchro transmitters.

(*b*) A base assembly which provides an anti-vibration mounting for the case, and houses certain electrical components and connectors.

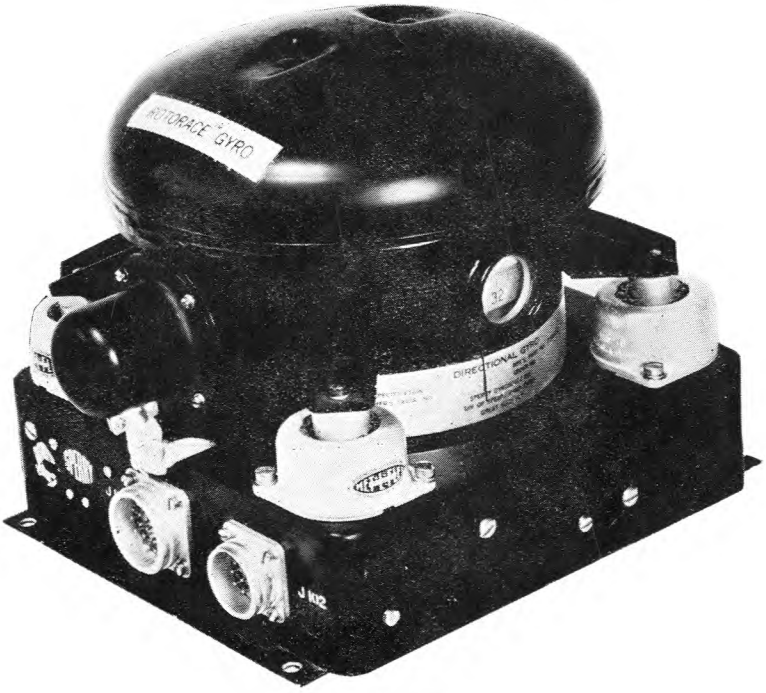


Fig. 1. C.L.11 Gyro

5. **Gyro.** The spinning element of the gyro is the spherical rotor of a 3-phase squirrel cage induction motor. It rotates at approximately 22,500 r.p.m. The rotor is supported by conventional ball races in a circular band which is part of the inner gimbal. The stator of the induction motor is also housed in this band. The whole sub-assembly is covered by two hemispherical bowls, which are filled with helium gas and hermetically sealed. All electrical connections are brought out through slip rings.

6. **Inner Gimbal.** The inner gimbal ring is pivoted on two Rotorace bearings within the outer gimbal and at right-angles to the gyro spin axis, stops giving $\pm 87^\circ$ of freedom about a horizontal axis. A levelling torque motor system maintains the inner gimbal, and thus the gyro spin axis, in the horizontal plane.

7. **Outer Gimbal.** The outer gimbal is supported in the vertical axis by two conventional ball races. The upper ball race is located in the synchro stator assembly at the top of the case and the lower ball race in the bottom of the case. This gimbal has complete freedom in azimuth. A bearing scale, which may be viewed through an inspection window on the side of the case, is mounted on the outer gimbal. Arbitrary heading information for test purposes may be read against a vernier mounted on the synchro stator assembly. Fig. 2 illustrates the mechanism with the case removed.

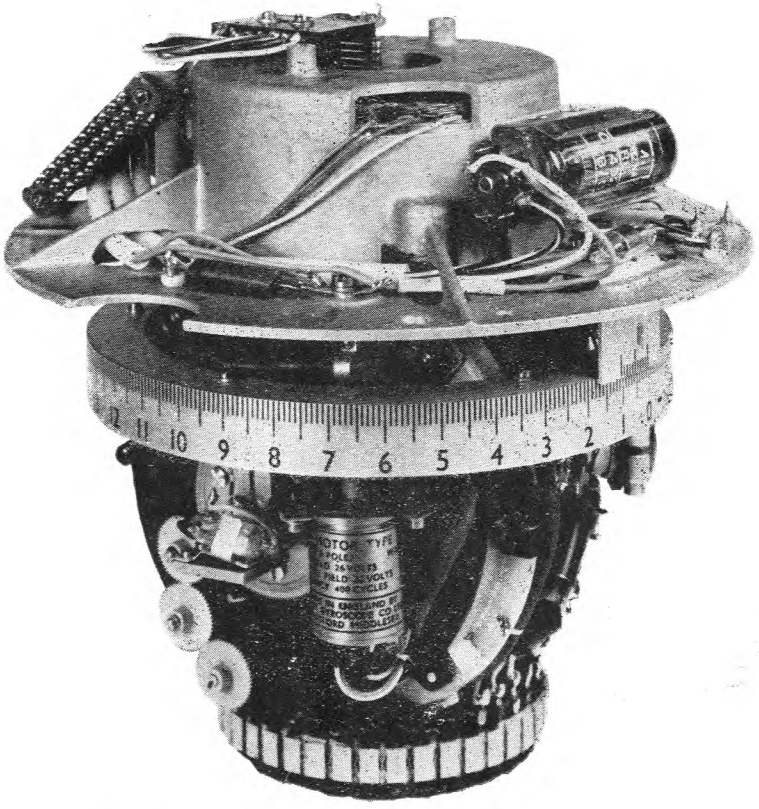


Fig. 2. C.L.11 Gyro—Gimbal and Synchro Stator Assembly

8. **Synchro Control Transmitters.** There are two synchro control transmitters mounted together; the stators are enclosed in the synchro stator assembly which is attached to the case of the instrument, while the rotors are mounted on the outer gimbal axis. One of the transmitters is used for autopilot monitoring and the other, known as the servo synchro transmitter, passes signals proportional to heading to the stator coils of the synchro control transformer in the master indicator of the compass system. The autopilot data transmitter rotor has an a.c. supply fed from the autopilot, while the servo synchro transmitter rotor is normally energized from the compass a.c. supply.

9. **Slaving Torque Motor.** When the gyro is being used in a gyro-magnetic compass system its spin axis is maintained in a constant angular relationship to the magnetic meridian, *i.e.* it is slaved to the meridian. This is achieved by a slaving torque motor which precesses the gyro in azimuth by applying a torque about the inner gimbal axis. The spin axis is not actually aligned with the meridian. Initially the gyro is allowed to assume any direction in relation to the magnetic meridian. Thereafter the angular difference between the two is maintained by precession of the gyro to counteract drift. The two-phase winding of the slaving torque motor is carried on the outer gimbal and its squirrel cage rotor is mounted on the inner gimbal shaft. The control field winding is energized by signals from the magnetic element of the compass system.

10. **Levelling Torque Motor.** The levelling torque motor system, which is described in more detail in paragraphs 15 and 16, serves to maintain the gyro axis in the horizontal. A liquid level switch is used to define the horizontal position.

11. **Outer Case.** All the components described above are enclosed in an outer case which is attached by four lugs to anti-vibration mountings on the base assembly (*see* Fig. 1). The case is filled with a mixture of helium and nitrogen gases at a pressure of 20 in. of mercury and hermetically sealed. Connections are brought out to the base assembly which has two electrical connectors on the outside.

Rotorace Bearings

12. The principal source of random drift in ball-bearing gyros is bearing friction due to the drag of the viscous lubricant, the configuration of the balls, and imperfections in the surfaces of races, balls and pivots. This friction has a much greater effect in bearings which have small angular movements during operation, gimbal suspensions for example. The average torque exerted by a bearing when it is rotated follows a curve of the form shown in Fig. 3. This torque appears as friction opposing the motion of the mechanism. As the graph shows, a high torque is required to start a bearing in motion, *i.e.* to overcome starting friction, but the torque falls rapidly as the bearing speed increases. Over the section A-B of the graph the torque required is fairly uniform, but as the speed increases further the torque increases again owing to viscous and dynamic forces. The torque levels for a given bearing will vary according to temperature, lubricant properties, load, state of wear and cleanliness.

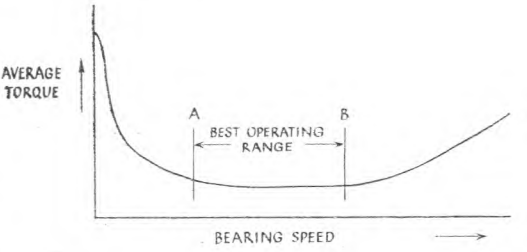


Fig. 3.
Graph of Average Torque against Bearing Speed

13. The Rotorace principle is that of rotating the ball races of the gimbal bearings in opposite directions, and reversing the direction of rotation at regular intervals. This rotation uniformly distributes about the axis of rotation the various frictional forces that arise in the bearings, and the speed of rotation is such that the bearings operate on the low friction section A-B of the graph in Fig. 3.

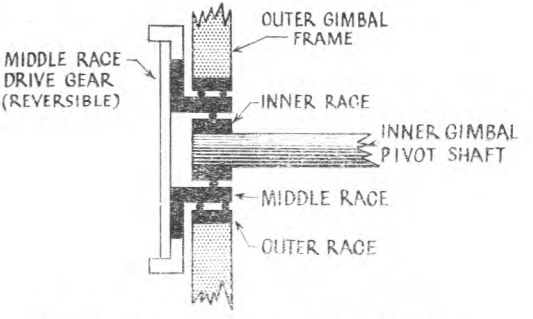


Fig. 4. Rotorace Bearing—Cross Section

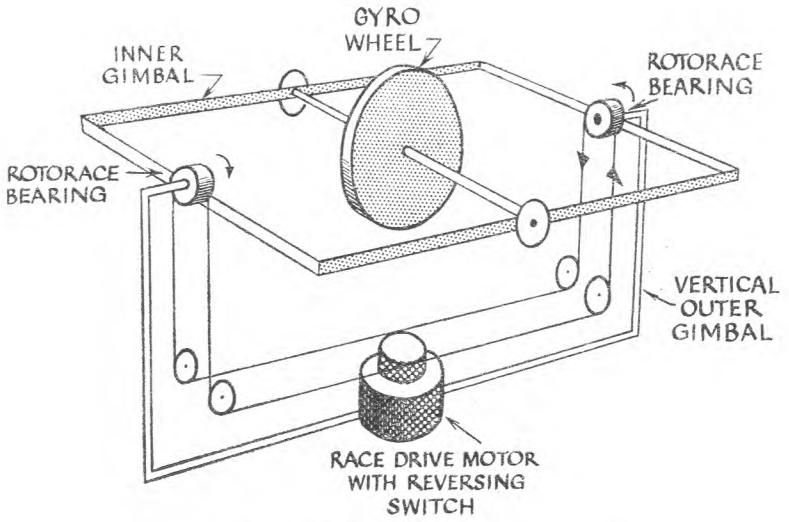


Fig. 5. Rotorace Principle—Schematic

14. The Rotorace bearing itself is shown diagrammatically in Fig. 4. It is a composite bearing consisting of an inner race, a middle race and an outer race. Two sets of balls separate the three races. The middle race is the one which is constantly rotated by a race drive motor. The inner gimbal of the gyro is supported by two such Rotorace bearings, and their middle races are rotated in opposite directions with the effect that a great deal of the frictional torques which would cause drift are cancelled out. The direction of rotation is regularly changed to eliminate any drift which might be introduced through mis-match of the two bearings. The speed of rotation of the bearings is not critical but the number of revolutions in each direction must be equal. Current models have a race speed of about 120 r.p.m. Fig. 5 is a schematic diagram which illustrates the Rotorace principle.

Levelling System

15. The spin axis of the gyro is maintained horizontal to within a half-degree by a levelling torque motor controlled by a liquid level switch. This motor produces torque about the outer gimbal axis which results in precession about a horizontal axis. The motor, which is located at the bottom of the case, has a split control field winding, the two halves of the winding being in opposition to each other. The reaction of these fields with the fixed field produces torques in opposite directions. One half of the control field is continuously energized, producing a constant torque in one direction. The other half is only energized when the liquid level switch is made, but it produces twice the torque given by the continuously energized section of the control field.

16. The liquid level switch, containing an electrolytic fluid, is mounted on a bracket attached to the inner gimbal shaft. When the gyro spin axis is displaced from the horizontal in one direction a torque, due to the continuously energized section of the control field, is exerted on the outer gimbal. This precesses the gyro to bring the spin axis back to the horizontal. When it is displaced in the other direction the liquid level switch is made and the other more powerful section of the control field is energized. The resultant torque precesses the gyro again to bring the spin axis back to the horizontal. As there is always a torque exerted in one direction or the other on the outer gimbal, the gyro continuously hunts about the horizontal. The amplitude of these oscillations is very small owing to the low precession rate of approximately 3° per minute.

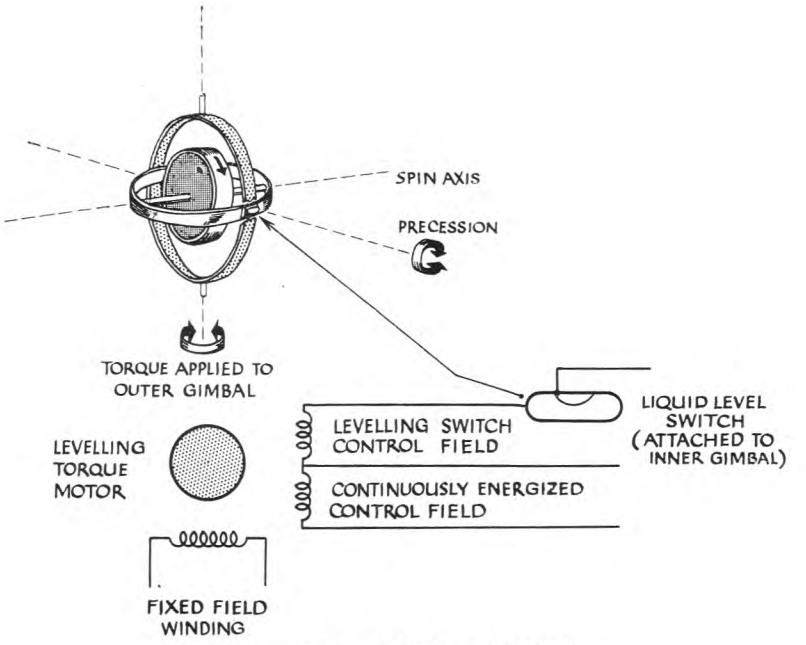


Fig. 6. Levelling System—Schematic

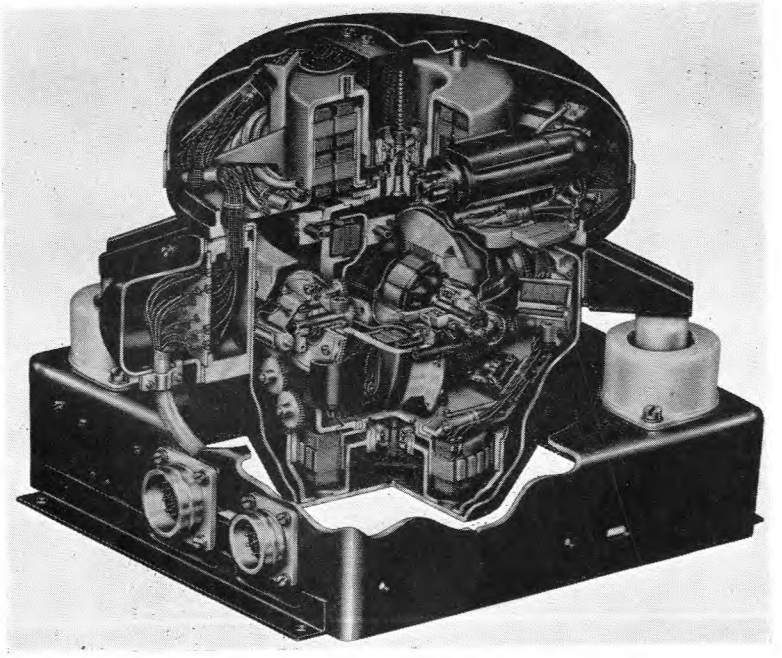


Fig. 7. C.L.11 Gyro—Sectional View

Compensation for Apparent Drift

17. Compensation for the apparent drift of the gyro due to the rotation of the earth is made by the application of a controlled torque to the inner gimbal axis. This torque, the value of which is proportional to the sine of the latitude, causes the gyro to precess in azimuth to cancel out the apparent drift.

18. Two small d.c. electromagnets are built into the outer gimbal, one on either side of the inner gimbal bearing. As the gyro rotor turns through the magnetic field of either coil, eddy currents are induced in the rotor which result in an induced torque on the inner gimbal. The sense of the torque depends upon which electromagnet is energized. One coil, marked N, is used in the northern hemisphere and the other, marked S, in the southern hemisphere. The current to the electromagnets depends upon a N/S switch selection and a potentiometer setting. This is normally catered for by the provision of a control box with a switch and a dial calibrated in latitude for in-flight use, or it may be set on the base assembly to suit the latitude of operation of the aircraft.

Advantages

19. The principal advantage of the C.L.11 gyro is its low drift rate, which makes it a very stable heading reference. The gyro is also relatively cheap and simple for the increase in performance which it gives over more conventional gyros. In addition it does not require the precise temperature control which other advanced types of gyro need.

CHAPTER 4

SPERRY TWIN GYRO PLATFORM (M.R.G. MK. 2)

CONTENTS

	Paras.
Introduction	1
Gyro Reference Unit	2– 3
Electronic Unit	4
Levelling Control	5– 6
Pitch and Roll Outputs	7– 8
Azimuth Output	9–10
Orthogonality	11–12
Erection Cycle	13

Introduction

1. The Sperry Twin Gyro Platform is a gyroscopic attitude and heading reference which can be used with various flight instrument and compass systems. The main components are the gyro reference unit, an all-attitude flight reference unit consisting basically of two directional gyros mounted in servoed pitch and roll gimbals; and the electronic unit, whose purpose is to energize and control the gyro reference unit and to repeat and apply corrections to the platform outputs. The two units together are also known as the M.R.G. Mk. 2 (Master Reference Gyro Mk. 2). This chapter is concerned with the instrument only as a basic reference and not with any particular application.

Gyro Reference Unit

2. The gyro reference unit consists of two identical directional gyros mounted as shown diagrammatically in Fig. 1. The gimbals are in the form of successive spherical shells, minimum amounts of which are removed for access. The outer roll gimbal is pivoted inside the case. The pitch gimbal is designed as two separate shells one around each gyro, but they are mechanically linked to give the effect of one common gimbal. Both the pitch and roll gimbals are servoed by error signals from the gyro pick-offs and the outputs of pitch and roll are derived from the attitude of the gimbals with respect to the case, which is fixed to the aircraft and aligned accurately with the fore-and-aft axis. The case itself is filled with helium and nitrogen and hermetically sealed, electrical signals being transmitted through connectors on top of the outer cylinder. Fig. 2 shows one of the gyros inside its pitch gimbal.

3. The gyros, which are the spherical rotors of two 3-phase induction motors, are each mounted within their respective vertical and inner gimbals. This arrangement within the pitch and roll gimbals makes the system an all-attitude stable reference. The spin axes are kept horizontal by means of torque motors actuated by liquid levels. Each gyro has $\pm 72°$ of freedom about the inner gimbal axis and complete freedom in azimuth about the vertical gimbal axis. The two vertical gimbals, and in consequence the gyro spin axes, are maintained orthogonal to each other. Any departure from this orthogonal condition due to gyro drift is detected and immediately rectified by precession of the gyros. Rotorace bearings are used on the sensitive axes of the gyros, giving a low random drift rate in both azimuth and vertical as well as greater tolerance to vibration and shock. The Rotorace principle is described in Chapter 3 of this Section, which deals with the Sperry C.L.11 Gyro. The maximum random drift rate of the platform under normal operating temperatures is 0.25 degrees per hour.

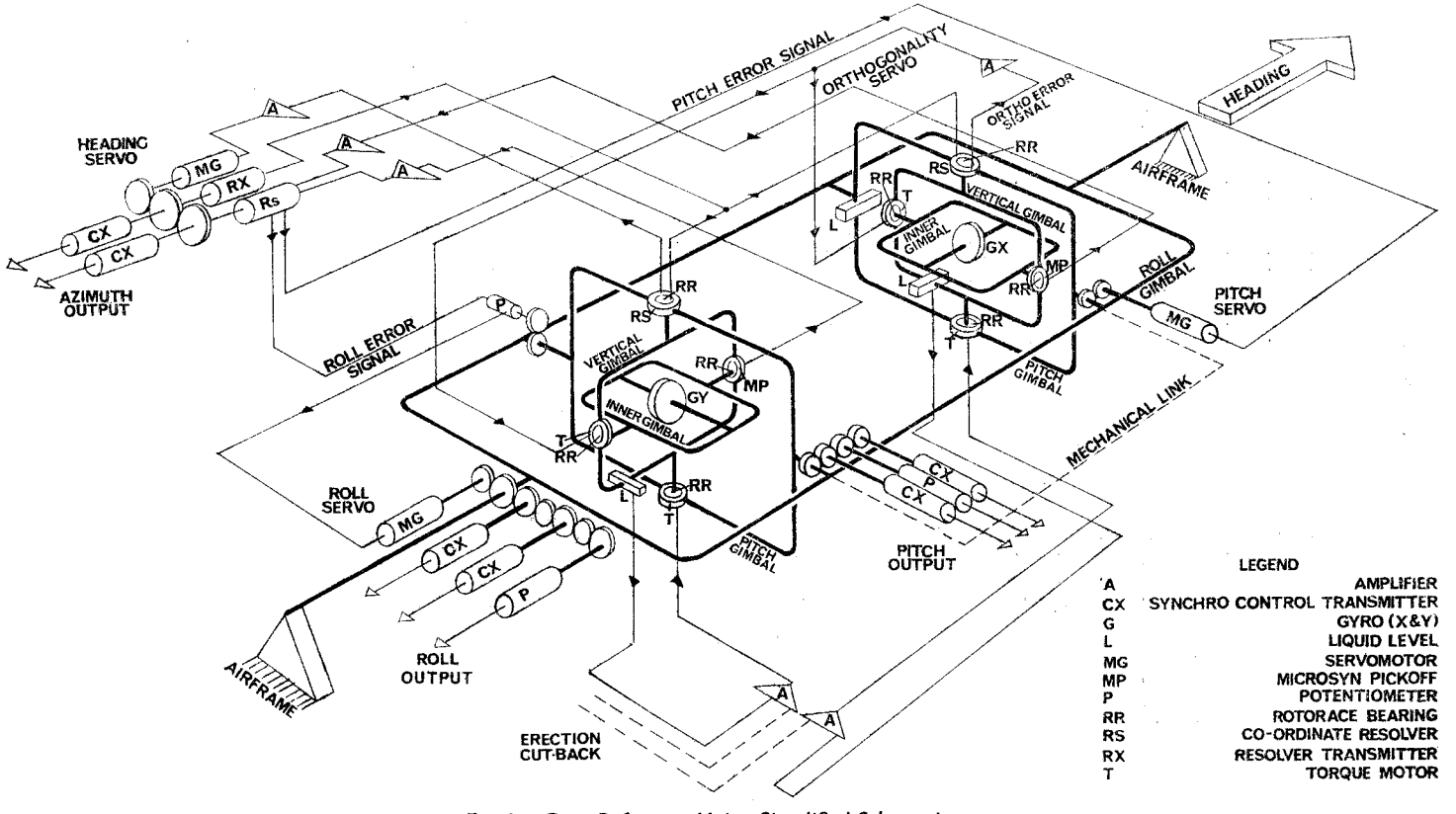


Fig. 1. Gyro Reference Unit—Simplified Schematic

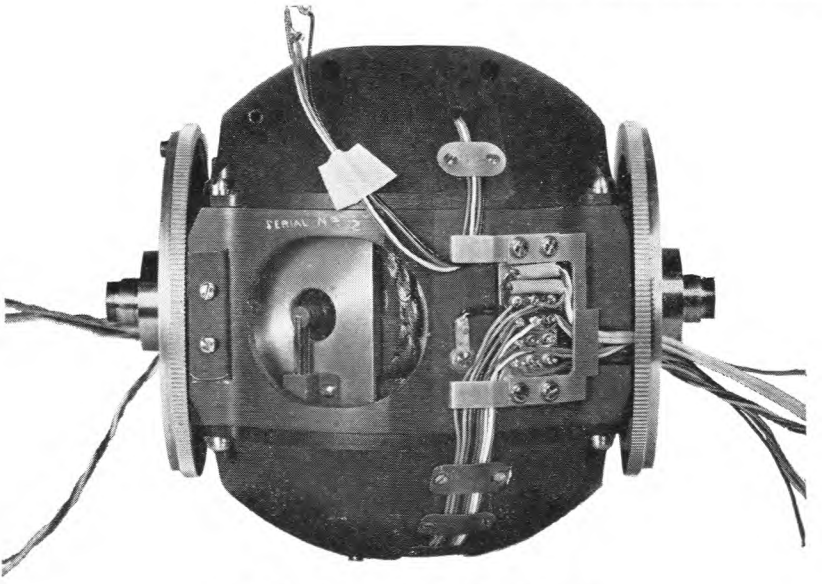


Fig. 2. Platform Gyro in its Pitch Gimbal

Electronic Unit

4. The electronic unit is designed with a particular application in view. Its main functions in most applications are as follows:—

(*a*) Power supply to the gyros.

(*b*) Control of the erection cycle.

(*c*) Servo control of the gimbals.

(*d*) Erection cut-back during aircraft accelerations.

(*e*) Correction of the azimuth output for earth rate and gyro bias drift.

(*f*) Correction signals to the vertical to compensate for movement over the spherical earth.

(*g*) Magnetic monitoring, utilizing signals from a fluxvalve.

(*h*) Transmission of heading information.

Levelling Control

5. Each gyro has a liquid level transducer mounted on the vertical gimbal. Deviation of the spin axis from the horizontal will produce an error signal from the liquid level and this will activate a torque motor to precess the gyro back to the horizontal. The spin axes are orthogonal and therefore a horizontal plane is established from which pitch and roll may be determined. The displacement of the spin axis with reference to the vertical gimbal will activate the appropriate roll and/or pitch servo so as to follow the gyro in its precession. As the platform becomes quiescent again in the horizontal the liquid level signal is eliminated.

6. When subject to horizontal accelerations the platform will erect to the dynamic vertical, *i.e.* the vector resultant of gravity and the additional acceleration, either linear or centripetal. To minimize the effect of short duration random accelerations the erection rate is low (approx. 1.5 degrees/minute). During turns the gyro erection rates are cut back to 0.25 degrees/minute at predetermined conditions based on the combined effect of bank angle and rate of turn. Provision is also made for similar erection cut-back during fore-and-aft accelerations such as at take-off and landing.

Pitch and Roll Outputs

7. Pitch and roll are detected by pick-offs at the junction of the vertical gimbal and the inner gimbal axes. The pick-offs provide error signals through a co-ordinate resolver to pitch and roll servomotors on the outer gimbals. These turn through the appropriate angles until the pick-off signals are zero again. The actual pitch and roll outputs are derived from synchro transmitters and potentiometers on the pitch and roll gimbals.

8. When the gyros are arranged as in Fig. 1 with the axis of one in the fore-and-aft direction and the axis of the other athwartships, then the forward gyro (X) is sensitive to pitch and the rear

gyro (Y) sensitive to roll. If the two axes are turned through 90° in the horizontal then gyro X is sensitive to roll and gyro Y to pitch. In fact the gyros are allowed to assume initially any random direction in space and are left to drift, the only precession applied to move them in azimuth being that necessary to maintain orthogonality. The gyro axes and the aircraft axes will therefore very seldom coincide and consequently each gyro pick-off will detect components of both pitch and roll, the magnitude of which will depend upon the angle between the particular gyro spin axis and the aircraft fore-and-aft axis. The signals from the pick-offs are therefore passed through a coordinate resolver mounted on the heading follow-up shaft in the electronic unit. The separate pitch and roll signals are then fed to the gimbal servomotors.

Azimuth Output

9. Gyro Y is used to provide the azimuth output of the platform. A change in heading produces an output from the rotor of the synchro resolver (RS) mounted at the junction of the pitch gimbal and the axis of the vertical gimbal. This error signal is fed to the heading follow-up servo in the electronic unit. The servomotor moves and thereby turns a resolver transmitter (RX) which changes the signals in the stator coils of the RS on the pitch gimbal until a new null condition of the rotor is obtained and movement ceases. The rotation of the servo-motor is a measure of the change in heading, and this is transmitted by synchro control transmitters (CX). The signals from the RX are also passed to the stators of the RS on the pitch gimbal of gyro X.

10. The azimuth output is initially purely arbitrary since it only indicates the change in the angular relationship between the spin axis of gyro Y and the fore-and-aft axis of the case. Provision is made in the electronic unit for the relative heading to be given significance, by magnetic monitoring for example.

Orthogonality

11. The two resolvers on the pitch gimbals are zeroed when the vertical gimbals, and therefore the gyro spin axes, are orthogonal. Whenever this orthogonal condition tends to change owing to drift of either or both gyros an output is obtained from the rotor of the RS on the pitch gimbal of gyro X. This signal is passed through the orthogonality servo system and the gyros are precessed by means of torque motors on their inner gimbal axes into an orthogonal condition again. It is arranged that the gyros are precessed in opposite directions simultaneously to achieve this. The azimuth servo and the orthogonality servo loop are closely inter-related; it is stated in paragraph 9 that the signals in the stators of the RS associated with gyro X depend upon the signals transmitted from the RX in the heading servo system. In operation both loops are quiescent only when both resolver rotors are in their respective nulls at 90° to each other.

12. Because the gyro spin axes are maintained orthogonal and precessed in opposite directions the random azimuth drift of the system is the average random drift of the two gyros. If for example gyro X drifts at 0.2 degrees/hour and gyro Y at 0.1 degrees/hour both in the same direction then, since orthogonality is maintained, gyro X is restrained, as it were, by the slower drift rate of gyro Y, and conversely gyro Y is accelerated by gyro X. The drift rate of the system is therefore about 0.15 degrees/hour. If the two gyros drifted at the same rate but in opposite directions then the drift rate of the system would be zero.

Erection Cycle

13. The erection cycle of the platform which takes place after switching on is as follows:—

(*a*) The pitch and roll gimbals are brought to their zero position inside the case.

(*b*) Full power is applied to the gyros for 30 seconds to bring them up to about one-third full speed, then reduced power is applied to maintain this speed until orthogonality slaving is complete.

(*c*) After 30 seconds fast orthogonality slaving commences. At 60 seconds the orthogonality error signal is sampled and fast erection to the vertical is applied when the error signal is that of a 9° angular difference.

(*d*) Thirty seconds after fast erection commences the orthogonality slaving goes from fast to normal and full power is restored to the gyros to accelerate them up to their operational speed.

(*e*) After a further 40 seconds fast erection is switched to normal erection and a signal indicating platform operational readiness is made available to operate a lamp or flag in the aircraft.

The readiness time for the platform is therefore between 2 and 3 minutes from switch on, the actual time depending on the relative vertical gimbal positions when orthogonality slaving is first applied.

CHAPTER 5

TURN AND SLIP INDICATORS

CONTENTS

	Paras
Introduction	1– 4
Air Driven Indicators	5
Electrically Driven Indicators	6– 8
Errors in Turn Indicators	9–11
Errors in Slip Indicators	12
Aircrew Serviceability Checks	13
Instrument Indications	14

Introduction

1. The turn and slip indicator consists of two instruments contained in the same case. The turn indicator is a vertical rate gyro mounted in the instrument panel with its spin axis parallel to the pitch axis of the aircraft. It is used to indicate the rate at which the aircraft is turning about the vertical (yaw) axis. The slip indicator consists of a curved liquid-filled tube mounted parallel to the pitch axis. A ball contained in the tube indicates any tendency of the aircraft to slip or skid along the axis of the instrument.

2. **The Turn Indicator.** The principle of operation of a rate gyro is described in Chapter 1. The turn indicator is calibrated for set rates of turn. A rate 1 turn corresponds to a 180° per minute turn, rate 2 to 360° per minute, and rate 3 to 540° per minute. The rate gyro is calibrated for a limited range of TAS and angles of bank; outside this range the indicated rates will be in error. Constructional details of the specific marks of instrument are contained in AP 112. It should be noted that the rate gyro shown in Chapter 1 has two springs; current gyros have only one spring.

3. **The Slip Indicator.** On some types of instrument a damped pendulum is used as the slip indicator instead of the ball in tube. In both types the ball or pendulum is kept in the central position by the force of gravity, or in balanced turns by the resultant of gravity and centrifugal force. The movement of the ball is damped by the viscosity of the liquid in the tube.

4. **Types of Turn and Slip Indicator.** The following paragraphs describe the main features of the instruments currently in use.

Air Driven Indicators

5. **Mk 1B.** The Mk 1B turn and slip indicator is an air driven instrument. Suction is provided either by an engine driven vacuum pump or by a venturi head. Air is exhausted from the instrument by suction, and the replacement air enters the case through a jet which causes it to impinge on slots cut in the periphery of the gyro rotor, driving it round. On the face of the instrument (*see* Fig 1) a needle indicates zero rate of turn when central, and is deflected to right or left according to the rate and direction of turn experienced. The gyro gimbals are damped magnetically to ensure smooth operation, and to reduce unwanted oscillations caused by flight in turbulent conditions. The instrument markings are luminous, and the black ball in the slip indicator is backed by a fluorescent lining to the rear of the tube.

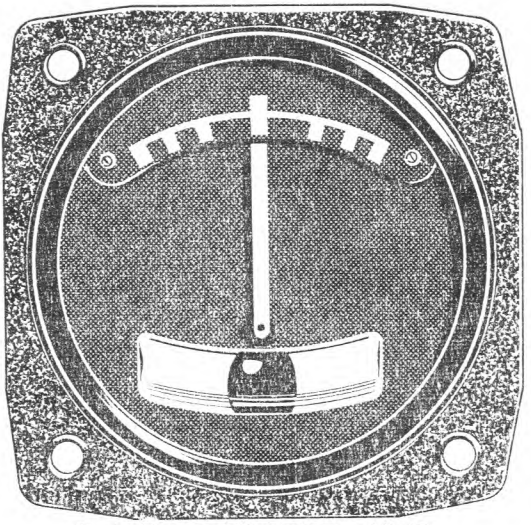


Fig 1 Turn and Slip Indicator Mk 1B

Electrically Driven Indicators

6. The Mark 2 series onwards are equipped with electrically driven gyro rotors in place of the air driven rotors of the Mk 1B.

7. **Kelvin and Hughes Type—Mk 2, 2A and 3.** These instruments are powered by a nominal 28 volt direct current supply. The presentation of the Mk 2 and 2A is similar to that of the Mk 1B apart from the power failure indicator (*see* Fig 2). The word OFF in luminous letters is shown if the gyro wheel speed reduces below 2,000 rpm. In the Mk 2A indicator the slip tube is black and the ball is luminous. The Mk 3 is a development of the Mk 2A in which the tilt angle of the gyro has been reduced in order to lessen looping errros.

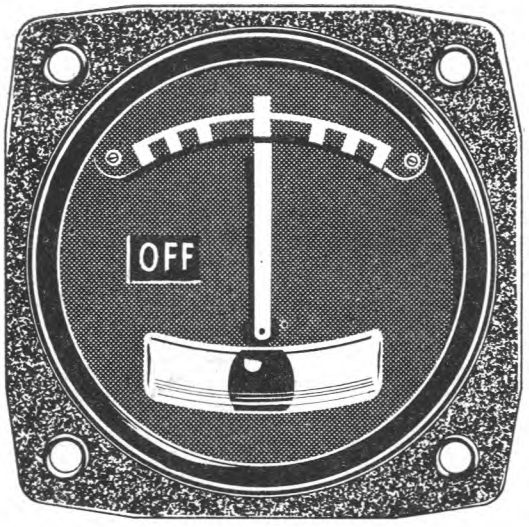


Fig 2 Turn and Slip Indicator Mk 2

8. **RB Pullin Type Mk 2A and 3.** These instruments are similar to the Kelvin and Hughes instruments. The OFF flag appears when power fails or rotor speed drops below 2,000 rpm. The slip indicator is of the pendulous type, and has a second inverted pendulum added for increased sensitivity. The Pullin type Mk 2A is shown in Fig 3. The slip indicator is luminous.

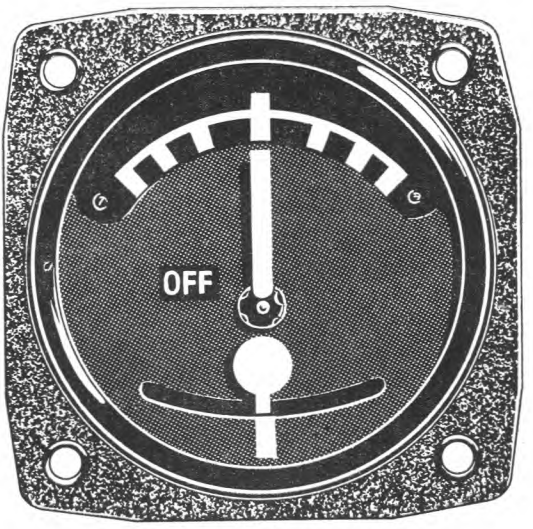


Fig 3 Turn and Slip Indicator Mk 2A

Errors in Turn Indicators

9. **Looping Error.** The turn indicator measures the rate of turn about the vertical (yaw axis). Movement of the aircraft about the pitch axis (which normally coincides with the spin axis of the gyro) does not normally produce any gyro precession. If the aircraft is yawed so that the gyro axis tilts to indicate the rate of turn, and simultaneously pitched, the gyro axis is no longer parallel to the pitch axis, and the pitching moment causes a torque to be applied to the gyro. The rate of turn indicated will therefore exceed the true rate of turn. The resultant error, which is called looping error, is dependent on the rate of yaw and the rate of pitch, and in some circumstances can cause full scale deflection of the rate of turn indicator. The indicators are calibrated for rate 1, 2 and 3 turns at specified angles of bank and TAS, and looping error is taken into account in this calibration. Therefore in any manoeuvre in which the rate of pitch is greater or less than that associated with a particular angle of bank the instrument indication will be in error.

10. **Rotor Speed Errors.** The expression for the angle of tilt Ø of a rate gyro was derived in Chapter 1, where it was shown that:

$$\text{Ø} \propto \text{Rate of Turn} \times I\,\omega$$

where: I = Moment of inertia of the gyro

ω = Angular velocity of the gyro.

It can be deduced that if the angular velocity (rotor speed) of the gyro is increased then a greater angle of tilt is produced by the same rate of turn. Conversely if the rotor speed reduces then the rate of turn indicated will be less than the true rate of turn.

11. Details of the calibration of the various types of instrument are given in AP 112.

Errors in Slip Indicators

12. The slip indicator acts as a rudimentary form of accelerometer and is not subject to observable error.

Aircrew Serviceability Checks

13. The turn and slip indicator may be checked whilst taxying by turning the aircraft. The turn needle should indicate the direction of the turn, whilst the slip indicator should indicate skid in the opposite direction. The instrument can also be checked in the air by timing a rate 1 turn.

Instrument Indications

14. In Fig 4 the various indications given by a turn and slip indicator in flight are summarized.

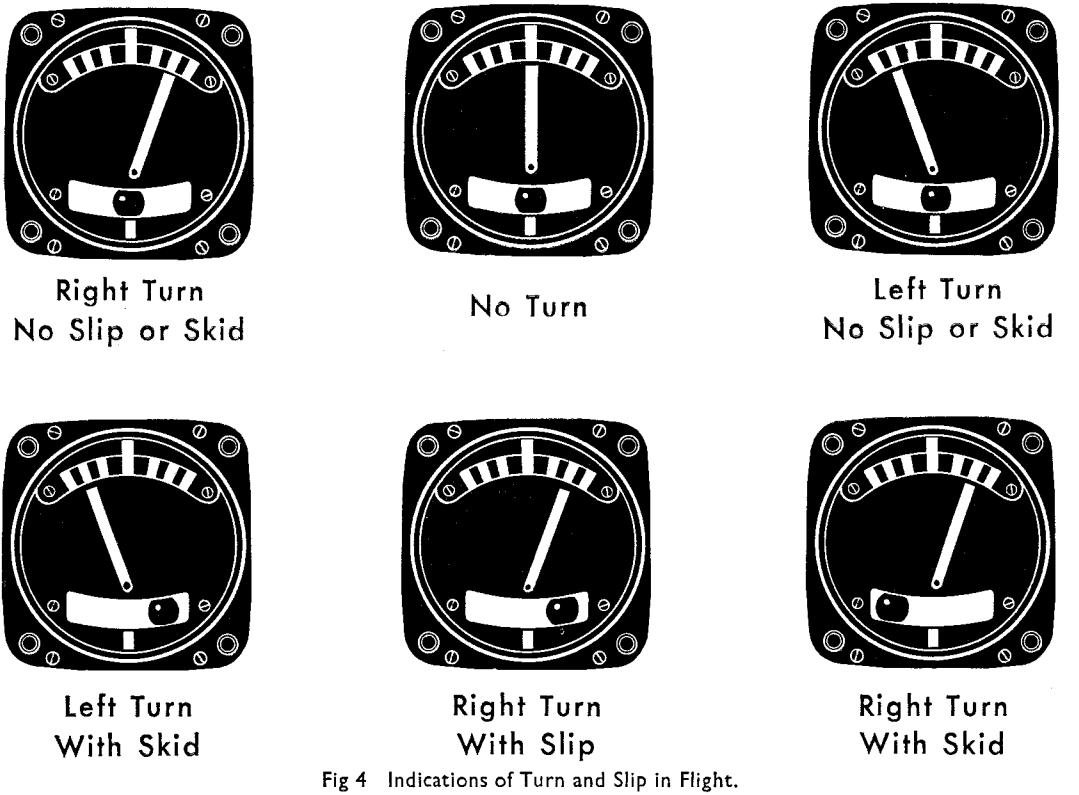


Fig 4 Indications of Turn and Slip in Flight.

CHAPTER 6

THE DIRECTION INDICATOR Mk 1

CONTENTS

	Paras
Introduction	1– 2
Description	3– 5
Gyro Erection	6
Errors	7– 9
Serviceability Checks	10
In-Flight Procedure	11

Introduction

1. The Direction Indicator Mk 1 (DI Mk 1) is an air driven vertical gyro mounted on the pilot's instrument panel. It is used as a heading reference in light aircraft in conjunction with a pilot type magnetic compass. The properties of vertical gyros are described in Chapter 1.

2. The DI Mk 1 is aligned with the magnetic heading of the aircraft by setting it to the reading obtained from the magnetic compass. Thereafter the instrument may be used as the primary heading reference during level flight, provided that it is checked at regular intervals against the magnetic compass and reset after aerobatic flight.

Description

3. The instrument consists of an air driven vertical gyro rotating at about 10,000 rpm. It is operated by an engine driven suction pump which draws air from the instrument case at a suction of 4 inches of mercury. Replacement air discharges into the case through a pair of nozzles impinging as a jet on a series of buckets cut in the periphery of the gyro.

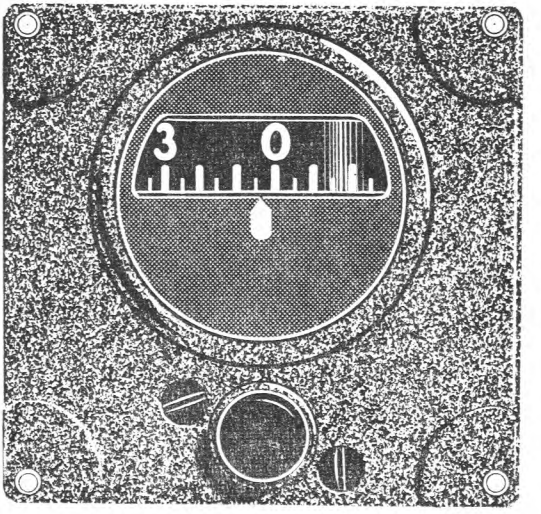


Fig 1 The Direction Indicator Mk 1

4. **External Detail.** Fig 1 shows the front of the instrument. The cylindrical scale is attached to the vertical (outer) gimbal of the gyro. The scale is calibrated every 30° with 10° divisions and 5° sub-divisions. The aircraft heading is read off against a fixed lubber line. The markings are all luminous for night flying. Underneath the window is the caging knob which is used to cage the vertical and horizontal gimbal rings so that they are mutually orthogonal, and to reset the vertical gimbal and its attached scale to the desired heading.

5. **Internal Detail.** Fig 2 shows the mechanism of the instrument. The horizontal gimbal has a freedom of movement of $\pm 55°$ when it is uncaged. The nozzles of the air jet are attached to the vertical gimbal.

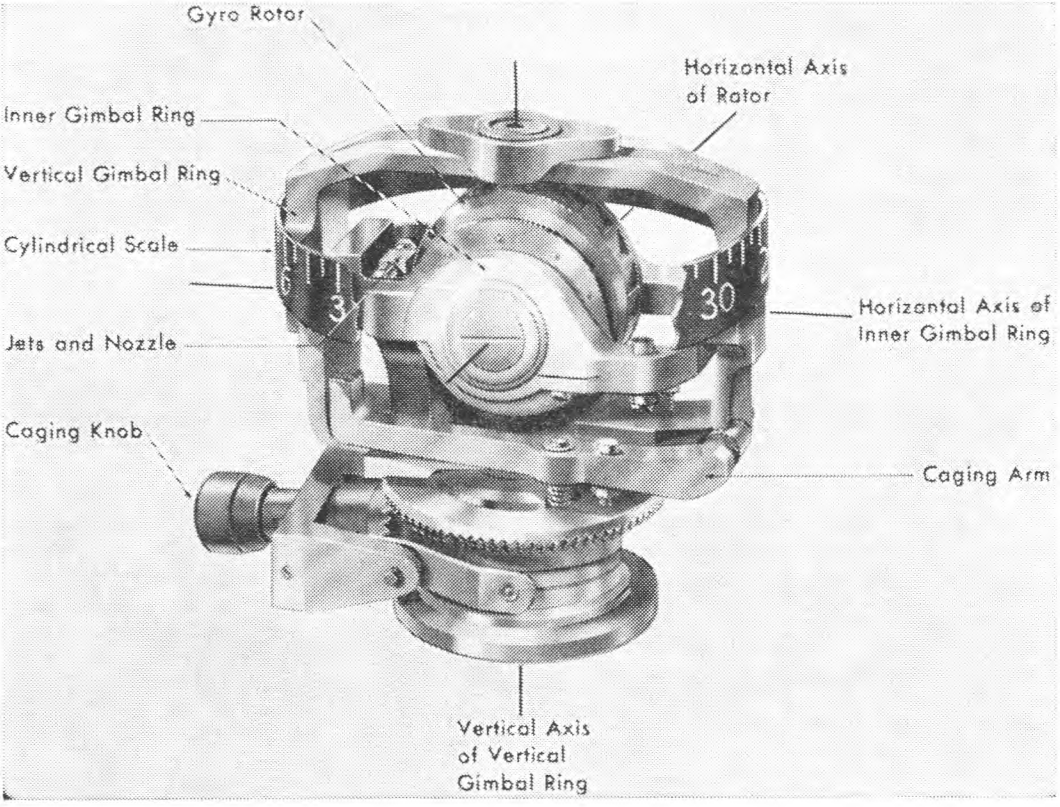


Fig 2 The Mechanism of the DI Mk 1

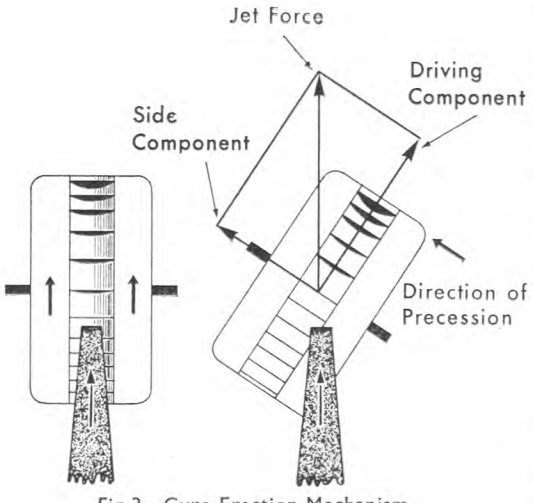


Fig 3 Gyro Erection Mechanism

Gyro Erection

6. The gyro is initially erected by the caging mechanism so that the gyro spin axis is aligned with the plane defined by the aircraft pitch and roll axes. The gyro rotor is spun by the action of the air jet, attaining operating rpm about five minutes after full suction is developed by the vacuum pump. The air jet also maintains the rotor in alignment with the vertical gimbal. If the gyro topples a component of the jet force will act at right angles to the rotor and produce a precession which tends to erect the gyro (*see* Fig 3).

Errors

7. **Apparent Drift.** The gyro is subject to apparent drift caused by earth rate and transport wander (*see* Chapter 1). A fixed rate of compensation is obtained by means of balancing nuts attached to the gimbals. The rate of compensation

is calculated for the latitude of operation of the aircraft. There is no compensation for transport wander apart from the periodic resetting of the instrument to the magnetic compass heading.

8. **Real Drift and Topple.** The gyro is subject to real drift and topple rates caused by imperfections in manufacture. The drift rate can be as much as 16 degrees per hour on a single heading. Topple is corrected by the air jet erection system.

9. **Gimballing Error.** The gyro is subject to gimballing error during manoeuvres (*see* Chapter 1). Once level flight is resumed after a manoeuvre the error disappears.

Serviceability Checks

10. Five minutes after full suction has been applied the instrument should be checked for erection by depressing the caging knob and noting that there is a resistance to movement in azimuth. The scale should then be aligned with the reading on the magnetic compass. During turns on the ground the instrument should be checked for operation in the correct sense, ensuring that the scale ceases to rotate as soon as a turn is completed.

In-Flight Procedure

11. The following procedures should be adhered to in flight:

a. The instrument should be aligned with the magnetic heading derived from the compass every 15 minutes and after prolonged climbs and descents.

b. The instrument should only be aligned when the aircraft is in straight and level flight.

c. The instrument should be caged before starting any manoeuvre likely to exceed pitch or roll angles in excess of 55 degrees.

CHAPTER 7

DIRECT READING ATTITUDE INDICATORS

CONTENTS

GENERAL INFORMATION

	Paras
Introduction	1–2
Gyro Principle	3–4
Control of Gyro Spin Axis	5–9
Acceleration Errors in Erection Systems	10–15
Fast Erection Systems	16
Controlled Precession	17–22
Controlled Precession Limitations	23–28
Effect of Instrument Tilt	29–30

ARTIFICIAL HORIZONS

Artificial Horizon Mk 1B	31–35
Artificial Horizon Mk 1E	36–37
Artificial Horizon AMI 300	38
Artificial Horizon Mk 3C	39–44
Artificial Horizon Mk 4	45–51
Artificial Horizon Mk 4A	52
Artificial Horizon Mk 4B, C, D and E	53
Artificial Horizon Mk 5	54–58
Artificial Horizon Mk 5A	59
Artificial Horizon Mk 6 Series	60–68

HORIZON GYRO UNITS

Introduction	69
HGU Type B Series	70
HGU Mk 1	71
HGU Mk 2 and 2C	72
HGU Mk 3 and 3A	73
HGU Mk 5 and 5A	74

FH 30 SERIES HORIZON GYRO UNITS

Introduction	75–78
Erection System	79–82
Erection System Operating Characteristics	83–86
Caging Mechanism	87–93
Summary of FH 30 Series HGU	94

(AL 33, Sept 86)

GENERAL INFORMATION

Introduction

1. Aircraft have three degrees of rotational freedom and to be flown accurately require some form of attitude reference. In its simplest form this reference can be the visible horizon, but for unrestricted all weather flying a cockpit instrument indication is required to substitute for the natural horizon. This indication is provided by either direct reading self-contained instruments or by displays controlled by outputs from other aircraft systems *eg* IN platforms. This chapter deals with the direct reading self-contained instruments.

2. The current artificial horizons, horizon gyro units or attitude indicators as the instrument may be called, employ an air or electrically driven displacement gyro with its spin axis maintained vertical to the earth by gravity sensing devices. Indication of pitch and bank attitude is presented by the relative positions of two elements, a fixed symbol representing the aircraft and a bar or line stabilized at right angles to the earth vertical representing the natural horizon. Supplementary indication of bank angle is presented by the position of a gyro stabilised pointer against a fixed bank angle scale (see Fig 1).

Gyro Principle

3. The principle of operation of the gyroscope may be understood with reference to Fig 2. The gimbal system is arranged so that the inner gimbal forms the rotor casing and is pivoted to the outer gimbal ring parallel to the aircraft's athwartships or pitch axis (Y Y_1). The outer gimbal ring is pivoted to the front and rear of the instrument case parallel to the aircrafts fore and aft or roll axis (Z Z_1). This arrangement ensures that with the gyro spin axis maintained vertical to the earth, all three axes of the gyro are mutually at right angles when the aircraft is in straight and level flight and are coincident with the three aircraft axes.

4. From Fig 2 it can be seen that any change in pitch attitude will result in the

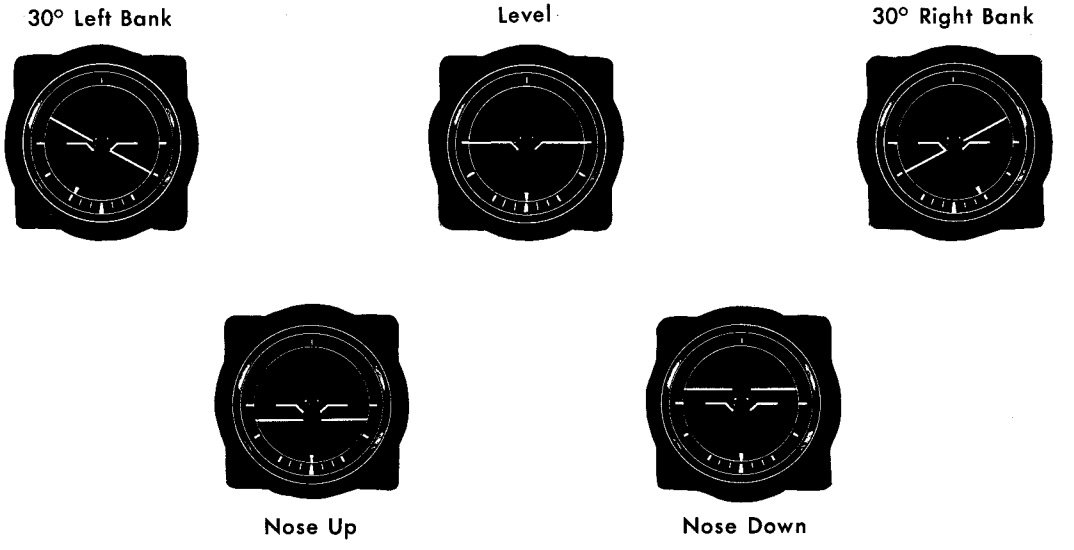


Fig 1 Attitude Indications

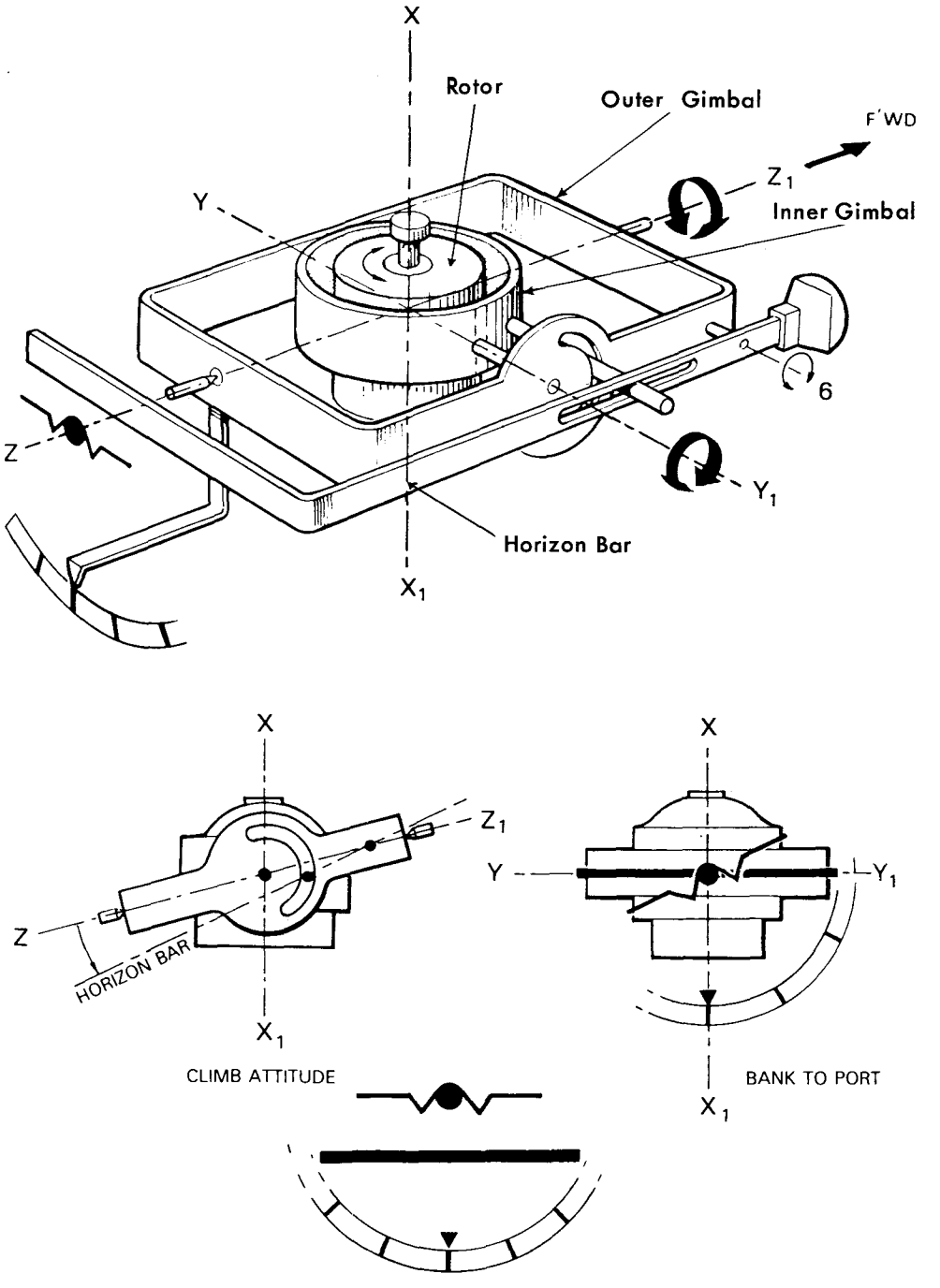


Fig 2 Gyro Principle

instrument case and the outer gimbal rotating round the Y Y_1 axis of the gyro and any change in bank attitude will result in the instrument case rotating round the Z Z_1 axis. These rotating movements are linked mechanically to an indicator, in this case an horizon bar, and may also be sensed electrically to provide pitch and bank information to other systems.

Control of the Gyro Spin Axis

5. To provide accurate indications the gyro spin axis must be maintained vertical to the earth, therefore correcting torques must be applied to compensate for three main effects:—

a. Earth rotation.

b. Transport error.

c. Internally generated torques due to any gimbal imbalance.

6. A common way of sensing any deviation of the gyro spin axis away from the earth's vertical (termed verticality error), is to use a simple pendulum type device which will respond to the earth's gravitational pull to initiate a correcting torque. The systems used are either pendulously suspended vanes that mechanically control air jets, or mercury switches that control electric torque motors (see Chap 1 of this section), to provide the correcting torques. These devices are discussed in more detail in later paras.

7. Such systems, being pendulous, will however also respond to any horizontal reactive force generated by accelerations (and decelerations) in flight. This force will displace the pendulous system in the opposite direction to the acceleration and the torque generated will align the spin axis with the resultant of the reactive force and true vertical vectors to produce a verticality error. To prevent the generation of significant gyro verticality errors with high accelerations, a cut-off device can be incorporated to inhibit the erecting system above a predetermined level of horizontal acceleration. When in this erection cut-off condition the gyro is free to wander only under the influences noted in para 5.

8. **Effect of Gyroscopic Inertia.** Any verticality error is seen by the pilot as a combination of pitch and roll errors. During a turn in the cut-off condition, because of gyroscopic inertia any error present will appear to transfer from one axis to the other, eg a purely pitch error will be wholly transferred to the roll axis as a 90° turn (see Fig 3). This effect can be compensated for by incorporating a Pitch/Bank erecting system described in para 49.

9. Under prolonged horizontal accelerations below the cut-off condition the verticality error will be generated at the same rate as the correcting errors in unaccelerated flight. Therefore the erection rates must be optimised to achieve a balance between compensating for the effects in para 5 sufficiently quickly whilst preventing errors caused by accelerations being built up too rapidly.

Acceleration Errors in Erection Systems

10. **Introduction.** Below the cut-off level the gyro is subject to verticality errors due to the fore and aft and turning accelerations experienced in flight. These errors can be considerable after continuous periods of accelerating flight below the cut-off level.

11. **Fore and Aft Axis Acceleration.** If an acceleration in the fore and aft axis is made the pendulous vanes or the mercury switch controlling pitch error will be displaced towards the rear. This results in a torque that precesses the top of the spin axis away from the vertical in the fore and aft axis to falsely indicate a change in pitch. A deceleration will result in a false indication of pitch in the opposite sense. The direction of the error is determined by the direction of gyro rotation and the characteristics of the erection system.

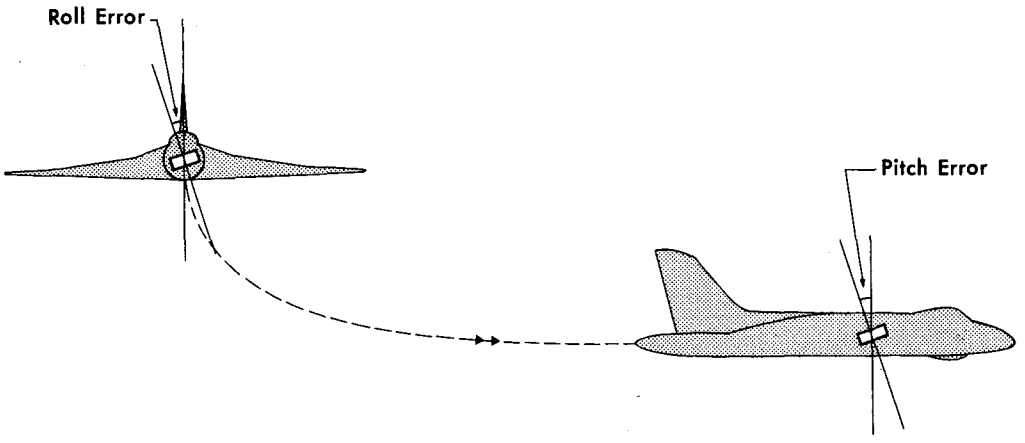


Fig 3 Roll Tilt Changing to Pitch Tilt in a Turn

12. **Turning Errors.** During a turn the gravity controlled erection devices will be influenced by the acceleration force acting towards the centre of the turn (see AP 3456K). This force will displace the pendulous vanes or mercury switch controlling bank error away from the centre of the turn. This results in a torque precessing the top of the spin axis away from the vertical in the athwartships axis ie heading +/− 90°. In a constant turn the strength of the force will be constant but the direction it acts in will be changing at the same rate as the direction of the aircraft. The effect of this acceleration force can be explained by comparing the action of the gyro and the aircraft in a constant rate 360° turn as illustrated in Fig 4.

13. Fig 4 (a) illustrates the movement of an aircraft in a 360° constant rate turn commencing at A. Fig 4 (b) shows the movement of the top of the gyro spin axis during the same turn. First we will consider the movement of the gyro spin axis throughout the turn. At position A before the turn begins, the spin axis is vertical with no acceleration force to cause any precession. As soon as the aircraft starts to turn the acceleration force will act along the athwartships axis of the gyro displacing the pendulous vanes or the mercury switch controlling bank error and a false tilting of the spin axis to the left will be sensed. The resulting torque will precess the top of the spin axis to the right, in a direction of aircraft heading +90° at a rate depending on the gyro characteristics and the rate of turn. As the rate of turn is constant the precession rate will be constant and the top of the gyro spin axis will describe a circle at the same rate as the aircraft heading changes, in this case leading the aircraft heading by 90° as shown in Fig 4(b). If we now superimpose the gyro head movement onto the aircraft movement we can see in Fig 4(a) that as the aircraft has moved from A to B the gyro head has moved from A to B_1 resulting in a false indication to the pilot of both pitch and bank. At aircraft position C the gyro head has moved to C_1 giving no bank error but a large pitch error, and at D both pitch and bank error will again be present, reducing to zero error on return to A.

14. **Compensation for Acceleration Errors in Erection Systems.** The effects of acceleration errors in erection systems cannot be

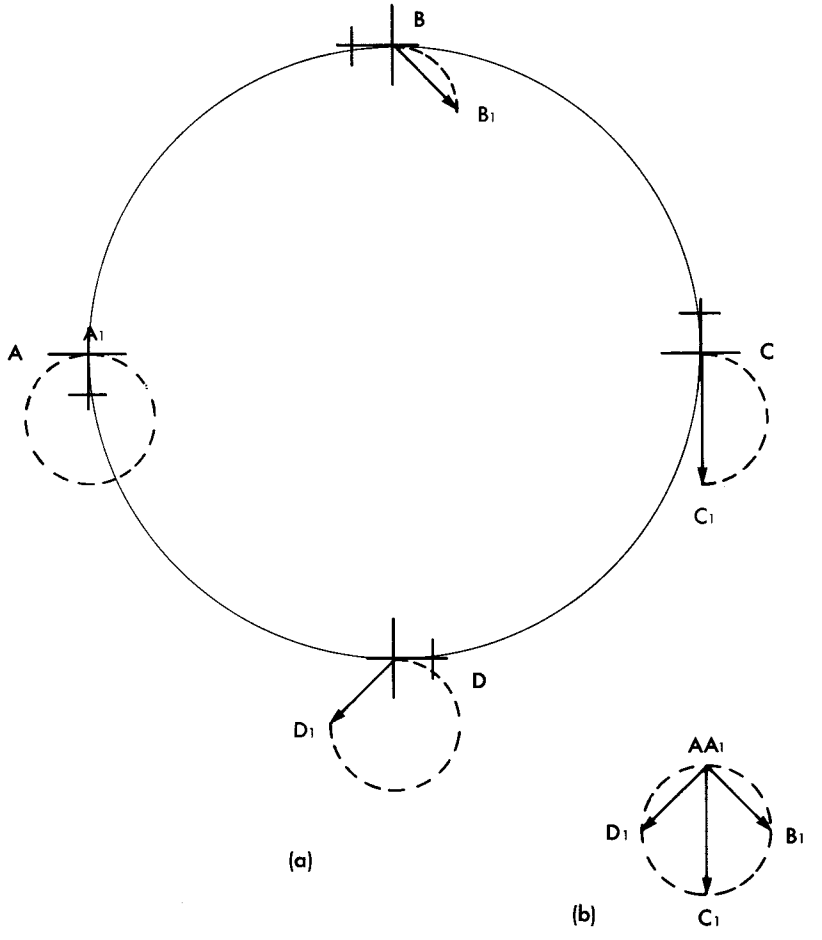


Fig 4 Turning Error

fully eliminated but may be reduced by inclining the spin axis from the vertical. In its application the method is mechanical in form and varies with the type of instrument, but in all cases the result is to impart a constant forward (rearward in some instruments) tilt to the gyro axis from the true vertical. The angle of tilt varies but is usually in the region of 1.6° to 2.5°. In air driven systems the pendulous vanes controlling pitch error are balanced so that the gyro is precessed to the tilted position; in electrical switch systems the pitch mercury switch is adjusted for the same effect. The linkages between the gyroscope gimbal system and the indicator are modified so that in level flight zero pitch is indicated. The effect of the inclined spin axis is illustrated in Fig 5(a) where point A represents the end of the true vertical through the centre of the rotor and A A_1 represents the direction of tilt. During a turn the top of the spin axis describes a circle about point A at the same rate as the aircraft changes heading. The error and its direction in relation to the aircraft during the turn are constant.

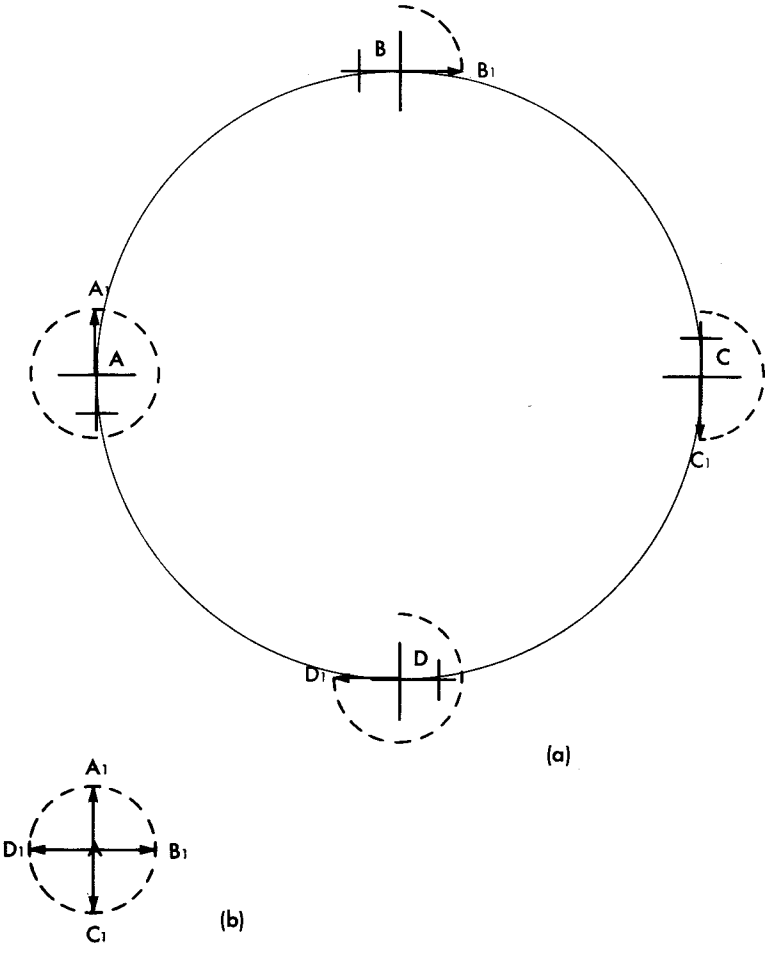


Fig 5 Turning Error Compensated

15. **Pendulosity Error.** Some gyroscopes are constructed with their rotor assemblies slightly bottom heavy to ensure quick initial erection. The assembly itself is therefore also sensitive to acceleration errors. When acceleration takes place the base of the rotor assembly tends to lag behind, ie it tends to swing in the opposite direction to the acceleration and a torque is applied direct to the assembly in the axis of the acceleration. This results in a precession of the spin axis in the remaining axis. Thus a fore and aft axis acceleration creates a bank error and turning creates a pitch error. In a turn the top of the spin axis describes a circle synchronised with the aircrafts heading as shown in Fig 6. Compensation is usually made by tilting the spin axis in the athwartships axis, the direction being dependent on the direction of rotor rotation. The effect is the same as that produced in correcting for erection system errors, ie the top of the spin axis traces out a circular path about itself to produce a single constant error as in Fig 6.

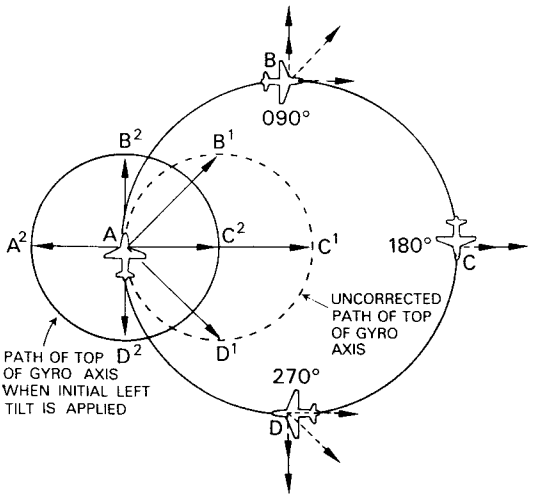


Fig 6 Pendulosity Error

Fast Erection Systems

16. After extended periods of high accelerations, on return to level flight the gyro may have very large verticality errors, or in the extreme case have toppled. To restore the gyroscope to its normal operating position as quickly as possible a fast erection mechanical or electrical system may be provided.

Controlled Precession

17. **Requirement.** It is required that the attitude indication should be consistent and coherent over the full flight envelope of an unrestricted manoeuverability aircraft, irrespective of how any particular attitude is achieved. For example an inverted flight, zero pitch indication may result from a 180° roll or a 180° pitch manoeuvre, starting from wings level flight.

18. Physically, the outer gimbal is required to rotate 180° relative to the airframe in both cases. During the 180° roll manoeuvre, the rotation of the airframe and instrument case around the gyro stabilized outer gimbal provides the necessary inverted flight indication.

19. For a 180° pitch manoeuvre there is no roll rotation of the airframe, but to retain the correct attitude display a rapid 180° rotation of the outer gimbal has to occur on passing through the vertical in climb or dive. Controlled Precession is the name given to the means of ensuring the outer gimbal rotates 180° as the aircraft fore and aft axis passes through the vertical.

20. **Method.** Inner gimbal resilient limit stops are incorporated in the instrument to cause the pitch rotation of the airframe to apply a torque to the gyro, about the inner gimbal (Y Y_1) axis, just before the outer gimbal axis becomes coincident with the gyro spin axis. This prevents gimbal lock occurring and causes the gyro to precess about an axis mutually orthogonal to the spin and torque axes and in so doing causes the outer gimbal to rotate, as this is the only axis of freedom remaining to the instrument. After 180° rotation the continuing pitch rate of the aircraft results in the inner gimbal moving away from the stop and the instrument regains its second axis of freedom. It is important that the inner gimbal stop acts before the gyro spin axis and the outer gimbal axis become coincident. If

that were to occur the plane of gyro precession would intercept the outer gimbal axis and no outer gimbal rotation would be induced.

21. During the 180° degrees outer gimbal rotation, the gyro spin axis has to move conically around the outer gimbal axis and is therefore subject to transient verticality errors at least equal to the magnitude of the offset of the inner gimbal stop. The gimbal stop is usually set to act between 5° and 8° before the +/− 90° position is reached, ie it permits the gyro spin axis to have between +/− 85° and +/− 82° of freedom from its normal orthagonal position relative to the outer gimbal.

22. After the complex transient disturbance ceases on completion of the rapid 180° rotation about the outer gimbal axis the gyro will usually restore to within a degree or so of the vertical after a single loop, due to the conservation of angular momentum of the gyro and the gimbal structure. There are however certain flight manoeuvres that can result in significant errors.

Controlled Precession Limitations

23. **Vertical Flight.** If the aircraft is held in the region of vertical flight more than momentarily the normal rapid 180° outer gimbal rotation observed in a loop may be replaced by a slow and irregular rotation of the outer gimbal in either direction. Such rotation causes a conical oscillation of the gyro spin axis around the vertical equal to the 5° to 8° degrees offset angle of the inner gimbal stops. On restoring aircraft pitch rate the inner gimbal departs from the stop but may have a verticality error of up to 8°.

24. **Inclined Pitch Plane.** If the aircraft is looped in a pitch plane inclined to the vertical by more than the maximum stop angle offset of 8° the inner gimbal never reaches the stop and the gimbals are free to adopt their normal two degree of freedom positions. Under such conditions the direction of the 180° outer gimbal rotation will depend upon whether the looping plane is left or right of the vertical.

25. When looping under the conditions where the inner gimbal stop is encountered, the imposed controlled precession rotation of the outer gimbal is always in the same direction for a given direction of rotation of the gyro wheel.

26. The combination of the factors explained in the two preceding paragraphs produces an error profile for looping manoeuvres inclined to the vertical plane. Trials have shown that the errors are small outside the region where the gimbal stops are likely to be encountered, ie where the looping plane is inclined greater than 8° from the vertical. Errors are also small from 8° to vertical on that side where the geometric and controlled precession rotations of the outer gimbal are in the same direction.

27. Looping manoeuvre errors may however become significant in that region between 8° and vertical where the two gimbal rotation directions are in opposition as the outer gimbal hesitates before deciding which way to turn. The FH30 Series attitude indicators for example, have a clockwise spin axis rotation looking down on top. Their peak controlled precession errors occur when looping in a pitch plane inclined 5° to 8° left of vertical, where considerable errors can be introduced.

28. **Effect of Stop Resilience.** Whilst the stop load/deflection characteristics have some influence on performance during the one second or thereabouts periods taken to complete the controlled precession rotation of the outer gimbal in a normal continuous pitch rate looping manoeuvre, their primary task is to reduce the peak shock loads on the gryo and gimbal bearings and gimbal

structure when the stops are impacted on other occasions. In flight the maximum gimbal stop impact velocity is limited by the pitch rate of the aircraft, which seldom exceeds 0.5 radians per second. In comparison, stop impact velocities of up to 20 radians per second might be encountered in transit and other handling away from the aircraft. Higher stop impact velocities approaching 50 radians per second are theoretically possible by abuse of the mechanical caging knob on an unpowered instrument to induce rapid oscillation of the inner gimbal between its climb and dive limit stops. This is potentially damaging to the instrument.

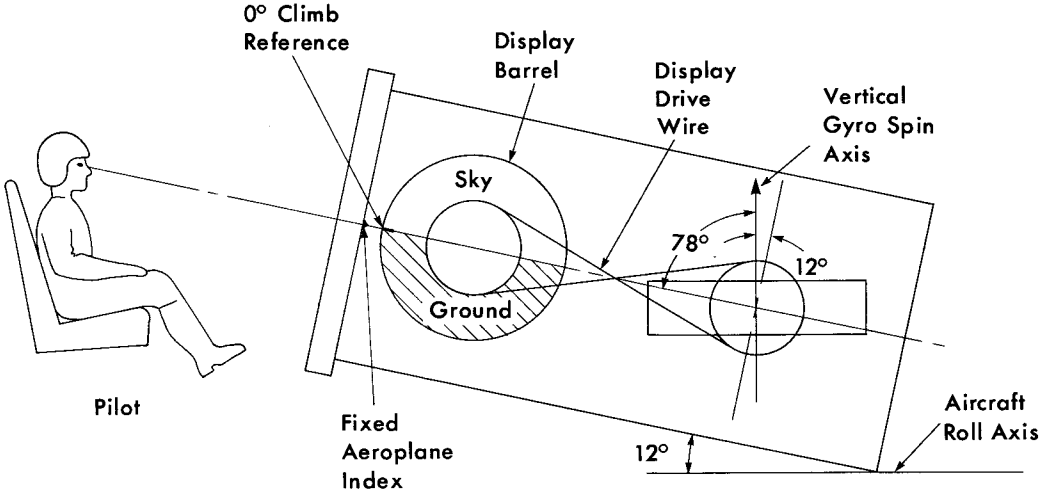


Fig 7a Level Flight Before and After a 360° Loop or Bank

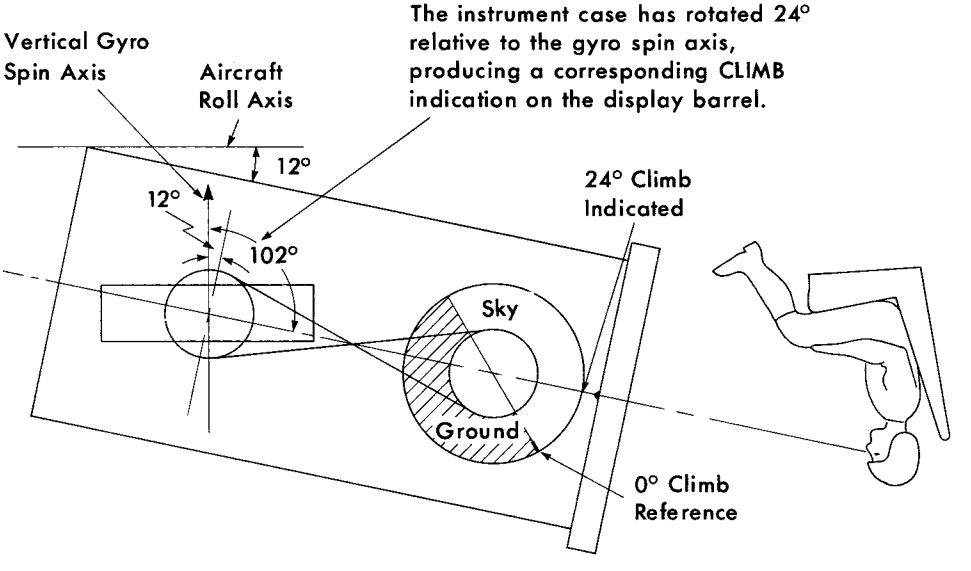


Fig 7b Level Inverted Flight After a 180° Loop or Bank

Fig 7 Effect of Instrument Tilt

Effect of Instrument Tilt

29. **Geometric Error.** Most modern military aircraft have instrument panels inclined from the vertical in normal cruise flight. Zero pitch attitude indication is restored by adjusting the linkage to the indicators to correct for the tilt. The gyro spin axis is retained vertical, and hence is biased away from the orthogonal position in cruise flight by the magnitude of the panel tilt. This means that the inner gimbal stops are intercepted early in dive and late in climb relative to the attitude of the airframe. This geometric offset produces known errors (termed geometric errors) in the displayed attitude which vary as a function of the true pitch and bank angles. These geometric errors can be large, are due entirely to the effect of the panel tilt and are not indicative of any gyro verticality error. The effects as seen by the pilot are summarised as follows:

a. The effect of the panel tilt is compensated for within the instrument such that under wings level conditions there is no geometric error displayed in pitch or bank.

b. If the aircraft is then looped or rolled inverted the effect of the panel tilt compensation built into the instrument is to produce a geometric pitch error of twice the panel tilt angle (see Fig 7).

c. At intermediate bank angles the geometric error in pitch varies as a function of the actual bank angle, increasing from zero at 0° bank to + (nose up) Tilt Angle and + 2 × Tilt Angle at +/− 90° and 180° bank angle respectively. This is shown in Fig 8 for a 9° Tilt Angle.

d. The geometric error in roll cannot be expressed in such a simplified manner but can be considered insignificant (>5°) at pitch angles less than +/− 30°.

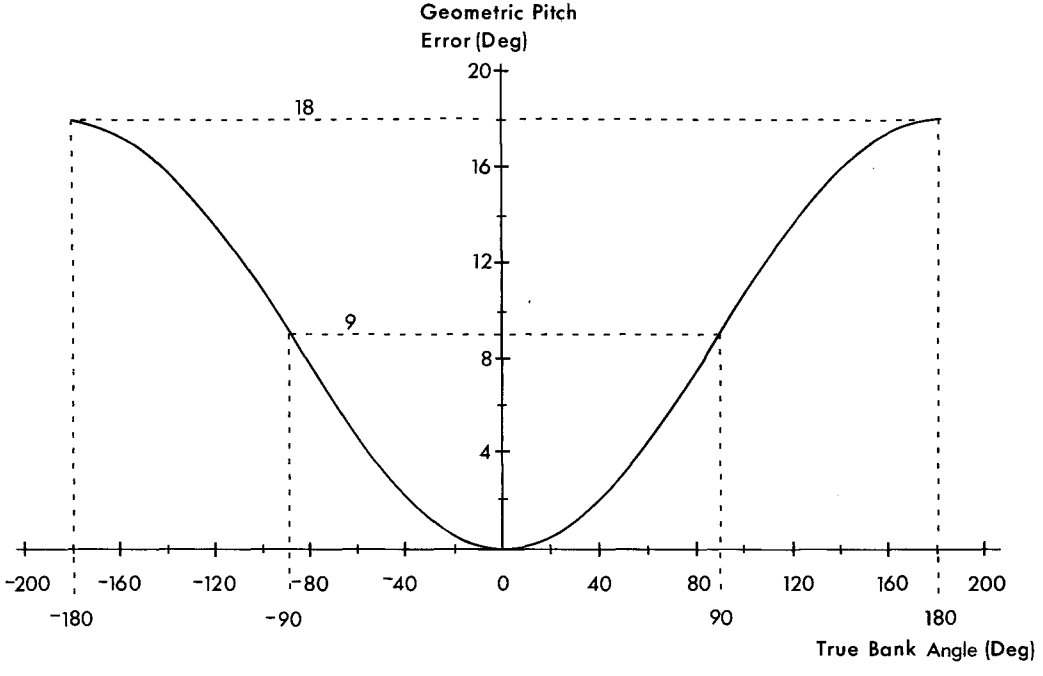


Fig 8 Geometric Error Variance

30. Pilots should be aware of the geometric errors during any manoeuvre involving large pitch and bank angles particularly when they can combine with the controlled precession errors discussed in para 27 to give significant attitude indication errors.

ARTIFICIAL HORIZONS

Artificial Horizon Mk 1B

31. The Mk 1B artificial horizon is shown in Fig 9 and its mechanism in Fig 10. The gyro rotor is air driven at approximately 12,000 rpm by air jets impinging on buckets cut in the periphery of the rotor. The air flows from the rotor casing through jets in the pendulous unit at the base of the casing.

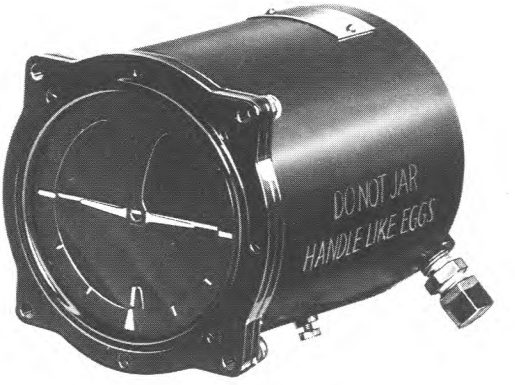


Fig 9 Artificial Horizon Mk 1B

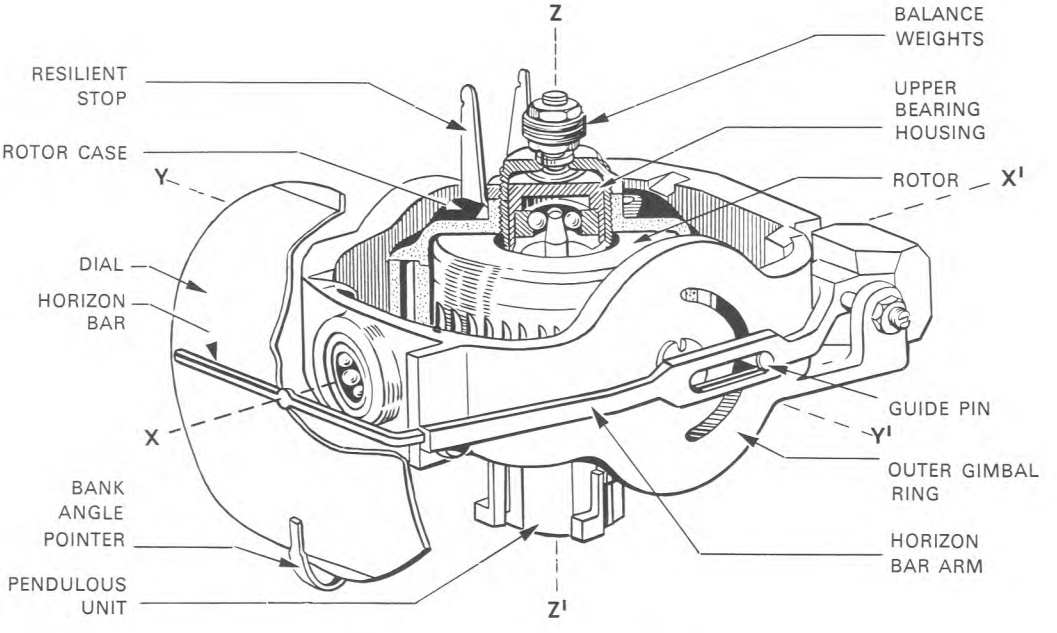


Fig 10 The Mechanism of the Mk 1B

32. **Erecting System.** The gyro is erected when at rest by the weight of the pendulous unit. Once the rotor is spinning it is precessed to the vertical by means of the air jets in the pendulous unit. In Fig 11 the base of the gyro casing is shown with four apertures or jets spaced equally around the circumference. When the gyro is erect and operating the air exhausts through each of the jets at an equal rate into the instrument case (whence it is drawn by a suction pump). The size of each jet is controlled by a pendulous vane. The vanes are mounted in pairs on common spindles, and hang vertically under the influence of gravity, bisecting each of the apertures. If the gyro axis departs from the vertical the jets in the plane of displacement remain half open, and whilst one of the other two jets acting at right angles to the plane of displacement opens the other closes. Thus a reaction occurs at 90° to the plane of displacement, which causes the gyro axis to be precessed back to the vertical (*see* Chapter 1 for explanation of the forces).

33. **Errors.** The instrument is subject to errors caused by the effect of horizontal forces acting on the pendulous unit and on the vanes controlling the air jets. The pendulous unit tends to align itself with the direction of the resultant of gravity and the horizontal force, causing the gyro to precess from the vertical. The errors occuring in a rate 1 turn at 260 knots are compensated for by adjusting the controlling vanes so that the rotor axis is erected 2½° forward in pitch and ¾° to the left in roll. The horizon bar is set level when the rotor axis is in this attitude. At other rates of turn there will be small errors, but these will be less than those in an uncorrected instrument.

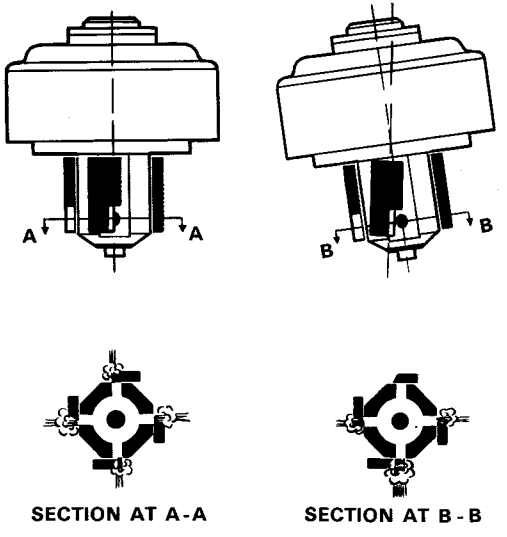


Fig 11 Operation of Pendulous Unit Line

34. **Performance.** The instrument has freedom in pitch of ±60° and in roll of ±110°. Movement in excess of these limits is prevented by spring stops. Erection after toppling takes between 10 and 15 minutes.

35. **Serviceability Checks.** The instrument should be checked for signs of external damage before start-up. After start-up the horizon bar should be checked to ensure that it indicates the aircraft ground attitude. In the air the instrument should be checked for correct and immediate indication of attitude in roll and pitch.

Artificial Horizon Mk 1E

36. The Mk 1E ariffícial horizon works on the same principle as the Mk 1B. A gull wing (*see* Fig 12) represents the aircraft, and the horizon bar is split to accommodate the outer gimbal support. The gyro rotor speed is approximately 15,000 rpm, and erection after toppling takes 11 minutes. The gyro has full freedom in roll, but is limited to ±85° of pitch indication.

37. **Errors.** The Mk 1E instrument is subject to errors caused by the effect of horizontal forces acting on the pendulous unit and on the vanes controlling the air jets (*see* para 33). The errors occuring in a rate 1 turn at 260 knots are compensated for by adjust-

Fig 12 Artificial Horizon Mk 1E

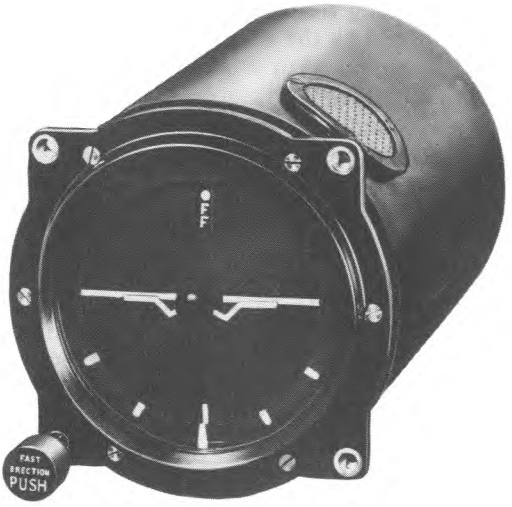


Fig 13 Artificial Horizon Mk 3C

ing the controlling vanes so that the gyro exists is erected 2½° forward and ¾° left. The serviceability checks are given in para 35.

Artificial Horizon AIM 300

38. The AIM 300 artificial horizon is very similar to the Mk 1E, with controlled precession (*see* paras 17–22) after ±85° of pitch.

Artificial Horizon Mk 3C

39. The Mk 3C artificial horizon is an electrically driven instrument which provides a continuous indication of aircraft attitude in pitch and roll. The instrument is designed to operate from an aircraft power supply of 115V, 400 Hz 3 phase ac. The gyro rotor speed is approximately 22,500 rpm. The instrument incorporates a power failure indicator and a fast erection system. It has full freedom in roll but is limited to ±80° of pitch indication.

40. **Power Failure Indicator.** The power failure indicator is attached to the gimbal ring immediately behind the sky plate (*see* Fig 13). The indicator flag is marked OFF and becomes visible when it moves into alignment with the aperture in the sky plate. When the power supply is on and all three phases are available, three solenoids act to keep the flag to one side. Should any one of the phases fail a control spring positions the flag in alignment with the aperture.

41. **Normal Erection System.** The gyro assembly is constructed so that it is slightly bottom heavy and the gyro axis settles in the vertical when the instrument is at rest. When power is applied, gravity sensitive mercury switches (*see* Fig 14) feed a 20V output from a transformer to activate the roll and pitch torque motors which correct any residual tilt. If the gyro topples in flight, the action of the torque motors is to re-erect the gyro at a rate of 5 deg/min. A roll cut-out switch is attached to the gimbal, isolating the roll mercury switch when the angle of bank exceeds 10° (*see* Fig 15). This prevents the gyro being erected to a false vertical during a turn.

42. **Fast Erection System.** Pressing the fast erection button connects the 115V supply to the erection circuits and bypasses the roll cut-out (*see* Fig 16). The fast erection rate is 180 deg/min. The fast erection button

Printed in the UK for HMSO Dd8914835 1/87 NB40500 3347

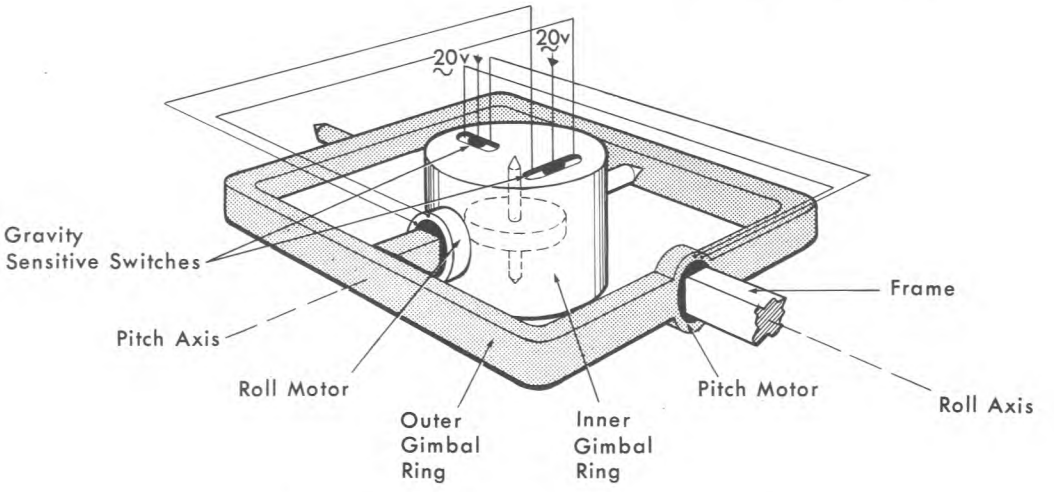


Fig 14 Operation of the Gravity Sensitive Switches

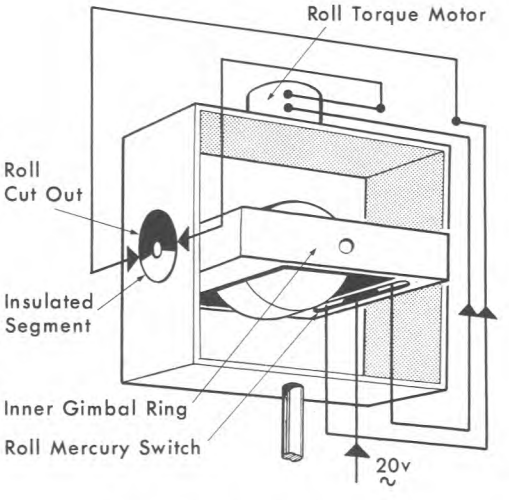


Fig 15 The Roll Cut-out Switch

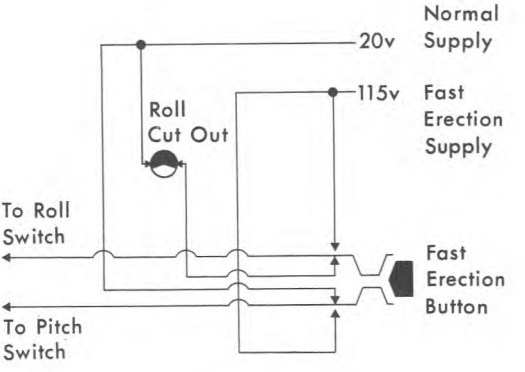


Fig 16 Fast Erection Circuit

should not be pressed until at least 15 seconds after power has been switched on, as the gyro must be rotating at speed before high precession rates are applied. Fast erection must be applied to obtain gyro erection if the axis is tilted by more than 10° in roll, in order to bypass the cut-out.

43. **Errors**. The pendulosity of the gyro assembly gives rise to errors in a turn. The error occuring in a rate 1 turn at 260 knots is compensated for by the roll mercury switch which is tilted .6° from the level, causing the spin axis to be erected with a .6° tilt to starboard. The horizon bar and bank indicator are set level when the axis is so tilted.

44. **Serviceability Checks**. Before flight check that the instrument is undamaged.

After switching on power check that the OFF flag disappears and the horizon bar erects. If the horizon bar fails to erect and the OFF flag is not visible, press the fast erection button. In flight check that the instrument gives correct and immediate indication of changes in the pitch and roll attitude of the aircraft.

Artificial Horizon Mk 4

45. The Mk 4 artificial horizon operates from an aircraft supply of 115V, 400 Hz 3 phase ac. The gyro rotor attains its operating speed of 23,000 rpm after one minute. The instrument incorporates a power failure indicator, roll and pitch cut-outs, fast erection circuits and a pitch-bank erection system. It has full freedom of roll but is limited to $\pm 85°$ of pitch indication.

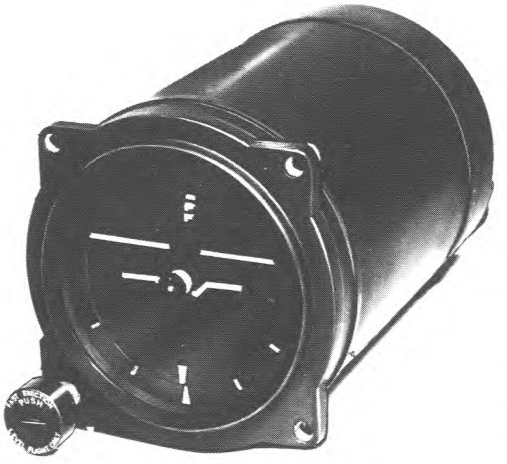


Fig 17 Artificial Horizon Mk 4

46. **The Power Failure Indicator**. If the power supply or any one of its phases fails an OFF flag appears in an aperture in the sky plate (*see* Fig 17). The OFF flag is restrained by eddy currents which act on a disc, creating sufficient torque to overcome the action of a spring when full power supplies are available.

47. **Normal Erection System**. The normal erection circuit is connected to a reduced voltage supply from an auto-transformer and operates as shown in Fig 14. If either the pitch or the roll mercury switch is tilted between 5″ and 10° along its principle axis the respective torque motor is activated, and the gyro axis is precessed back to the vertical at 5 deg/min. If the tilt angle exceeds 10° the input electrodes to the mercury switches are isolated as shown in Fig 18. Accelerated flight equivalent to .18g also operates the cut-out. The cut-outs prevent the gyro being erected to a false vertical.

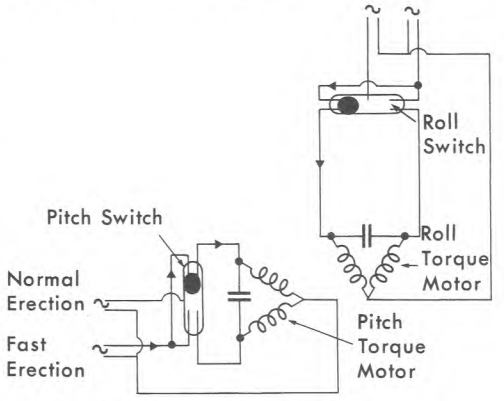


Fig 18 Normal and Fast Erection Circuits

48. **Fast Erection System**. When the fast erection button is pushed the full 115V supply is applied to the erection circuits, giving a precession rate of 120 deg/min. The circuits marked with arrows in Fig 18 represent the live connections when the gyro axis is tilted more than 10° left and 10° forward and the fast erection button is pressed. Since the gyro assembly has no pendulosity the fast erection circuits must generally be used for initial erection. The button must not be pressed until the gyro is approaching its operating speed (about 15 seconds after switch-on) nor must it be pressed for longer than one minute, otherwise the torque motors may be damaged. The fast erection button should only be used when the aircraft is level and not accelerating.

49. **The Pitch-Bank Erection System**. If the gyro axis is displaced from the vertical,

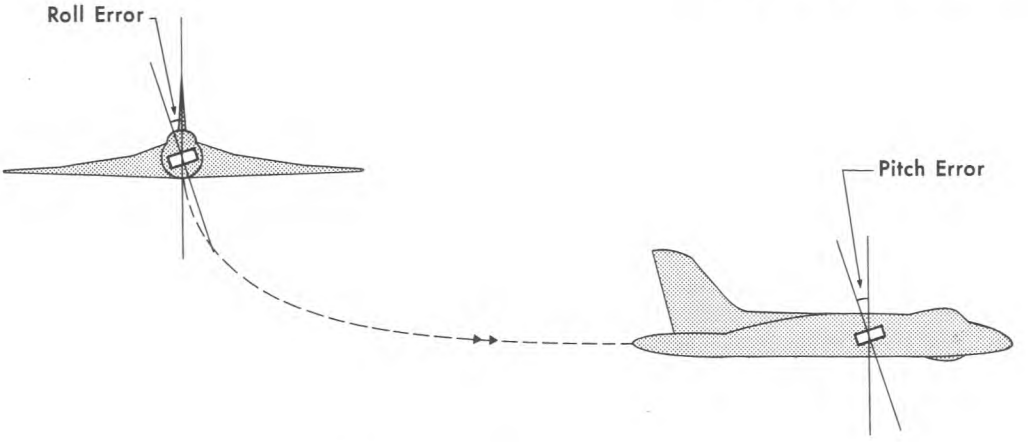


Fig 19 Roll Tilt Changing to Pitch Tilt in a Turn

tilt in roll at the beginning of a turn will appear as a tilt in pitch after an aircraft has turned through 90° (*see* Fig 19). The Mk 4 series instruments incorporate a pitch-bank erection system which transfers control of the roll torque motor to the pitch mercury switch during turns. The system is shown in Fig 20. Changeover switches actuated by lateral accelerations of .18g (equivalent to 10° of bank) transfer control of the roll torque motor so that the pitch switch actuates both the pitch and the roll torque motors simultaneously. In Fig 20 the aircraft is turning to the left and the gyro axis is tilted forward in pitch by less than 10°. During a sustained turn roll error progressively appears as pitch error and vice-versa. The erection system causes the gyro axis to be precessed back to the vertical in a spiral path, the connections being arranged to give the following sequence:

a. *Left Turn*. When the pitch torque motor is precessing the gyro forward the roll torque motor precesses it to the left. When the pitch torque motor is precessing the gyro axis aft the roll torque motor precesses it to the right.

b. *Right Turn*. The roll torque motor precesses the gyro axis in the opposite directions to those given for a left turn.

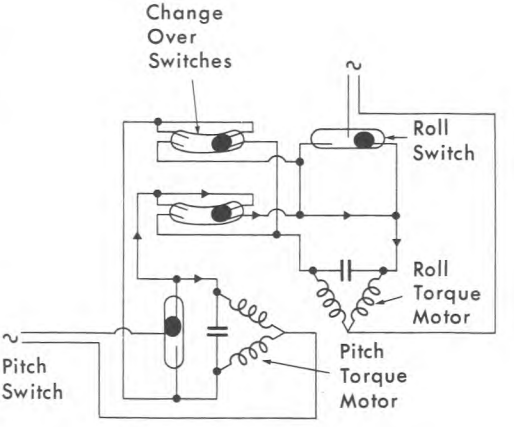


Fig 20 Operation of the Changeover Switches

50. **Errors**. The Mk 4 instrument is subject to acceleration errors in a turn. The maximum error in a sustained rate 1 turn is 3.8°.

51. **Serviceability Checks**. The serviceability checks are as given in para 44 except that the fast erection button is pressed to obtain initial erection.

Artificial Horizon Mk 4A

52. The Mk 4A artificial horizon is similar to the Mk 4. It has been modified by the

addition of a levelling controller which erects the gyro after switch-on at 200 deg/min. After the initial starting cycle, lasting 20 seconds the instrument functions in the same way as the Mk 4, and the fast erection button must be pushed if the gyro axis topples more than 10° from the vertical.

Artificial Horizons Mk 4B, C, D and E

53. The Mk 4B, C, D and E artificial horizons are all similar to the Mk 4, the main differences being in the appearance of the instrument face:

a. *Mk 4B*. The Mk 4B is identical to the Mk 4, but has been modified to incorporate an improved type of push switch assembly.

b. *Mk 4C*. The Mk 4C is illustrated in Fig 21. It has a two colour half inch deep horizon bar. The two colours, together with the white divergent lines on the lower half, give positive indication of inverted flight. Pitch circles are etched on the glass at 20° and 40°, with intermediate arcs at 10°, 30° and 50°. There are additional roll scale markings at 10° and 20°.

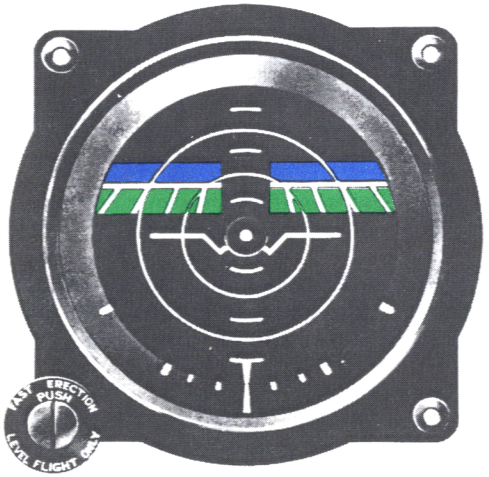


Fig 21 Artificial Horizon Mk 4C

c. *Mk 4D*. The Mk 4D is illustrated in Fig 22. It has a two colour half inch deep horizon bar, the two colours being

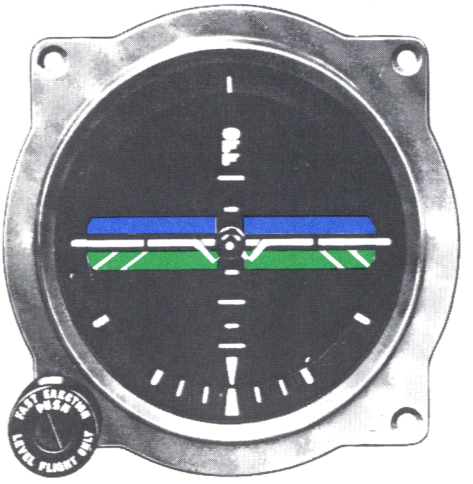


Fig 22 Artificial Horizon Mk 4D

separated by a fluorescent white line edged with black. Superimposed on the green (lower) part of the bar are two divergent pairs of fluorescent white lines. Fluorescent pitch marks indicate 10°, 20°, 30°, 40° and 50° nose up sense, and 10° and 20° in the nose down sense. The roll scale is calibrated at 10°, 20°, 30°, 60°, 90° and 180°.

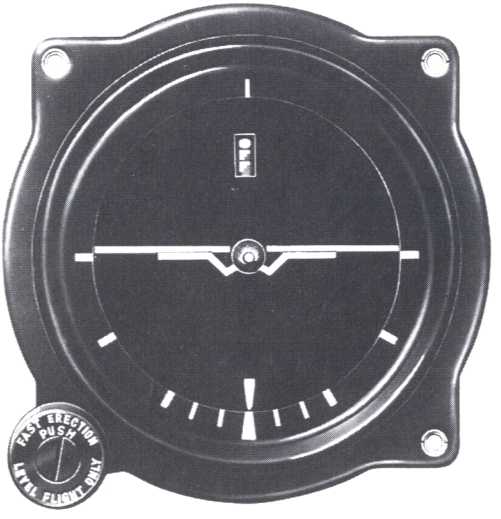


Fig 23 Artificial Horizon Mk 4E

d. *Mk 4E.* The Mk 4E is similar to the Mk 4. The roll scale has additional markings at 10°, 20°, 90° and 180°. The pitch-bank erection system operates when the angle of bank exceeds 7°. Post Mod Inst A/301 versions of the Mk 4D have been similarly modified.

Artificial Horizon Mk 5

54. The Mk 5 artificial horizon is an electrically driven instrument designed for use in helicopters, and is mounted on a sloping instrument panel. It operates from an aircraft power supply of 115V, 400 Hz 3 phase ac. The gyro rotor speed is 22,500 rpm. The instrument incorporates a power failure indicator, pitch and roll cut-outs, a pitch cut-out bypass and fast erection circuits. The gull wing representing the helicopter is mounted on a perspex bridge which can be raised or lowered, by turning the pitch datum knob, to give a variation in attitude datum from 5° nose up to 10° nose down (*see* Fig 24). It has full freedom in roll but is limited to +80° and −70° of pitch indication.

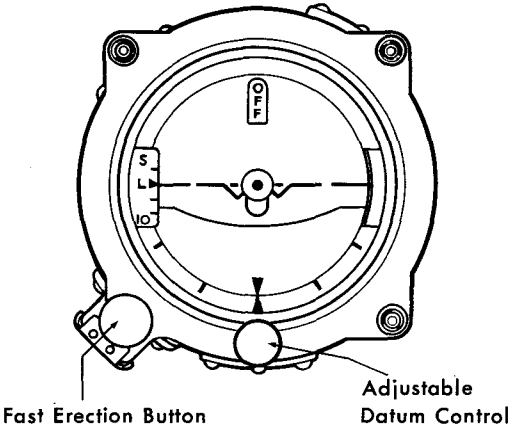


Fig 24 Artificial Horizon Mk 5

55. **Power Failure Indicator.** The power failure indicator consists of an OFF flag which appears in an aperture in the sky plate when any of the following malfunctions occur:

a. The power is off.

b. The voltage is below the specified limit.

c. An open circuit occurs in one of the three phases in the gyro stator winding.

d. The phase rotation is incorrect.

The OFF flag is restrained from appearing in the aperture by eddy currents acting on a rotor which create sufficient torque to overcome the action of a control spring.

56. **Normal Erection System.** The normal erection voltage (20V) is fed to the centre electrodes in the mercury switches from a transformer. When the switches are tilted by between 5′ and 10° along their principal axes, their respective torque motors are energized and precess the gyro back to the vertical at 5 deg/min. If the tilt angle exceeds 10° (equivalent to .18g) the centre electrodes are isolated and no precession takes place. When the roll angle exceeds 35° the normal erection supply is connected to the fast erection electrodes in the pitch switch (*see* Fig 25) via the pitch cut-out bypass switch, and pitch erection is maintained even when the pitch switch is subject to an acceleration in excess of .18g along the principal axis.

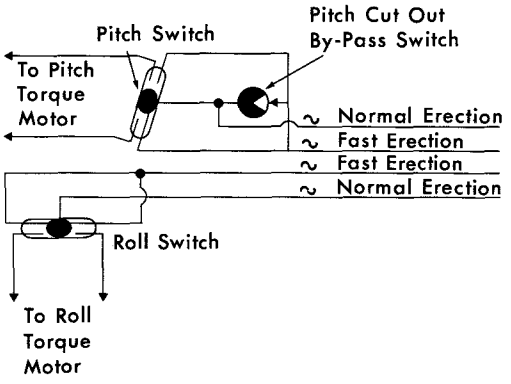


Fig 25 Normal and Fast Erection Circuits

57. **Fast Erection System.** Pushing the fast erect button energizes the fast erection circuits with 115V (*see* Fig 25). The gyro is

erected at 120 deg/min. The button should only be pressed when the aircraft is level and not accelerating.

58. **Errors**. The gyro assembly is constructed so that it is slightly pendulous. Centrifugal force acting on the assembly in a turn is compensated for by misaligning the pitch switch 10° starboard (*see* Fig 25). Thus during a turn centrifugal force acts on the pitch switch and the resulting precession of the gyro axis corrects the error. The purpose of the pitch cut-out bypass is to ensure that the compensation continues to be applied during steep turns, when the component of centrifugal force acting along the principal axis of the pitch switch may exceed .18g. Additional compensation is provided by tilting the roll mercury switch by .3°; the horizon bar is set level when the gyro axis is tilted .3°.

Artificial Horizon Mk 5A

59. The Mk 5A instrument is similar to the Mk 5 but is limited in pitch indication to $+80°$ and $-65°$.

Artificial Horizons Mk 6 Series

60. The Mk 6 series instruments are similar in appearance and method of operation (apart from the variants described in paras 66 and 67). They operate from a 115V, 400 Hz 3 phase ac aircraft power supply at 23,000 rpm and are controlled by control inverter boxes. The instruments incorporate power failure indicators, roll and pitch cut-outs, fast erection circuits and a pitch-bank erection system (except for the Mk 6J, K and L). The aircraft is represented by a fixed gull wing and pitch markings show 10°, 20°, 30° and 40° nose up sense (apart from the Mk 6E which has pitch markings at $\pm 10°$ only). The instruments have full freedom of roll but are limited to $\pm 85°$ of pitch indication (apart from the Mk 6S which is limited to $+72\frac{1}{2}°$ and $-77\frac{1}{2}°$). AP 112G—0306-16 provides full details of each variant and the facilities available from the compatible control boxes.

Fig 26 Artificial Horizon Mk 6E

61. **Control Inverter Boxes**. The control inverter boxes associated with the MK 6 series instruments provide combinations of some of the following facilities:

a. Fast erection supplies (115V).

b. Normal erection supplies (20V).

c. Power failure indication and gimbal centring circuit.

d. Radio interference suppression circuits.

e. Automatic lock-in of the fast erection circuit.

f. A transistorized inverter to provide continuous operation of the instrument utilizing dc from an emergency battery, the changeover being automatic.

62. **Power Failure Indicator**. The power failure flag is orange with black diagonal stripes. It is displayed when the power supply is off or when it is not of the correct value.

63. **Normal Erection System**. The normal erection rate is 3 deg/min, the method of operation being similar to that described in para 47).

64. **Fast Erection System**. The fast erection circuits are energised by pushing the fast erection button. If the control inverter box provides automatic lock-in the fast erection circuits remain energised until the gyro is erect; in other cases the button must be held in. The fast erection rate is 105 deg/min, and the circuits are similar to those

described in para 48. The aircraft must be level and not accelerating when the button is pressed.

65. **Pitch-Bank Erection System.** The pitch-bank erection system described in para 49 is fitted to Mk 6 series instruments other than the Mk 6J, K and L.

66. **Mk 6J, K and L.** These instruments are designed for mounting on helicopter instrument panels which are tilted during cruising flight. The erection system is arranged so that cut-out switches isolate both the roll and pitch erection circuits when a lateral acceleration greater than .18g is experienced.

67. **Mk 6S.** The Mk 6S instrument is designed for mounting on a helicopter instrument panel. It has a pitch-bank erection system which is energised when a lateral acceleration of .12g (equivalent to 7° of tilt) is experienced.

68. **Serviceability Checks.** Before flight check that the instrument is undamaged. After switching on power check that the power failure flag disappears and the horizon bar erects. The instrument should give correct and immediate indication of any change in the pitch and roll attitude of the aircraft. Errors are similar in value to those found in the Mk 4 (*see* para 50).

HORIZON GYRO UNITS

Introduction

69. Artificial horizons fitted with pick-offs on the roll and pitch axes are called horizon gyro units (HGUs). They provide outputs of aircraft attitude in roll and pitch for flight director instruments such as the zero reader.

Horizon Gyro Units Type B Series

70. The type B series HGUs are basically improved MK 3 artificial horizons (*see* Fig 27). The pitch mercury switch is misaligned

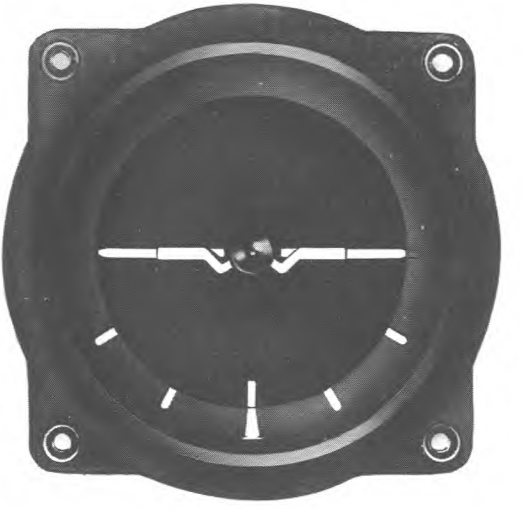


Fig 27 HGU Type B9

with the fore and aft axis so that it produces an error signal in sustained turns which causes the pitch torque motor to correct errors caused by horizontal forces acting on the gyro assembly. The pitch indication is limited to ±80° and there is full freedom in roll. There is no roll cut-out, fast erection or power failure indicator in this instrument.

Horizon Gyro Unit Mk 1

71. The HGU Mk 1 is similar in construction to the Mk 3C artificial horizon (*see* Fig 13). It has fast erection circuits and a power failure indicator. The erection system is controlled by mercury switches, and a roll cut-out commutator isolates the roll torque motor if the roll angle exceeds 10°. In addition the roll mercury switch is tilted so that the gyro spin axis is erected with a tilt of 1° starboard to compensate for horizontal forces acting on the gyro assembly in a turn. The pitch indication is limited to ±80° and there is full freedom in roll.

Horizon Gyro Units Mk 2 and 2C

72. The HGUs Mk 2 and 2C are similar in construction to the Mk 5 and 5A artificial horizons. They are designed for use in helicopters.

Fig 28 HGU Mk 1

Horizon Gyro Units Mk 3 and 3A

73. The HGU Mk 3 is similar in construction to the HGU Mk 1 except that the pitch and roll mercury switches are fitted with cut-outs operating at 10° of tilt (equivalent to .18g). In addition the pitch mercury switch is misaligned with the fore and aft axis so that it produces an error signal in sustained turns which causes the pitch torque motor to correct errors caused by horizontal forces acting on the gyro assembly. The roll and pitch erection rates are lower than in the HGU Mk 1 with a consequent reduction in acceleration errors. The Mk 3A instrument is identical to the Mk 3 apart from the addition of an external quick starting control.

Horizon Gyro Units Mk 5 and 5A

74. The HGUs Mk 5 and 5A are similar in construction to the Mk 6 artificial horizon. These instruments are used in conjunction with control boxes to provide erection supplies and emergency circuits. A pitch-bank erection system similar to that described in para 49 is actuated at a roll angle of 7° (eqivalent to a lateral acceleration of .12g). The pitch cut-out is actuated at 10° tilt (equivalent to a fore and aft acceleration of .18g). The roll cut-out is actuated at 7° of tilt. Pitch indications are limited to ±85°. A fast erection push button is used for initial erection (except when a type B control box is used) and for re-erection after toppling. The power failure flag is actuated by power supply failure; it is orange with black diagonal stripes.

FH 30 SERIES HORIZON GYRO UNITS

Introduction

75. The Type FH 30 Series horizon gyro is an AC powered, pneumatically erected and monitored, gyroscopic instrument, which pictorially displays and electrically transmits information proportional to the aircrafts pitch and roll attitude.

76. The instrument consists of a gyroscope which spins about the vertical axis, mounted in an inner gimbal which pivots about the pitch axis of an outer gimbal. The outer gimbal is pivoted about the roll axis of the instrument (*see* Fig 29).

77. A two coloured, quasi-spherical attitude display barrel is suspended in the outer gimbal and is coupled to the inner gimbal by a drive wire. The display consists of the two coloured barrel in front of which a shroud and roll pointer are fitted. The roll scale and aircraft symbol are positioned behind the glass dial. Aircraft attitude is indicated by the position of the horizon line (the intersection of the two colours of the barrel) and the roll pointer, in relation to the aircraft symbol and roll scale. Pitch angle is indicated by a graduated scale over the range 90° climb to 90° dive in 5° increments, the graduations being annotated at 30°, 60° and 90°.

78. **Gyro Operating Limits.**

a. Rotor speed: 21,500 rev/min (nominal).

b. Run up time: 90% full speed within 100 sec (maximum).

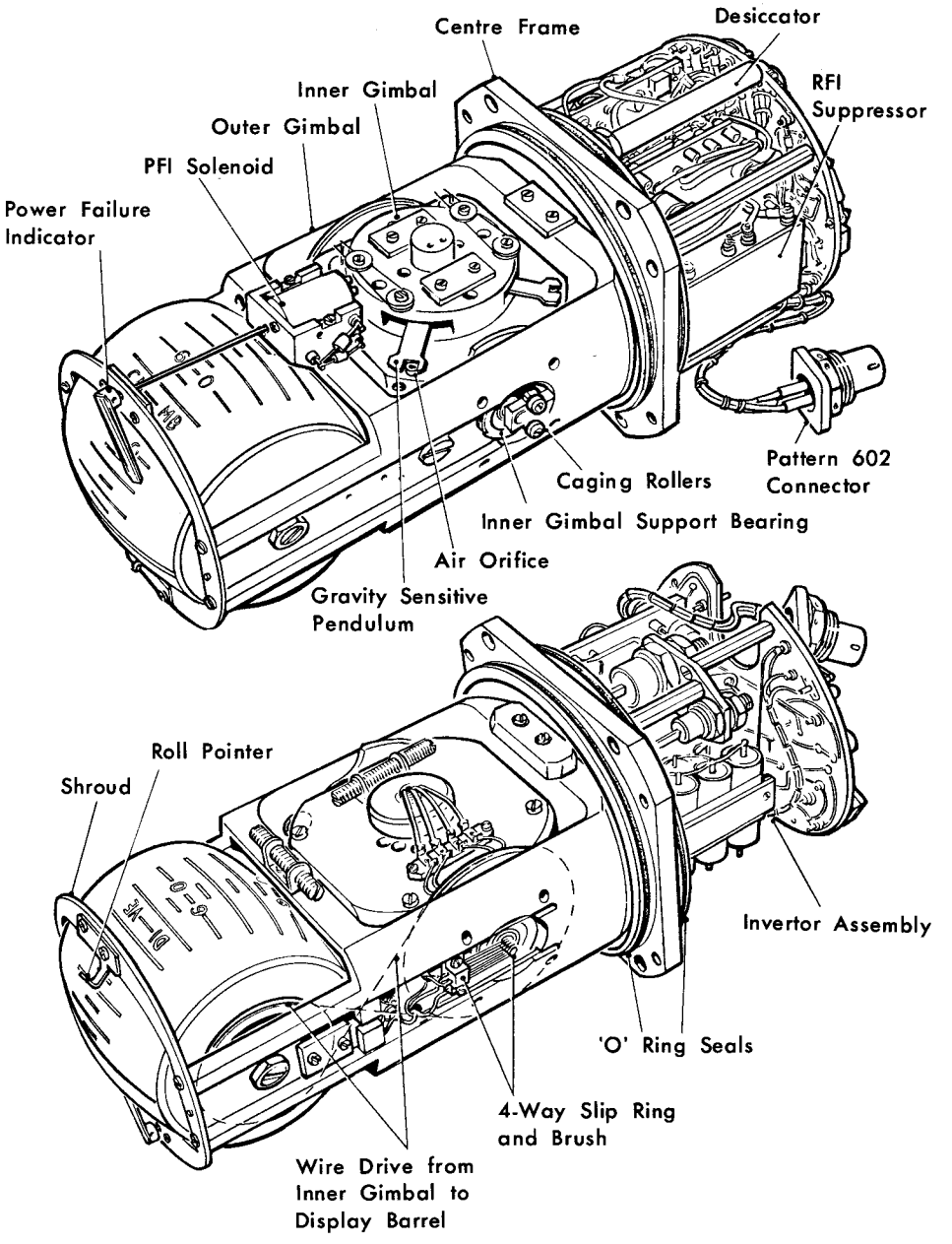


Fig 29 FH 30 Series Instrument Details

equal in both aircraft axes one vane will adopt its neutral position and only one jet will be left operating to correct the residual errors in both axes simultaneously. Since now only one jet is providing an erecting torque for both axes the errors in pitch and roll will only be corrected at half the nominal rate.

The specified precession rate for the FH 31 for example, is 3° ± 1°/min about the pendulum axis. Considering two instruments at the tolerance limits, the actual precession rates about the pitch and roll axis can vary between a maximum of 5.7°/min for a tilt in one axis on the "high" instrument to a minimum of 1.4°/min for an equal tilt in both axes on the "low" instrument. Thus an apparently slow erection rate of 1.4°/min can be experienced with a fully serviceable instrument.

84. To illustrate this further, if both the instruments had a verticality error of 5° in one axis and 3° in the other, the time taken to correct these errors would vary between 1.4 mins and 2.8 mins ie double the time.

85. **Single Vane Cut-off.** The tolerances on the settings of the two cut-off springs can give rise to the situation where one vane is in the cut-off position (closing two diagonally opposite jets) whilst the other vane is still controlling. This situation can only occur in the transition region between full erection control and complete cut-off (ie a gyro tilt angle of 7°–12°). The effect of this phenomenom depends on which vane is in cut-off in relation to the direction of the verticality error. It can result in the error in one axis being corrected whilst appearing to increase the error in the other axis at the same rate until the errors are equal in both the pitch and roll axes when the precession will cease and the gyro will be left with a residual error in both axes. The instrument may recover itself from this situation but the only way to ensure recovery is to manually cage the system.

86. **Effect of Gyroscopic Inertia.** Any verticality error is seen by the pilot as a combination of pitch and roll errors. During a turn in the cut-off condition, because of gyroscopic inertia, these errors will appear to transfer from one axis to the other, eg a pitch error will be wholly transferred into the roll axis after a 90° turn (*see* Fig 19).

Instrument Caging

87. **Caging Mechanism.** The FH 30 series instruments employ a mechanical caging system whereby the gyro is caged to a datum position referenced to the instrument case. Pressing the caging knob causes an annular cam to slide forward and engage two rollers on the inner gimbal (*see* Fig 32). These rollers then follow the cam profile until a detente is reached. The rollers (and hence the whole gyro/gimbal assembly) will remain in this caged position until the caging knob is released. A fairly high push force (nominally 8 lbf) is required to fully depress the caging knob. The amount of cam travel must allow for the maximum possible gimbal displacement and therefore when caging with relatively small errors there is a large travel of the caging knob before caging takes effect. To ensure full caging the caging knob must be fully depressed.

88. The rate at which the gimbals achieve their caged positions depends on their starting positions but may take as long as 45 secs. Caging with typical errors of say 10° in pitch and bank should take approximately 10 secs. The tolerance in the caged position is ±2°.

89. Depending on their initial positions, the gimbals and the display may describe a damped oscillatory motion with possibly several transitions through zero, particularly in roll, before reaching the fully caged position. The caging knob *must* be held fully depressed until all motion of the display has ceased. Releasing the knob on one of the transitions through zero could induce large errors.

90. **Caging Flag.** Pressing the caging knob causes the caging flag to appear as the cam

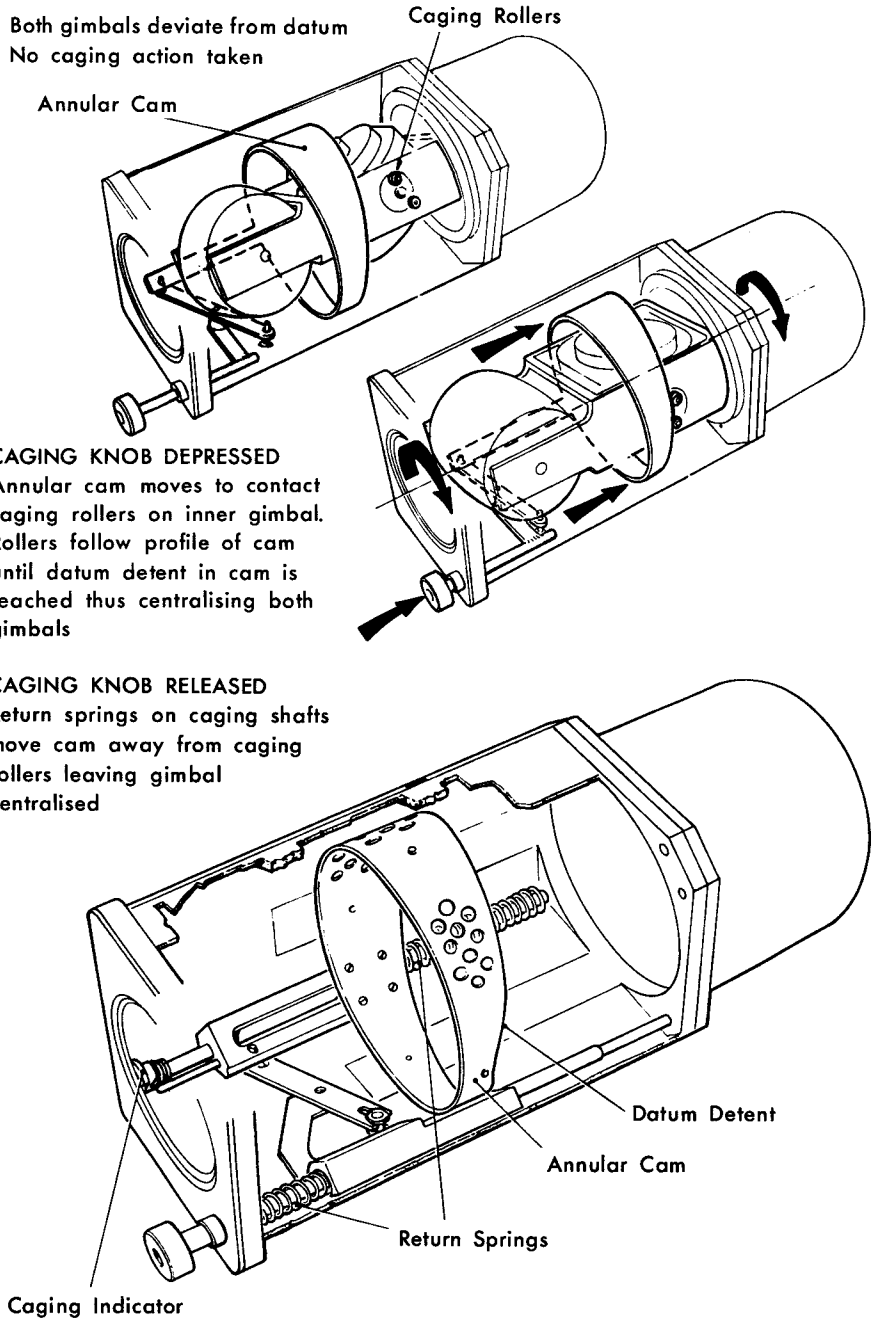


Fig 32 Caging Mechanism

starts to slide and may be fully in view before the cam engages the rollers. The purpose of the caging flag is to show that the caging mechanism is not in its fully "parked" position. It should not be used as an indication that the instrument is fully caged. Should the flag appear in view without the caging knob having been pressed then the caging mechanism may be stuck and inhibiting full gimbal movement. In this case large errors could be induced and the instrument is unserviceable.

91. **Caging in Flight.** Caging can be initiated in flight at any aircraft attitude, the gyro will always cage to the datum position, but the aircraft *must* be in S and L flight when the caging knob is released. The caging datum takes account of any panel tilt so that if the aircraft is at zero pitch and bank angles the gyro spin axis will be set to the true vertical $\pm 2°$ with the normal erection system removing any small residual error.

92. If the aircraft is not at zero pitch and bank angles the actual aircraft attitude at the instant the knob is released will immediately be translated into a display error, eg if the caging knob is released at 6° nose up and 10° left bank the display will aquire a 6° nose down and 10° right roll error. As explained in para 7 if the error exceeds 12° then the instrument will definitely be in erection cut-off and the error will not be corrected.

93. The pitch angle due to aircraft incidence will also be translated into pitch error. In 1g flight this error is unlikely to exceed the limit for full erection control (7°) therefore this "incidence induced" pitch error will be corrected at the normal erection rate provided unaccelerated flight is maintained.

Summary of FH30 Series HGU's.

94. The following are the main characteristics of the FH30 Series Attitude Indicators:

a. *Normal Erection Rates* in flight can vary between 5.7°/min and 1.4°/min in aircraft axes.

b. *Erection Cut-off* will occur between 7° and 12 to 14.5° (depending on Mark). The instrument can exhibit the characteristic of correcting an error in one axis whilst simultaneously increasing the error in the other axis. If this occurs the instrument should be caged in *S and L* flight.

c. *Pitch/Bank Errors.* Any attitude error present during a turn above cut-off will transfer between pitch and roll axes, full transfer between axes for each 90° of heading change.

d. *Tilted Instrument Panels* cause displayed (or Geometric) errors which vary with bank angle ie $2 \times$ Tilt Angle when inverted.

e. *Caging Mechanism.* The instrument *must* be caged in S and L flight otherwise the actual aircraft attitude (including aircraft incidence) at the instance of releasing the caging knob will be translated into an attitude error. A high push force is required to fully cage the instrument which may take up to 45 secs. The Caging Flag indicates the caging mechanism is not in its operating position and the instrument should be treated with extreme caution.

f. *Maneouvering.*

(1) *Gentle Maneouvres* below the full erection cut-off "g" can, over a period, generate significant errors.

(2) *Frequent Hard Maneouvres* above cut-off "g" may result in errors being corrected very slowly.

(3) *Extreme Pitch Maneouvres* approaching or passing through the vertical can generate significant attitude errors particularly if small amounts of left roll are applied during the maneouvre.

Printed in the UK for HMSO by Hobbs the Printers Ltd Southampton
(2089/86) Dd8908650 9/86 G436

PART 2

SECTION 3

REMOTE INDICATING COMPASSES

Chapter

1 **Introduction to Gyro-Magnetic Compasses**
Annex: **Fluxgate and Smiths Detector Unit Theory**

2 **Mk 4 and Mk 5 Series Gyro-Magnetic Compasses**

3 **Military Flight System, Mk 1 Series — Compass Element**

4 **Mk 7 Series Gyro-Magnetic Compasses**

5 *Spare*

6 **Polar Path Compass**

7 *Spare*

8 **C2G and C2J (GM8A) Gyro-Magnetic Compass Systems**

9 *Not Used*

10 **C12 Gyro-Magnetic Compass System**

CHAPTER 1

INTRODUCTION TO GYRO-MAGNETIC COMPASSES

CONTENTS

	Paras
Introduction	1—4
General	5
Basic Components	6—9
Fluxvalve Theory	10—22
The Transmission/Display System	23—25
HEADING ERRORS INDUCED BY THE FLUXVALVE	
General	26
Detector Tilt Error	27—39
THEORY OF THE GYRO-MAGNETIC COMPASS	
General	40
Mechanization	41—49
GYRO-MAGNETIC COMPASS SYSTEM ERRORS	
Fluxvalve Tilt Errors	50—56
Northern Instability	57—58
Hang-Off Error	59—64
Gimbal Error	65
Transmission Errors	66
Compass Swinging Errors	67
Variation and Deviation Errors	68
SIMPLE ANALYSIS OF ERRORS	
Hang-Off Error	69
Vehicle Movement	70
Sample Error Analysis	71—76
METHODS OF ERROR REDUCTION	
General	77—83
A Refined Compass System	84—85
Annex: Fluxgate and Smiths Detector Unit Theory	

Introduction

1. The direct indicating compass is subject to errors due to two main causes, magnetic fields of the aircraft structure and flight accelerations. In the case of direct indicating compasses, magnetic fields due to aircraft magnetism are accentuated by the necessary positioning of the compass so that it can be read by the pilot/navigator, *ie* in the cockpit where the deviating effects due to hard iron (including DC fields) and soft iron fields are large. The pendulously suspended magnet system is subject to errors due to accelerations.

2. In addition to these errors, the effect of reduction in the directional force acting on the detecting element renders the direct reading instrument unreliable in high magnetic latitudes where the horizontal component of the Earth's magnetic field is weak. This has the effect of making the compass sluggish in indicating a change of heading. After an alteration of heading, the detecting element will oscillate for a considerable time before settling down.

3. A further disadvantage of the direct indicating compass is that indications of direction can be given at only one position in the aircraft. Since the Earth's magnetic field strength cannot provide sufficient torque for driving repeater indicators from one master detector element, separate compass systems must be provided for each crew member requiring a heading readout.

4. The remote indicating compass was developed to reduce the errors of the direct indicating compass and to evolve an instrument giving automatic continuous direction which could be fed to other instruments. Although a number of these systems have been designed using different detecting and stabilizing techniques, the gyro stabilized remote indicating (gyro-magnetic) compass gradually evolved.

General

5. The gyro-magnetic compass consists essentially of a magnetic compass whose indications are stabilized gyroscopically so that the effects of turning and acceleration errors are reduced. A gyroscope is unaffected by changing magnetic fields or by normal aircraft accelerations but its heading indications may be inaccurate due to the effect of precessional forces caused by friction, incorrect balance *etc*. Since the commonly used detecting element, the flux-valve, is pendulously suspended, it is affected by accelerations. Therefore, the principle underlying the gyro-magnetic system is to integrate the heading indication of the magnetic compass with the directional properties of a gyroscope so that a compromise between the two is achieved. The net result is to reduce the individual errors of each. The technique most commonly used is to reference the azimuth gyroscope initially to the magnetic meridian and to maintain the relationship by applying precessional forces to the gyroscope based on long term magnetic azimuth information from the fluxvalve detector. The degree of control of the fluxvalve over the gyroscope, or the monitoring rate, is of considerable importance. For example, in a turn the fluxvalve heading is likely to be in error so the control rate must be engineered so that the induced heading is that of the gyro. At the same time there must be sufficient control to correct the gyro drift.

Basic Components

6. When considering the various units associated with the design of gyro-magnetic compass systems, it is logical to break them down into three basic components, the fluxvalve, the transmission and display system, and the gyroscope.

7. **The Fluxvalve.** A fluxvalve is the detecting element of many remote indicating compasses and it provides the long term azimuth reference for the gyroscope. It is usually remotely located in a wing tip or fin in an area relatively free from aircraft magnetic disturbances. Other devices which function in essentially the same manner are the fluxgate and the Smiths magnetic detector unit. The Annex to this chapter deals briefly with the theory of the fluxgate and the Smiths unit.

8. **The Transmission and Display System.** The transmission system provides data transmission between compass system components and to associated equipments. Control synchros are usually used for this purpose. For a heading display, the rotor of a control receiver (CT) can be attached to a digital counter, a movable pointer against a fixed card or a movable card against a fixed lubber line.

9. **The Gyroscope.** Short term azimuth stability is typically provided by a two degree-of-freedom gyro with the input axis vertical, *ie* the spin axis in the local horizontal plane.

Fluxvalve Theory

10. The fluxvalve, *see* Fig 1, consists of a sensitive pendulous element which is free to

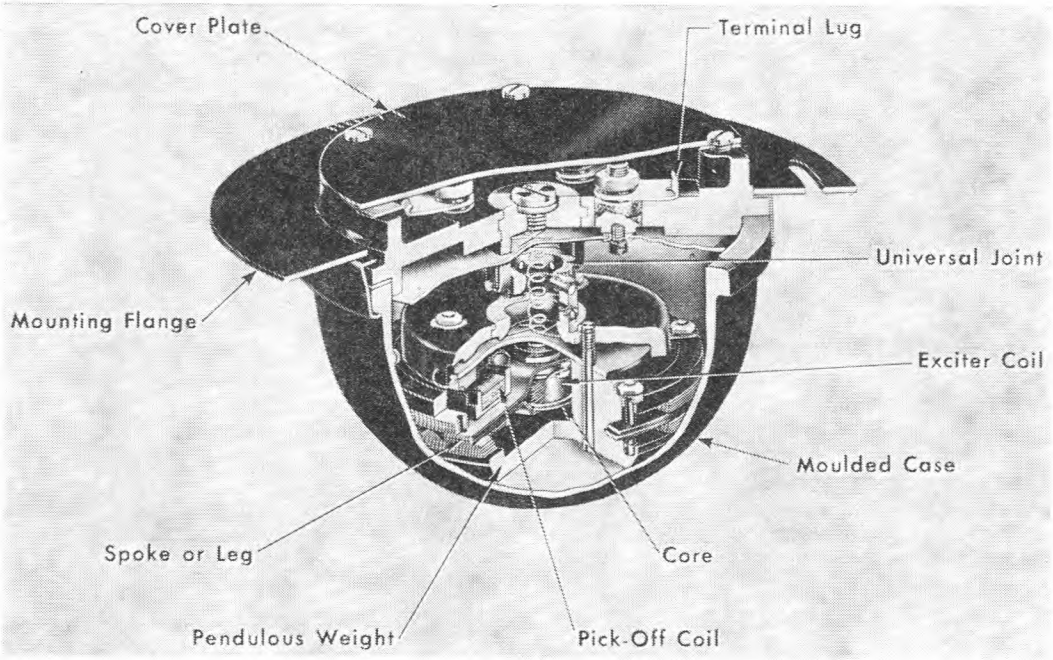


Fig 1 Fluxvalve

swing within limits (usually ± 25°) but fixed to the aircraft in azimuth. The element is suspended by a Hook's Joint with the whole assembly being hermetically sealed in a case partially filled with oil to dampen oscillations. A deviation compensator is usually mounted on top of the unit.

11. The pendulous detector element resembles a three-spoke wheel with the spokes 120° apart and slotted through the rim. The rim forms a collector horn for each spoke. The horns and spokes are made up of a series of metal laminations having a high magnetic permeability. Each spoke has a vertical cross-section similar to that shown in Fig 2. The spoke consists of two superimposed legs which are separated by plastic material and opened out to enclose the central hub cone. This cone has an exciter coil wound round it on a vertical axis, and each spoke has a pick-off coil wound round both legs on a horizontal axis. The exciter coil is fed with 400 Hz single phase AC. The output of the secondary or pick-off coil is an 800 Hz single phase AC current, the amplitude and phase representing the relationship of magnetic North to the aircraft longitudinal axis (magnetic heading).

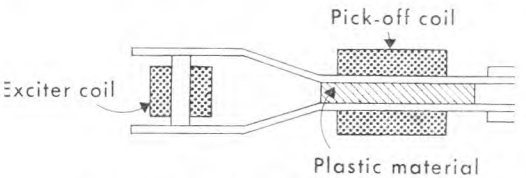


Fig 2 Vertical Cross-Section of Spoke

12. In order to appreciate the operation of the flux valve it is necessary to consider an individual spoke. The function of a spoke will be developed in a series of diagrams (Figs 3 to 10).

13. If a single coil is placed in a magnetic field, the magnetic flux passing through the coil is maximum when the axis of the coil is in line with the direction of the field, zero when the coil lies at right angles to the field, and maximum but of opposite sense relative to the coil when turned 180° from its original position. For a coil placed at an angle θ to a field of strength H (*see* Fig 3) the field can be resolved into two components, one along the coil equal to H cos θ and the other at right angles to the coil equal to H sin θ. The H cos θ component is parallel to the coil and is the effective flux producing element. Therefore, the total flux passing through the coil is proportional to the cosine of the angle between the direction of the coil axis and the direction of the field. The coil output curve is

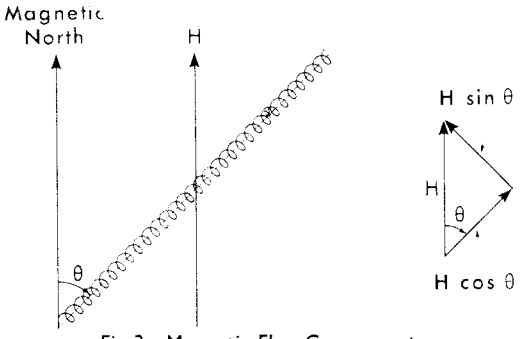


Fig 3 Magnetic Flux Components

shown at Fig 4. If the coil is in the horizontal plane with its axis parallel with the aircraft longitudinal axis, its output is affected by the horizontal component of the Earth's magnetic field and the flux passing through the coil is proportional to the magnetic heading of the aircraft.

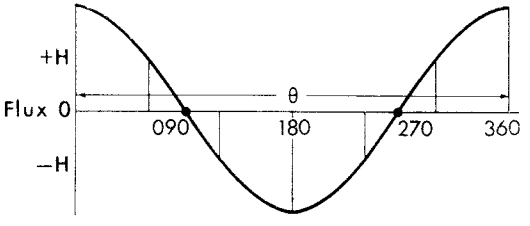


Fig 4 Variation of Flux with **0**

14. Unfortunately, the simple concept just described cannot be used without modification as a heading reference system for two important reasons. Firstly, the voltage induced into a coil depends on the rate of change of flux. Therefore, once established on a heading, there would be no change of flux and, consequently, no induced voltage. Secondly, the output of the simple detection device would be subject to heading ambiguity, *ie* there are always two headings which cause the same induced output voltage. Therefore, the problem that must be solved is how to produce an output waveform which is proportional in some way (frequency, phase or amplitude) to the components of the Earth's field and linked with the coil. This is achieved in the fluxvalve by introducing an alternating magnetic field in addition to the static field caused by the horizontal component of the Earth's magnetic field.

15. Fig 5 shows the relationship between flux density (B) and magnetizing force (H) known as the hysteresis loop for the permalloy commonly used in the legs of the flux valve spokes. Permalloy has a very high magnetic permeability (μ = B/H) and a corresponding low hysteresis loss. In the following discussion the hysteresis loop is represented by a single line curve.

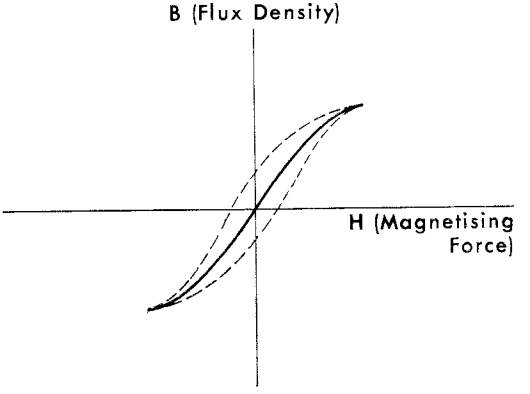


Fig 5 Hysteresis Curve for Permalloy

16. One spoke of the three-spoke fluxvalve is shown diagramatically in Fig 6. It consists of a pair of soft iron (usually permalloy) cones each wound with a primary coil. The winding on one core is the reverse of that on the other. The coils are fed from the same alternating source so that the direction of current flow in one core is equal and opposite to that in the other. The AC supply is just sufficient, at peak power, to magnetically saturate each of the parallel soft iron cores. A secondary coil, wound round the two primaries, is linked with the circuit, and any change of flux through it induces a voltage and current flows.

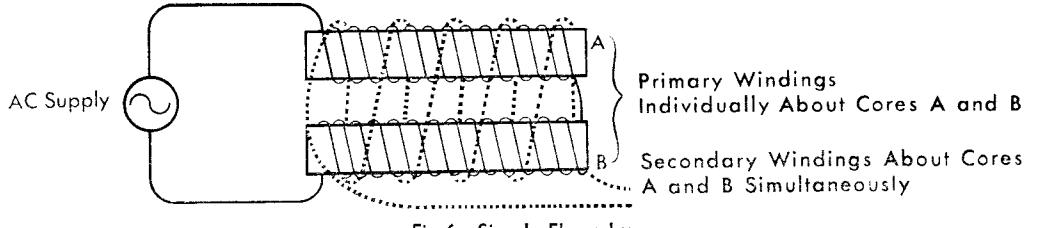


Fig 6 Simple Fluxvalve

17. Fig 7 shows the 400 Hz alternating flux induced in the top leg by the excitation current considering only the top leg of the spoke and the effect of the excitation.

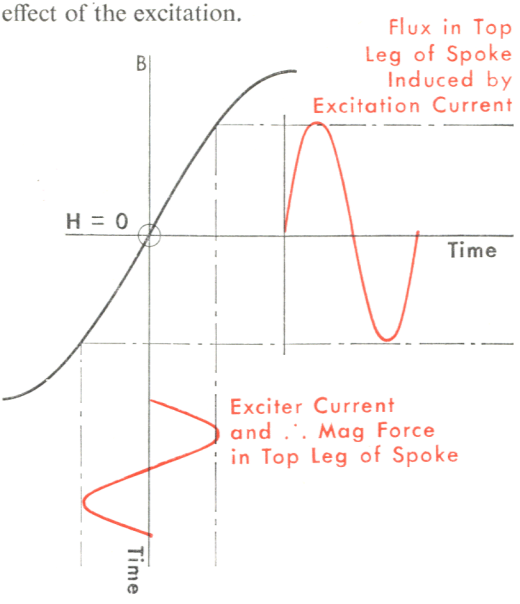


Fig 7 The Effect of the Excitation Current in the Top Leg Only

18. Now considering the bottom leg only; the flux induced in this leg by the excitation current will at any instant be in the opposite direction to that induced in the top leg, *ie* the flux in the bottom leg is 180° out of phase with the flux in the top leg as shown in Fig 8.

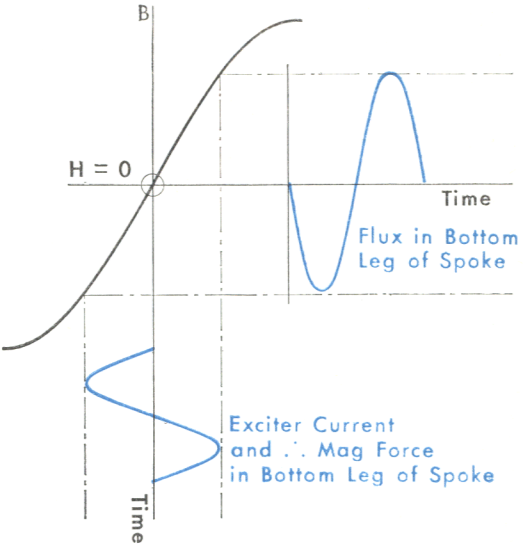


Fig 8 The Effect of the Excitation Current in the Bottom Leg Only

19. Since the top and bottom legs are identical, the amplitudes of the flux in the two legs are equal but 180° out of phase with each other relative to the pick-off coil, which is wound round both legs. Therefore, the resultant flux cutting the pick-off coil, which is the algebraic sum of the flux in the top and bottom legs is zero as shown in Fig 9.

20. If the horizontal component of the Earth's magnetic field (H) is now added in line with the

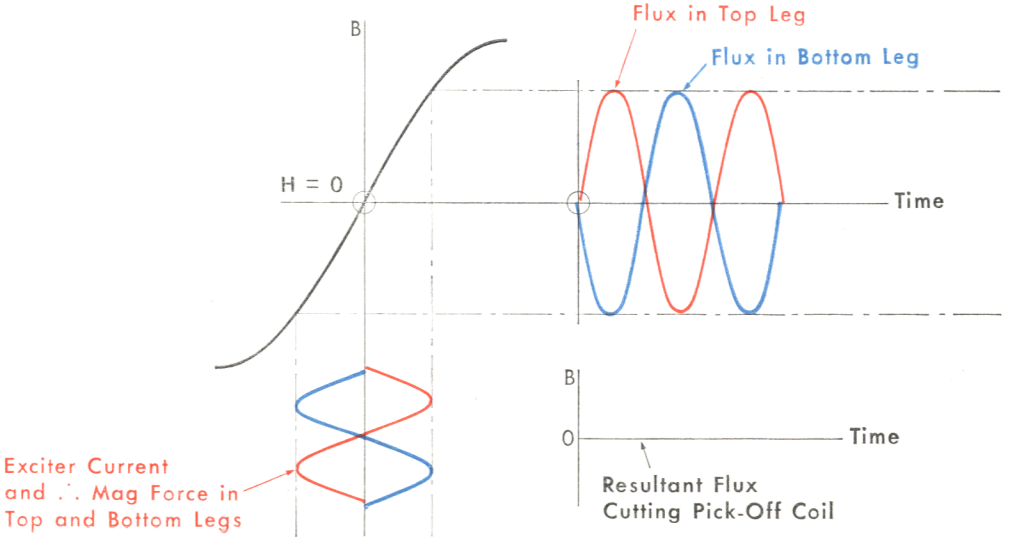


Fig 9 The Effect of the Excitation Current in Both Legs

spoke, it will induce a steady flux in both legs of the spoke which will be added to the flux due to the excitation current. The effect, as shown in Fig 10, will be to bias the datum for the magnetizing force, due to the excitation current, on the B-H curve by an amount equal to H. The strength of the excitation current is so arranged that the effect of the introduction of the Earth's magnetic field component is to bring the flux density curves in Fig 10 onto the saturation part of the hysteresis curve. The resultant flux cutting the pick-off coil, which is the algebraic sum of the fluxes in the top and bottom legs, will no longer be zero but will have a resultant proportional in amplitude to heading. The emf induced in the pick-off coil is proportional to the rate of change of flux cutting the coil and therefore will have a waveform approximating to a sine wave at 800 Hz, *ie* twice the frequency of the excitation current as shown in Fig 10. It has been found by experiment that the amplitude of the emf is proportional to H. Therefore, the emf in the pick-off coil is a measure of H, *ie* the horizontal component of the Earth's magnetic field in line with the spoke. This should be apparent from Fig 10 in that, if a greater H is detected, the excitation current is biased further from the mid-point of the hysteresis curve, and the imbalance between the upper and lower leg fluxes will increase. Therefore, a greater resultant flux exists which will induce an emf of greater amplitude in the pick-off coil. A plot of the amplitude of the pick-off coil output voltage would show that it varies as the cosine of the magnetic heading.

21. **Limitations of the Simple Single Spoke Detector.** It should be apparent that there are two magnetic headings corresponding to zero flux (90° and 270°) and two headings corresponding to a maximum flux. The two maximum values give the same reading on an AC voltmeter since the instrument cannot take into account the direction of the voltage. For any other value of flux (other than zero), there will be four headings corresponding to a single voltmeter reading. This ambiguity is overcome by using a fluxvalve having three spokes (each spoke being similar to the single spoked device previously discussed) with 120° separation as shown in Fig 11. Regardless of the heading, at least two of the spokes will have a voltage induced and their vector sum points to magnetic North (*see* Fig 12). The simple one-spoke detector suffers from another limitation in that the value of H changes with magnetic latitude. This produces a change in the static flux linking the spoke, even though the heading may remain unchanged. This limitation is overcome in the three-spoke fluxvalve because the flux associated with each spoke will change in

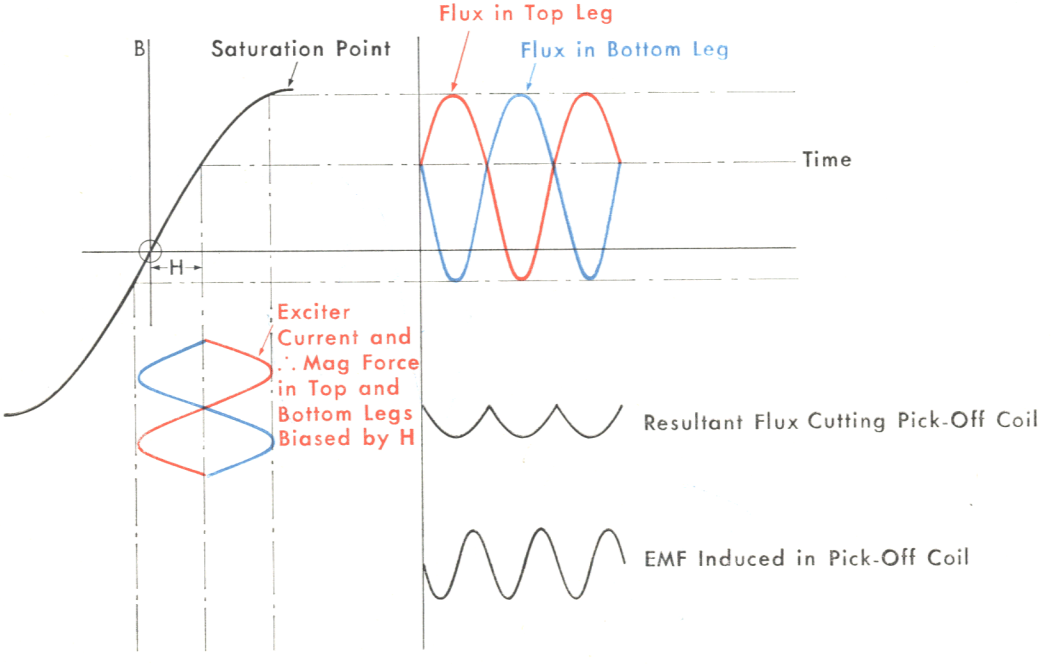


Fig 10 The Combined Effects of the Excitation Current and the Component of the Earth's Field

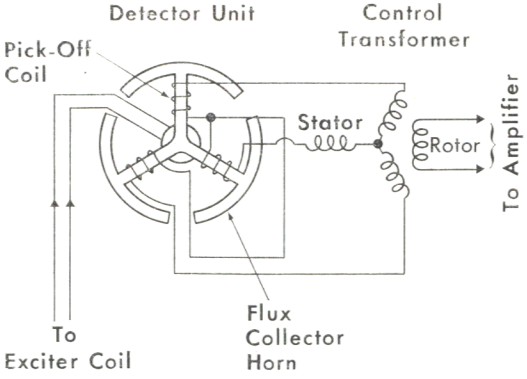


Fig 11 Detector Unit and Transmission System-Schematic

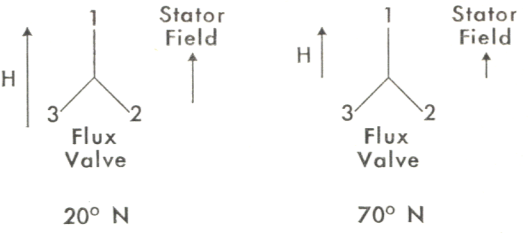


Fig 13 Eliminating Latitude Ambiguity

proportion to the change in H. The resultant field across the receiver stator is still aligned with H (*see* Fig 13).

22. In the three-spoke fluxvalve a single primary coil excites all six cores. If a single arm of the fluxvalve is considered, it will be apparent that the top and the bottom of the exciter coil have opposite polarity. The flux induced in the upper core of the spoke is equal and opposite to that induced in the lower core and this is exactly the effect produced by the primary windings in the simple fluxvalve. The three arms of the fluxvalve are wound with secondary or pick-off coils which are star connected. The exciter coil is fed with 400 Hz single-phase current so that each of the three pick-off coils has an emf at 800 Hz induced in it whose amplitude is proportional to the magnetic heading of the aircraft. Each core of the fluxvalve is fitted with a flux collector horn to concentrate the Earth's lines of force through the core. This increases the static flux and therefore the induced voltage.

The Transmission/Display System

23. It has been shown that the resultant field

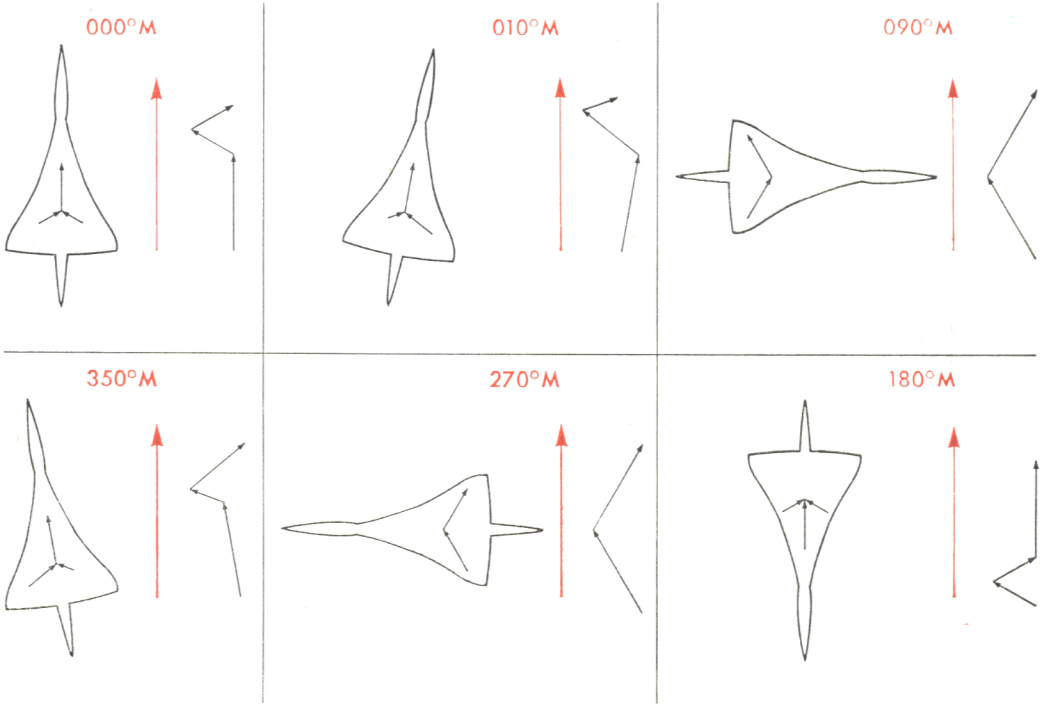
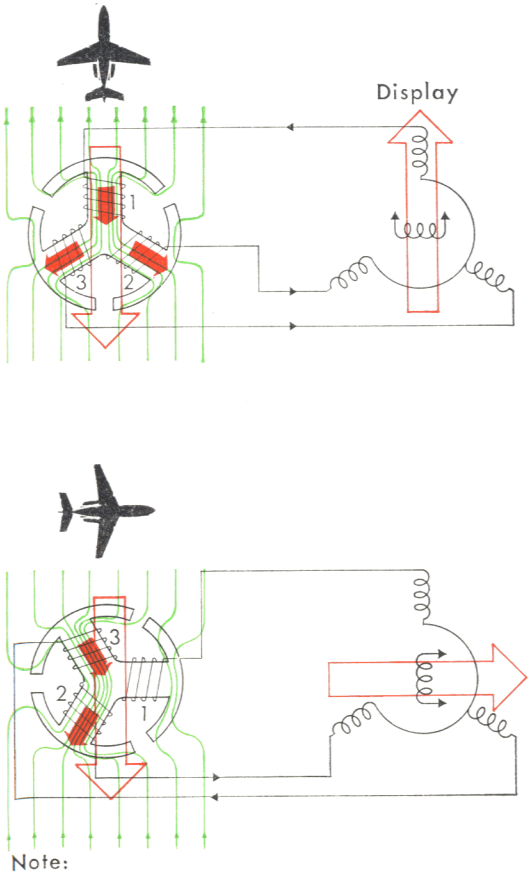


Fig 12 Operation of the Three-Spoke Fluxvalve

produced by the three pick-off coils is directly related to the direction of the horizontal component of the Earth's magnetic field. It is now necessary to convey this heading information from the detector unit to those positions in the aircraft where the information is required. This is achieved by means of the transmission system.

24. The fluxvalve can be likened to a control transmitter (CX) where the transmitter rotor field is represented by the horizontal component of the Earth's magnetic field. The voltage induced in the fluxvalve pick-off coils cause a current to flow along the connecting lines to the receiver stator (*see* Fig 14). A field is set up across the receiver stator in a direction determined by the resolution of the current flowing in each of the receiver stator coils. When the pattern of current flow changes in the receiver stator, as a result of the effects of a heading change in the fluxvalve, the direction of the induced field will change accordingly. A null seeking rotor will follow this field change since it remains at right angles to the field and may be used to transmit any change in aircraft heading.



Note:

Current Flow is AC it is shown in One Direction for Simplicity.

Red Arrows represent Stator Fields.

Fig 14 Action of the Fluxvalve and Transmission System

25. The outputs from the second and third fluxvalve spokes may be wired to the second and third receiver stator coils respectively or vice versa. The wiring will depend on whether it is necessary to drive a compass needle or a compass card. If the aircraft alters heading to starboard, the field across the fluxvalve (which always points to magnetic North) will rotate in an anti-clockwise direction. In this case a compass needle must rotate clockwise (therefore 2 to 3 and 3 to 2), but a card rotating against a stationary lubber line must rotate anticlockwise in which case the second and third fluxvalve spokes are attached to their respective receiver stator coils.

HEADING ERRORS INDUCED BY THE FLUXVALVE

General

26. The errors discussed under this section are limited to those evident in a magnetic compass system without gyroscopic azimuth stabilization, *ie* the fluxvalve is connected directly to the indicator. This approach will simplify the presentation of the errors associated only with the fluxvalve without having to consider gyro behaviour. It can be said at this point that those errors are present to some extent even in gyro-magnetic compass systems. Since most compass systems in use have refinements which to some extent compensate the errors outlined here, the following discussion considers a single system without compensation or refinement of any sort apart from deviation correction. Such a system is illustrated in Fig 15.

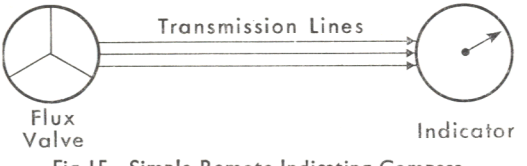


Fig 15 Simple Remote Indicating Compass

Detector Tilt Error

27. The fluxvalve will provide a correct output of magnetic heading only if the detecting element is maintained in the local horizontal plane, *ie* only detecting the horizontal component of the

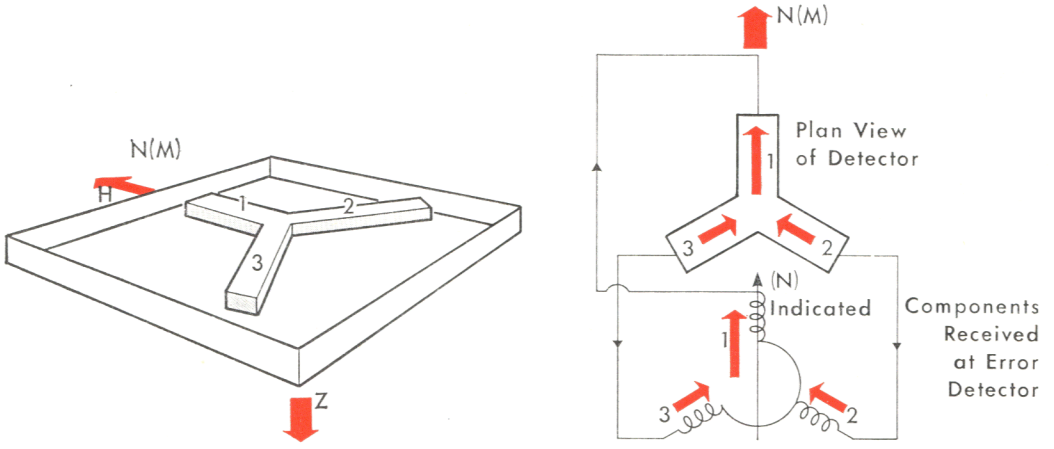


Fig 16 Indication of Magnetic North

Earth's magnetic field (H). Any vertical component of the Earth's field (Z) linked through the fluxvalve coils will cause an error in the output heading. At this stage it is sufficient to note that even small tilts can cause significant errors in heading. In ostensibly straight and level flight, accelerations act upon the fluxvalve which tilt it slightly and small errors result. During manoeuvres the accelerations, and hence the tilts and errors, can be quite large.

28. Fig 16 illustrates a fluxvalve fitted in an aircraft on a heading of magnetic North. The currents induced in spokes 1, 2 and 3 are such that they produce component magnetic fields in the error detector which compound to produce a resultant magnetic field in a direction indicating magnetic North. Only the horizontal component (H) threads the fluxvalve spokes to produce this result.

29. In Fig 17 the fluxvalve is tilted through 90° to port. The induced currents in the spokes change as the components of the total field through them change. Therefore, in this case the component in spoke 1 remains unchanged while that in 2 increases and 3 decreases. The resultant field in the error detector is displaced and an error in heading results. In this case the direction of magnetic North is rotated anti-clockwise and the heading indication is an over reading. At intermediate tilts the error would be less.

30. The error also depends on magnetic dip for, if the case at Fig 17 is repeated with a different dip, the components threading the spokes will

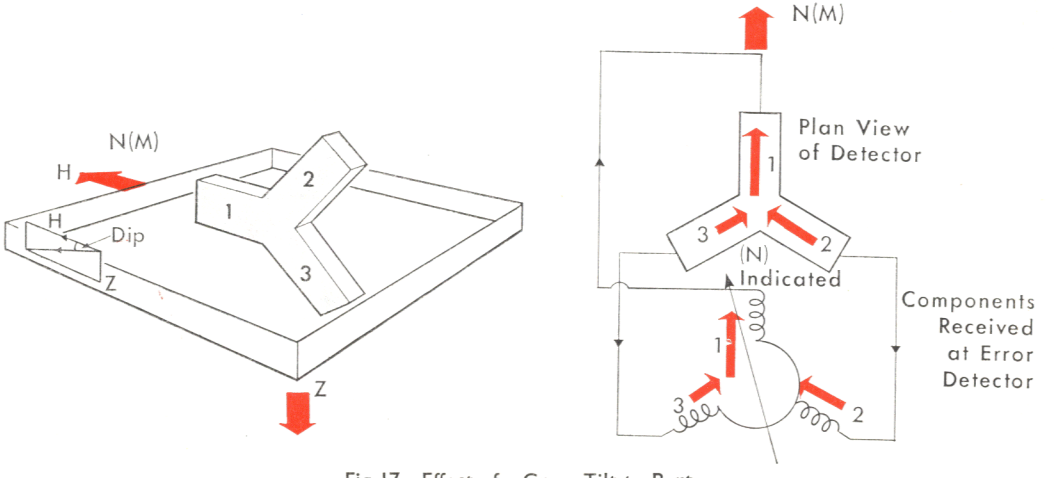


Fig 17 Effect of a Gross Tilt to Port

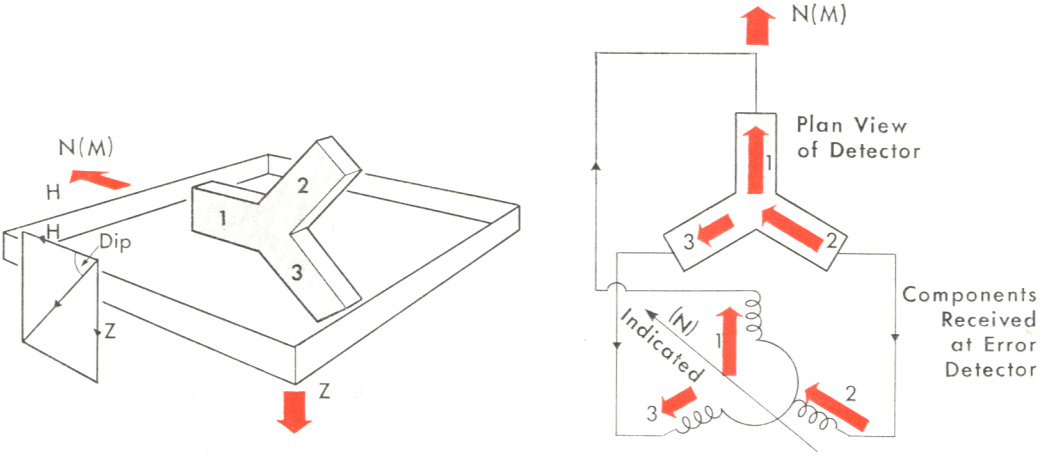


Fig 18 Effect of Change of Dip

alter. In Fig 18 the dip is increased, thereby increasing the error and reversing one component in this particular case.

31. The direction of tilt relative to the total field is also important. Fig 19 shows how a tilt in the direction of the total field may produce no error. In this case the flux flow through each spoke changes but the proportion of one to the other remains unchanged. The intensity of the resultant field increases but the direction remains the same. A second case exists in which the tilt is in the opposite sense as in Fig 20. Here, if the tilt exceeds 90° — dip, the flux flow in each spoke is reversed and the error is 180°.

32. Therefore, the error produced by tilting depends on the following factors:

a. Angle of tilt (α).
b. Direction of tilt.
c. Magnetic dip (δ).

Typical values of the error in fluxvalve output are shown against the direction θ of the axis of tilt for various values of tilt in Fig 21. In general, the bigger the tilt and the dip, the larger the error. Gross errors occur when tilt is greater than 90 — δ due to field reversal (*see* para 31).

33. A number of factors exist during flight which can cause fluxvalve tilts; these include:

a. Central acceleration caused by aircraft turns.
b. Coriolis accelerations.
c. Vehicle movement (rhumb line) acceleration.
d. Fluxvalve vibration.
e. Aircraft linear acceleration.

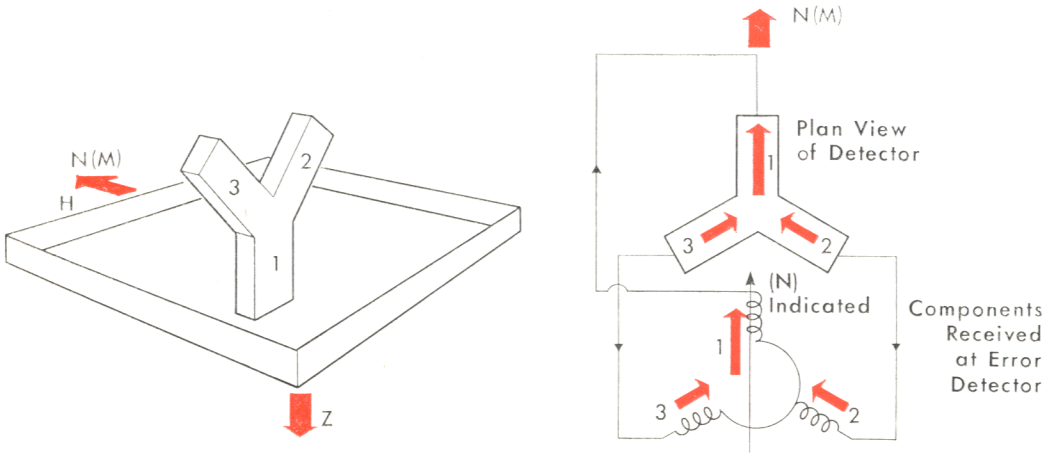


Fig 19 Effect of Direction of Tilt

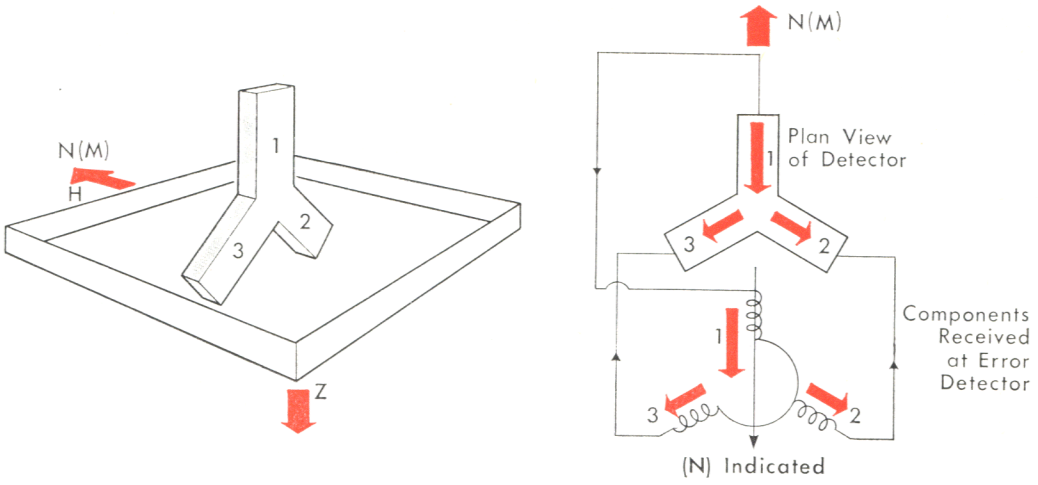


Fig 20 Tilt Exceeds 90°—Dip

34. **General Formula for Tilt Error.** For fluxvalve tilts of less than 10°, the error in magnetic heading can be expressed as:

$$\text{error} = \frac{-57{\cdot}3 \ \alpha \ \cos \theta \ \tan \delta}{1 + \alpha \ \sin \theta \ \tan \delta} \text{ (degrees)} \ldots (1)$$

Where α = tilt in radians

θ = angle in degrees between magnetic North and the axis of fluxvalve tilt.

δ = magnetic dip angle in degrees.

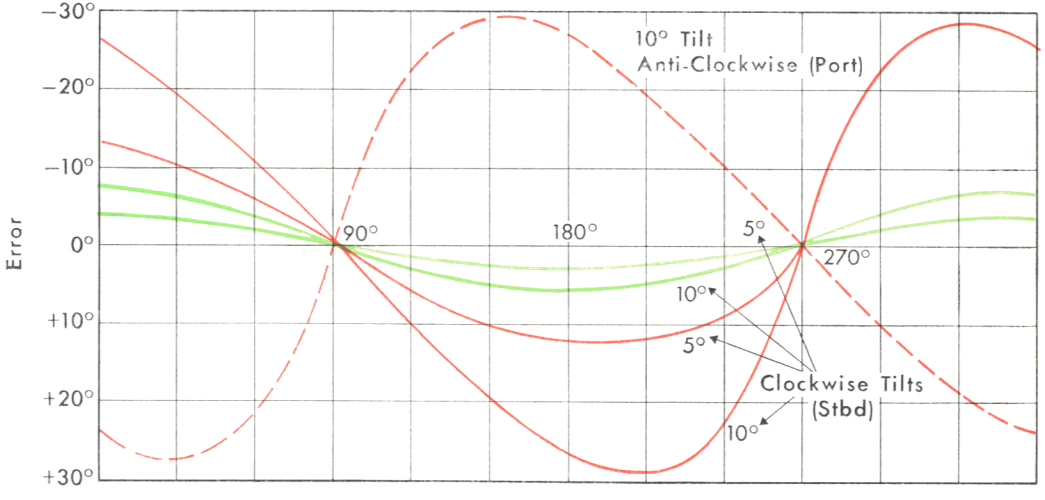
α is the result of horizontal acceleration and can be calculated as follows:

$$\alpha = \frac{\text{horizontal acceleration}}{\text{acceleration due to gravity}} \text{ (radians)} \ldots (2)$$

The sign of α will be +ve if the tilt is clockwise looking along the axis of tilt. A simpler approximation of heading error valid for very small tilt angles and for dip up to 85° is given by:

$$\text{error} = -57{\cdot}3 \ \alpha \ \cos \theta \ \tan \delta \text{ (degrees)} \ldots . (3)$$

Generally, formula (1) will be used for the fluxvalve errors discussed in the following paragraphs. Neither formula (1) or (3) may be used to calculate fluxvalve heading errors if α is greater than 10°; for such cases a much more



Dip = 70°

Dip = 30°

Fig 21 Typical Errors in Magnetic Heading Due to Tilt

complicated formula is necessary from which these are derived.

35. **Turning Error.** During a correctly executed turn the aircraft is banked at an angle which causes the total force to cut through the aircraft's apparent vertical. Thus the fluxvalve tilts at an angle equal to the angle of bank and the tilt varies with the rate of turn and speed. Referring to Fig 21, the instantaneous fluxvalve error can be interpreted by calling θ the instantaneous heading. (This is because the axis of the tilt is in the aircraft fore and aft axis.) Therefore, if the aircraft turns port through North the heading error is positive, *ie* the compass over reads. If the turn is starboard through South the effect is again an over reading.

36. **Coriolis Acceleration Error.** An aircraft flying on a spherical rotating Earth flies a curved path in space. In consequence, a central force acts to displace all pendulous elements from the vertical. The error in heading is calculable and depends on ground speed, latitude, dip and track. For an aircraft on a track of 045°, at 50°N00° at 600 kts, the error is approximately 0·5°. The value of coriolis acceleration is given by $2\ \Omega\ G\ \text{Sin}\ \lambda$. In the northern hemisphere it acts to the left of track, tilting the fluxvalve anti-clockwise about an axis along magnetic track. Therefore, the tilt, α, is given by:

$$\alpha = \frac{\text{horizontal acceleration}}{g} = \frac{2\ \Omega\ G \sin \lambda}{g} \text{(radians)}$$

where Ω = Earth rate in radians per sec.
G = ground speed in ft per sec.
g = gravity acceleration in ft per sec per sec.

Substituting more conventional units and simplifying we obtain

$$\alpha = \frac{2 \times 15 \times 6080\ G \sin \lambda}{3600 \times 3600 \times 32{\cdot}2}$$
$$= -\ 4{\cdot}37 \times 10^{-4}\ G \sin \lambda \text{ (degrees)}$$
where G = groundspeed in kt.

To find the error in magnetic heading caused by this tilt:

$$\text{error}(\varepsilon) = -\ \alpha \text{ (degrees) Cos } \theta \text{ tan } \delta \text{ (degrees)}$$
$$= -\ (-4{\cdot}37 \times 10^{-4}\ G \sin \lambda) \times \text{Cos mag track tan } \delta \text{ (degrees).}$$

Note: λ is negative in the southern hemisphere, as is δ, so that the + sign applies in both hemispheres. The sign of the error also changes cyclically with magnetic track.

37. **Vehicle Movement Acceleration Error.** An acceleration, in general, acts on the pendulous fluxvalve when the aircraft tries to maintain a constant track with reference to meridians which are converging. For example, if an aircraft flies a rhumb line then it must turn to maintain its constant direction relative to the meridian and a central acceleration acts to displace all pendulous elements. The rate of turn is the rate at which true North changes direction *ie* the rate of meridian convergence. When the path flown is of constant direction relative to magnetic North then an additional rate (the rate of change of variation) must be added. We may proceed as follows:

$$\alpha = \frac{\text{central acceleration}}{g} \text{ (radians)}$$
$$= G \left(-\frac{U}{60} \tan \lambda + W_{vm} \right) \times 0{\cdot}146 \times 10^{-4} \text{ (degrees)}$$

where U is the easterly component of groundspeed in kts, G is ground speed in knots and W_{vm} is the rate of change of magnetic variation in degrees per hour. The expression $\frac{U}{60} \tan \lambda$ gives W_{tm}, the rate of heading change caused by geographic meridian convergence in degrees per hour. Therefore:

$$\text{error } (\varepsilon) = \alpha^{\circ} \text{ Cos } \theta \text{ tan } \delta \text{ (degrees)}$$
$$= -\alpha^{\circ} \text{ Cos mag track tan } \delta \text{ (degrees)}$$
$$= -\ 0{\cdot}146 \times 10^{-4} \times G \left\{ -\frac{U}{60} \tan \lambda + W_{vm} \right\} \times \text{Cos mag track tan } \delta \text{ (degrees)}$$
$$\text{or } -\ 0{\cdot}146 \times 10^{-4} \times G\ ..(W_{tm} + W_{vm})\ .. \times \text{cos mag track tan } \delta \text{ (degrees)} \quad ....(5)$$

38. **Fluxvalve Vibration.** Oscillation of the aircraft in height and speed take place in even the most stable aircraft. These oscillations are small in amplitude and fairly small in period but may induce oscillation of the fluxvalve. One other source of fluxvalve vibration is wing-tip oscillation. These oscillations may combine to produce a small heading vibration. It is worth noting that even if the fluxvalve is swinging with equal amplitudes about the true vertical, the heading may in consequence be out by a constant amount.

This is because (*see* Fig 21) equal tilts in opposite senses do not produce equal magnitude heading errors. The effect is in general very small because of the damping action of the detector unit fluid and the errors may usually be ignored. However, in helicopters the vibration can have a large amplitude and so produce an error which is quite large.

39. **Linear Velocity Changes.** The effects of longitudinal acceleration are not great since they are not often prolonged. The most effect is probably experienced during take-off. A sustained acceleration such as this on East or West can induce errors of 2° to 3°.

THEORY OF THE GYRO-MAGNETIC COMPASS

General

40. To overcome the inaccuracies in magnetic heading obtained from a tilted fluxvalve, a gyro must be added to the system. The incorporation of a gyro introduces a number of new errors in the heading output of the system, but these errors are more than offset by the improvement in accuracy which results from having an accurate mechanical datum about which any change of heading may be measured. Any tendency for the gyro to drift away from its alignment datum may be checked by slaving it to the fluxvalve when the aircraft is straight and level.

Mechanization

41. The simple schematic at Fig 22 shows a basic, uncorrected and uncompensated gyro-magnetic compass system. The fluxvalve magnetic heading is compared with gyro heading at an error detection device. If the two headings are not equal, an error signal is developed, amplified and used to precess the gyro. This precession continues until the two headings are equal and the correct heading is displayed. An important principle is illustrated here. Since gyro heading is displayed, if an error exists in gyro heading, the displayed heading must also be in error.

42. The method of mechanizing the gyro precession loop is of extreme importance. Three methods of accomplishing the task are as follows:

a. Step function (bang-bang) correction.
b. Linear function correction.
c. Limited linear function correction.

43. The step function correction technique requires the gyro-fluxvalve error signal (ε) to be removed at a fixed rate (W_c) whenever it is generated (*see* Fig 23a). Not only is such a system difficult to engineer, but gyro behaviour suffers severely from nodding or nutation and secondary precession.

44. The linear correction technique appears to be ideal since the correction rate (W_c) is pro-

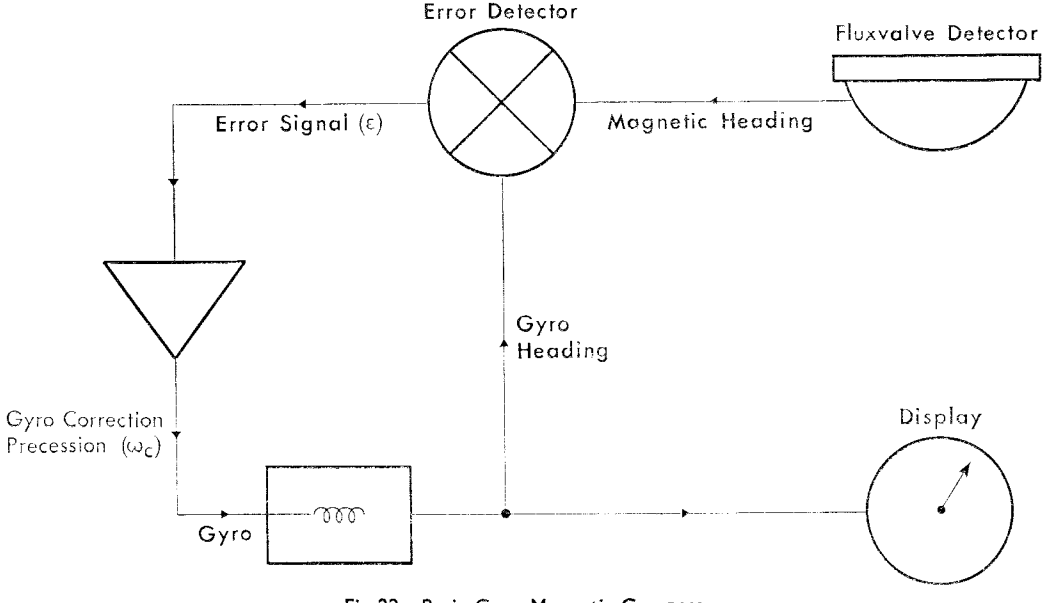


Fig 22 Basic Gyro-Magnetic Compass

portional to the error signal (ε), *ie* for small errors, small torques are applied and vice versa. A problem exists when very large errors occur. For example, modern gyro-magnetic compasses commonly use the random gyro azimuth technique in which the gyro spin axis can point in any direction relative to magnetic North or aircraft heading. When the system is initially switched on, 180° can exist between gyro and magnetic heading. If the system were mechanized to provide an adequate rate of precession for small errors, 180° would demand an excessive precession rate. Therefore, the purely linear system also has its limitations.

45. The common solution to the precession mechanization problem is a compromise between the step function and the linear function techniques—namely the method shown in Fig 23c, the limited-linear technique. In a gyro-magnetic compass system in which the gyro is controlled by the limited-linear concept, gyro precession rates are proportional to the error signal for small discrepancies but gyro precession is limited by the manufacturer to a predetermined value for large discrepancies. For example, in Fig 23c, the gyro precession rate (W_c) is proportional to ε, where ε is $\leqslant 2°$, however, W_c cannot exceed 2° per min regardless of the size of ε.

46. **Time Constant.** The rate of precession in a limited linear system is controlled by the amplified error signal and, for the linear portion of the curve, is arranged to be proportional to the error. Therefore, assuming small errors, the rate of precession multiplied by a constant is equal to the gyro-fluxvalve discrepancy or $W_cK = \varepsilon$ (degrees). If W_c is in degrees per minute and ε is in degrees, the dimension of K must be time. Therefore, if τ is substituted for K and it has the dimension of time (commonly minutes), τ is referred to as the time constant of the system.

$$\varepsilon = W_c \tau \qquad \ldots.(6)$$

Therefore, if $\varepsilon = 2°$ and $\tau = 0{\cdot}5$ minutes, the rate of precession (W_c) is given by:

$$W_c = \frac{\varepsilon}{\tau} = \frac{2°}{0{\cdot}5 \text{ min}} = 4° \text{ per min.}$$

Obviously the larger the time constant, the slower is the rate of precession. Notice that τ does not express explicitly the time to correct a given error since the rate of correction reduces as the error reduces so it takes much longer than τ minutes to correct the error. Since the error reduces exponentially, τ directly gives the time it takes to remove 63% of the error. It would require approximately 5 τ to remove all the error in a step error function. Therefore, for an initial error of 2° and a τ of 2 minutes, the error will reduce exponentially until at the end of 5 τ (10 mins) the error is effectively reduced to zero.

47. **Significance of τ.** The authority of the fluxvalve over the gyro is effectively controlled by τ. If the compass system contains a poor quality gyro, it would be expected that any discrepancy between gyro and fluxvalve was caused by the gyro; therefore, a short τ should be anticipated. Conversely, if a high quality gyro with a low real drift rate is incorporated, the gyro should be less closely tied to the fluxvalve and a large time constant anticipated.

48. **Typical Gyro Slaving Mechanization.** The method by which the manufacturer has implemented limited linear control in the G4B is typical and fairly straightforward. With reference

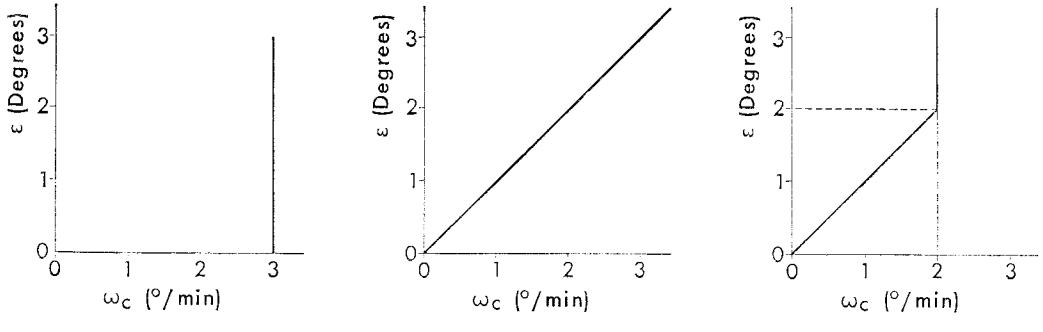


Fig 23 Gyro Correction Techniques

to the block diagram at Fig 24 and the schematic of the G4B at Fig 25, the currents induced in the spokes of the fluxvalve are passed to a receiver synchro (CT) and produce a field across the rotor from which the aircraft magnetic heading can be determined. The electrical output of the rotor is taken to the gyro azimuth precession coils which are threaded by a permanent magnet. If the rotor is not at right angles to the field set up by the stator coils, a current will flow through the precession coils setting up a magnetic field which will set up a force on the permanent magnet. This rotational torque will be translated through 90° by the gyro and will cause it to precess in azimuth. As the gyro precesses, the rotor is repositioned by mechanical feedback until eventually it reaches its null position. Since the compass needle is driven by the gyro, when the receiver rotor is lying in the null position, the fluxvalve, gyro and compass needle will all be correctly aligned. If an error occurs between gyro and fluxvalve, the rotor will be misaligned causing a current to flow in it which is fed to the precession coil to correct the gyro. As the rotor approaches the null, the current flowing in it will reduce. The current flowing through the precession coil will also reduce, therefore, the rate of gyro precession decreases as the error diminishes.

49. **The Change in τ with H.** Fig 26 illustrates the relationship between H field strength and gyro precession rate in the G4B compass system. As the H field strength decreases due to northward movement, the amplitudes of the voltages induced in the fluxvalve spokes are reduced proportionally. Although the direction of the resolved voltages remains the same, the size of the currents transmitted to the receiver synchro are smaller. Therefore, the field strength across the receiver stator will be reduced and the rotor current flow for any given misalignment will decrease. Since the amount of torque applied to the gyro azimuth precession device depends on rotor current, the precession will also decrease. The reduction in gyro correction rate with a decrease of H field strength (or an increase in magnetic latitude) results in effectively the same phenomenon as would be achieved by increasing τ. An increase in τ will make the system sluggish and will also tend to magnify any hang-off error present (*see* para 59). However, if the aircraft is operating at high latitudes, the fluxvalve is less reliable due to the reduction of H field strength and an automatic increase of τ is acceptable. Since τ changes with H field strength, the H field strength must be quoted with τ to make τ meaningful. The H field strength at Greenwich (nominally 0·18 oersteds) is the common datum quoted by British gyro-magnetic compass system manufacturers.

GYRO-MAGNETIC COMPASS SYSTEM ERRORS

Fluxvalve Tilt Errors

50. All of the horizontal accelerations which

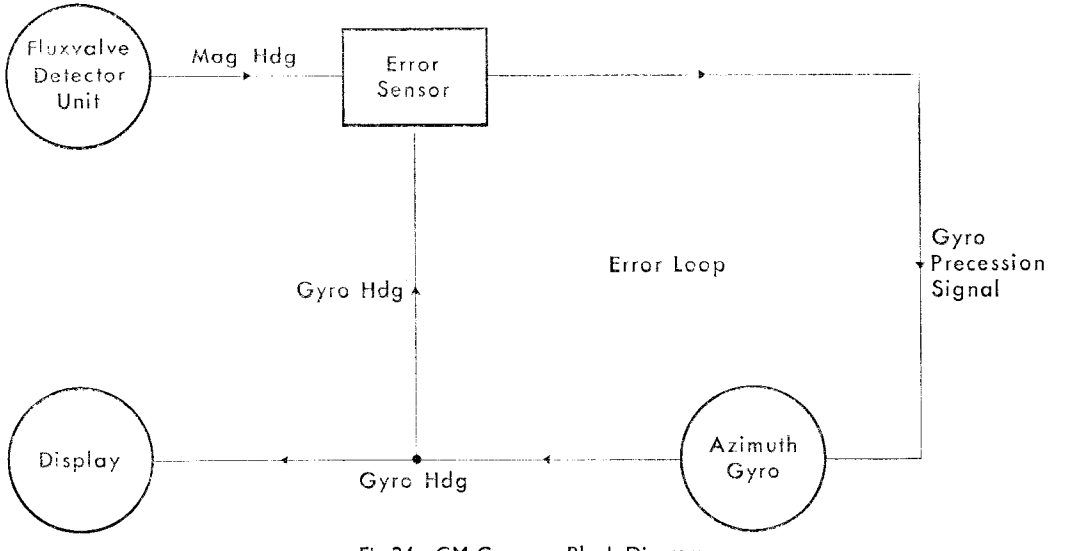


Fig 24 GM Compass Block Diagram

cause fluxvalve tilt can cause heading errors in a simple uncompensated gyro-magnetic compass system. As discussed previously, accelerations are caused by coriolis, vehicle movement (rhumb line), aircraft turns, linear changes of velocity (*eg* take-off) and fluxvalve vibrations.

51. It must be noted that fluxvalve induced heading errors will not appear immediately in the displayed heading of a gyro-magnetic compass. The rate of heading error incorporation depends on the limiting precession rate and the length of τ.

52. **Turning Error.** The amplitude of the displayed heading error in a gyro-magnetic compass due to co-ordinated aircraft turns is less than that shown in Fig 21. Although a high rate of turn in a fast aircraft would show the greatest fluxvalve heading error, since the time spent on the turn is minimal, little of the error is displayed. Fig 27 illustrates that slow prolonged turns at high speeds give greatest errors, *eg* for a speed of 520 kt the peak occurs when the bank angle is 10°. It should be noted that the values in Fig 27 are the maximum values for the conditions stated. The angle of turn varies from 90° in a slow turn to 180° for a fast turn from an Easterly heading. The errors will, of course, decay after level flight is resumed.

53. **Coriolis Error.** Coriolis acceleration affects the fluxvalve whenever the aircraft is moving. Therefore, when established on a given heading for approximately 5 τ, the entire error would be included in the gyro magnetic compass heading display. (*See* para 36 for a discussion on Coriolis error.)

54. **Vehicle Movement Error.** Whenever flying a true or magnetic rhumb line, an acceleration deflects the detector element from the local horizontal plane. The entire resultant heading error would appear in the displayed gyro magnetic compass heading after 5 τ. (*See* para 37 for a discussion on vehicle movement error.)

55. **Fluxvalve Vibration.** Fluxvalve vibration results in a fluxvalve heading oscillation, the mean of which is not the actual mean heading (*see* Fig 21). Since the gyro slaving loop tends to average fluxvalve headings over a period of time, the gyro would eventually be precessed to the erroneous fluxvalve mean heading.

56. **Linear Velocity Changes.** Small errors would be induced into the gyro due to longitudinal accelerations, the size of which would depend on the length of τ and the duration of the accelerations.

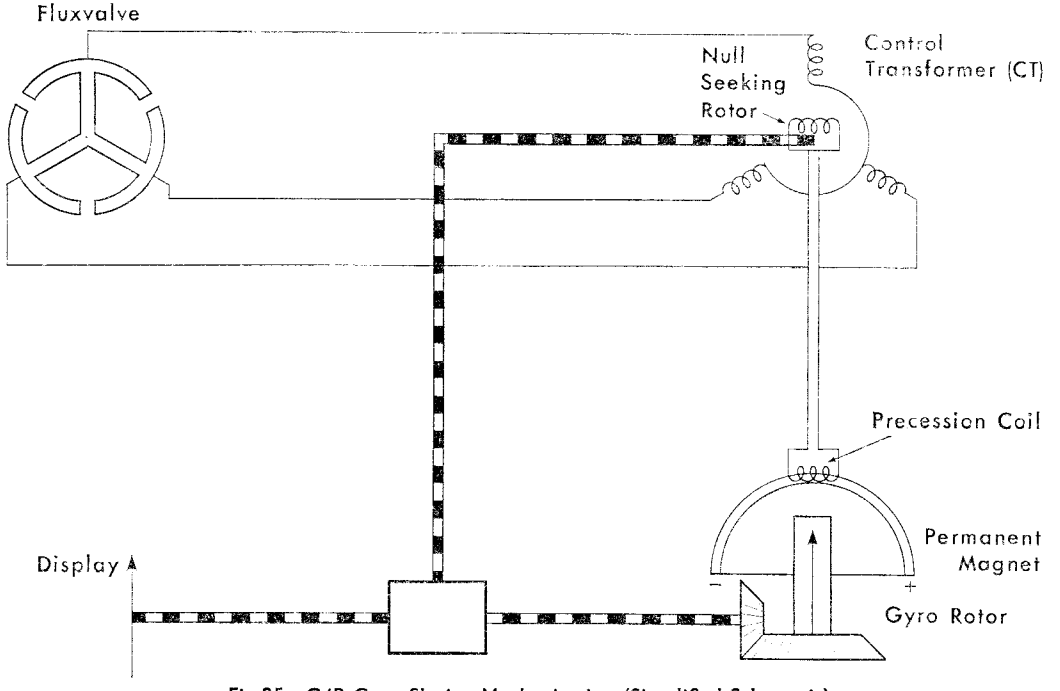


Fig 25 G4B Gyro Slaving Mechanization (Simplified Schematic)

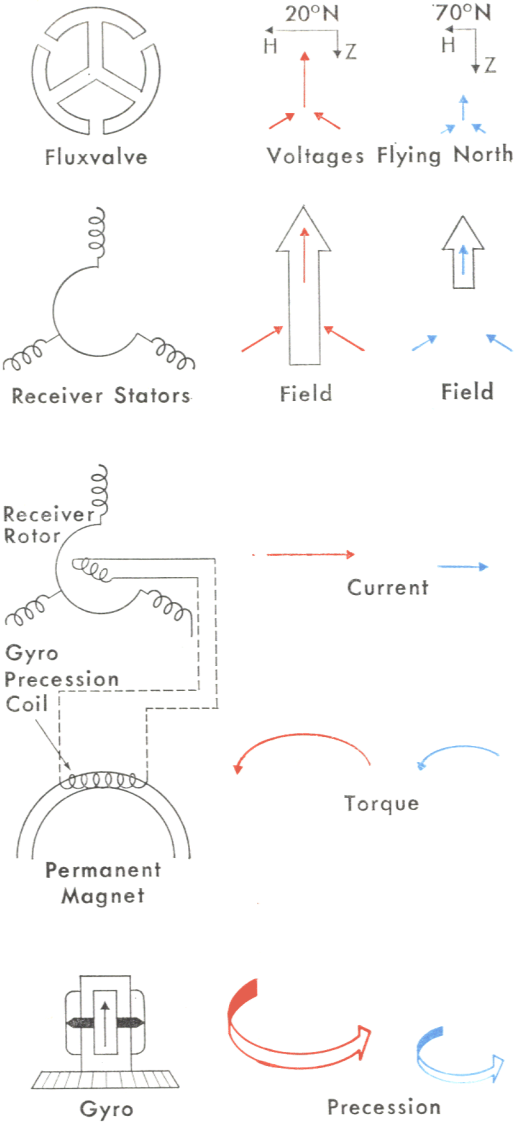


Fig 26 Effect of a Change in H on the Time Constant

Northerly Instability

57. Northerly instability or weaving is a heading oscillation experienced in high-speed aircraft attempting to fly straight and level at or near a heading of magnetic North. Starboard bank of the aircraft induces starboard tilt, and this causes an under reading of the heading. Another way of saying this is that the magnetic meridian appears to rotate clockwise. Thus, if an aircraft on North banks to starboard, to correct some small error, the magnetic meridian rotates in the same direction. The aircraft continues to turn and eventually reaches the false meridian. On levelling out, the fluxvalve senses the true meridian and starts to precess the gyro towards it. The indicated heading changes and the aircraft is banked to port to achieve a northerly indication. This tilts the fluxvalve which rotates the magnetic meridian to port. The aircraft again chases the false meridian, and upon resuming level flight, the fluxvalve will sense the true meridian. This time the precession of the gyro to starboard will be followed by the aircraft. The pattern will be repeated, the oscillation amplitude often being as great as 6°. If the autopilot is engaged the weave will still occur but the compass will transmit an apparently steady heading to user equipments. The amplitude of the weave may be expected to increase with an increase in dip angle, and increase in aircraft velocity and a decrease in τ.

58. The condition for weaving to take place is that the rate of precession (W_c) is greater than the rate of turn (ω) of the aircraft, assuming a rigid coupling to the aircraft controls. Therefore $W_c > \omega$. The tilt produced by the rate of turn, ω, is given by:

$$\alpha = \frac{\omega V}{g} \text{ radians, where V = TAS,}$$

and if α is small the resulting small heading error (ε) will be:

$$\varepsilon = \frac{\omega V}{g} \tan \delta \cos \text{mag hdg}$$

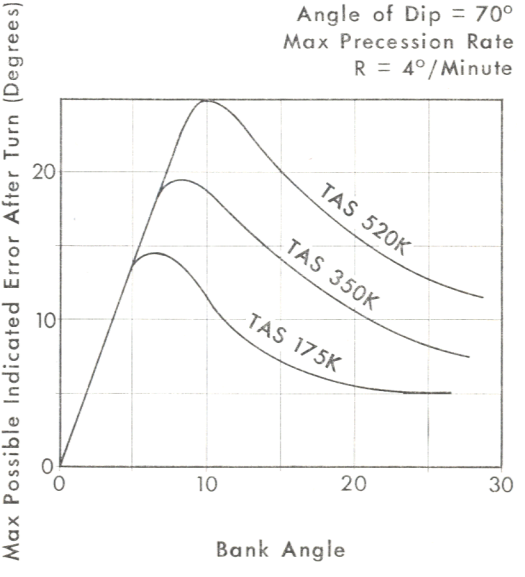


Fig 27 Maximum Turning Errors

The rate of precession produced by ε is given by W in $\varepsilon = W\tau$ and from these equations for ε:

$$W\tau = \frac{\omega V}{g} \tan \delta \cos \text{mag hdg}$$

If the condition for weaving exists, that is if

$$W_c > \omega, \text{ then } \tau < \frac{V}{g} \tan \delta \cos \text{mag hdg}.$$

Using this formula it is possible to estimate the propensity to instability under particular circumstances. For example, at 60°N 05°E, 550 kts TAS, on a heading of 000°, the right hand side becomes:

$$\frac{550 \times 3 \times 6080}{32{\cdot}2 \times 3600} = 86{\cdot}9 \text{ sec}$$

If $\tau = 30$ sec, then weaving will almost certainly take place. By choosing a larger τ the danger can be avoided.

Hang-off Error

59. The limited linear control system ensures that any given difference between gyro and magnetic headings will be eventually removed. Consider, however, a difference which is being continuously created. A gyro will wander from its set Earth direction due to Earth rotation, movement of the gyro round the Earth and from random drift due to imperfect gyro construction. To prevent accumulation of these errors, a gyro must be precessed back towards the magnetic heading at the same rate as it is drifting. To cause such a rate of precession (W_c° per min), an error signal (ε) must be passed to the precession coils of the gyro. Obviously, an error must exist before this signal can be generated. This error, equivalent to $W_c \times \tau$, is known variously as hang-off error, stand-off error or velocity lag.

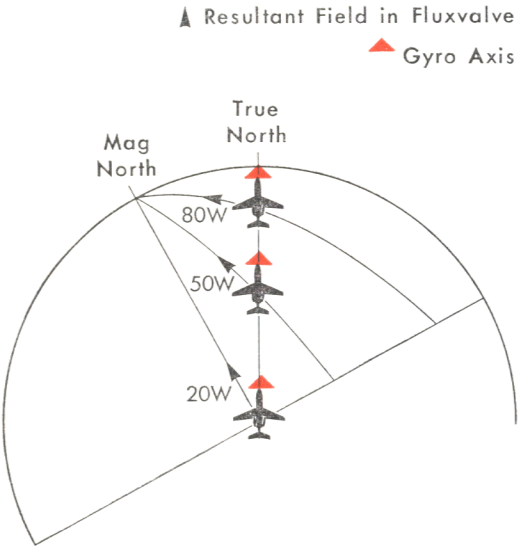
60. The causes of gyro drift will now be reviewed.

61. **Earth Rate.** Earth rate describes the rate of apparent wander of a gyro due to the Earth's rotation. A compass uses an azimuth gyro, and Earth rotation causes an apparent drift (W_c).

62. **Vehicle Movement.** When an aircraft flies over the Earth, the gyro drifts from its original setting relative to the meridian. The value of this drift from the magnetic meridian is given by adding the rate of drift from the true meridian (W_{tm}) to the additional drift due to the magnetic meridian "wandering" from the true (W_{vm}). Whereas apparent gyro drift due to convergence of geographic meridians is covered elsewhere in this AP, more discussion is required concerning the problem of magnetic meridian convergence (*see* Fig 28). If a perfect gyro (no real drift) is corrected for Earth rate and transport wander, it will always maintain its alignment with a point on the Earth's surface. However, the output from the fluxvalve is always magnetic North. Therefore, as the aircraft flies across the Earth, there will be movement between fluxvalve and gyro since the variation is changing (unless the aircraft flies along an isogonal). The rate of change of variation is not easily expressed analytically, but an approximation to it can be obtained by measuring the actual change along the pertinent part of track and relating this to the time to fly that part, *eg:*

position	60°N 00°
groundspeed	600 kt
track	045°

Measure ½hr at ground speed each way along track. This shows that variation changes from 12½°W to 5½°W in 1 hr, *ie* the rate of change is 7° per hour easterly. Therefore the rate of change of variation = W_{vm} = + 7° per hr. A rule for the size of the error is simply obtained. If variation changes westerly then anti-clockwise precession is necessary and the hang-off is negative. Similarly, easterly changes produce positive errors.



Westerly Variation Increasing Westerly Causing an Apparent Clockwise Rotation of Gyro Axis from Fluxvalve and Resulting in a Negative HOE

Fig 28 Hang-Off Error Caused by Convergence of Magnetic Meridian

63. **Random Drift.** Random drift (W_{cr}) is due to imperfect gyro construction. Because it is random it cannot be calculated or predicted other than in terms of statistical parameters.

64. **Hang-Off Error Formula.** Since the amplitude of hang-off error depends on total gyro drift (W_T) and the length of the time constant, the formula for the hang-off error can be expressed as:

hang-off error = $W_T \tau$

where $W_T = W_e + W_{vm} + W_{tm} + W_R$.

Gimbal Error

65. Another form of error introduced during flight is associated with the gimbal pick-offs of the azimuth gyro. Gimbal error is a function of the gyro and is therefore independent of the magnetic reference. This error is the result of the geometry of the gimbal system in that unless the gyro frame in the aircraft is rotated about one of the gyro axes, the outer gimbal ring must itself move if the direction of the rotor axis is to be undisturbed. This movement will be detected by the outer gimbal synchro and portrayed as a change in heading. This may be more readily understood from a study of Fig 29a, b, c and d when the aircraft is heading West. The rotor axis in the illustration is pointing North-South and the outer bearing is fixed to the aircraft so that when the aircraft banks as in b, or dives as in c, there is no error indication. In d however, the aircraft banks and dives and it will be seen that the outer gimbal must rotate to keep the rotor axis pointing in the same direction. Therefore, a change in heading is signalled from the synchro and yet no change of heading has occurred. Exactly the same thing happens if the aircraft is flying on an intercardinal heading, as in e (with rotor axis North-South), and then simply banks to port, as in f, whilst retaining its level pitch attitude. Gimbal error is directly related to the angle of bank and the angular difference between the spin axis and the longitudinal axis. Fig 30 shows the error to be a maximum on 045°, 135°, etc with a value of about 12° for

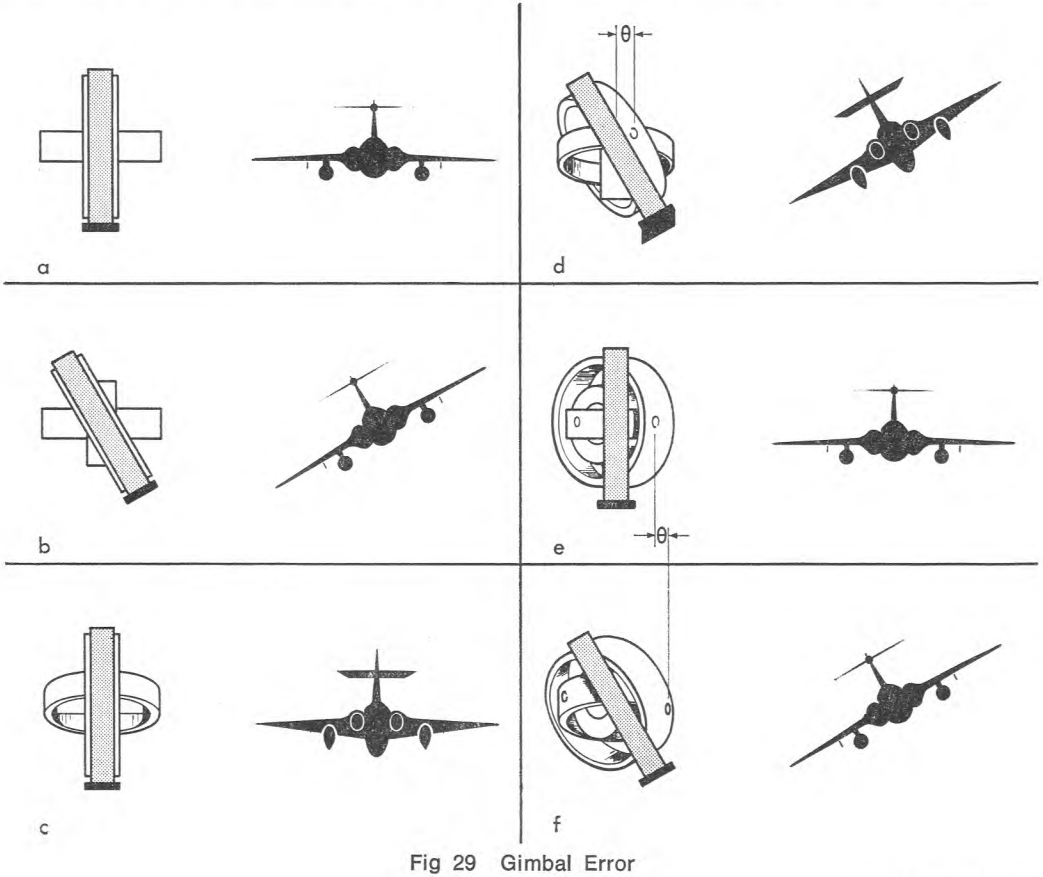


Fig 29 Gimbal Error

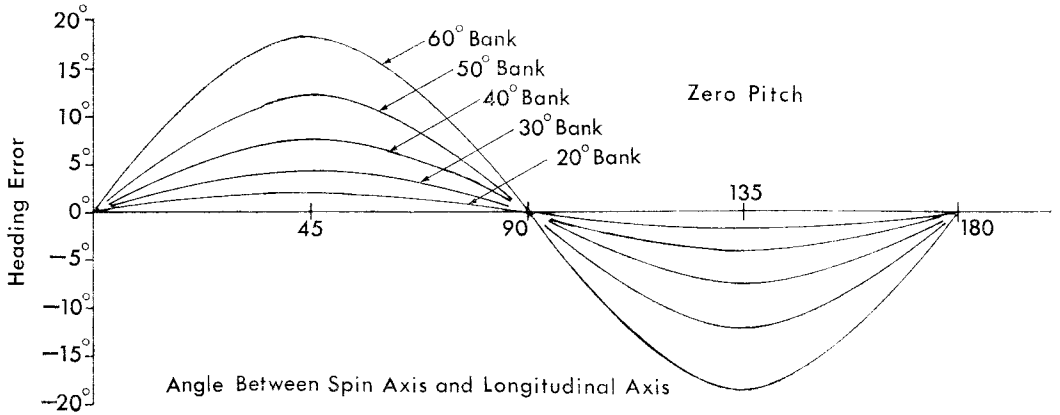


Fig 30 Graphical Analysis of Gimbal Error

50° of bank. As the error is geometrical, it will disappear when the aircraft is levelled but it will have accumulated in GPI equipments to produce an error in computed positions. It should be noted that the spin axis in most compass systems may take up any direction relative to North and the error may be in any direction such that the effect on a given heading is not easily calculated.

Transmission Errors

66. Overall system accuracy is lowered by the errors in the synchro systems. These errors can usually be limited to 0·2° (2σ) for each synchro and errors of 0·1° are becoming more general at this level. The total transmission error shows in a compass swing as a D or E error where the E error is mechanical error in the construction of synchros and D error is the incorrect winding of the coils so that voltages or resistances are unbalanced. An overall error of 0·5° (2σ) is suggested.

Compass Swinging Errors

67. It is not possible to obtain absolute accuracy in compass swinging, and even refined methods are considered to be only accurate to 0·2° (2σ).

Variation and Deviation Errors

68. Various estimates of the error in the value of charted variation have been made; they vary between 0·1° and 2°, but these are inspired guesses rather than reliable statistics. Over the UK at height, magnetic variation uncertainty is considered to be 1·0° (2σ level). The error must vary with height and locality—it being large where local anomalies prevail. The actual setting of deviation and variation values is usually accurate to 0·25° (2σ). This inaccuracy gives rise to a further source of error.

SIMPLE ANALYSIS OF ERRORS

Hang-Off Error

69. **Earth Rate.** The drift of an azimuth gyro (W_c) due to Earth rotation, as derived in Part 2, Sect 2, Chap 1, is given by:

$$\omega = -\Omega \sin \lambda \ (^\circ \text{ per hour})$$

where Ω = Earth rotation in ° per hour
λ = Latitude (− ve in the southern hemisphere).

The sign of the error due to Earth rate can be seen from Fig 31. The spin axis drifts clockwise in the northern hemisphere and the correction required is anti-clockwise. Because a signal is necessary to maintain the correcting precession, gyro North never quite coincides with true North but is 'held-off' by a small amount, hang-off error, and the induced heading is always an under reading. Therefore it is negative in the northern hemisphere and positive in the southern heimsphere.

Vehicle Movement

70. **True Meridian Convergence.** The rate of true meridian convergence is derived in Part 2, Sect 2, Chap 1 as:

$$W_{tm} = -\tan \lambda \frac{U}{R} \text{ (radians per hour)}$$

where U = eastings ground speed in knots
R = Earth radius in nm.

Another form is obtained by multiplying by 57·3:

$W_{tm} = -\frac{U}{60}\tan\lambda$ (degrees per hour)

If the error is East the error is, as for Earth rate, –ve in the northern hemisphere or +ve in the southern hemisphere. Therefore the correct result is obtained by ascribing a +ve sign to U when motion is easterly and a –ve sign when westerly.

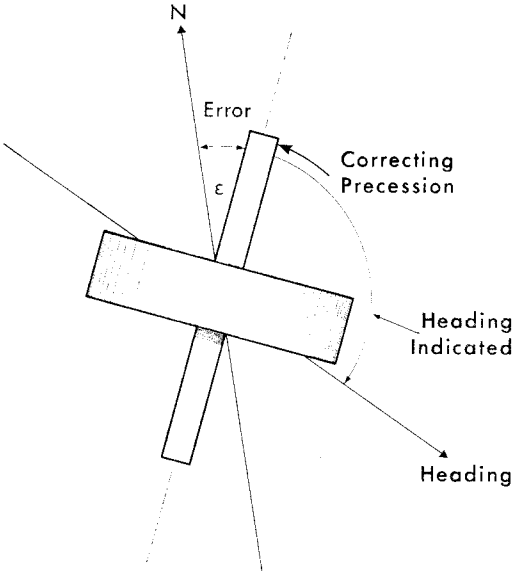


Fig 31 Sign of Hang-Off Error

71. Consider an aircraft at 45°N90°W flying a true track of 135° at a groundspeed of 600 kt. The compass system time constant is 1·0 mins at 45°N. The magnetic variation is increasing westerly along track at a rate of 8° per hour. The manufacturers specification for gyro random drift is ± 1° per hour (2 σ).

72. **Hang-Off Error (HOE).** HOE is given by:

HOE = $W_T\,\tau$ (degrees)

where $W_T = W_e + W_{tm} + W_{vm} + W_R$

$W_e = -15 \sin 45 = -11{\cdot}6°$ per hour

$W_{tm} = -\frac{U}{60}\tan 45 = \frac{-600 \sin 135 \tan 45}{60}$

$= -7{\cdot}1°$ per hour.

$W_{vm} = -8°$ per hour

Total (disregarding W_R) = W_T = –26·7° per hour

HOE = $-26{\cdot}7°$ per hour × 1 min × $\frac{1}{60}$

= –0·45°

HOE due to real drift = $W_R\,\tau$

= ± 1° per hour × 1 min × $\frac{1}{60}$

= ± 0·017°

∴ Total HOE = –0·45 ± 0·017° (2σ)

73. **Coriolis Error.** The formula for coriolis error as given in para 36 is:

ε = 4·37 × 10^{-4} × G Sin λ Cos Mag track tan δ (degrees)

but in the UK this may be modified to:

ε = 4·37 × 10^{-4} × G Sin λ Cos true track tan δ (degrees)

without serious error if it is more convenient. In the following example, the error is:

= 4·37 × 10^{-4} × 600 Sin 45° Cos 135° × tan 74°

= 4·37 × 10^{-4} × 600 × 0·707 × (–0·707) × 3·49

= –0·456°

∴ Coriolis error (ε) = –0·46°.

74. **Vehicle Movement.** The error caused by fluxvalve tilt due to vehicle movement is given in para 37 as:

ε = –0·146 × 10^{-4} × G (W_{tm} + W_{vm}) Cos Mag track tan δ (degrees)

however, as with corilois, true track may be substituted in the UK without significant error. Since W_{tm} and W_{vm} have already been found in this example (*see* para 72), the error is:

= 0·146 × 10^{-4} × 600 × (–15·1) × (–0·707) × 3·49

= –0·326

∴ vehicle movement error (ε) = –0·33°.

75. **Other Predictable Errors.** Other errors, such as those discussed in paras 65 to 68, can be estimated from data such as manufacturing tolerances and trials results. Therefore, system transmission errors may contribute less than ± 0·50 (2σ) and compass swinging errors to less than ± 0·2° (2σ). Lack of knowledge of variation is less well defined and little information is available but errors as great as 2° should be anticipated. However, over the UK, knowledge of variation is considered to be ± 1° (2σ). Using a statistical method, the root-sum-square (RSS) technique, the statistical errors (including the real drift contribution to HOE) may be summed giving an error which on 95% of occasions will be less than ± 1·1° (2σ).

76. **Total Error.** Under the flight conditions given in para 72, the error will be within the total given by:

$$\varepsilon = \text{HOE} + \text{coriolis} + \text{vehicle movement} \pm \text{estimated errors}$$
$$= 0{\cdot}45^\circ + (-0{\cdot}46^\circ) + (-0{\cdot}33^\circ) \pm 1{\cdot}19^\circ$$
$$= -1{\cdot}24 \pm 1{\cdot}19\ (2\sigma)$$

METHODS OF ERROR REDUCTION

General

77. **Turning Error.** The most common practice in limiting fluxvalve induced turning errors is to revert the system to an unslaved directional gyro mode whenever turns are required. Turn sensing can be automatically accomplished by vertical gyros, rate gyros or gravity detection devices, *eg* liquid levels.

78. **Coriolis Error.** The coriolis error can be calculated and compensated automatically by supplying a computer with the error formula parameters.

79. **Vehicle Movement Error.** The same technique applies for vehicle movement error as applied to coriolis error.

80. **Fluxvalve Vibration.** The effects of fluxvalve vibration can be limited to small values through careful design of the pendulous detector damping mechanism and through the location of the detector in the aircraft to ensure a low sympathetic vibration.

81. **Northern Instability.** Weaving can be reduced to a certain extent by increasing the time constant of the compass system. However, it should be borne in mind that this leads to a sluggish response and to a large hang-off error. A balance is struck by a time constant of about 40 to 90 sec for most systems in UK latitudes, and by using the limited linear system with a maximum precession rate at about 3° per min.

82. **Hang-Off Error.** HOE is caused by gyro drift in a gyro-magnetic compass system in the slaved mode of operation. To reduce the HOE, gyro drift (real and apparent) must be reduced. Whenever gyro corrections are applied, they are best done "down-stream" of the gyro, *ie* the gyro performs best when left undisturbed.

a. *Real Drift.* Real drift can only be reduced by the incorporation of a high quality azimuth gyro having a low real drift rate.

b. *Earth Rate.* Apparent azimuth gyro drift due to Earth rotation can be countered by correcting the gyro at a rate of 15° per hour $\times$ sin λ. The correction can be supplied through a hand-set latitude correction mechanism, automatically from a computer which utilizes GPI latitude, or through a constantly biased gyro. The latter technique is accomplished in some current systems by applying a constant torque to the gyro through the use of a lead weight. The mass imbalance thus created will precess the gyro at a predetermined rate, usually to compensate for W_e at 51°N.

c. *Vehicle Movement.* To compensate a system for convergence of geographic meridians, the gyro or gyro output must be corrected at a rate equal to $\frac{U}{60} \tan \lambda$ (° per hour). This correction can be applied through a hand-set precession mechanism which is calibrated in degrees per hour or through a computer correction procedure which receives inputs of ground speed, heading and latitude. The correction for apparent azimuth gyro drift due to magnetic meridian convergence is more difficult to achieve. Originally moving the compass needle by the amount of variation was the most obvious way of applying variation in an aircraft compass system (*see* Fig 32), but this did not remove the hang-off error caused by a change in variation since the slaving system still had to continually correct the true North seeking gyro (corrected for ER and TW) towards magnetic North. If the variation is added to the output of the detector unit before the gyro error loop (*see* Fig 33), both gyro and fluxvalve give directional information about true North and there will be no movement between the two sensors, hence, no hang-off error. Small hang-off errors will occur in this latter system if the variation is not constantly updated. An Automatic Variation Setting Control Unit (*see* Part 2, Sect 3, Chap 5, Appendix) is installed in some aircraft to ease the navigator's workload and to reduce hang-off errors.

83. **Gimbal Error.** Gimbal error in a gyro-magnetic system can be eliminated by keeping the gyro assembly (gyro spin axis and azimuth output synchro) in the local horizontal plane at all times. This is achieved by using a four gimbal gyro suspension. Less effective, but less expensive, are correction techniques in which a three gimbal system is roll stabilized or in which gimbal error is computed and electronically compensated. Such a computer must be fed with roll angle (and pitch if full compensation is desired) and the

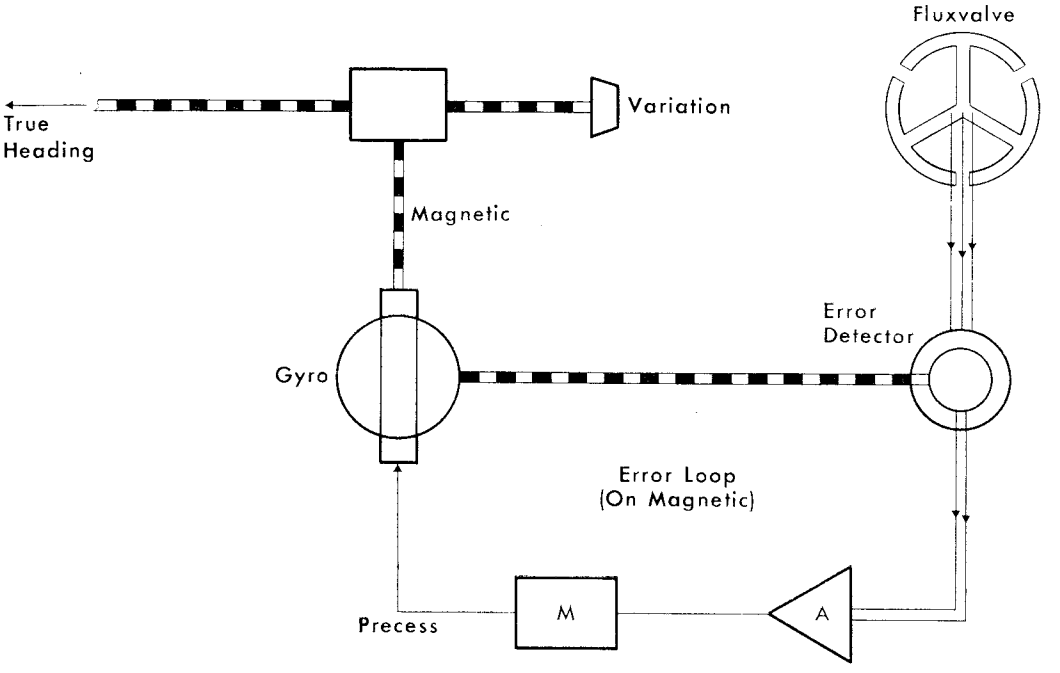


Fig 32 Variation Incorrectly Applied—HOE Present

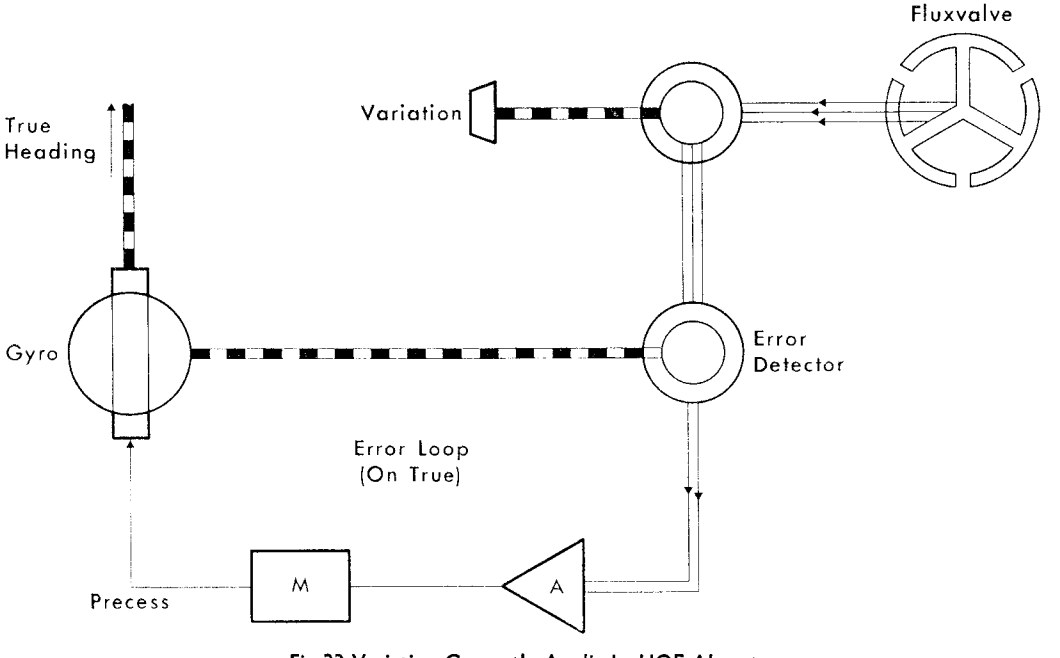


Fig 33 Variation Correctly Applied—HOE Absent

angular relationship between the gyro spin axis and the aircraft longitudinal axis.

A Refined Compass System

84. Fig 34 depicts some of the methods of error reduction. Different methods of correction are possible for some of the errors depending on the whims of the individual manufacturer and the users considerations of experience and accuracy. Note that corrections may be made "up" or "down-stream" of the gyro or a combination of both; there are advantages and disadvantages to all approaches.

85. The following description applies to Fig 34:

a. *Hang-Off.* The computer supplies the quantities for Earth rate and meridian convergence to the error detector. Therefore, the rate of gyro drift sensed is reduced considerably and hang-off results from only random drift.

b. *Coriolis and Vehicle Movement Accelerations.* The corrections for coriolis and vehicle movement are applied at the fluxvalve by reducing or increasing the output from the athwartships spokes.

c. *Gimbal Error.* Gimbal error is eliminated by the use of a vertical gyro coupled with four gimbal suspension to keep the azimuth gyro and the azimuth pick-off synchro horizontal.

d. *Operation on DG.* The fluxvalve monitor and the computer rate of change of variation are cut out when on DG. The accuracy of the heading then depends on random drift error, the error in the gyro correction terms and the statistical error *ie* transmission error.

e. *Northern Instability.* Variable gain in the precession amplifier maintains the value of τ constant, for variable H, thus reducing weaving.

f. *Coefficient D and E.* A compensation is applied to counter coefficients D and E.

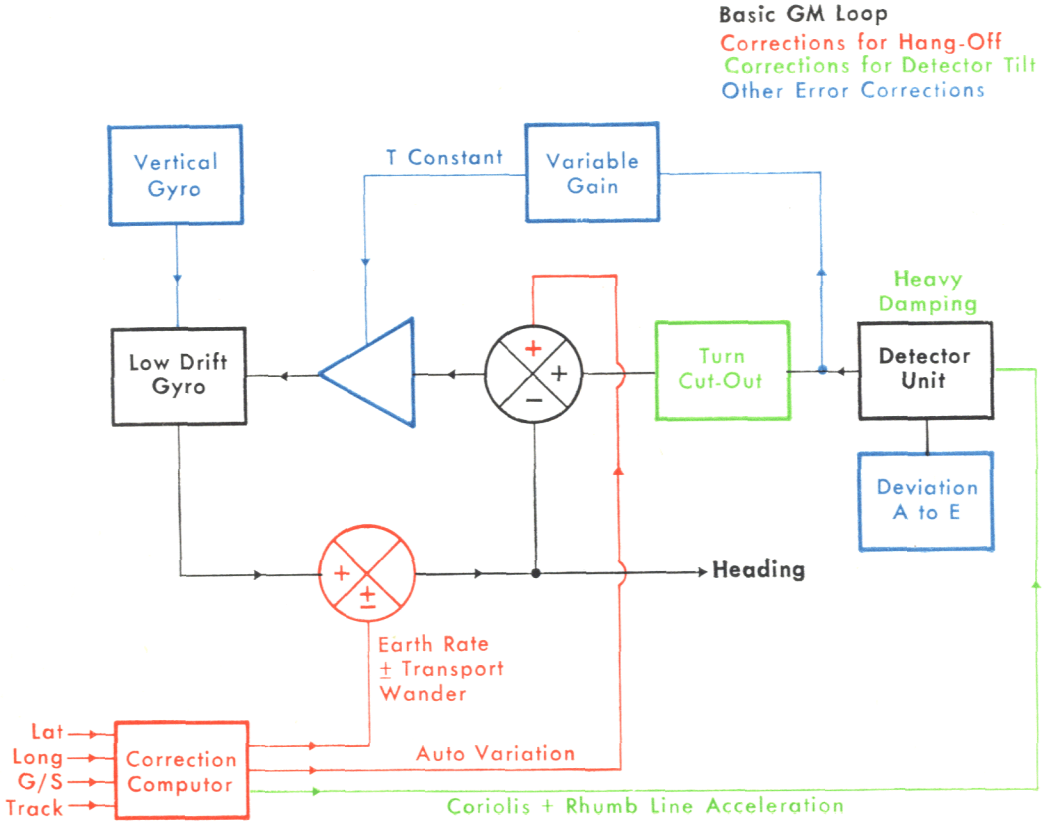


Fig 34 Ideal Gyro-Magnetic Compass

FLUXGATE AND SMITHS DETECTOR UNIT THEORY

CONTENTS

	Paras
Introduction	1—2
THE FLUXGATE	
General Description	3—4
Operation	5—9
SMITHS DETECTOR UNIT	
General Description	10—13
SUMMARY	
Comparison	14

Introduction

1. The fluxvalve theory of operation discussed in the main chapter does not hold exactly true for all gyro-magnetic compass systems. Different manufacturers have developed and patented remote magnetic detector units which have significant design differences from those previously discussed under fluxvalve theory. Before proceeding, the reader is advised to be familiar with the basic fluxvalve theory as detailed in the main chapter.

2. Regardless of the manufacturer, the remote magnetic detection unit attempts to sense the direction of the horizontal component of the Earth's magnetic field thereby establishing a magnetic heading reference for the related compass system.

THE FLUXGATE

General Description

3. The Fluxgate transmitter (*see* Fig 1) is mounted, as remotely as possible from stray magnetic fields, in a wing tip location. Mounting is by means of three legs equally spaced around the case of the unit. An elongated slot in each lug permits the correct orientation of the transmitter relative to the aircraft longitudinal axis. One of the lugs is engraved AFT and incorporates a scale calibrated over the range 10-0-10 degrees to facilitate correction for coefficient A during installation.

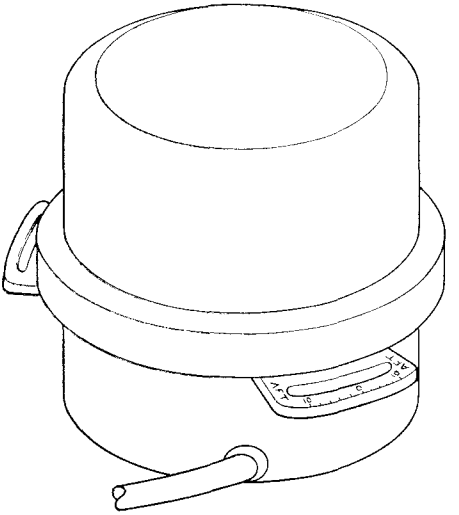


Fig 1 Fluxgate Transmitter

4. The basic components of the fluxgate transmitter are the fluxgate element and the magnetic suspension system (*see* Fig 2). The magnet assembly consists of two bar magnets suspended below a compass card. The card is pivoted on top of the vertical shaft which supports the fluxgate elements and is free to align itself with the Earth's field. The fluxgate sensor is pendulously supported by a gimbal system which permits freedom of movement in the fore-and-aft, and lateral direction within an arc of 20° from the vertical axis. The outer gimbal prohibits freedom of movement in azimuth thus establishing a fixed positional relationship between the fluxgate element and the housing. Consequently the fluxgate element turns with the aircraft in azimuth. The assembly, comprising both pendulous elements, one fixed in azimuth and the other free, is totally immersed in a fluid contained in the housing. The damped pendulosity of the assembly ensures that the elements always lie in the same plane as the horizontal component of the Earth's field during unaccelerated conditions of straight and level flight. The fluxgate element (*see* Fig 2) is essentially a transducer consisting of three split highly permeable cores, physically arranged in the form of an equilateral triangle. An exciter coil is wound in phase opposition to each half of each core. The exciter windings on the three cores are connected in series and supplied with a 400 Hz excitation voltage. An output coil is wound on each core over both exciter coils. The three output coils are star-connected, the outer connections providing the three-wire output for the fluxgate transmitter. The output signal is an 800 Hz current which is essentially the same as the output from a fluxvalve. This 800 Hz signal is used to set up a stator field in a receiver synchro in exactly the same way as a fluxvalve.

Operation

5. The operation of the fluxgate is basically the same as the fluxvalve but with two minor exceptions, viz the "delta" arrangement of the sensing element and the rotating compass card in the detector.

6. The "delta" connection ensures, as does the "Y" of the fluxvalve, that there is no heading ambiguity since there is always a voltage induced into at least two detector coils. Also, there is

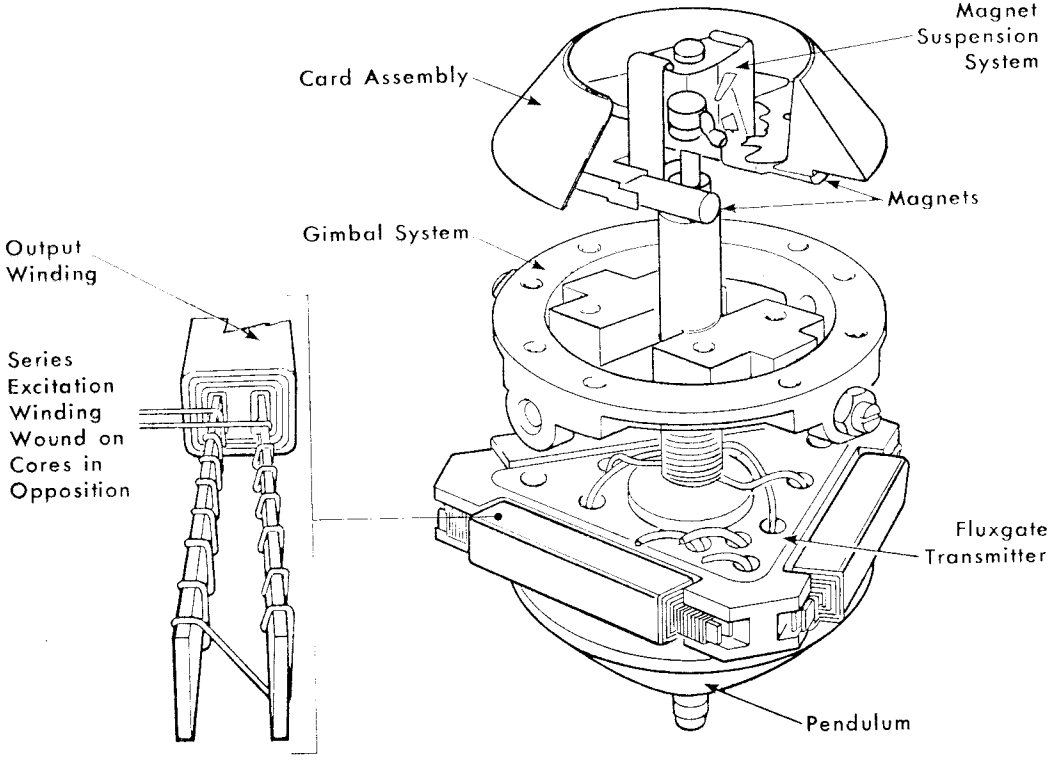


Fig 2 Fluxgate Transmitter Assembly

no heading error due to change in magnetic latitude. Therefore, there is no functional difference between the detectors because of the different winding configuration.

7. Mounted within the fluxgate unit, and free to rotate azimuth, is a compass card carrying two permanent magnets. The magnets are included to reinforce the Earth's field, both to increase the magnitude of the voltages induced in the output coils, and to provide a more uniform field. Augmenting the Earth's field in this manner results in reliable monitoring signals from the fluxgate transmitter theoretically being maintained at higher latitudes than normal, the increase in signal permitting the use of a smaller fluxgate unit.

8. The pendulous attachment of the fluxgate element ensures that the cores always lie parallel to the Earth's axis, but rotate in azimuth in accordance with changes in aircraft heading. The fluxgate is, therefore, subject to the same acceleration errors as the fluxvalve.

9. Due to the effects of the Earth's field on the cores, the output windings have different voltages induced in them, depending on their respective orientation in relation to the Earth's magnetic meridian. Only one combination of voltages is possible for a specific magnetic heading, this voltage pattern being reproduced, via a three-wire connection, in the stator windings of the fluxgate follow-up synchro. Since the Earth's magnetic field enters the fluxgate cores at each half cycle of the excitation voltage, the frequency of the signals induced in the fluxgate output windings is doubled and appears at a frequency of 800 Hz in the stators of the receiver synchros (*see* Fig 3).

SMITHS DETECTOR UNIT

General Descriptions

10. The Smiths detector unit, the elements of which are shown diagrammatically and schematically at Figs 4 and 5 respectively, is mounted in the aircraft in a position remote from local magnetic disturbance. Three holes, by which the unit is secured to the aircraft, are slotted to permit the unit to be turned to remove coefficient A. A scale of 10° is engraved on the forward edge of the mounting flange to facilitate adjustment. The function of the unit is to produce two 800 Hz currents, the amplitudes of which vary as the sine and cosine of the magnetic heading of the aircraft. The unit requires an excitation current of 400 Hz AC.

11. Each detector unit is completely sealed and is constructed in two parts, upper and lower. The upper part of the unit contains two electromagnetic corrector coils, one parallel and the other perpendicular to the longitudinal axis of the aircraft. These coils are fed with DC which is adjusted at the resolver unit to compensate for coefficients B and C.

12. The detector system consists of two pairs of detector elements, one pair at 90° to the other, pendulously suspended in a liquid-filled bowl. The assembly is suspended so that it is free to swing up to 22° from the horizontal, but it cannot turn in azimuth. One pair of elements is aligned with the longitudinal axis of the aircraft and the other with the lateral axis. Each of the elements consists of a simple one spoke fluxvalve. The primary coils of all elements are wound in series, as is each pair of secondary coils. The 400 Hz supply fed to each primary coil saturates the core during most of each half-cycle. In the absence of an external magnetic field, the change of flux in each section of the primary coil is equal and opposite, and the voltages induced in the secondary by the two sections cancel. Thus no current flows in the secondary coil. However,

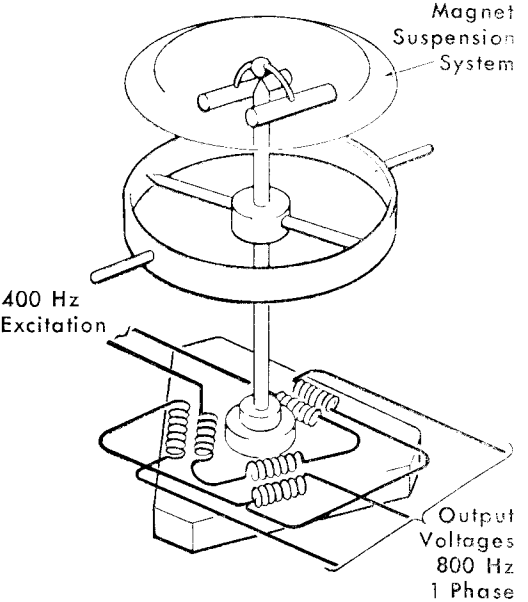


Fig 3 Fluxgate Transmitter, Simplified

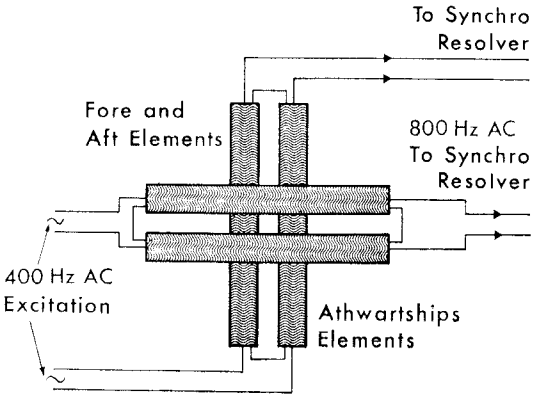


Fig 4 Configuration of Detector Elements

when the element is influenced by an external magnetic field such as the Earth's, the component of the field along the axis of the primary coil causes one section to become saturated before the other section. This momentary imbalance causes the voltages induced in the secondary by the two sections to be unequal and thus a current flows in the secondary. Since the imbalance in the primary occurs twice in each cycle, the resultant in the secondary is an 800 Hz current with an amplitude proportional to the component of the external magnetic field operating along the primary. Fig 6 shows that the lateral and longitudinal detector elements produce currents with amplitudes that are proportional, respectively, to the sine and cosine of the magnetic heading. These currents are fed to the synchro resolver in the heading indicator.

13. A synchro resolver (RS) consists of two stator coils mounted at right-angles to each other, and a rotor coil (Fig 6). The two stator coils are fed with the currents from the detector unit elements. The currents in the stators create a magnetic field, and since the direction of this field is controlled by the direction of the external magnetic field at the detector, it represents the direction of the local magnetic meridian. Unless the rotor is at right-angles to the magnetic field, *ie* in the null position, a current is induced in it. This current is used by the monitoring amplifier to drive the rotor to the null position, via the azimuth gyro and its azimuth synchro transmitter. The stator coils in the resolver can be rotated through 10° by a key inserted in the face of the heading indicator. Up to ± 5° of coefficient A can be corrected by this method.

SUMMARY

Comparison

14. Although the fluxvalve, fluxgate, and Smiths magnetic detector are all constructed differently,

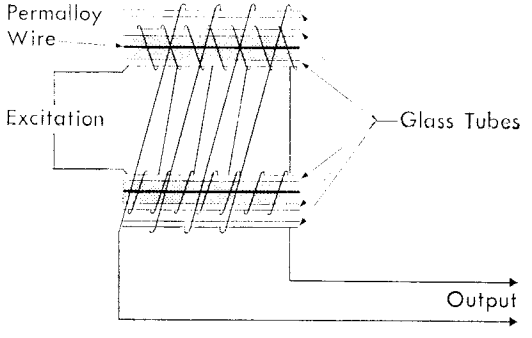


Fig 5 Single Detector Element

the function of the units remains basically the same, *ie* to provide a long term directional reference to a gyro-magnetic compass system azimuth gyro. All are subject to basically the same errors and limitations and all produce outputs of comparable accuracy.

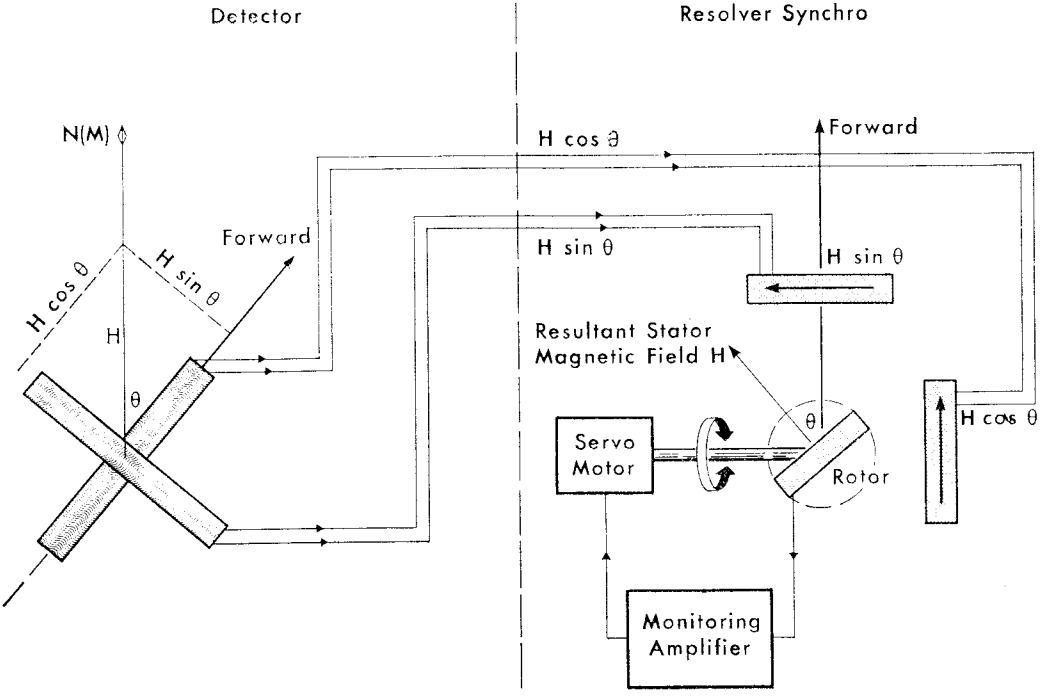


Fig 6 Interconnection of Detector and Synchro Resolver

CHAPTER 2

MK 4 AND MK 5 SERIES GYRO-MAGNETIC COMPASSES

CONTENTS

	Paras
Introduction	1—2
GYRO-MAGNETIC COMPASS, MARK 4B	
Components	3
Detector Unit	4—5
Gyro Unit	6—11
Master Indicator	12
Amplifier Unit	13—15
Control Panel	16—19
Principles of Operation	20—26
GYRO-MAGNETIC COMPASS, MARK 4F	
Components	27
Detector Unit	28
Gyro Unit	29—31
Amplifier Unit	32
Corrector Control Box	33
Principles of Operation	34—39
GYRO-MAGNETIC COMPASS, MARK 5	
Components	40
Detector Unit	41
Gyro Unit	42
Amplifier Unit	43—46
Indicator Units	47—55
Principles of Operation	56—61

Introduction

1. The Mk 4 series consists of two basic models, the Mk 4F and the Mk 4B, which are generally referred to as the G4F and G4B compasses respectively. The G4F is intended for use in single seater fighter aircraft and provides an accurate, dead beat indication of magnetic heading on a single indicator. The G4B is a more complex instrument for use in multi-seat aircraft. It has facilities for the setting of magnetic variation and a powered repeater system to provide accurate heading to remote indicators, ADRIS, bombsights, and radar systems.

2. The Mk 5 gyro-magnetic compass was designed to provide, for fighter aircraft, outputs of aircraft heading in all possible attitudes. Previous types of remote indicating compasses were restricted in pitch and roll movements by mechanical stops necessary in the particular type of gyro used in these systems. The G5 compass obtains its freedom from attitude restrictions by using the azimuth gyro element of the Master Reference Gyro Unit Mk 1 (MRG Mk 1).

GYRO-MAGNETIC COMPASS MK 4B

Components

3. There are five basic units to the G4B. These are:

a. Detector Unit with Deviation Corrector.
b. Gyro Unit.
c. Master Indicator.
d. Amplifier Unit.
e. Control Panel.

Detector Unit

4. The detector unit contains a pendulous sensitive element mounted on a Hooke's joint which enables it to swing within limits of 25° about the pitch and roll axes, but allows no rotation in azimuth. It is contained in a sealed case partially filled with oil to dampen oscillations. The complete unit, shown in Fig 1 is usually mounted in the wing tip or tail fin where magnetic disturbances are a minimum. The detailed operation of a fluxvalve of this sort is described in Chapter 1.

5. A remotely controlled electromagnetic corrector, mounted at the top of the detector unit, is used for compensating coefficients B and C so that corrections of up to ± 15° can be made from the flight deck. This unit is shown in Fig 2.

Gyro Unit

6. The gyro unit, which is the pilot's indicator, contains a gyroscope, a control transmitter (CX), a compass card, a heading setting facility, an annunciator and a compass synchronizing control. The unit is shown in Figs 3 and 4.

7. An erection mechanism maintains the gyro rotor axis horizontal; it comprises a levelling switch and a torque motor. The torque motor is mounted on the top bearing of the vertical ring. It has two windings, one of which is permanently energised, the other is controlled by the levelling switch. This levelling switch consists of a commutator which is fixed to the inner gimbal ring and is divided by an insulated strip. Brushes, diametrically opposite to each other, are fixed to the outer gimbal ring. The arrangement is shown in Fig. 5.

8. When the gyro axis is horizontal with respect to the airframe, the brushes are in contact with the insulating strip and no current flows to

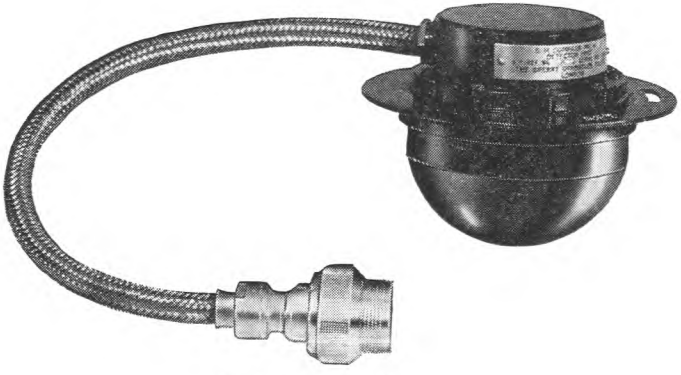


Fig 1 Detector Unit

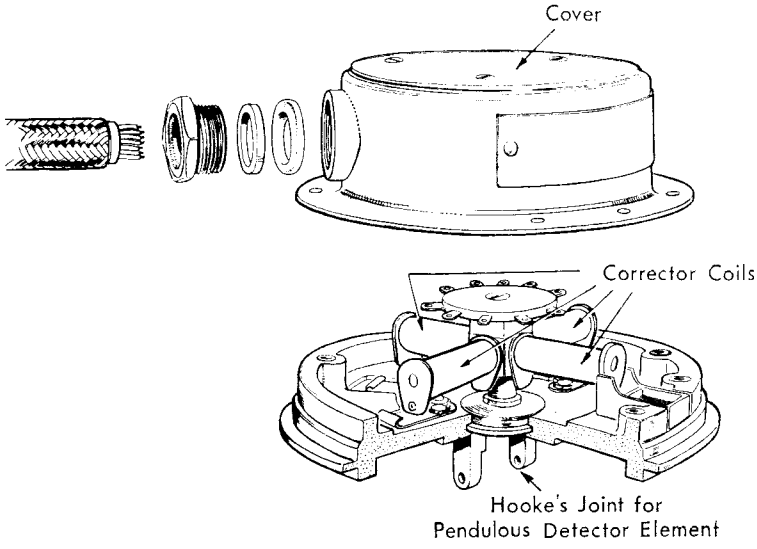


Fig 2 Section Drawing of Deviation Corrector

the second winding of the torque motor. When the gyro axis departs from the horizontal, the brushes make contact with the segments of the commutator and the torque motor is energized. The torque is applied to the inner ring. The normal action of the gyro is to rotate as if the force was applied 90° removed in the direction of rotation (precession), thus the rotor axis returns to the horizontal. The polarity of the current energizing the torque motor controls the direction of the torque applied to the vertical ring to precess the rotor axis in the correct direction.

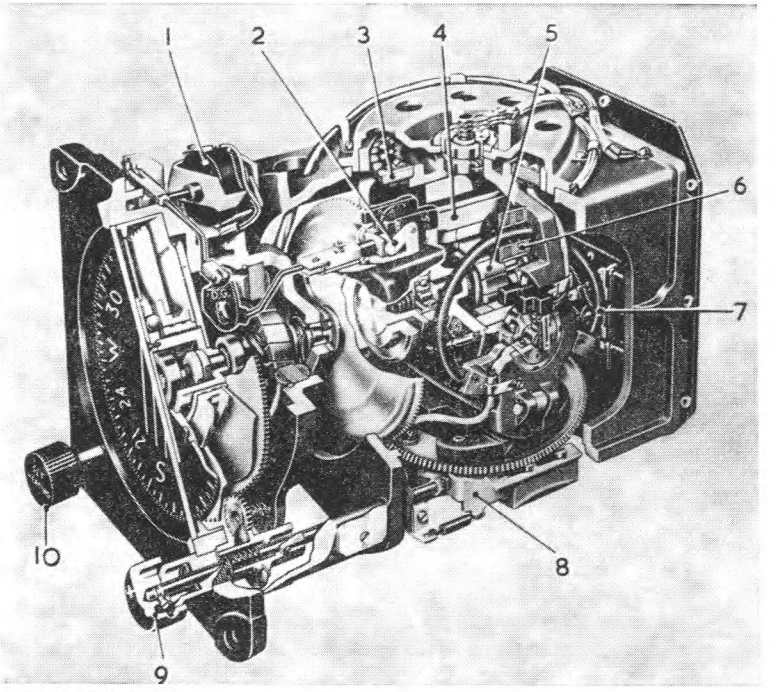
9. **Precession.** Precession in azimuth is achieved by a precession motor mounted on the vertical ring, *see* Fig 5. One of the two coils is permanently energized, the other is energized when the rotor of the control transformer (CTB) in the master indicator and is not in its null position. Signals are then fed via the precession amplifier to the second precession coil. Torque is then applied to the inner gimbal which causes the gyro to precess in azimuth.

10. **Annunciator.** The DC output of the precession amplifier is also fed to the annunciator circuit, which indicates whether or not the compass is synchronized. The annunciator consists of a pivoted arm, one end of which carries a flag marked with a dot and a cross, and the other a small permanent magnet. The magnet is placed between two annunciator coils and is attracted to one or the other depending on the values of the currents in the two coils. These depend on the amount and direction of the misalignment of the rotor of the CTB. The indication is either a dot or a cross being shown in the annunciator windows of the pilot's repeater and the master indicator. When the compass is synchronized, the flag lies midway between the annunciator coils and a point midway between the dot and cross should be seen. However, owing to aircraft vibration, the synchronized position is generally indicated by a slow oscillation between dot and cross. The annunciator operation is shown schematically in Fig 6.

11. **Synchronization.** To synchronize the compass, push in the synchronization knob of the gyro unit selected at the control panel and turn in the direction corresponding to the indication of the annunciator. Continue turning the knob until an indication midway or alternating between dot and cross is obtained. It will not normally be necessary to re-synchronize the compass in flight unless the operating limits of the gyro are exceeded causing the gyro to topple, or the compass has been used in the DG mode. Pressing in the synchronizing knob automatically erects the gyro.

Master Indicator

12. The master indicator, which mounted at the navigator's position, is shown in Fig 7. It comprises a motor, a control transformer with a



1 DG Solenoid
2 Annunciator Coil Assembly
3 Torque Motor
4 Gyro Rotor
5 Gyro Stator
6 Precession Coils
7 Precession Armature Ring
8 Caging Mechanism
9 Synchronizing Knob
10 Set Heading Knob

Fig 3
Gyro Unit Type B, G4B Sectional View

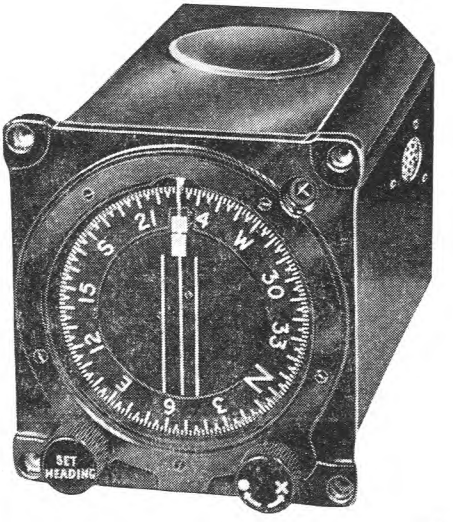


Fig 4 Gyro Unit Type B, G4B

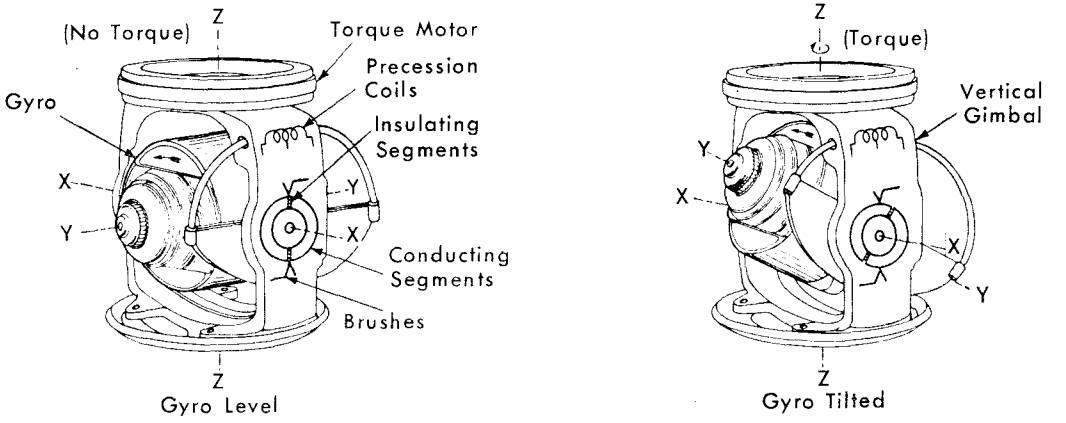


Fig 5 Erection Mechanism

Note:

1 The levelling Switch is Indicated outside the vertical gimbal ring for the convenience of explanation. It is actually on the inside.

2 The gyro housing serves as the inner or horizontal gimbal.

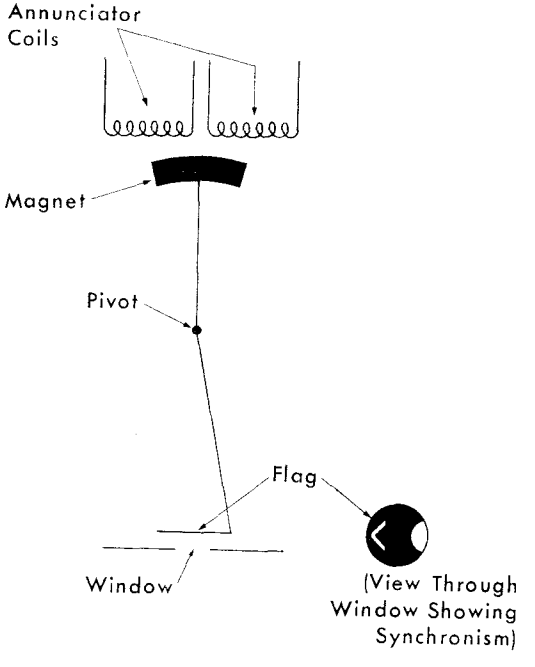


Fig 6 Annunciator Operation

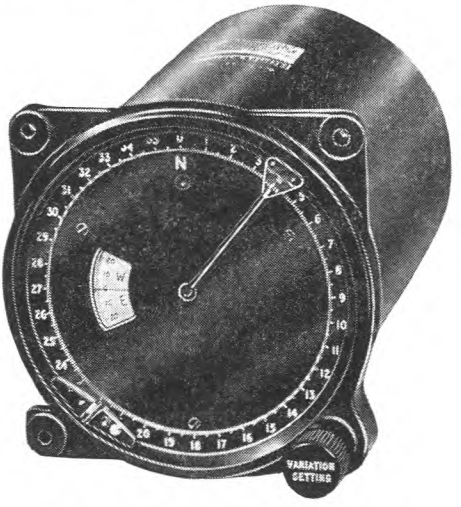


Fig 7 Master Indicator Type B, G4B

rotatable stator (CTB), a control transmitter (CX), a heading pointer, an annunciator, a step-by-step transmitter and a variation setting facility. The value of variation set is shown on a degree graduated scale through a window on the front of the instrument. Variation up to 180° East or West can be set.

Amplifier Unit

13. The amplifier unit comprises two separate amplifiers, the precession amplifier and the follow-up amplifier. The former is employed to amplify, rectify and phase detect the monitoring signals relayed from the detector unit before they are applied to the precession coils on the gyro. The follow-up amplifier provides suitable AC power, related in amplitude and phase to the signal output from the rotor of the CTB in the master indicator, which is used to energize the follow-up motor and so maintain synchronism between the master indicator and the gyro unit.

14. Incorporated in the amplifier unit is a DC voltage stabilizing circuit and two centre-tapped potentiometers for adjusting the current to the electromagnetic deviation compensator. When the corrector circuit is operating correctly, two lamps in the amplifier are illuminated. As these bulbs are a matched pair, the compass must be swung if either or both lamps are replaced. A jack socket is provided on the amplifier for an external centre-zero reading voltmeter, used for calibrating the compass.

15. A relay is included in the precession amplifier to enable the compass monitoring to be interrupted. This facility is used when it is necessary to reduce the errors arising at the detector unit during turns.

Control Panel

16. The control panel is used to control the monitoring signals to the gyro unit. A type A panel is illustrated in Fig 8, and the type B differs only in that:

a. Panel illumination is provided by two bulbs, using the Plastek lighting principle.
b. The shape of the selector switch has been changed.

17. The three-position selector switch enables either the port or starboard gyro units to be selected for monitoring by the flux valve. When the switch is in the OFF position, the monitoring signals are cut off and both gyro units function as directional gyros; the DG flag appears in the annunicator windows of the gyro units and the master indicator; the precession amplifier is switched off, only the follow-up amplifier being operative. In the OFF position, the master indicator is synchronized with the port gyro.

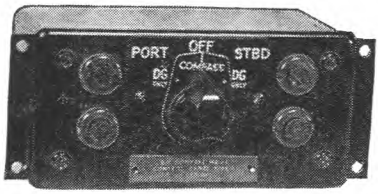


Fig 8 Control Panel Type A

18. When the switch is at PORT or STBD, the corresponding gyro unit receives a monitoring signal from the detector unit and functions as a magnetic compass. The annunciator flag appears in the annunciator window of the appropriate gyro unit and the master indicator, and the DG flag appears in the annunciator window of the other gyro unit.

19. The selector switch indicator plate is reversible and fitted with a stop so that on single gyro unit installations only two positions, GYRO COMPASS and COMPASS OFF are available. Four 2-amp cartridge fuses are fitted on the face of the control panel, two protecting the port and two the starboard gyro unit.

Principles of Operation

20. The following text should be read in conjunction with the signal flow diagram given in Fig 9, and the schematic diagram in Fig 10. The detector unit senses the Earth's magnetic field and sends signals, which vary in amplitude and sign according to the direction of the field, to the stator of the control transformer (CTB).

21. The heading pointer of the master indicator is mounted on the same shaft as the rotor of CTB, and rotates with it. The position of the heading pointer is aligned during manufacture so that a correct heading indication is given when the rotor lies in the null position at right angles to the field in the stators.

22. If during flight the pointer does not indicate the heading of the aircraft, the rotor on the same shaft will not be at right angles to the magnetic field set up in the stator by the signals from the detector unit. An error signal proportional to the amount of misalignment will then be induced in the rotor.

23. This error signal is fed to the precession amplifier where it is phase detected, rectified and amplified and then fed to a transducer which converts it to AC. The AC signals are then fed to the precession coils on the vertical gimbal of the gyro. The resultant precession of the gyro in azimuth is transferred through bevel gears into a movement of the shaft which carries the pilot's

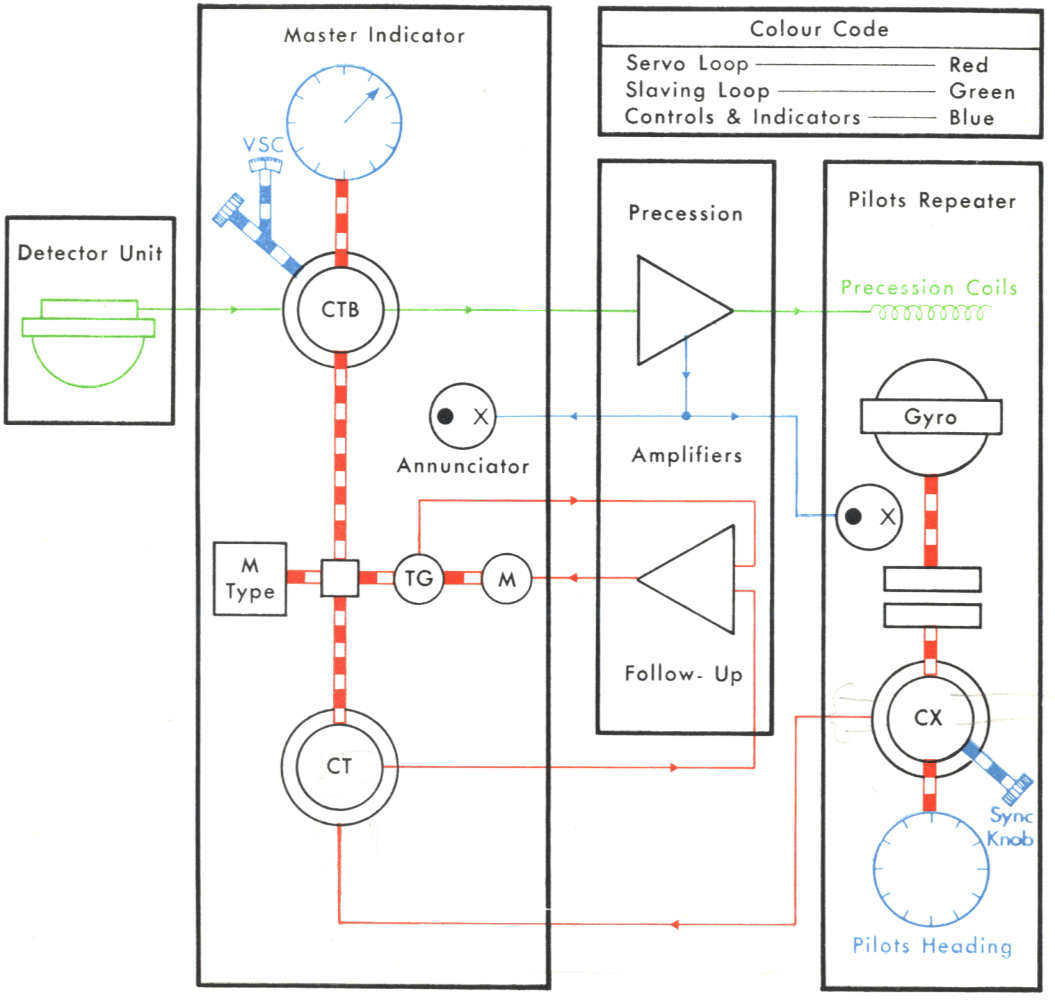


Fig 9 G4B Signal Flow Diagram

compass card and the rotor of a control transmitter (CX).

24. This CX, a control transformer (CT) in the master indicator, the follow-up amplifier and the follow-up motor constitute a control synchro transmission system, whose function is to rotate the compass card in the master indicator and the rotor of CTB, in unison with the precession of the gyro. This rotation moves the rotor of CTB towards the null position, at which position the precession signal ceases and the whole system is aligned with the magnetic meridian. Both compass cards indicate magnetic heading and the compass is said to be synchronized.

25. When the aircraft alters heading, the gyro axis, and therefore the horizontal bevel gear, remain fixed in space. The gyro unit case turns with the aircraft and the vertical bevel gear turns about the horizontal bevel gear. The resulting rotation of the shaft of the gyro unit, which carries the compass card, is equivalent to the amount of aircraft turn. The rotor of the CX is rotated with the shaft and this rotation is transmitted to CT and the follow-up motor. Simultaneously the signals from the detector unit change as the aircraft turns. The amount of the two changes are identical. The rotor of CTB turns, due to the effect of the gyro, at the same rate as the detector unit changes the magnetic field in the stators. The compass, therefore, remains synchronized. In practice a false signal may be passed to the precession coils as the fluxvalve will measure a component of the Earth's field other than the horizontal, and in a prolonged turn this may cause a slight precession of the gyro resulting in desynchronization. The compass will re-synchronize itself at approximately 2° a minute.

26. The above explanation assumes that the magnetic variation control is set to zero. When variation is set, the stator of CTB is moved through an amount equivalent to the variation. The null position is moved by the same amount, so the precession circuit is energized and the rotor is motored to the new null position. The true heading of the aircraft is now shown by the gyro unit and the master indicator and is transmitted to the compass repeaters and monitored equipment.

GYRO-MAGNETIC COMPASS, MARK 4F

Components

27. The G4F operates on principles similar to the G4B, but, being intended primarily for single seater aircraft, it is a simpler instrument. It comprises four basic units:

a. Detector Unit.
b. Gyro Unit.
c. Amplifier Unit.
d. Corrector Control Box.

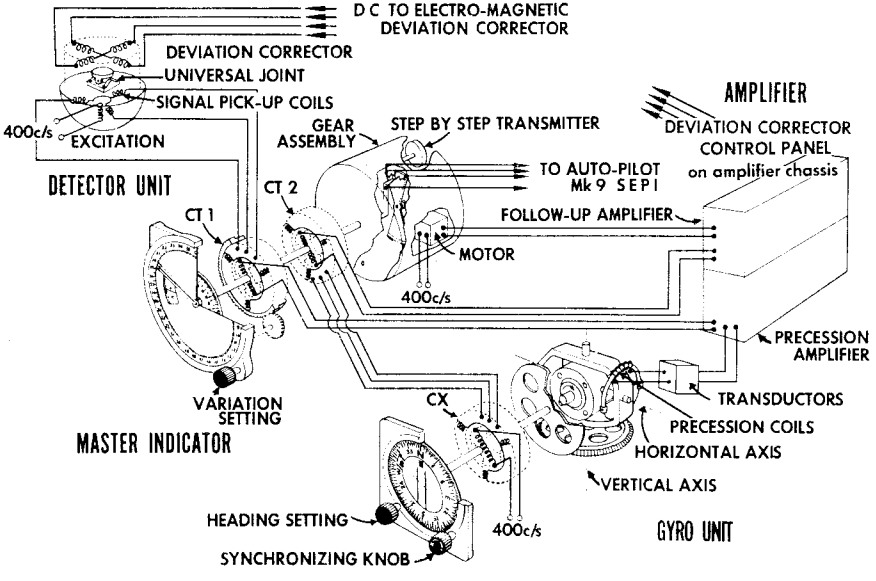


Fig 10 G4B Schematic

Detector Unit

28. The detector unit is identical with that in the G4B and the units are interchangeable. It is described in paras 4 and 5.

Gyro Unit

29. The gyro unit is similar to, but not interchangeable with the gyro unit of the G4B. The main difference is that the control transmitter of the G4B gyro unit becomes the control transformer of the G4F gyro unit. The control panel of the G4B is replaced in the G4F by a DG compass selector switch in the form of a simple left/right switch located at the top left-hand corner of the gyro unit face, as shown in Fig 11.

30. **Annunciator.** The annunciator is the same as that in the G4B and is described in para 10.

31. **Synchronization.** To synchronize the compass, push in the synchronization knob and turn in the direction shown by the indication of the annunciator until an indication midway or alternating dot/cross is obtained. It will not normally be necessary to re-synchronize the compass in flight unless the gyro exceeds its limits, $\pm 85°$ in pitch and roll, or the compass has been used in the DG mode. Pressing the synchronization knob automatically erects the gyro but should only be carried out in straight and level, un-accelerated flight.

Amplifier Unit

32. The amplifier unit is shown in Fig 12. This unit amplifies and rectifies the signals from the CT before they are applied to the precession circuits of the gyro.

Corrector Control Box

33. The corrector control box is shown in Fig 13. It operates in the same way as the deviation corrector mounted in the amplifier unit of the G4B. The current to the electromagnetic corrector is varied by means of the two controls on the face of the box, one for coefficient B and the other for coefficient C up to a maximum of $\pm 15°$. Two

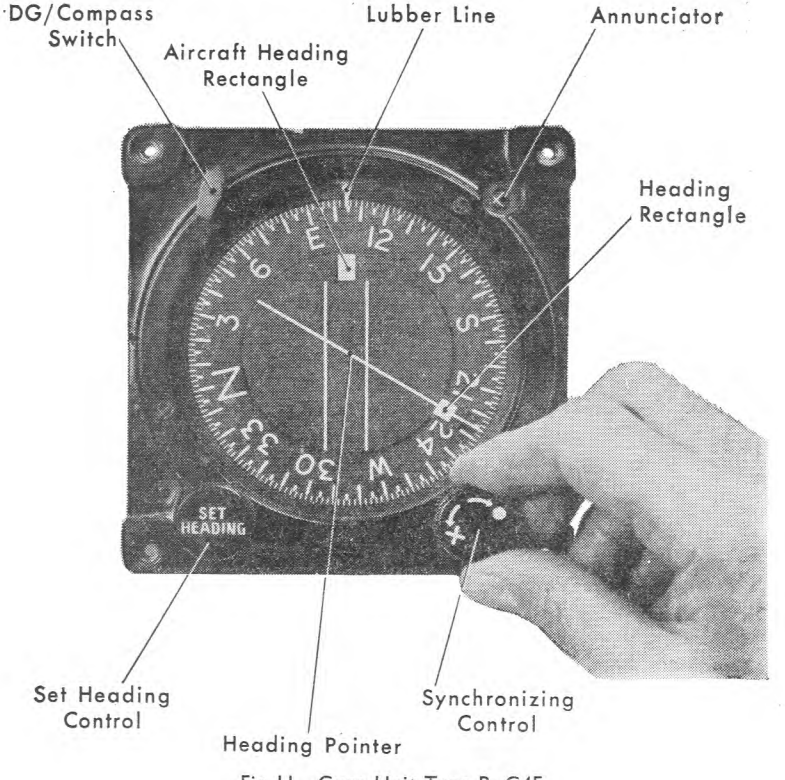


Fig 11 Gyro Unit Type B, G4F

lamps in the face of the box must be alight if the corrector circuit is to function correctly. If either lamp fails it must be replaced and a check swing carried out.

Principles of Operation

34. The following text should be read in conjunction with the signal flow diagram given in Fig 14 and the schematic diagram in Fig 15.

35. The detector unit senses the Earth's magnetic field and sends signals, which vary in amplitude and sign according to the direction of the field, to the stator of the control transformer (CT) in the gyro unit.

36. The compass card is on the same shaft as the rotor of CT and is aligned during manufacture so that when the rotor is at right angles to the field produced in the stators, the compass card correctly indicates the magnetic heading of the aircraft. If the compass card does not indicate the correct heading, the rotor on the same shaft will not be at right angles to the magnetic field set up in the stator by the signals from the detector unit. An error signal proportional to the amount of misalignment will then be induced in the rotor.

37. This error signal is fed to the precession amplifier where it is phase detected, rectified and amplified and then fed to a transducer which converts it to AC. The AC signals are fed to the precession coils on the vertical gimbal of the gyro. The resultant precession of the gyro in azimuth is translated into a movement of the shaft which carries the compass card and the rotor of the CT.

38. This rotation moves the rotor of the CT towards the null position, at which position the precession signal ceases and the system is aligned with the magnetic meridian.

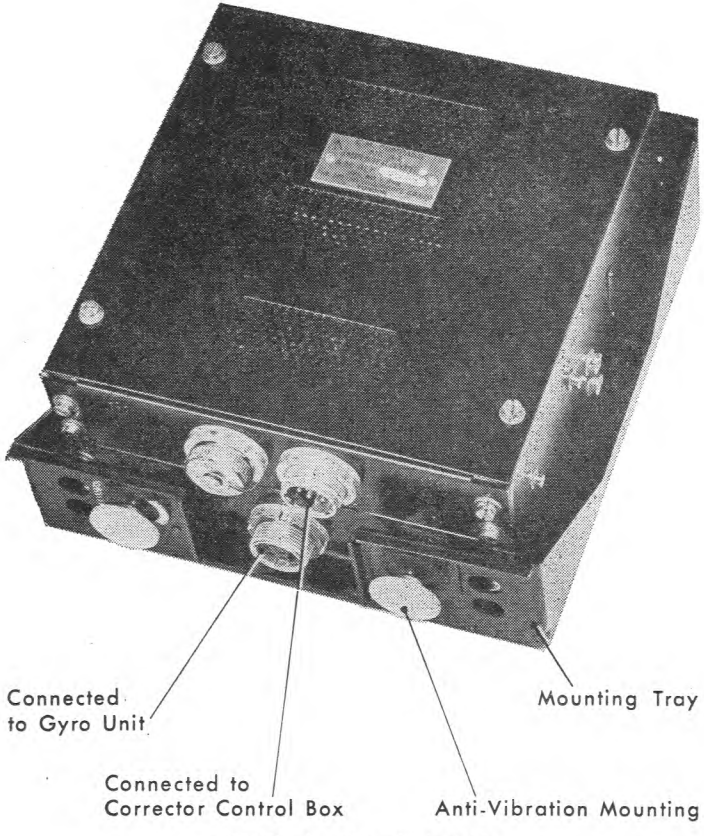


Fig 12 Amplifier Unit G4F

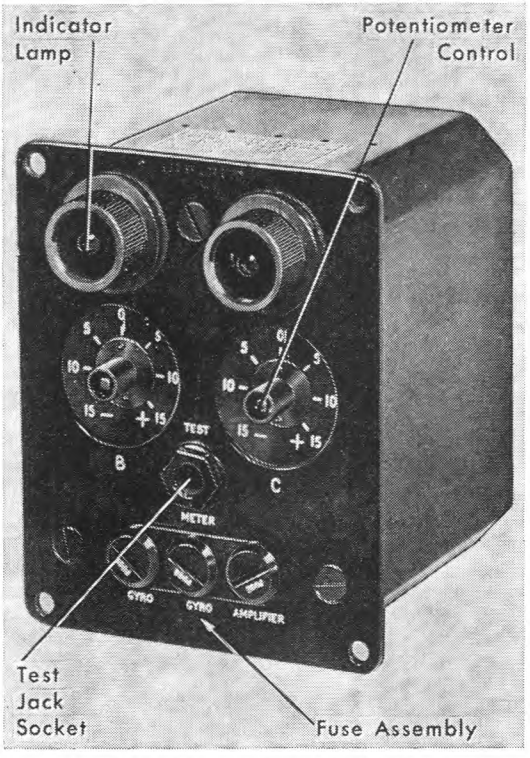


Fig 13 Corrector Control Box

39. When the aircraft alters heading, the gyro axis, and therefore the horizontal bevel gear, remains fixed in space. The gyro unit case turns with the aircraft and the vertical bevel gear turns about the horizontal bevel gear. The resulting rotation of the shaft of the gyro unit, which carries the compass card, is equivalent to the amount of aircraft turn. Simultaneously the signals from the detector unit change as the aircraft turns. The amounts of the two changes are identical. The compass, therefore, remains synchronized. In practice a false signal may be passed to the precession coils as the fluxvalve will measure a component of the Earth's field other than the horizontal, and in a prolonged turn this may cause a slight precession of the gyro resulting in desynchronization. The compass will resynchronize itself as described in paras 36 and 37.

GYRO-MAGNETIC COMPASS MARK 5

Components

40. The Gyro-Magnetic Compass Mk 5 (G5) is an integral part of the aircraft flight system. The following components are used in the compass system:

a. Detector Unit.
b. Gyro Unit.
c. Amplifier Unit.
d. Indicator Units.

Detector Unit

41. The detector unit of the G5 compass is similar to that in the G4B. It is illustrated in Fig 16. The sensitive element is suspended in a liquid filled bowl so that it is free to swing up to 22° from the horizontal in pitch and roll but it cannot turn in azimuth. The detector system consists of two pairs of detector elements one pair at 90° to the other as illustrated in Fig 17. One pair is aligned with the fore and aft axis of the aircraft. These elements operate as a simple fluxvave as is described in Chapter 1.

Gyro Unit

42. The G5 compass works in conjunction with the Master Reference Gyro Mk 1 which is fully described in Part 2, Sect 2, Chap 2. As the detector is pendulously mounted, any aircraft accelerations in the horizontal plane will result in the fluxvalve tilting and its coils sensing part of the vertical component of the Earth's magnetic field. Similarly, if the attitude of the aircraft is more than 22° from the horizontal, the detector will foul its mounting and the resultant tilt will again cause the vertical component to be picked up. To prevent these false signals from monitoring the azimuth gyro, cut-out switches, of the mercury in glass type, are incorporated in the MRG. When the following conditions of acceleration or attitude occur, magnetic monitoring is interrupted:

a. Sustained accelerations fore-and-aft in excess of 1/12g.
b. Bank angles in excess of 10°.
c. Pitch angles in excess of 20°.

These values are commonly used but they may be changed for different types of aircraft.

Amplifier Unit

43. The main amplifier unit contains a monitoring amplifier, a servo amplifier and components to produce correcting currents for coefficients B and C. The unit is shown in Fig 18.

44. **Monitoring Amplifier.** The monitoring amplifier demodulates, amplifies and rectifies the signals from the resolver synchro in the heading indicator to produce a DC output which is fed to the coil of the azimuth gyro in the MRG unit. This signal precesses the gyro in azimuth at up

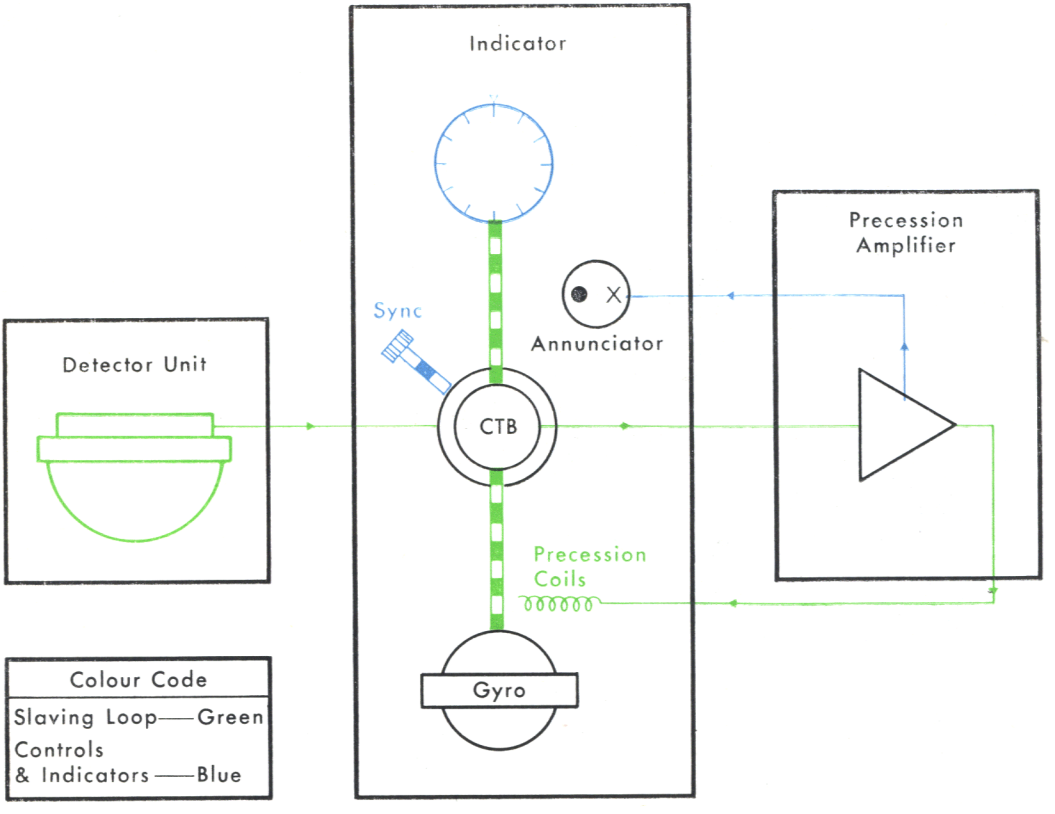


Fig 14 Signal Flow Diagram G4F

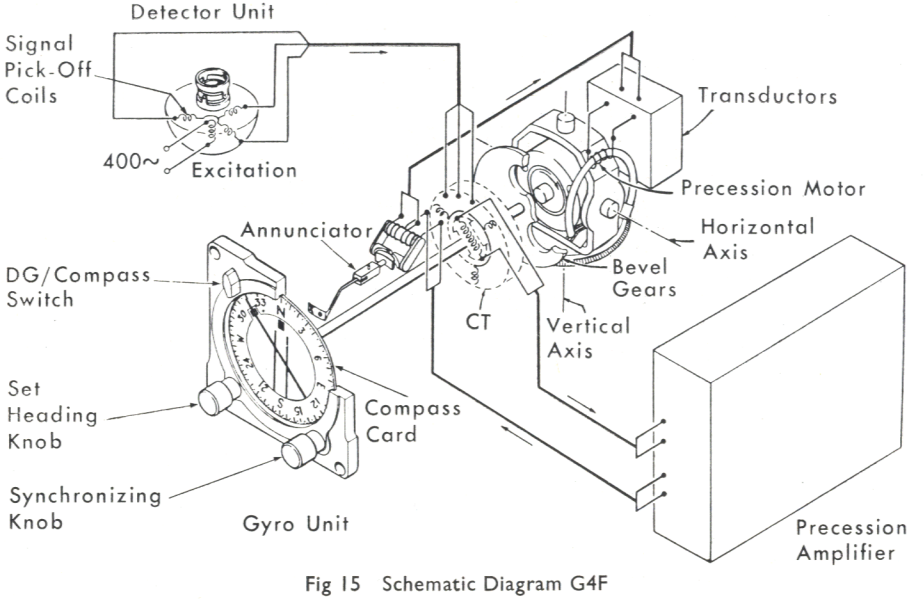
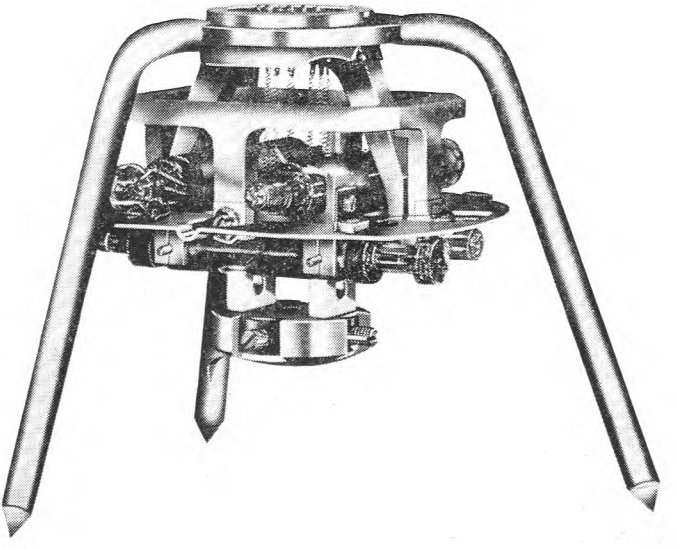


Fig 15 Schematic Diagram G4F

Fig 16 Detector Unit (2 pieces)



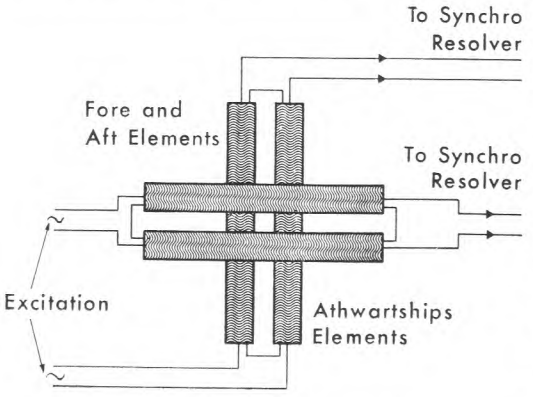


Fig 17 Detector Elements

to 3° per minute. The direction of the monitoring is indicated by the annunciator in the pilot's heading indicator.

45. Servo Amplifier. The servo amplifier provides suitable AC power, related in amplitude and phase to the signal output from the rotor of CT, to the pilot's indicator. This energizes the follow-up motor and so maintains synchronism.

46. **Coefficient Correction.** The coefficients B and C correction controls are on the amplifier, see Fig 18. Two of the controls vary the scale of the correction, giving the following ranges, 0 to +3°, 0 to −3°, 0 to +9° and 0 to −9°. Separate controls are provided for B and C correction.

Indicator Units

47. **Pilot's Indicator.** The heading indicator shown in Fig 19 is the pilot's indicator. It provides control for the system. The indicator dial displays a card, graduated at five degree intervals, which rotates against a fixed lubber line to indicate the heading of the aircraft. When the power supply to the compass is switched off, or fails, a flag marked OFF is displayed on the dial. The unit contains the controls for the system a resolver synchro (RS), a repeater synchro transmitter (CX_2), a synchro control transformer (CT_1) and a motor generator.

48. **Annunciator.** The annunciator consists of a dot/cross flag operated by solenoids. Any current from the monitoring amplifier to the precession coil of the azimuth gyro passes through these solenoids so that whenever the resolver synchro is out of the null the annunciator flag shows a steady dot or cross. However, owing to aircraft vibration, the synchronized position is generally indicated by a slow oscillation between dot and cross. When the monitoring is interrupted by the acceleration or attitude cut-out switches, or when DG is selected, the flag remains in the central position.

49. **Synchronizing Control.** The knob marked SYN at the base of the dial can be depressed and rotated to adjust the relationship between the position of the azimuth gyro and the heading pointer. The control is used to synchronize the compass to indicate the correct magnetic heading, or with the compass set at DG to set the heading card to any required reading.

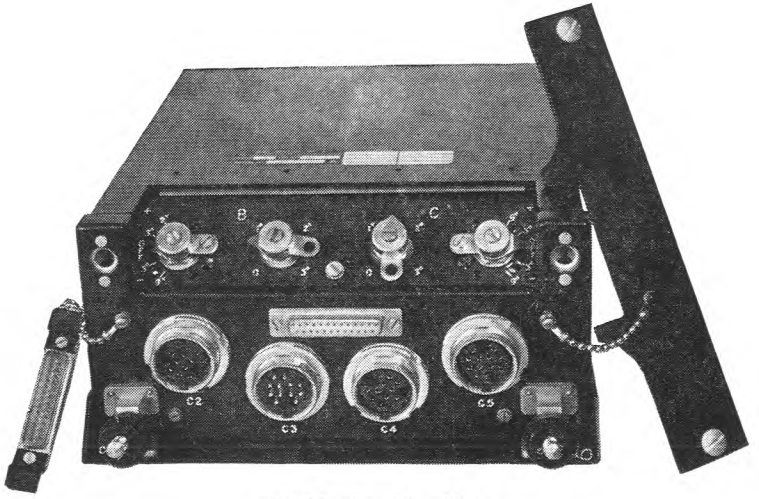


Fig 18 Amplifier Unit

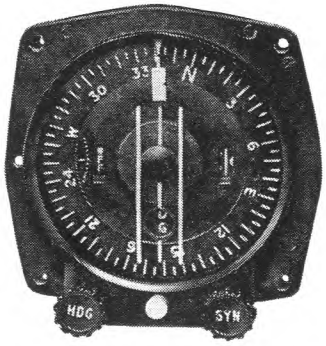


Fig 19 Pilot's Heading Indicator

50. **Required Heading Selector.** The required heading is displayed by a movable pointer which can be adjusted as required by the knob marked HDG. To fly the selected heading the aircraft is turned until the needle registers against the lubber line. This control also governs the autopilot when the autopilot is switched in, the aircraft is then maintained on the selected heading.

51. **Monitoring Cut-Out DG Switch.** The magnetic monitiring of the azimuth gyro can be cut out by depressing the button at the base of the dial. The indicator thereafter functions as a directional gyro and the letters DG appear in a window on the dial. Magnetic monitoring is resumed when the button is pressed again.

52. **Coefficient A Corrector.** Up to $\pm 5°$ of coefficient A can be removed at the heading indicator. To incorporate a correction a compass key is fitted to the triangular shaft on the left of dial and turned to set the pointer against the appropriate mark on the corrector scale. Movement of the shaft rotates the stator coils of the resolver synchro.

53. **Heading Outputs.** The pilot's heading indicator has a synchro control transmitter CX_2 geared to the servo motor. This transmits heading information to the navigator's repeater and to other equipment if required. An autopilot potentiometer may be fitted to the heading indicator. This is a fixed toroidal resistance mounted on the shaft of the synchro transmitter CX_2 and a wiper geared to the heading pointer shaft. When the aircraft is flying a selected heading and the pointer is aligned with the lubber line, the wiper is in its central neutral position. If the aircraft deviates from this heading, the wiper is rotated with the heading pointer and a voltage dependent on the heading error is fed to the autopilot. An alternative to the autopilot potentiometer is the fitting of a flight director synchro resolver transmitter to supply a heading error signal for the flight director or other equipment. When heading outputs are required in greater numbers or of different form from that of the basic system an azimuth repeater unit may be added. This consists of an additional control transformer, amplifier and servo motor aligned to the heading output of the heading indicator or the VSC. Linked to the rotor shaft of the repeater unit are a number of transmitters, M-type, potentiometer or synchro resolver which transmit heading to the further equipments requiring it.

54. **Navigator's Repeater.** The navigator's repeater is shown in Fig 20. The aircraft's heading is indicated by a moving pointer against a fixed scale graduated every five degrees. A triangular marker can be set to the desired heading by rotating the bezel ring.

55. **Variation Setting Control.** The variation setting control (VSC) is normally fitted to convert a magnetic output from the pilot's indicator to a true reading for the navigator. Variation up to 180° East or West can be set and is displayed on veeder counters. The VSC consists of a control differential (CDX) the stators of which are fed with currents from the stators of the synchro transmitter (CX_2) in the heading indicator. The rotor is turned manually through the angle of variation. The currents induced in the rotor coils are fed to CX_2 in the navigator's repeater or to other equipment requiring true heading inputs.

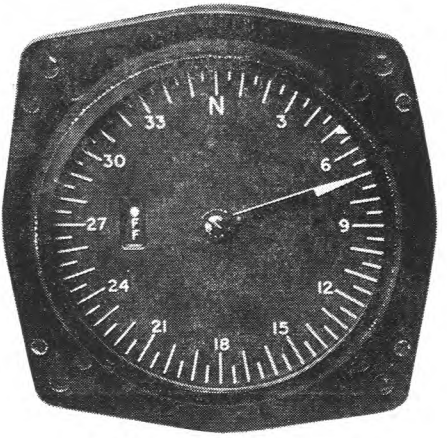


Fig 20 Navigator's Repeater

Principles of Operation

56. The following paragraphs should be read in conjunction with the signal flow diagram which is included as Fig 21.

57. The detector unit senses the earth's magnetic field and sends corresponding electrical signals, which vary in amplitude and sign according to the direction of the field, to a resolver synchro (RS).

58. The resolver synchro (RS) consists of two stator coils mounted at right-angles to each other, and a rotor coil, *see* Fig 22. The two stator coils are fed with the currents from the detector unit elements. The currents in the stators create

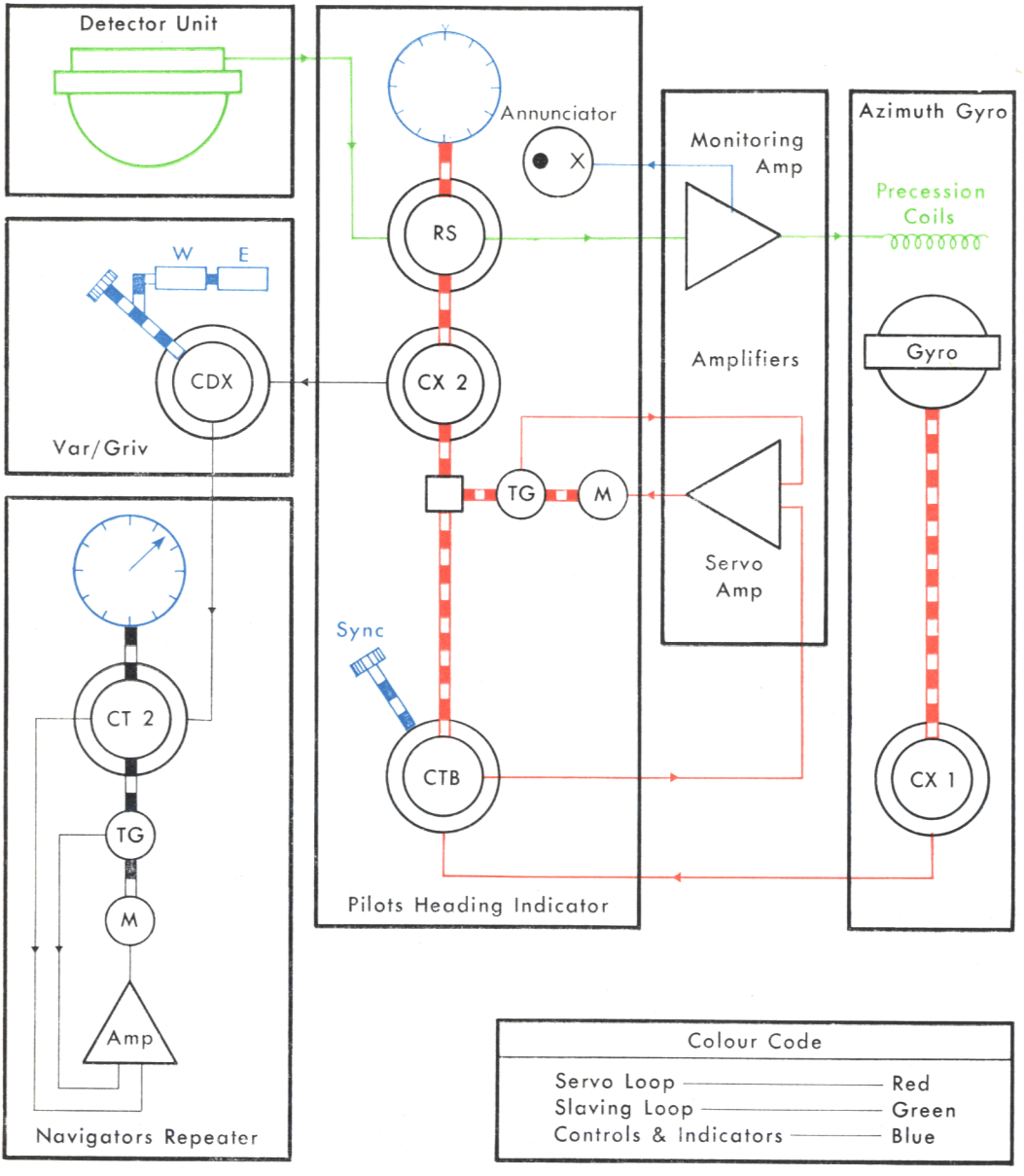


Fig 21 Signal Flow Diagram G5

a magnetic field, and since the direction of this field is controlled by the direction of the external field at the detector, it represents the direction of the local magnetic meridian. Unless the rotor is at right-angles to the magnetic field, in the null position, a current is induced in it. This current is fed via the monitoring amplifier, which rectifies, phase detects and amplifies the signal, to the precession coils of the gyro (MRG). The gyro is connected in azimuth to the rotor of a control transmitter (CX_1). This CX, a control transformer (CTB) in the indicator, the servo amplifier and the servo motor constitute a synchro control transmission system. Its function is to rotate the compass card of the indicator and the rotors of the CTB and RS in unison with the precession of the gyro. This rotation moves the rotor of the RS towards the null position. When the null position is reached the signal ceases and the system is aligned with the magnetic meridian.

59. The stator coils in the RS can be rotated through 10° by a key inserted in the face of the heading indicator. Up to ±5° of coefficient A can be corrected by this method.

60. When the aircraft alters heading, the gyro axis remains fixed in space. The synchro control transmission system, CX_1, CTB, amplifier and servo motor will cause the shaft of the indicator, which carries the compass card and the rotor of the RS, to turn by an amount equivalent to the aircraft change of heading. Simultaneously the signals from the detector unit change as the aircraft turns. The amount of these two changes is identical, therefore the compass remains synchronized.

61. **Manual Synchronization.** The synchronizing control rotates mechanically the stators of the CTB. The effect of this rotation is to alter the relationship between the direction of the azimuth gyro axis and the direction of the magnetic field of the CTB stators. When the heading indicator is being magnetically monitored, the SYN knob will only turn in the direction required to synchronize it with the correct reading. A ratchet wheel prevents synchronization to a reciprocal heading. Fig 23 illustrates the system. The ratchet wheel is engaged by a solenoid operated pawl. The solenoids are energized through contacts carried by the annunciator, and the direction of rotation is governed by the direction of misalignment. If the SYN knob is turned in the reverse direction, a slipping clutch prevents possible damage. When the magnetic monitoring is interrupted the annunciator will be in its central position and the solenoid circuits will be broken. The synchronizing shaft can then be turned in either direction. It should be noted that if the SYN knob is turned sharply, it is sometimes possible to turn the card in the wrong direction through about 10° before the ratchet engages. It is thus possible if the compass is misaligned by nearly 180° to synchronize to a reciprocal heading. To avoid this the knob should be turned smoothly.

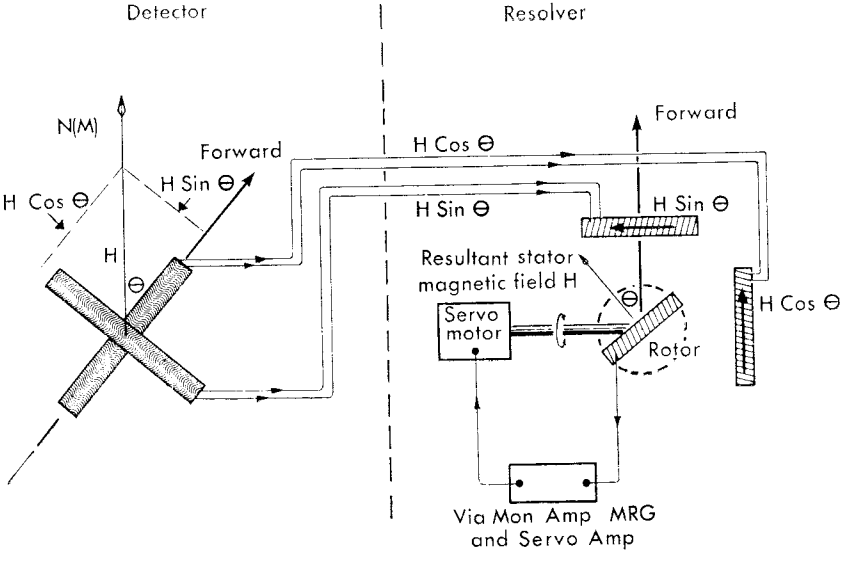
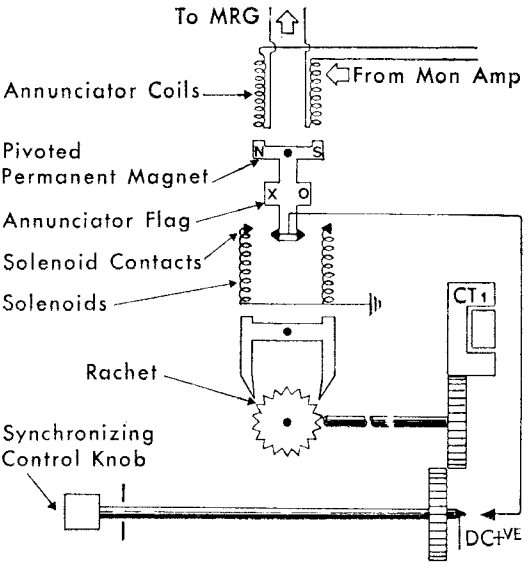
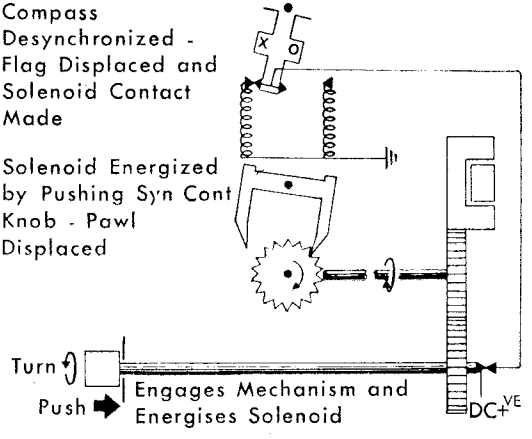


Fig 22 Interconnections of Detector and Synchro Resolver



a Compass Synchronizer



Pawl Engages in Sprocket
Allowing Rotation of Entire
Linkage in One Direction Only

b Action of Synchronizer

Fig 23 Manual Synchronization Scheme

CHAPTER 3

MILITARY FLIGHT SYSTEM, MK. 1 SERIES

CONTENTS

	Paras.
Introduction	1– 4
General Description	5– 7
PRINCIPLES	
Twin Compass System	8–15
Track Control Unit	16–19
Beam Compasses	20–22
Navigator's Repeater	23
Turn Switch...	24
Compass Comparator Operation	25–28
EQUIPMENT	
Power Requirements	29
Components	30
Detector Units	31–34
Twin Azimuth Gyroscopes	35–38
Twin Compass Amplifiers	39–44
Annunciator Units	45–48
Beam Compasses	49–57
M.F.S. Selector	58–61
Navigator's Repeater	62–63
Track Control Unit	64–73
M.F.S. Mk. 1A	74–77
M.F.S. Mk. 1C	78–79
OPERATION	
Navigational Techniques	80–83
Operating Instructions	84–85
Adjustment and Calibration	86–93

Introduction

1. The Military Flight System combines into an integrated system many of the functions previously left to individual and, for the most part, unconnected instruments such as the zero reader, artificial horizon, and gyro-magnetic compass. The M.F.S. offers far better correlation between the types of information offered to the pilot, and displays this information on only two instruments, the beam compass (Fig. 1) and the director horizon (Fig. 2). These instruments show:—

(*a*) Aircraft attitude, and heading or track.

(*b*) Deviation from an I.L.S. localizer and glide path beams.

(*c*) The change in attitude and heading required to follow a set flight path or pattern, or an I.L.S. approach and let-down. This function is known as flight direction.

2. In conjunction with the Mk. 10A autopilot the M.F.S. also provides for the control of automatic flight.

Fig. 1. Beam Compass

3. The M.F.S. includes, as part of the integrated whole, two gyro-magnetic compass systems and a navigation instrument called the track control unit (T.C.U.). Only these parts of the system are described in this chapter: for a description of the complete system reference should be made to A.P. 4686B, Vol. 1, Book 1.

4. The Mk. 1 series of M.F.S. consists of three systems designated Mk. 1A, Mk. 1B and Mk. 1C flight systems. Initially the Mk. 1A flight system was installed in aircraft fitted with M.F.S., but as more components became available this was converted to the Mk. 1B flight system in bomber aircraft and the Mk. 1C flight system in transport aircraft. This chapter is based upon the Mk. 1B flight system, but the differences in the Mk. 1A and Mk. 1C systems are given in paragraphs 74-79.

General Description

5. The twin compass system of the M.F.S. consists of two separate and identical gyro-magnetic compass systems operated from separate power supplies. One compass system supplies magnetic heading to the port beam compass and the navigator's repeater and is known as the port compass system. The other supplies magnetic heading to the starboard beam compass and is known as the starboard compass system. Either the port or starboard system can be selected to provide magnetic heading to the T.C.U.

6. The track control unit is a navigation instrument which receives and combines magnetic heading, magnetic variation, and Doppler or manually set drift. The instrument normally provides both true heading outputs and true track outputs. Fig. 3 shows, in block schematic form, the relationship between the units described in this chapter.

7. An a.c. synchro control transmission system is used throughout the twin compass system of the M.F.S. This type of data transmission is described in Part 3, Section 1, Chapter 1 of this Volume.

PRINCIPLES

Twin Compass System

8. Each compass system employs a magnetically monitored azimuth gyroscope as its reference. The two azimuth gyroscopes are mounted in sealed units remote from the instrument panel and are monitored separately by two detector units which sense the magnetic heading of the aircraft.

Fig. 2. Director Horizon

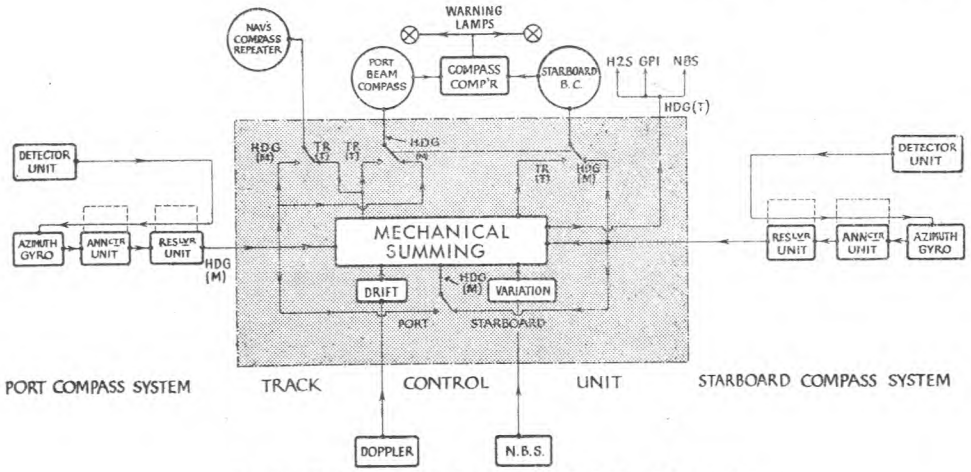


Fig. 3. Twin Compass System and T.C.U.—Schematic

9. The basic elements of a single compass system are illustrated in Fig. 4. The output of the detector unit is two a.c. signals, one proportional to the sine and the other to the cosine of the magnetic heading. These signals are fed to the stators of a synchro resolver, the rotor of which is coupled to the resolver shaft in the resolver unit, part of the compass amplifier. If the rotor is in the null position of the field created by the stators then the resolver shaft is aligned with the magnetic heading sensed by the detector unit. But if the rotor is not in the null position an alternating voltage is induced in it.

10. The output of the rotor of the synchro resolver is passed to the monitoring amplifier, where it is amplified and demodulated by a phase sensitive detector. The resultant d.c. output is passed through the annunciator unit to the precession coils of the azimuth gyroscope, causing the gyroscope to precess in a direction determined by the sense of the current.

11. The resolver shaft is servo-controlled, by means of an a.c. servo as indicated in paragraph 7 above, to follow the movements in azimuth of the gyro; for this, a synchro control transmitter (CX) on the gyro gimbals picks off displacements in azimuth and feeds the signals *via* a differential transmitter (CDX) in the annunciator to the control transformer (CT) in the resolver unit.

12. Thus it follows that the resolver shaft always indicates directly what the gyro is doing, and therefore the aircraft heading relative to the gyro, while the d.c. gyro precessing current is a measure of the error between this heading and the magnetic heading determined by the detector unit. The direction of precession of the gyro resulting from this current is such as eventually will make the "gyro heading" and magnetic heading the same. Once this synchronism is achieved, any slow wander of the gyro is countered by the long-term accuracy of the magnetic heading information, maintaining the gyro axis at a constant angle to the magnetic meridian. Conversely, short-term inaccuracies of the magnetic reference, which can only effect a maximum gyro precession rate of 3 degrees/minute, are countered by the gyro rigidity.

13. The compass system is synchronized when the rotor of the synchro resolver on the resolver shaft is in the null position of the stator field determined by the detector unit. When there is a large difference between the magnetic heading and the angular position of the resolver shaft, the normal rate of magnetic monitoring turns the resolver shaft too slowly for synchronization within a convenient time. The inclusion of the manually operated differential CDX allows the resolver shaft to be turned independently of the azimuth gyroscope. Thus if the resolver shaft is 30° from synchronism and the differential synchro is turned through 30°, the resolver shaft is driven into alignment with the magnetic heading. The adjustment is facilitated by an annunciator flag which is operated by a meter movement connected in the monitoring current circuit. The flag indicates the presence or absence of monitoring current and the direction in which the synchronizing knob should be turned to achieve synchronism.

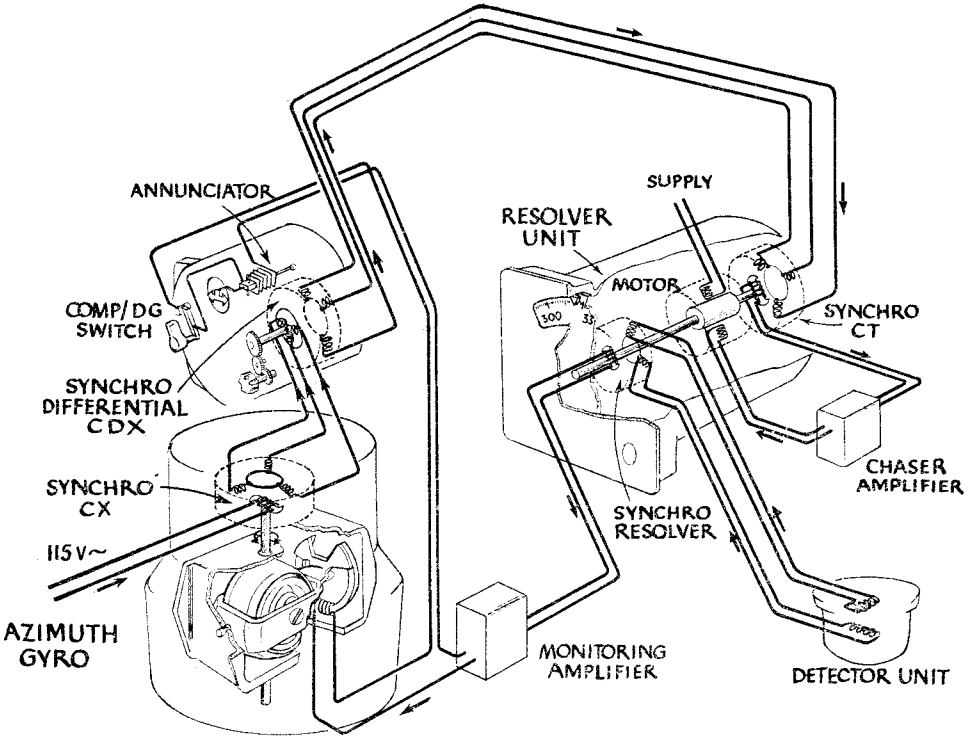


Fig. 4. Basic Elements of a Single Compass System

14. A selector switch is also connected in the monitoring current circuit, and is used to disconnect the monitoring current to the gyroscope, thus enabling the system to be operated as a directional gyroscope.

15. The two resolver shafts, one in each compass system, are the sources of magnetic heading for all other instruments. Five additional synchros, which are not shown in Fig. 4, are mechanically coupled to each shaft. These supply magnetic heading information to the track control unit, from which it is fed to other instruments.

Track Control Unit

16. The track control unit is illustrated schematically in Fig. 5. It contains three shafts: the drift shaft, the variation shaft, and the drift + variation shaft.

17. **Drift Shaft.** The drift shaft has a rotational freedom of $\pm 20°$ and is controlled either manually or by a synchronous transmission of drift from the Doppler equipment. Either method rotates the shaft through an angle equal to the drift angle.

18. **Variation Shaft.** The variation shaft has complete rotational freedom and is controlled either manually or by an M-type transmission of variation from the automatic variation setting corrector in the N.B.S. The M-type transmission requires reduction gearing and is not self-synchronous, making it therefore necessary to set the known variation manually before engaging the automatic drive. Either method of control rotates the shaft through an angle equal to variation. The rotors of three synchro differential transmitters (CDX) are coupled to the shaft and aligned so that when the shaft is at the position representing zero variation, the rotor coils are in line with the stator coils. When the shaft is rotated through an angle equal to variation, the rotor coils are turned relative to the stators through an equal angle. Since the stator of each synchro receives a synchronous transmission of magnetic heading from the resolver shaft, the output from each rotor is a synchronous transmission of true heading. The three outputs of true heading are fed to the G.P.I., H2S and N.B.C., which are not part of the M.F.S.

19. **Drift + Variation Shaft.** The drift shaft and the variation shaft are coupled to a differential

gear box which drives the drift + variation shaft. The shaft is coupled to the rotors of four synchro differential transmitters, which are aligned so that when the drift shaft and the variation shaft are at their zero positions the rotor coils are in line with the stator coils. Subsequent rotation of the drift shaft and the variation shaft rotates the drift + variation shaft through an angle equal to the algebraic sum of variation and drift. The rotor of each synchro is therefore turned relative to the stator by an equal angle. Since the stator of each synchro receives a synchronous transmission of magnetic heading from the resolver shaft, the output from each rotor is a synchronous transmission of true track. One of the differential synchros is spare; the other three can be included in the synchronous transmission of heading to the following instruments:—

(*a*) The two beam compasses. The method of inclusion is described in paragraph 22.

(*b*) The navigator's repeater. The method of inclusion is described in paragraph 23.

Beam Compasses

20. The port beam compass derives magnetic heading information from the port resolver shaft, and the starboard beam compass derives magnetic heading information from the starboard resolver shaft.

21. The heading pointer drive shaft is driven by an a.c. servo system, as indicated in paragraph 7, to follow-up the position of the resolver shaft.

22. When either REMOTE or BOMB is selected at the M.F.S. selector (described in paragraphs 58 to 61) a synchro differential transmitter (CDX) mounted on the drift + variation shaft of the track control unit is included in the link between the synchro CX and the synchro CT of the heading pointer servo. The inclusion of the synchro CDX causes the heading pointer to be offset from the angular position of the resolver shaft by the angular setting of the drift + variation shaft. Since this shaft is normally set at the sum of variation and drift the resulting indication of the heading pointer will be true track. Other indications such as magnetic track and true heading can be obtained by manipulation of the track control unit, but it should be noted that unless REMOTE or BOMB has been selected the heading pointers of both beam compasses will always indicate magnetic heading.

Navigator's Repeater

23. The navigator's repeater is controlled by a synchronous transmission of magnetic heading from the port resolver shaft. The principle of operation is illustrated in Fig. 5. The output of the synchro control transmitter (CX) mounted on the port resolver shaft is passed, *via* a synchro differential (CDX) mounted on the drift + variation shaft, to the stator of a synchro control transformer (CT) mounted on the pointer drive shaft of the navigator's repeater. The output of the rotor of the synchro CT is amplified and applied to the control winding of a two-phase motor. The motor drives the shaft and rotor to the null position of the stator field. Since the stator field defines the angular position of the port resolver shaft offset by the angular setting of the drift + variation shaft, the normal indication of the repeater is true track. The synchro CDX can, however, be by-passed by depressing a spring-loaded switch on the face of the track control unit. The repeater will then indicate the magnetic heading of the port compass system.

Turn Switch

24. A further function of the resolver shafts is to control the turn switch, which is operated when the rate of turn exceeds a fixed value—normally 30° per minute. The turn switch is used to:—

(*a*) Remove magnetic monitoring of the compass during turns, when the detector unit is not horizontal and is consequently detecting a false magnetic heading.

(*b*) Inhibit the compass comparator operation (*see* paragraph 28).

(*c*) Control the levelling of the gyroscopes that supply attitude information to the director horizons.

Compass Comparator Operation

25. The two compass systems should normally indicate the same heading; if they differ by 5° or more a fault is present in either one or both compass systems. In the M.F.S. a failure warning lamp is provided on each beam compass, and if the heading (or track) indications differ by more than five degrees then both lamps will light.

26. The compass comparator arrangement consists of two lamps and two synchro comparators (one in each beam compass), relay switches, and an amplifier, connected together as shown schematically in Fig. 6.

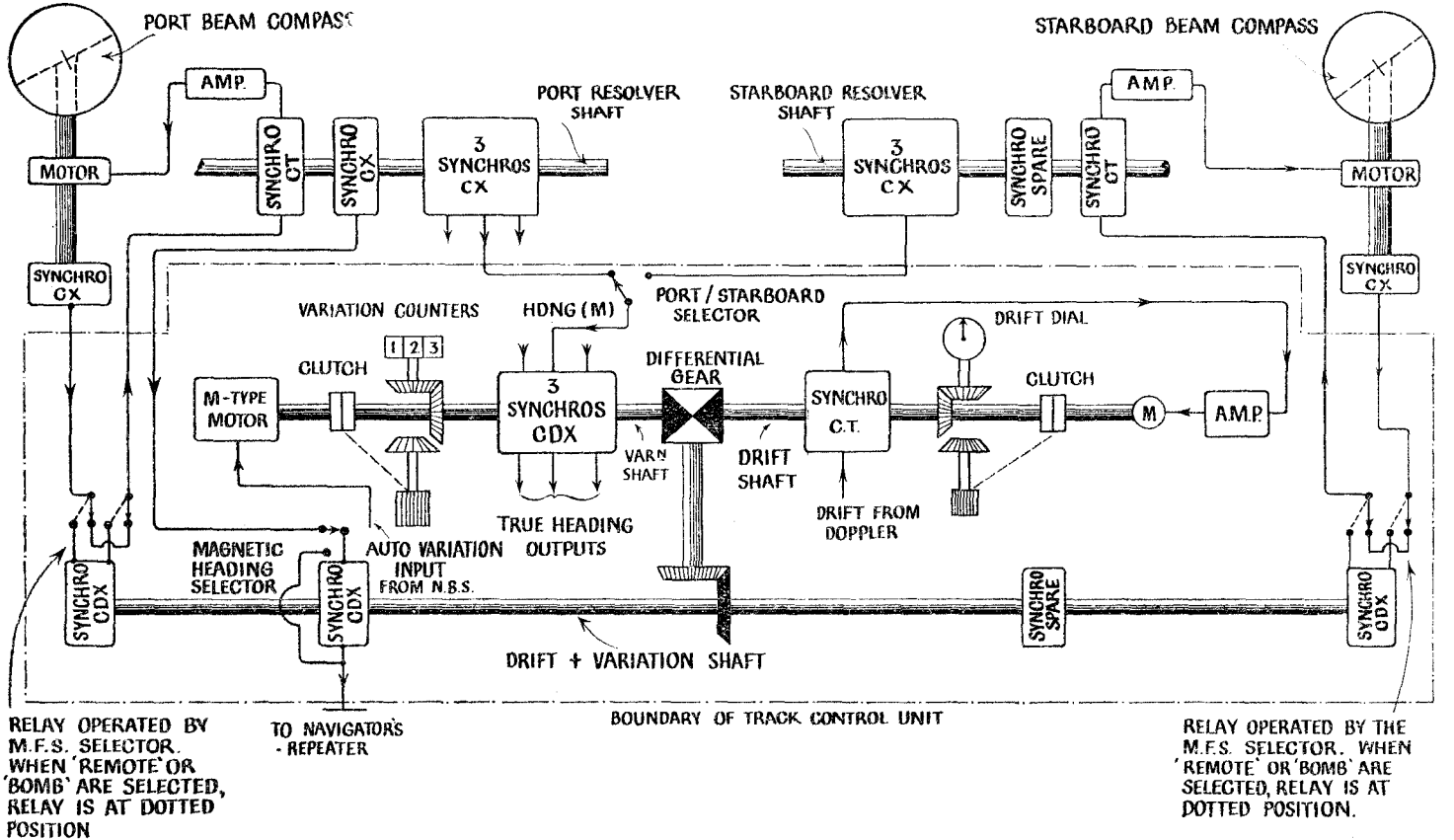


Fig. 5. Track Control Unit—Schematic

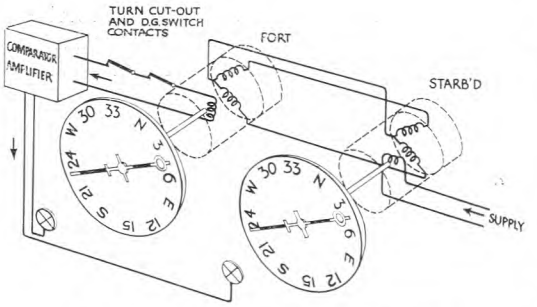


Fig. 6. Compass Comparator Circuit

27. In each beam compass the rotor of a synchro resolver is geared to the heading pointer shaft. The port and starboard beam compass resolver stators are interconnected and alternate rotors are utilized in the two beam compasses so that, if an a.c. supply is connected to the rotor of the starboard comparator synchro and the two heading pointers indicate the same reading, the rotor of the port comparator synchro will be at right-angles to the resultant field, and no voltage is induced in it. If the indications differ, the two rotors are no longer at right-angles and a misalignment signal is induced in the port comparator rotor. This signal is amplified and used to energize a relay which switches on the supplies to the warning lamps. The two compass systems, and their respective warning lamps, are supplied with power separately. If the power supply to one system fails, the warning lamp of the other system will light.

28. Both the turn switch and the COMP/DG switch operate relays to disconnect the signal inputs to the comparator amplifier. This means that the warning lamps will not light during sustained turning flight, or when either, or both, compass systems are selected to DG. In addition, a delay switch operating with a delay of thirty seconds ensures that the lamps will not light during misalignments of short duration.

Power Requirements

29. The M.F.S. operates on 115V, 400 c/s, three-phase a.c. and 28V d.c. supplies. The power supplies to the port and starboard halves of the installation are completely separate.

Components

30. The following components of the M.F.S. are either integral parts of the twin compass system or are concerned with the definition of the direction in which the aircraft is flying:—

(*a*) Detector Units.
(*b*) Azimuth Gyroscopes.
(*c*) Compass Amplifiers.
(*d*) Annunciator Units.
(*e*) Beam Compasses.
(*f*) M.F.S. Selector.
(*g*) Navigator's Repeater.
(*h*) Track Control Unit.

Detector Units

31. Two identical detector units, one for each half of the twin compass system, are fitted in each aircraft equipped with the M.F.S. They are usually mounted one in each wing tip or in some other position remote from magnetic interference. A single detector unit is shown in Fig. 7. The three holes by which the unit is secured to the aircraft are slotted so that the unit may be turned to remove coefficient A. The mounting flange is engraved with a datum mark and scale, and the words FORE and AFT to ensure that the unit is correctly orientated and aligned with the fore-and-aft axis of the aircraft. Each division on the scale represents 2°.

32. Each detector unit is completely sealed and is constructed in two parts, upper and lower. The upper part of the unit contains two electromagnetic corrector coils, one parallel and the other perpendicular to the fore-and-aft axis of the aircraft. These coils are fed with d.c. which is adjusted at the resolver unit to compensate for coefficients B and C.

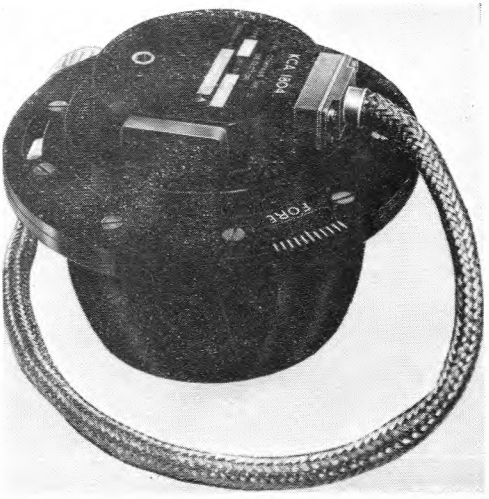


Fig. 7. Detector Unit

33. The lower part of the detector unit contains the detector assembly. The assembly consists of two pairs of magnetic flux detectors forming a cruciform pattern, with one pair parallel to the fore-and-aft axis of the aircraft, and the other pair parallel to the athwartships axis. The assembly is suspended in gimbals so that it is free to swing up to 22° from the horizontal, but cannot turn in azimuth. The bowl is filled with oil to damp out oscillations of the assembly. The construction of a single detector element is shown in Fig. 8.

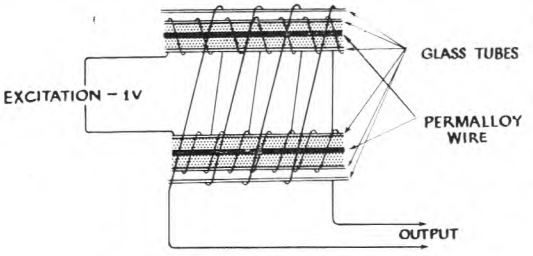


Fig. 8. Single Detector Element

34. The output of each detector unit is a voltage proportional to the cosine of the magnetic heading of the aircraft (fore-and-aft elements) and a voltage proportional to the sine of the magnetic heading (athwartships elements). The port detector unit output is transmitted to the synchro resolver on the resolver shaft of the port resolver unit, and the output of the starboard detector is similarly transmitted to the starboard resolver unit.

Twin Azimuth Gyroscopes

35. The two azimuth gyroscopes are each enclosed in hermetically sealed units which contain an inert atmosphere of helium and nitrogen. The two units are identical and interchangeable and are supported in a common anti-vibration mounting. The complete unit is shown in Fig. 9.

36. Each gyroscope is levelled by a torque motor controlled by mercury levelling switches. Another torque motor consisting of a horn magnet and a d.c. precession coil is used to precess the gyroscope so that it maintains a constant angle with the magnetic meridian.

37. Each gyroscope is balanced to compensate for apparent wander (due to the rotation of the earth) at a latitude of 51°N. When unmonitored by the magnetic element the random rate may be up to 12°/hr. In addition each gyroscope is subject to gimballing errors (*see* Appendix to Chapter 1 of this Section).

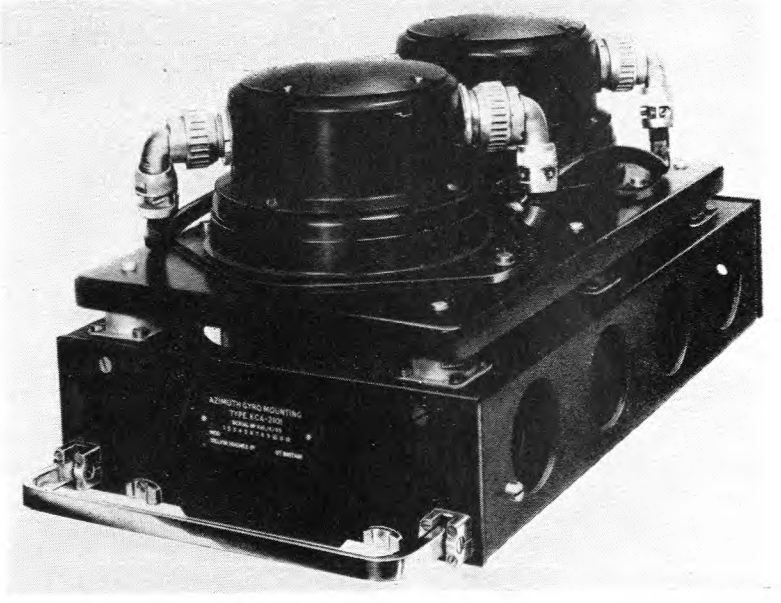


Fig. 9. Twin Azimuth Gyro Unit

38. A synchro transmitter attached to the outer gimbal ring transmits movement of the gyro axis in azimuth to a servomechanism which rotates the resolver unit shaft. The mechanism is illustrated in Fig. 4.

Twin Compass Amplifiers

39. The two compass amplifier units, one for the port compass system and one for the starboard compass system, are fitted in a common mounting. A single amplifier unit is illustrated in Fig. 10. Each amplifier unit contains a resolver unit sub-assembly which compares the position of the azimuth gyroscope with the direction of the Earth's magnetic field as determined by the detector unit. The method of comparison is described in paragraphs 9 and 10.

40. The resolver shaft, controlled by a synchro control transformer, controls four synchro control transmitters to repeat compass information to the beam compass and other equipment. It also drives a graduated compass card, a sector of which is visible through the window at the front of the unit. Owing to installation difficulties, the compass amplifier is not usually fitted so that this card will be visible to the navigator during flight; a compass repeater is provided instead. However, the resolver dial must always be used for maximum accuracy during calibration and adjustment of the compass system.

41. The compass system is calibrated by three correctors, for deviation coefficients A, B, and C, which are located on the front panel of the amplifier unit, and protected by a perspex panel. The A corrector mechanically rotates the synchro resolver stator in the resolver unit, to compensate for any constant angular difference between the magnetic heading of the aircraft and that indicated by the compass. The B and C correctors are potentiometers which remotely control the current flowing in the corrector coils of the detector unit. Since deviation can be either positive or negative, a polarity switch is associated with each corrector potentiometer. The process of calibrating the compass system is described in detail in paragraphs 86 to 93.

Fig. 10. Compass Amplifier Unit

42. In addition to the resolver unit with its associated servomechanism and the compass correctors, each compass amplifier contains the following components:—

(*a*) A beam compass repeater amplifier.

(*b*) A turn switch, the function of which is described in paragraph 24.

(*c*) A gyro monitor amplifier which amplifies and rectifies the output of the rotor of the synchro resolver on the resolver shaft.

(*d*) A compass comparator amplifier. Only the comparator amplifier of the port amplifier unit is included in the compass comparator circuit but each amplifier unit contains a comparator amplifier so that the units may be completely inter-changeable. The function of the comparator amplifier is described in paragraphs 25-28.

43. Below the perspex panel on the front of the amplifier unit are three jack plug sockets, a 2.5 amp fuse, and a spare fuse. When a centre-zero milliammeter is plugged into the left-hand socket it will indicate the value of the current flowing in the B corrector coils; when plugged into the centre socket, the value of the gyro monitor current; and when plugged into the right-hand socket, the value of the current flowing in the C corrector coils. The 2.5 amp fuse protects the supply to the warning lamp and DG flag.

44. Above the C corrector is the turn switch reset indicator. The indicator should rotate when the aircraft turns through more than about 8°.

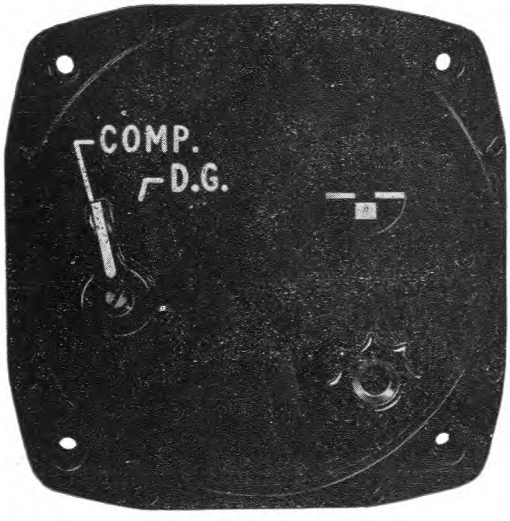


Fig. 11. Annunciator Unit

Annunciator Units

45. The two annunciator units, one for each compass system, are fitted in the pilot's instrument panels. A single unit is illustrated in Fig. 11.

46. Each unit contains a differential synchro, a meter movement, and a selector switch. The function of these components is described in paragraphs 13 and 14. The compass is synchronized by depressing and turning the synchronizing knob in the direction indicated by the arrow on the annunciator flag. When the compass is synchronized an equal portion of each arrow is visible.

47. If the indications of the annunciator flag are ignored, it is possible to set a compass system on the reciprocal of the magnetic heading so that the annunciator flag will show an absence of monitoring current and the apparent synchronism of the resolver shaft with the magnetic heading. It is, however, unlikely that this misalignment would escape notice, and it is in any case an unstable condition, *i.e.* the gyro would be gradually monitored away from this position and back to the true null.

48. The system can be operated as a gyro-stabilized magnetic compass or as a directional gyro, and the selector switch has two positions, engraved COMP and DG, to enable either function to be selected. Setting this switch to DG simply disconnects the monitoring current to the azimuth gyro and also disconnects the compass warning lamps. At the same time a flag inscribed DG will appear in the dial of the associated beam compass, but the heading indicated by the beam compass can still be reset by operating the synchronizing knob.

Beam Compasses

49. The two beam compasses, one operated by the port compass system and the other by the starboard compass system, are mounted in the port and starboard pilot's instrument panels. Each beam compass is essentially a remote indicator of its respective compass system, but it also provides a number of additional facilities concerned with the operation of the M.F.S. as a whole.

50. Aircraft heading (or track) is indicated by a heading pointer in the form of a miniature aircraft and ringsight moving over a compass card which normally remains fixed. The compass card can, however, be set to bring any desired scale marking to the top of the dial. Differential gearing

is employed to ensure that if the compass is reset, the heading pointer is moved at the same time, and will continue to indicate the aircraft heading.

51. A single beam compass is illustrated in Fig. 1. The compass card is graduated every five degrees on the outer scale, and the whole card can be rotated by depressing and turning the setting knob on the lower right-hand face of the instrument. Moving over the compass card are the heading pointer, and the heading index which can be set to any required position by pulling out and turning the setting knob. The angular difference between the heading pointer and the heading index is used to produce a heading error signal which is displayed on the azimuth director pointer of the director horizon. In this way the pilot is shown the correct bank angle at which to fly the aircraft to turn accurately on to the selected heading. Alternatively if the automatic pilot has been selected, the aircraft will turn automatically until the heading pointer and the heading index are coincident. The heading index is thus the heading selector both for the flight director and for the automatic pilot.

52. The beam compass also displays deviation from the localizer beam of an I.L.S. installation by means of the beam bar which is the equivalent of the vertical pointer of the normal I.L.S. meter. If the compass scale of the beam compass is correctly orientated, *i.e.* if the localizer QDM is set to the top compass datum, then the beam bar and the heading pointer together will indicate pictorially the displacement and heading of the aircraft in relation to the beam. This is shown in Fig. 12.

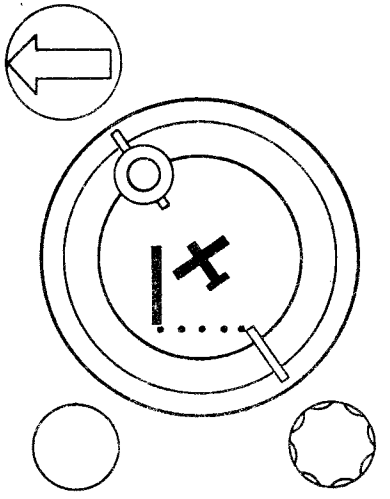


Fig. 12. Beam Displacement Indication

53. Radio beam displacement is also mixed with heading error signals (obtained from the angular difference between the heading pointer and the heading index) to provide a steering signal to intercept and align the aircraft with a selected I.L.S. localizer beam. This system can be most easily understood by considering an aircraft on a heading of 090°M approaching an I.L.S. localizer beam with a QDM of magnetic north. If the compass scale is turned so that north is at the top of the scale, the beam compass display will show the aircraft approaching the beam at right-angles. If the heading index is now turned to north then a heading error signal will indicate that the aircraft should turn to port. When the aircraft does turn 90° to port the beam displacement signal indicates that the aircraft should turn to starboard in order to intercept the beam, the heading error signal and the beam displacement signal are therefore in opposition. In the M.F.S. the two signals are mixed so that the aircraft is directed to intercept the beam at an angle of about 45° to the beam centre-line. When the aircraft enters the beam the beam displacement signal begins to decrease, and the aircraft is directed to turn to reduce the heading error by an equivalent amount. Consequently the beam displacement signal and the heading error signal progressively decrease together, and both become zero when the aircraft is correctly established on the beam centre-line.

54. Radio coupling, in the manner described in the previous paragraph, can only be obtained when the heading index is set within 30° of the top or bottom compass datum. This angular range is indicated by markings on either side of each datum. If the heading index is set outside this range then the radio beam displacement signals are disconnected and only heading error signals are available. In addition, when the heading index is set within the top range the beam displacement signals are in the correct sense for flying on the beam QDM, but when the heading index is set within the bottom range the sense of the beam displacement signals is reversed and is then correct for flying on the beam QDR.

55. **Sense Switch.** Sometimes radio coupling is required on beam QDM only, and at other times on beam QDR only. The required coupling is selected by the sense switch located at the top left-hand corner of the beam compass. The switch is marked with an arrow, and has three positions—with the arrow pointing across, up,

and down the case. These three positions respectively permit radio coupling in both ranges, in the top range only, and in the bottom range only. The switch is normally set with its arrow pointing across the case to permit coupling on both ranges and is turned to one of the other positions when coupling in only one direction is required.

56. **Compass Warning Lamp.** The compass comparator circuit has been described in paragraphs 25-28. If the heading indications of the two compass systems differ by more than 5°, the warning lamps on both beam compasses will light after a delay of about 30 seconds.

57. **DG Flag.** A flag marked DG will appear on the dial of the beam compass when DG has been selected at the appropriate annunciator unit. The appearance of this flag indicates that magnetic monitoring of the gyroscope has been discontinued and that the beam compass is operating as the indicator of a directional gyroscope.

M.F.S. Selector

58. The M.F.S. selector (Fig. 13) is the primary control panel of the complete system and is fitted in the instrument panel to be accessible to both pilots. Mounted on the M.F.S. selector are three selector switches. The upper switch is the navigational selector switch, the lower switch is the pitch selector switch, and the switch on the left-hand side of the instrument is the compass selector switch.

59. **Navigational Selector Switch.** The navigational selector switch selects the steering signals required for different modes of operation of the M.F.S. It has five positions: the central position; clockwise from the central position, LOC and LOC & G.P; and, anticlockwise from the central position, REMOTE and BOMB.

(*a*) *Central Position.* When the switch is in its central position, both beam compasses indicate the magnetic heading of their respective compass systems.

(*b*) *LOC, LOC & G.P.* When either LOC or LOC & G.P. is selected, I.L.S. localizer deviation signals are switched into the M.F.S. and are displayed on the beam bars of both beam compasses. The heading pointers will continue to indicate magnetic heading and, as described in paragraph 53, a steering signal is displayed on the director horizon which directs the pilot to align the aircraft with the localizer beam.

(*c*) *REMOTE.* When REMOTE is selected the angular setting of the drift + variation shaft of the track control unit is included in the synchronous transmission to both beam compasses. These will then indicate the magnetic heading of their respective compass systems offset by the setting of the drift + variation shaft. This indication will normally be true track.

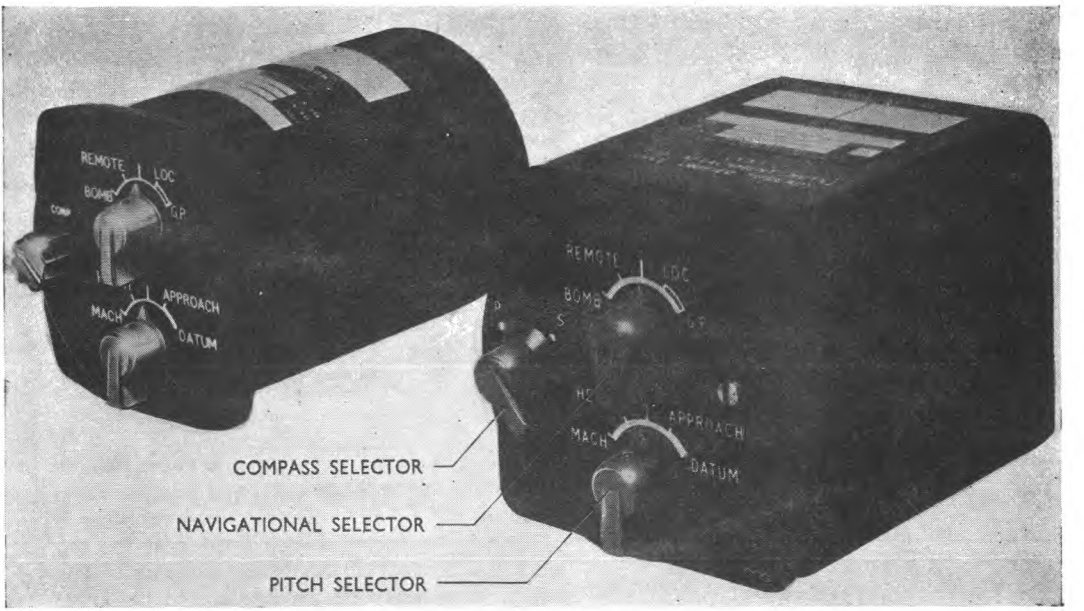


Fig. 13. M.F.S. Selectors, Types A (left) and B (right)

(*d*) *BOMB.* When BOMB is selected steering signals derived directly from the N.B.S. are displayed on both director horizons. The beam compasses will have the same indication as when REMOTE is selected.

60. **Pitch Selector Switch.** The pitch selector switch does not affect the functioning of the twin compass system and track control unit in any way, and is consequently not described in this chapter. For a description of its function reference should be made to A.P. 4686B, Vol. 1, Book 1.

61. **Compass Selector Switch.** The compass selector switch selects either compass as a datum to provide the heading reference for flight direction and the heading monitor of the automatic pilot. The selector knob is marked with an arrow to indicate whether the port or starboard compass system has been selected.

Navigator's Repeater

62. The navigator's repeater is illustrated in Fig. 14. Aircraft track, or heading, is indicated by a pointer moving over a fixed compass card which is graduated at 2° intervals. A knob situated at the lower right-hand corner of the instrument controls the marker which can be set against any desired track.

Fig. 14. Navigator's Repeater

63. The principle of operation of the repeater is given in paragraph 23 where it is explained that the normal indication of the repeater is the magnetic heading of the port compass system offset by the angular setting of the drift + variation shaft of the track control unit. This indication is thus true track. The track control unit can however be by-passed by depressing a switch on the face of the track control unit. The repeater will then indicate the magnetic heading of the port compass system.

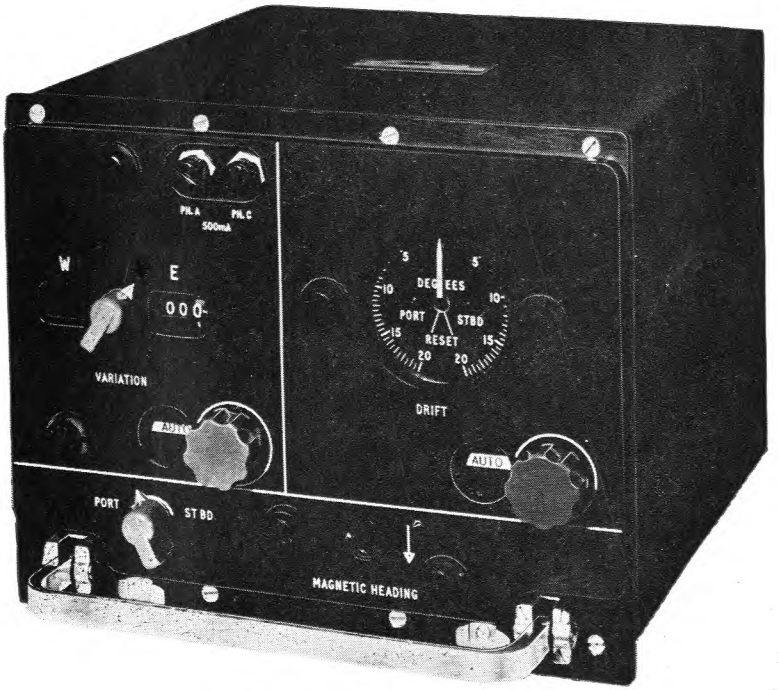


Fig. 15. Track Control Unit, Type A

Track Control Unit

64. The principles of operation of the track control unit, illustrated in Fig. 15, are explained in paragraphs 16 to 19. The T.C.U. receives magnetic heading from either compass system, drift information from the Doppler equipment, and magnetic variation information from an automatic source such as the N.B.S. It provides the following outputs:—

(*a*) Three synchronous outputs of true heading from the variation shaft. These are transmitted to equipments such as the G.P.I. which are not parts of the M.F.S.

(*b*) Four synchronous outputs of true track from the drift + variation shaft. These are distributed as follows:—

(i) True track to the port beam compass when REMOTE or BOMB is selected at the M.F.S. selector.

(ii) True track to the starboard beam compass when REMOTE or BOMB is selected at the M.F.S. selector.

(iii) True track to the navigator's repeater.

(iv) One output is normally spare, except in certain special installations.

65. The setting of the drift shaft is indicated by a pointer moving over a dial marked DRIFT which is graduated at 0.5° intervals. The maximum indication of drift is 20° port or starboard. The drift shaft can be rotated manually by pulling out and turning the knob below the drift dial. When the knob is pulled out a flag marked MANUAL appears at the window next to the knob. When the knob is pushed in, the flag is removed, leaving the sign AUTO in the window, and the automatic drive is engaged. The drift shaft is then automatically aligned with the drift sensed by the Doppler equipment. If the power supply to the T.C.U. is interrupted for some time while a large change of drift occurs, it may be possible for the drift servo to become de-synchronized; the drift pointer will then move into the area marked RESET on the dial when power is restored. This indicates that the pointer should be manually re-synchronized before engaging AUTO.

66. It can be seen from the normal presentation of Doppler drift on an instrument such as Green Satin, that the drift is continually fluctuating. This would make it difficult for a pilot to steer a particular track if Doppler drift were used without any form of damping. In the T.C.U. drift fluctuations are damped out electrically to provide a dead-beat indication of drift. This means that a momentary comparison of the drift indicated on the T.C.U. with the drift indicated on the Doppler equipment may show a slight discrepancy.

67. The variation shaft has complete rotational freedom so that a 360° rotation in one direction represents 360° of westerly variation and a 360° rotation in the other direction represents 360° of easterly variation. The setting of the shaft is indicated on two identical sets of counters, one marked E(east), reading the reciprocal of the other marked W(west). Selection of either easterly or westerly variation is made by the two-position switch which separates the counters. When one set of counters has been selected, the other is covered so that it cannot be seen.

68. The variation shaft can be set manually by pulling out and turning the knob below the counters. When the knob is pulled out a flag marked MANUAL appears at the circular window next to the knob. When the knob is pushed in the flag is removed, leaving the sign AUTO in the window, and the automatic variation drive is engaged. This is a non-synchronous M-type transmission, so the variation shaft must be set manually at the known value of variation before engaging the automatic drive.

69. Below the variation setting knob is a two-position switch, the positions being marked PORT and STBD. This switch selects which compass system supplies magnetic heading to the true heading synchros of the T.C.U. It has no control over the supply of magnetic heading to either of the beam compasses or to the navigator's repeater.

70. On the lower right-hand side of the T.C.U. is a two-position switch marked MAGNETIC HEADING. The switch is spring-loaded in the up position. When the switch is depressed, the navigator's repeater will indicate the magnetic heading of the port compass system instead of the normal indication of true track.

71. The suppressed lighting of the T.C.U. is controlled by a rheostat which is normally mounted close by. Two 500mA fuses guard the a.c. supply. They are marked PH.A (phase A) and PH.C. (Phase C). Phase B is earthed.

72. **Track Control Unit, Type C.** The T.C.U. described above (Type A) will eventually be

replaced by the Type C which is basically similar but which has:—

(*a*) Manual input of deviation, up to 3°E or W, feeding *via* a differential on to the variation shaft: the amount of deviation set is indicated on a dial on the face of the instrument.

(b) No "spare" output of true track.

73. **Track Control Unit, Type B.** The T.C.U. Type B is the track control unit fitted in the Mk. 1C flight system. Designed for drift input from the computer of the AD 2300B Doppler equipment instead of the input from the G.P.I.4 of the Type A and C units, it differs from the Type A unit in that:—

(*a*) Variation input is manual only.

(*b*) The "spare" output of true track drives a compass repeater in the supply aimer's position.

M.F.S. Mk. 1A

74. The twin compass system of the M.F.S. Mk. 1A is basically the same as that of the M.F.S. Mk. 1B, but a track control unit is not fitted.

75. The track control unit is replaced by a variation setting control which contains a manually operated synchro differential coupled to two counters. One set of counters is inscribed from 0-180 degrees East and the other 0-180 degrees West. They are arranged so that when one set of counters is visible the other set is covered by a shutter.

76. The variation setting control receives magnetic heading from a synchro transmitter on the resolver shaft of the port compass system, adds on the variation set by means of the manual setting knob, and provides an output of true heading to those external systems which would normally receive this from the T.C.U. This output of true heading is fed *via* an azimuth repeater unit, which is a synchronous-to-M-type converter, for those systems which require this form of input. In some aircraft installations a changeover switch may be mounted adjacent to the navigator's compass repeater and variation setting control, to enable either port or starboard compass transmissions to be used for these instruments.

77. In Mk. 1A flight system installations in transport aircraft a compass repeater is provided in the supply aimer's position. Like the navigator's repeater it receives magnetic heading from the resolver shaft of the port compass system (port or starboard system if a changeover switch is fitted) and like the beam compasses and navigator's repeater in the Mk. 1A flight system it can only indicate magnetic heading.

M.F.S. Mk. 1C

78. The Mk. 1C flight system is distinguished from the Mk. 1B by the installation of a Type B track control unit in place of the Type A or C, and by the fitting of an additional compass repeater. This compass repeater is fitted in the supply aimer's position and its system is identical with that providing the navigator's compass indication except that no provision is made for isolating the variation + drift input by means of the magnetic heading switch (paragraph 70).

79. A Type C M.F.S. selector is included in the installation. This is similar to the Type B but the navigational selector switch has no BOMB position. When REMOTE is selected on this switch the across-track error signal from the AD 2300B computer is connected to the beam bars of the beam compasses thereby indicating displacement of the aircraft from the selected track.

OPERATION

Navigational Techniques

80. The M.F.S. makes it possible to adopt navigational techniques which are different from those which have previously been used in the R.A.F. While such techniques are decided in detail by the operational role of the aircraft, the following possibilities exist:—

(*a*) The steering of true track.

(*b*) The steering of grid track.

(*c*) The steering of gyro track.

81. **True Track Steering.** When REMOTE has been selected at the M.F.S. selector, and the automatic drives to the T.C.U. are engaged and functioning, the aircraft can be flown along a selected track. The navigator measures the required rhumb line track which is then set on the heading index of the beam compass. The aircraft can then be flown either manually or automatically to maintain this track. It should be noted that when this mode of operation is used the heading changes with the Doppler drift. The track is the constant factor, and the track flown will be accurate within the limits of accuracy of the Doppler and compass systems.

82. **Grid Track Steering.** The only difference between steering a true track and steering a

grid track is that the automatic variation feed is not used. Instead, the variation shaft of the T.C.U. is set manually to the value of grivation. Since the value of grivation is only altered at intervals, it is not correct to say that the aircraft can be steered to maintain a perfect grid track *i.e.* a great circle. The track steered will, however, approximate to a great circle.

83. **Gyro Track Steering.** In high-speed aircraft considerable acceleration errors are introduced into astro observations. One cause of these errors is the acceleration which occurs when an aircraft is flown on a curved path relative to the earth. It has been found in practice that this error can be reduced by flying on a constant gyro heading during the astro observation. This is because the heading given by a perfect gyro which is correctly balanced for earth rotation is a true great circle. With the M.F.S. a further improvement can be made by steering a track which is close to a great circle, *i.e.* a gyro track. This can be done without effectively altering the outputs of true heading which are required for other equipments, if the following settings are made:—

(*a*) *M.F.S. Selector.* Select REMOTE and the port compass system.

(*b*) *Port Annunciator Unit.* Select DG.

(*c*) *Autopilot.* Select the autopilot in. The aircraft is now being steered from the port compass system.

(*d*) *Track Control Unit.*

(i) *Variation.* Set the variation to MANUAL so that the gyro heading will not be altered by a change in variation. When the observations are completed the control should be reset to AUTO.

(ii) *Drift.* The drift control should be set to AUTO.

(iii) *Compass System.* Select the starboard compass system. Outputs of true heading to other equipments will then be derived from the starboard compass system which is still magnetically monitored.

Operating Instructions

84. **Before Flight.**

(*a*) Switch on the main system inverters and then set the port and starboard compass isolation switches to NORMAL (*Note.*—In some aircraft the compass systems may be switched on automatically with the main M.F.S. instrument inverters).

(*b*) Check that each compass system is selected to COMP. at the annunciator unit. Observe that the heading pointer of each beam compass settles to a steady reading, and that the annunciator flag settles to a steady position.

(*c*) After about twenty seconds synchronize each compass by pushing in and turning the synchronizing knob in the direction of the arrow shown on the annunciator flag. The compass is synchronized when an equal length of each arrow is visible.

(*d*) Set the navigational switch on the M.F.S. selector to the central (vertical) position. Check that the two compass readings are the same and that they agree with the reading of the standby compass.

(*e*) Make the following settings on the track control unit:—

(i) Select PORT compass system.

(ii) Select MANUAL drift and set the drift to zero.

(iii) Select MANUAL variation and set the variation to its present value.

(*f*) The navigator's repeater will now read true heading. Synchronize other equipment with the navigator's repeater.

(*g*) Set the drift pointer on the Doppler equipment to zero drift and the drift pointer on the T.C.U. to 10° port or starboard drift.

(*h*) Select AUTO drift on the T.C.U. The T.C.U. drift pointer should pull in quickly to 4° of drift, and then more slowly towards zero drift which should be indicated in about 30 seconds.

(*j*) Switch the variation to AUTO when the source of automatic variation has been set to the co-ordinates of the present position.

85. **In Flight.** The control of the twin compass system and T.C.U. is divided between the pilot and the navigator. This can lead to confusion if crew co-operation is not of a high order. The following examples illustrate how confusion may occur:—

(*a*) The navigator may pass a track to steer. If the pilot has not selected REMOTE then the beam compasses will indicate not track but magnetic heading.

(*b*) The navigator has no indication of compass synchronization or whether DG has been selected on either annunciator unit. It is thus possible for the pilot to select DG on the same compass system that the navigator has selected on the T.C.U. Outputs of true heading to other equipment will then become incorrect.

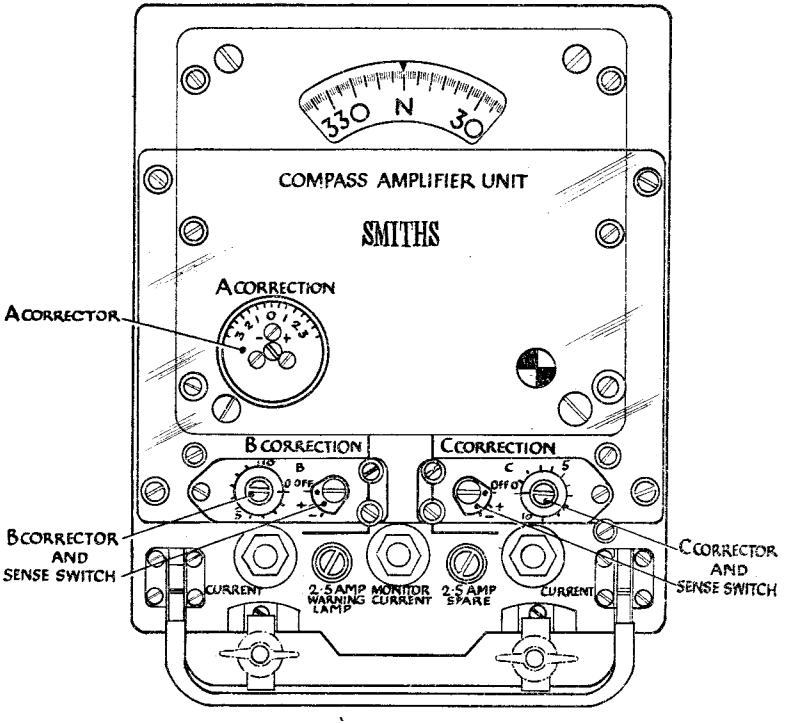


Fig. 16. Compass Correction Controls

(*c*) The navigator must be told if one compass is faulty. Otherwise it is possible to have the faulty compass system selected at the T.C.U. although the pilot's instruments have been selected to the serviceable system.

Adjustment and Calibration

86. Both compass systems of the M.F.S. are calibrated by the usual method of calculating and correcting coefficients A, B, and C.

87. The compass correction controls are grouped on the front panel of each amplifier unit, and are illustrated in Fig. 16. They are preset by screwdriver adjustment and protected by a clear perspex panel. The coefficient A corrector scale is graduated in degrees, but the B and C corrector scales are graduated arbitrarily in equal divisions. The scales are intended only for guidance when resetting the correctors. In each instance the actual correction applied must be determined by observing the resolver dial. This is graduated at one degree intervals and provides more accurate readings of magnetic heading than the beam compasses or the navigator's repeater.

88. The sense in which A correction is applied is indicated by plus and minus engraved on the corrector dial; anticlockwise rotation gives plus correction, and clockwise rotation gives minus correction. The B and C correction signs are determined by a switch located at the side of each corrector. Each switch has three positions; zero, positive, and negative corrector current.

89. The A corrector mechanically rotates the stator windings in the synchro resolver and allows correction of up to $\pm 3°$ coefficient A. If coefficient A is larger than three degrees itmust be removed at the detector unit. The normal range of the B and C correctors is $\pm 3°$, but, by reversing the engraved plates behind the sense and corrector switches, correction up to $\pm 6°$ may be obtained.

90. A two-point jack socket is associated with each corrector. A centre-zero milliammeter can be plugged into each socket in turn to determine the exact values of the corrector currents. When a swing has been completed the values of the corrector currents must be recorded. At some time it may be necessary to replace a compass

amplifier in circumstances where a proper compass swing is impossible. The replacement unit can then be adjusted to give the same values of corrector currents as the unit it replaces, so providing an appropriate compensation.

91. During a compass swing it is essential to ensure that magnetic monitoring has ceased before a compass reading is taken. The annunciator flags are not sufficiently precise for this purpose, and it is necessary to use a centre-zero milliammeter plugged into the socket marked MONITOR CURRENT. If the azimuth gyroscope has no drift, monitoring has ceased when the meter oscillations are evenly balanced about the zero position. Any drift however must be balanced by a small amount of monitoring current which will cause the meter to oscillate evenly about some datum other than zero. This datum should not exceed 1.2mA in the northern hemisphere and should never exceed 2mA.

92. **Power Supplies.** During a compass swing the a.c. supply voltage must be 115V $\pm$3V at a frequency of 400 c/s $\pm$ 8 c/s. The d.c. supply voltage must be between 26V and 28V.

93. **Swinging Procedure.** Both the compass systems are calibrated and corrected during the same swing. Separate values of coefficients A, B and C are calculated and applied to the respective compass systems. Two separate deviation curves are then drawn up. The following procedure describes the calibration and correction of one compass system only.

(*a*) When the power supply is connected to the aircraft, switch on the compass.

(*b*) Ensure that the a.c. and d.c. supply voltages are correct (*see* paragraph 92).

(*c*) In order to assess any changes in coefficients B and C it is advisable to set the corrector sense switches to zero.

(*d*) Carry out a correcting swing measuring the compass deviations on the cardinal headings. Calculate the values of the coefficients.

(*e*) If coefficient A is three degrees or less, remove it by rotating the A corrector on the resolver unit. The value applied must be checked on the resolver dial; the markings on the corrector are only a guide. If coefficient A is greater than three degrees it must be removed by rotating the detector unit.

(*f*) Coefficients B and C are removed in the following way:—

(i) Set the corrector sense switch to give correction of the same sign as the calculated coefficient.

(ii) Rotate the appropriate corrector fully clockwise to obtain maximum monitoring rate. Closely observe the rotating resolver dial. Immediately the dial attains the required heading, quickly turn back the corrector until the milliammeter once more indicates datum monitoring current. Verify that the correct reading has been achieved and adjust the corrector as necessary.

(*g*) Carry out a calibration swing on the cardinal and quadrantal headings and plot a deviation curve. For greater accuracy calibrate on twelve headings and calculate a deviation curve by Fourier analysis.

CHAPTER 4

MK 7 SERIES GYRO-MAGNETIC COMPASSES

CONTENTS

	Paras
Introduction	1—4
GYRO-MAGNETIC COMPASS MARK 7	
Introduction	5
Components	6
Detector Unit	7—8
Remote Correction Unit	9—11
Gyro Unit	12—16
Master Indicator	17—19
Pilot's Indicator	20—21
Controller	22
Mounting Rack	23—24
OPERATION OF THE SYSTEM	
Introduction	25
Basic Servomechanism	26—27
Magnetic-Slaving Servomechanism	28—29
Magnetically Slaved Mode	30
Variation Setting	31
Operation in a Turn	32—33
Pilot's Indicator Servomechanism	34
DG Mode of Operation	35—36
GYRO-MAGNETIC COMPASS MARK 7A	
Introduction	37
Components	38
Detector Unit	39
Remote Correction Unit	40
Directional Gyroscope	41—42
Master Indicator	43—45
Navigator's Controller	46
Pilot's Indicator	47
Pilot's Control Panel	48—49
Mounting Rack	50—51

OPERATION OF THE SYSTEM

Introduction 52
Follow-Up Servomechanism 53
Magnetic Slaving Servomechanism ... ... 54—55
Initial Synchronization ... ... 56—60
Operation in a Turn 61
Heading Outputs and Variation ... ... 62—66
Pilot's Heading Repeater 67

Introduction

1. The Gyro-Magnetic Compass Mark 7 Series (GM 7), in addition to providing direct visual indication of heading, can provide remote heading outputs to aircraft equipments such as autopilot, radio magnetic indicators and ADRIS. The gyro output is usually slaved to magnetic heading information, but can be used alone in a directional gyro (DG) mode.

2. A CL 11 rotorace gyro provides the azimuth reference for each variant of the series. The gyro can be precessed continuously to compensate for Earth-rate gyro drift.

3. Magnetic heading is derived from a pendulous fluxvalve. In the slaved mode the heading output of the gyro is continuously slaved to the fluxvalve heading. The maximum follow-up rate of the magnetic slaving system is approximately 1·5° per minute in normal operation.

4. No new principles are used in the GM 7 compass series. The theory of the fluxvalve is covered in Chap 1 and the servo mechanism principles are covered in Part 3, Sect 1. The reader is advised to revise his knowledge of these subjects if necessary, before continuing.

GRYO-MAGNETIC COMPASS MARK 7

Introduction

5. The Gyro-Magnetic Compass Mark 7 (GM 7) installation is a twin compass system in which each compass system operates independently of the other. The two compasses are identical and the description which follows is of one half of the twin system. The CL 11 gyro is roll stabilized to reduce gyro gimballing errors during aircraft turns.

Components

6. The basic components of a single compass are:

a. Detector Unit.
b. Remote Corrector Unit.
c. Gyro Unit.
d. Master Indicator.
e. Pilot's Indicator.
f. Controller.
g. Mounting Rack.

Detector Unit

7. The detector unit, shown in Fig 1, senses aircraft heading relative to the magnetic meridian. The theory is covered in Chapter 1.

8. The fluxvalve unit is a conventional three-spoke assembly, suspended in gimbals which permit ± 30° of freedom about the aircraft pitch and roll axes, but no movement in azimuth relative to the aircraft fore-and-aft axis. This assembly is mounted in a hemispherical bowl which is filled with oil to damp out oscillations, and which can be rotated at its mounting to compensate for compass deviation coefficient A.

Remote Correction Unit

9. The earth's magnetic field in the vicinity of the detector unit is likely to be distorted by permanent and induced aircraft magnetism. The magnetic heading sensed by the detector unit is therefore subject to compass deviation. A remote correction unit enables the navigator to introduce equal and opposite distortion to compensate for deviation coefficients B and C. The unit is shown, with the cover removed, at Fig 2.

10. The three detector coils of the fluxvalve are permanently connected to three variable potentiometers in the correction unit. Com-

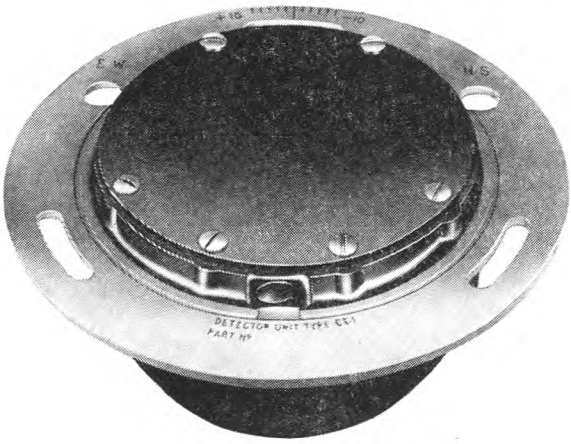


Fig 1 Detector Unit

pensation is made for coefficients B and C by applying small correcting DC voltages directly to the detector coils from the associated potentiometers.

11. The magnitude of the correcting voltages is altered whenever the navigator rotates the coefficient corrector dials on the face of the unit during compass swings. The graduations on the dials represent fractions of the total correction available, though the effect of any given amount of dial rotation will vary with magnetic latitude since the strength of the signals induced by the Earth's field in the coils of the detector unit is itself a function of magnetic latitude. For example, the total B or C voltage correction available represents a deviation correction of approximately $\pm 6°$ in a magnetic field of 0·18 oersted (UK), whereas the same voltage in a field of 0·40 oersted (Singapore) would represent a deviation correction of approximately $\pm 2{\cdot}7°$.

Gyro Unit

12. The directional gyro assembly is primarily

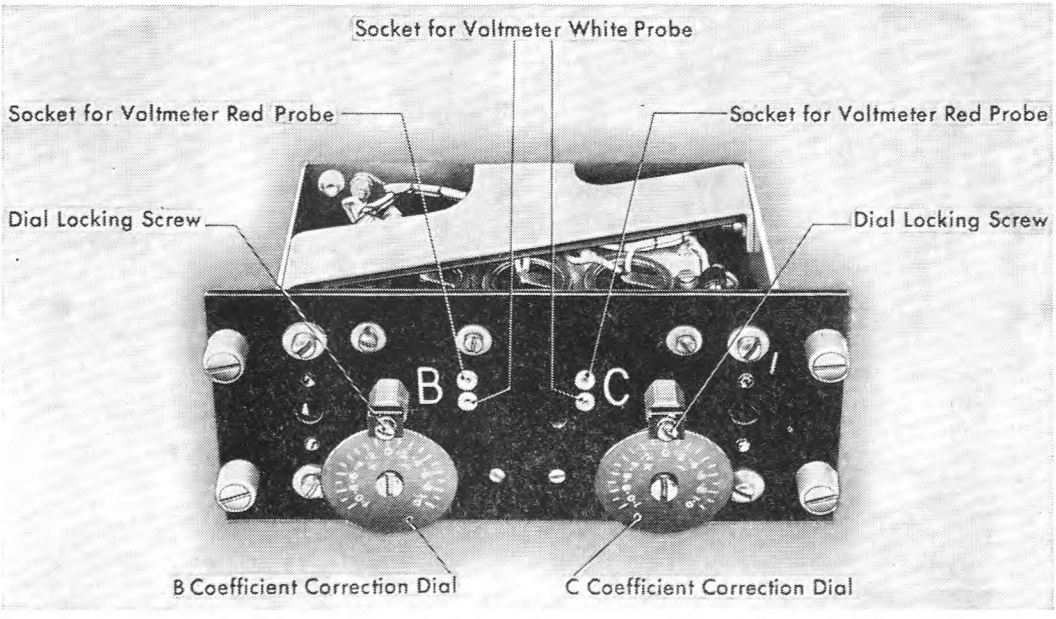


Fig 2 Remote Correction Unit

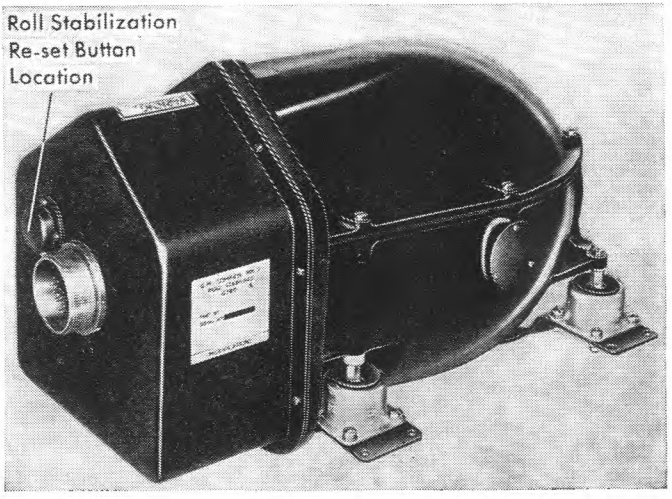


Fig 3 Azimuth Gyro Unit

a CL 11 gyro mounted in inner and outer gimbals The unit is shown in Figs 3 and 4. The gyro has $\pm 87°$ of freedom in the pitching and rolling planes, and is levelled by a levelling torque motor, actuated by a liquid-level switch on the inner gimbal. Magnetic slaving of the gyro is achieved by feeding magnetic heading information to a slaving torque motor which precesses the gyro in azimuth by applying a torque about the inner gimbal axis. Azimuth precession is also induced to compensate for earth-rate gyro drift, by the application of a separate controlled torque about the same inner gimbal axis. For more detailed information on the CL 11 gyro reference should be made to Part 2, Sec 2, Chap 3.

13. Initially the gyro erects to indicate a random heading, since the spin axis is not aligned physically to any specific directional datum. The gyro is subsequently precessed, as described above, by magnetic-slaving and earth rate correction signals, to maintain this random alignment with respect to Earth. Aircraft heading, relative to the gyro spin axis, is transmitted electrically through the stator of a synchro control transmitter (CX), attached to the gyro case, to the stator of a synchro control receiver (CTB) in the master indicator. It will be seen later that this relative heading is given significance by mechanical orientation of the CTB stator.

14. **Roll Gimbal Servomechanism.** To minimize gimballing errors, the directional gyro is provided with servo-controlled roll-stabilization, and is mounted in a roll gimbal. The drive to the roll gimbal is so designed that, whenever the aircraft rolls, the gimbal, and thus the directional gyro case, is kept horizontal. This is achieved by a roll servo loop in which the gimbal-drive motor is activated by the output of a servo-amplifier. Two signal inputs, shown in Fig 5, are compared in the servo-amplifier. One of the voltage inputs is proportional to the aircraft's roll attitude, and is taken from the roll potentiometer wiper of an external Horizon Gyro Unit. The other comes from the roll feedback potentiometer, whose wiper is positioned by the gimbal drive-shaft. Its voltage therefore represents the roll position of the roll gimbal. When the two voltages are equal, there is no output from the servo-amplifier and the gimbal-drive motor remains passive. Any subsequent change of roll attitude will be reflected as a change in the voltage input from the Horizon Gyro Unit to the servo-amplifier. Thus the two inputs will no longer be equal, and the resulting error signal from the servo-amplifier will immediately cause the drive motor to turn the roll gimbal in the appropriate direction. Movement will continue until the error voltage returns to zero, when the directional gyro will again be horizontal. The servo-loop follow-up rate is extremely fast (minimum 150° per second) and damping is provided by velocity feedback from a tachogenerator mounted on the gimbal drive-shaft.

15. **Roll Gimbal Locking Device.** A roll-gimbal locking device is incorporated, so that the roll-servo loop is automatically isolated in the event of a failure in the loop or its inputs. The device is

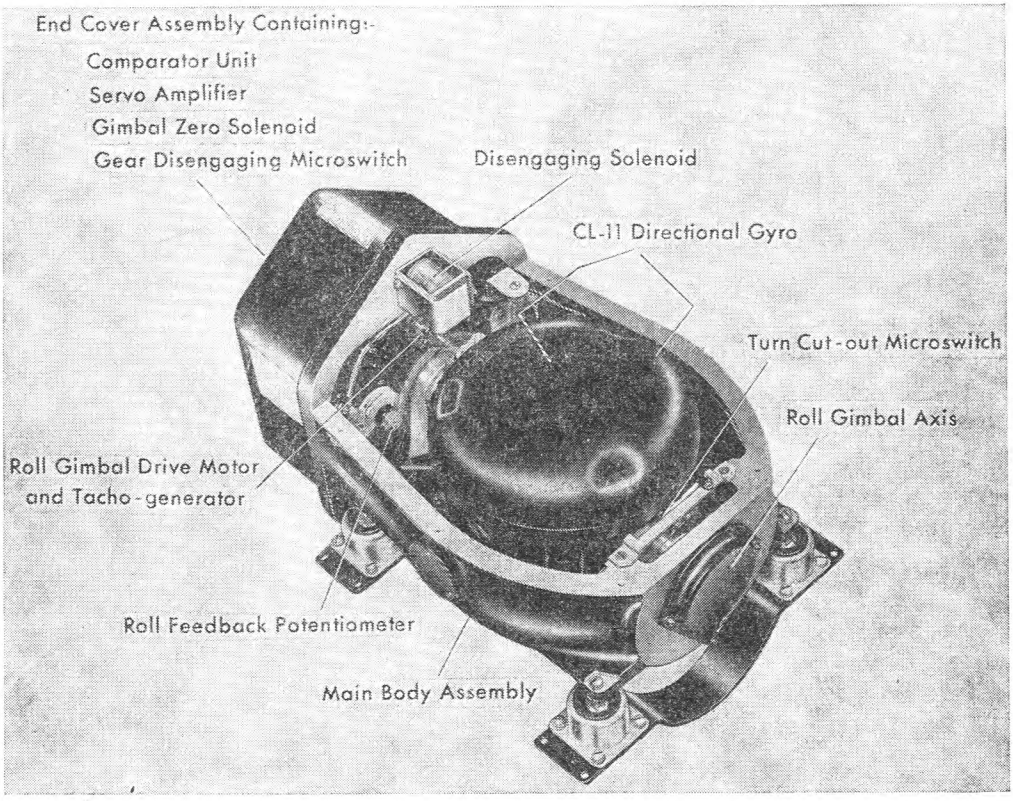


Fig 4 Azimuth Gyro Unit (Top Cover Removed)

actuated by a comparator unit which is fed with the same two roll voltage inputs as the servo amplifier. Whenever these two voltages differ for more than one eighth of a second by an amount equivalent to 5° or more of roll, the comparator unit generates a failure signal. This causes the gimbal to be centralized and locked. The drive is disengaged and the power to the servo-amplifier is removed. Thus the roll servo loop is completely isolated, and the directional gyro, which continues to operate normally, will therefore be subject to gimballing errors. The comparator circuit can be reset, and roll stabilization resumed, by the operation of a reset button on the body of the azimuth gyro unit, *see* Fig 3, or of a reset switch at the navigator's position. For test purposes, a cut-out switch is also fitted at the navigator's position, whereby the roll stabilization circuit can be deliberately disconnected.

Note: Aircraft which carry this twin compass installation also have twin Horizon Gyro Units. Each feeds either the compass system or the aircraft's zero reader. Thus at any one time, both compass systems are fed by one HGU. A remote change-over switch enables the navigator to select either of the HGUs to provide the requisite roll-attitude voltage for the roll stabilization circuits of the azimuth gyro units.

16. **Turn Cut-Out Cam.** When aircraft roll exceeds 6° in either direction, a cam on the pivot of the roll gimbal operates a cut-out microswitch which removes the power supply to the slaving amplifier and cuts off the magnetic slaving signals to the gyro. In other words, the compass system, during most turns is made to operate in the DG mode. In the event of roll stabilization failure, this cut-out facility will be inoperative, since the roll gimbal, permanently locked in its central position, can give no indication of roll.

Master Indicator

17. The master indicator is mounted at the navigator's position. The unit is shown in Fig 6.

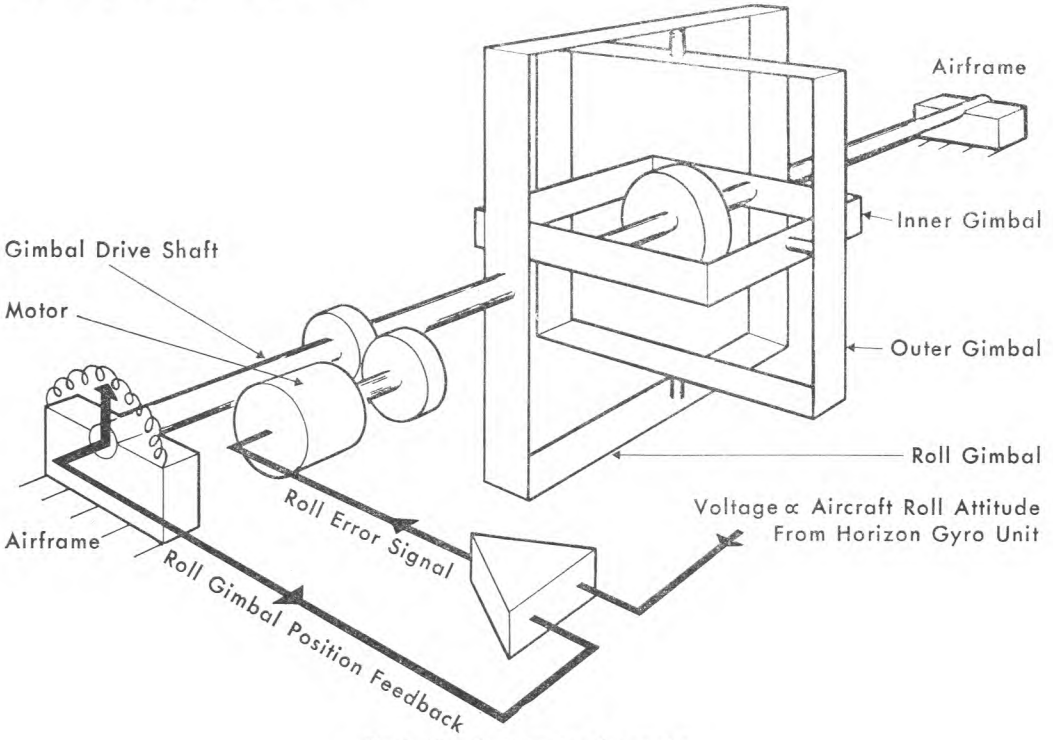


Fig 5 Roll Stabilization Schematic

18. **Description.** Compass heading is indicated on the front of the dial, which is calibrated in increments of one degree, against a rotatable heading pointer. Two windows in the face reveal an annunciator flag and a sector of a variation setting dial. The indicator has two controls, a variation setting knob and a manual synchronizing knob. The synchronizing knob provides a facility either for rapidly synchronizing the compass in the slaved mode, or for setting any desired gyro heading indication in the DG mode. Compass heading is fed from four synchro control transmitters in the master indicator to:

a. The Pilot's Indicator.
b. ADRIS equipment and airborne radar, via an Azimuth Repeater Unit, where the synchro heading signals are first converted to an M-type output.
c. A Radio Magnetic Indicator (RMI), when fitted.
d. The autopilot coupler.

19. **Deviation Correction.** The compass may be corrected for up to $\pm 5°$ of deviation coefficient A by using the variation setting knob to offset the stator of CTB 2, and subsequently rezeroing the variation indicator lubber-line with a compass key through a keyway in the side of the master indicator.

Pilot's Indicator

20. The pilot's indication is read from a card,

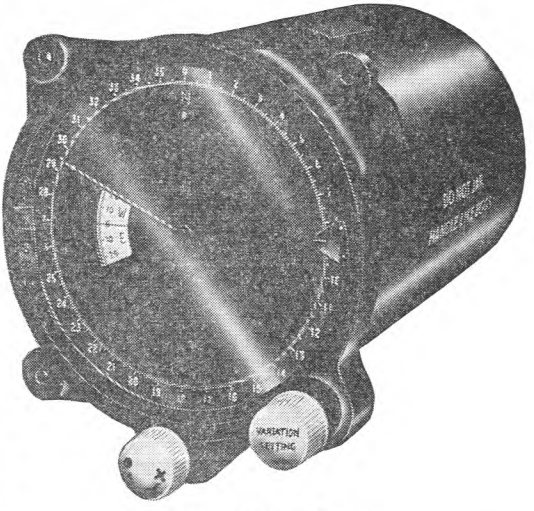


Fig 6 Master Indicator

graduated in increments of two degrees, which rotates against a fixed lubber line. The instrument is shown in Fig 7.

21. The indicator is driven by a servo-loop from the master indicator. It has no annunciator or function flag but has two control knobs. A set heading pointer is normally locked to the rotatable card, but can be rotated over the face of the card by the use of the knob marked SET HEADING. The second knob marked 1-2, is in effect a change-over switch, by which the pilot can select for display the heading output of either of the two compass systems.

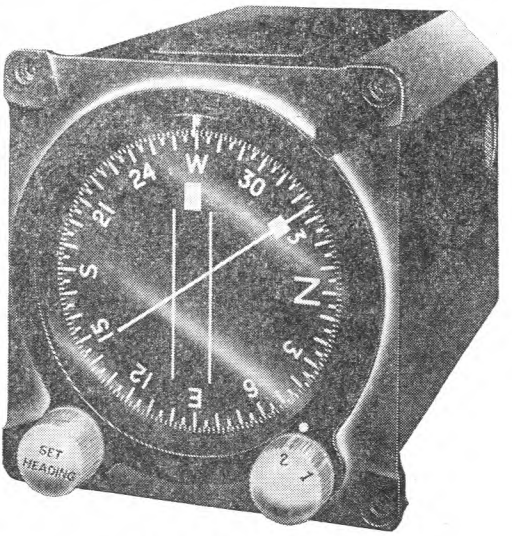


Fig 7 Pilot's Indicator

Controller

22. The controller, shown in Fig 8, carries the following controls:

a. A two-position mode switch, marked SLAVED/DG, by which the navigator selects the desired mode of compass operation.

b. A North/South hemisphere switch (N/S) which determines the sense of the Earth-rate precession correction applied to the directional gyro.

c. A dial, graduated in increments of two degrees of latitude, which determines the magnitude of the Earth-rate precession correction.

Mounting Rack

23. The mounting rack provides a protective environment for the following three elements of the compass system:

a. *Power Supply Unit.* The power supply unit provides power for the Earth rate compention circuits in the Controller and Azimuth Gyro Unit.

b. *Servo Amplifier.* As part of the heading output servo-loop, the servo amplifier amplifies an error signal to provide follow-up alignment between the master indicator and directional gyro.

c. *Slaving Amplifier.* The slaving amplifier is a part of the magnetic slaving servo-loop, and amplifies an error signal to provide a follow-up alignment between the detector unit and the master indicator.

Fig 8 Controller

24. A socket on the front of the mounting rack is the point where a Precise Headings Test Set is plugged into the compass system during compass swinging.

OPERATION OF THE SYSTEM

Introduction

25. In both modes of operation, DG and slaved, directional gyro information is reproduced at the equipment's heading outputs and the gyro can be continuously precessed to compensate for Earth-rate gyro drift. The gyro heading output is initially slewed in the master indicator to give either a desired gyro heading (DG Mode) or magnetic heading (Slaved Mode); no attempt is made to align the gyro to any specific azimuth datum. In the slaved mode, after an initial manual synchronization has been carried out, the gyro is subsequently precessed in azimuth by a magnetic-slaving loop to maintain synchronism between the detector unit heading and the gyro heading output. Transmission of gyro heading to the compass heading outputs by a servo mechanism is common to both modes of operation.

Basic Servomechanism

26. The basic servomechanism is illustrated in Fig 9. It is a position control servo. The basic theory and background information on servo-mechanisms are discussed in Part 3, Sec 1. The rotor of the synchro control receiver (CTB1), when displaced from its electrical null position by a change of input to the stator (during a change of heading for example), transmits an error signal whose phase depends on the direction of rotor displacement. This signal passes to the servo amplifier which activates the servo motor (M) turning in the direction necessary to turn the CTB rotor to its new null position and in so doing rotates the heading output shaft through an angle equal to the heading change. The output of the servo loop is damped against oscillation by velocity feedback from a tachogenerator (TG).

27. There are four servo loops in the GM 7 equipment. These are:

a. Azimuth gyro roll-stabilization servo.
b. Gyro follow-up or output servo.
c. Magnetic slaving servo.
d. Pilot's indicator servo.

The azimuth gyro roll-stabilization servo has already been described in para 14. The gyro

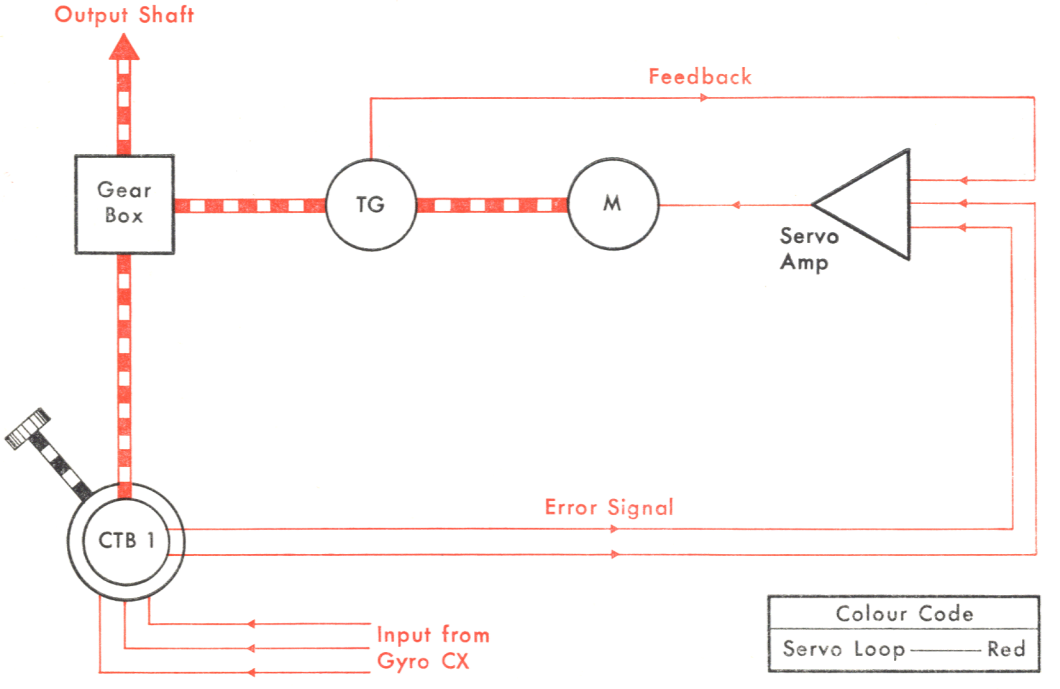


Fig 9 Basic Servomechanism

follow-up or output servo has so far been called the basic servomechanism. It passes the modified gyro heading as a shaft rotation, via the CTB, to the rotors of control transmitters (CXs 2, 3, 4, and 5) which transmit the equipment's heading output to the pilot's indicator and to ancillary equipments.

Magnetic-Slaving Servomechanism

28. The magnetic-slaving servo is the loop through which the system heading output is slaved to the magnetic heading sensed by the detector unit. A diagram of the GM 7 is given in Fig 10. Magnetic heading information is fed electrically to the stator of CTB 2 from the flux-valve detector coils, which themselves behave as a synchro CX.

29. When displaced from its electrical null, the rotor of CTB 2 transmits an error signal, via the slaving amplifier, to energize the slaving torque motor which then precesses the directional gyro. Any movement of the gyro is transmitted by the gyro follow-up loop and the output shaft of CTB 1 will be turned, reproducing the apparent heading change at the output CXs 2, 3, 4 and 5. It can be seen that during this operation the follow-up servo is being used as part of the magnetic slaving loop. The output shaft of CTB 1 will continue to turn until the rotors of CTB 1 and CTB 2 are both nulled. When this occurs the compass heading output will have become synchronized with the heading sensed by the detector unit.

Magnetically-Slaved Mode

30. **Initial Synchronization.** When the compass has been switched on, with the controller mode switch set to SLAVED, the compass heading will usually be found to be desynchronized. This is indicated by a steady dot or cross in the annunciator window, showing that an error signal from CTB 2 is being transmitted by the slaving amplifier to precess the gyro. The compass will synchronize itself but this is a slow process as the maximum rate of the slaving loop is only 1·5 degrees per minute. Manual synchronization can be achieved by pressing in and turning the synchronizing knob. This physically turns the stator of CTB 1 creating an error signal which precesses the gyro. The response of the follow-up servo is much more rapid than that of the slaving servo being approximately 45 degrees per second. When the annunciator shows half a dot and half a cross; the slaving-amplifier output will then be zero and the compass is therefore synchronized.

Variation Setting

31. When the variation setting knob is pressed and turned, this again has the effect of displacing the null position at CTB 2, and the resulting error signal activates the slaving servo which will slowly re-synchronize the system. If several degrees of variation are inserted at once, the system should be manually synchronized.

Operation in a Turn

32. In considering the operation of the system during a turn, the reader is asked to ignore for a moment the turn cut-out circuit. During the turn the follow-up servo follows the action of the gyro, transmitting the turn to the shaft of the master indicator. On this shaft are the heading pointer and the rotor of CTB 2. When the aircraft steadies at the end of the turn, the rotor of CTB 2 will still be in its null position, for it will have been turned through the same angle and the same rate as the stator field of CTB 2 transmitted from the detector unit.

33. During a turn the detector unit may be tilted and sense a false magnetic heading. A cut-out circuit is therefore incorporated. The magnetic slaving is cut off by removing the power supply to the slaving amplifier whenever the aircraft's angle of bank exceeds 6°. This is achieved by a micro switch activated by a cam on the azimuth gyro unit roll gimbal. In a slow turn, where the angle of bank is less than 6°, or in the event of roll stabilization failure, the sensed magnetic heading will be incorrect and an error will be left at the end of the turn. This will be subsequently removed by the normal action of the slaving servo at the slow rate.

Pilot's Indicator Servomechanism

34. The fourth servomechanism is simply a means of repeating the compass output at the pilot's heading dial. CX 2 transmits the compass heading, via a system changeover switch, to the stator of a control transformer (CT), which is the error detector in the pilot's heading indicator servomechanism. As in the basic servo loop, any displacement of the CT rotor from its null position generates an error signal which, after amplification activates a servo motor. The motor drives the shaft with the pilot's heading card and the rotor of the CT in such a direction as to eliminate the error signal. The pilot's heading dial is thus automatically synchronized with that of the master indicator.

DG Mode of Operation

35. The method by which the gyro heading is

picked off from the synchro CX 1 on the gyro and transmitted via the follow-up servo to the system heading outputs, has already been described. The magnetic slaving servo is of course inoperative in the DG mode, since, when the navigator selects DG at the controller, the power supply to the slaving amplifier is disconnected, and this amplifier can therefore transmit no error signal. Consequently, if the variation setting dial is turned, it will have no effect on the compass heading output.

36. **Initial Heading.** After the gyro has been switched on, it will erect to the horizontal and will indicate a random heading. In the DG mode the navigator can now use the manual synchronization knob, on the master indicator, to set his desired initial heading. The compass system can be used confidently for grid navigation techniques, provided the appropriate correction for Earth-rate gyro drift is maintained at the dial on the controller. Real drift at the gyro has been found to be approximately one degree per hour.

GYRO-MAGNETIC COMPASS MARK 7A

Introduction

37. The Gyro-Magnetic Compass Mark 7A (GM 7A) is in many ways similar to the GM 7 compass described earlier, but will be seen to have fewer refinements. The compass has two modes of operation, the slaved mode and the DG mode.

Components

38. The main components of the equipment are:
   a. Detector Unit.
   b. Remote Correction Unit.
   c. Directional Gyroscope.
   d. Master Indicator.
   e. Navigator's Controller.
   f. Pilot's Indicator.
   g. Pilot's Control Panel.
   h. Mounting Rack.

Detector Unit

39. The detector unit is illustrated in Fig 11. It is identical to the unit used in the GM 7 compass which is described in paras 7 and 8. It senses and transmits electrically, to the master indicator, the aircraft heading relative to the magnetic meridian. The unit can be rotated in its mounting to compensate for deviation coefficient A.

Remote Correction Unit

40. The remote correction unit, *see* Fig 12, is again the same as that used in the GM 7. It is described in paras 9 to 11. This unit enables the navigator to compensate electrically, the detector unit's output of magnetic heading for coefficients B and C of the local deviations affecting the detector unit.

Directional Gyroscope

41. The directional gyroscope, *see* Fig 13, is a CL 11 gyroscope. It is identical to the gyro component of the GM 7 except that it is not roll

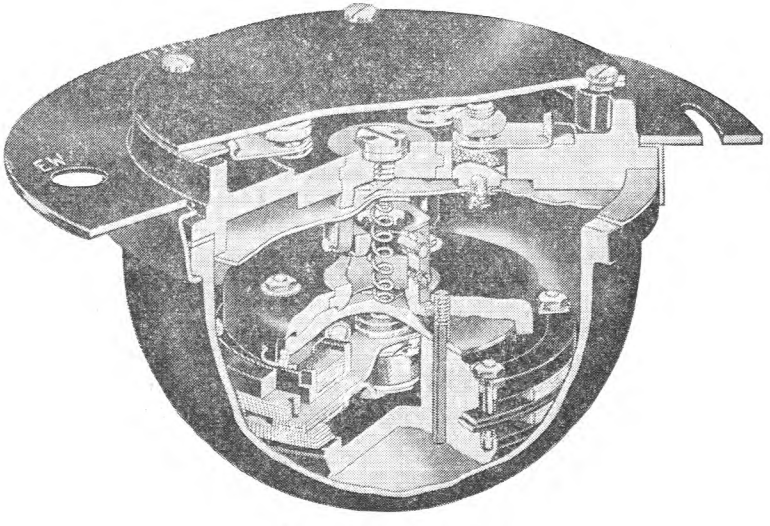


Fig 11 Detector Unit

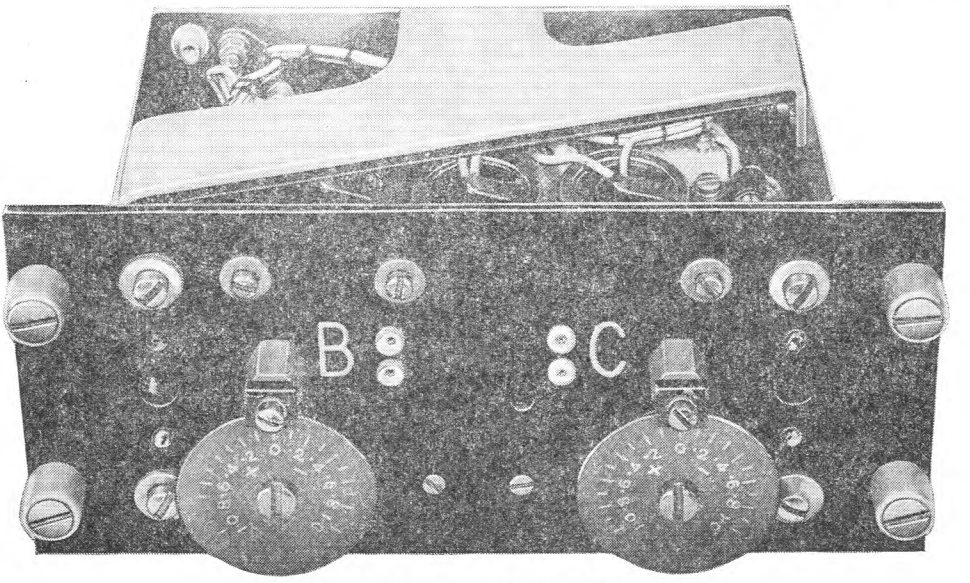


Fig 12 Remote Correction Unit

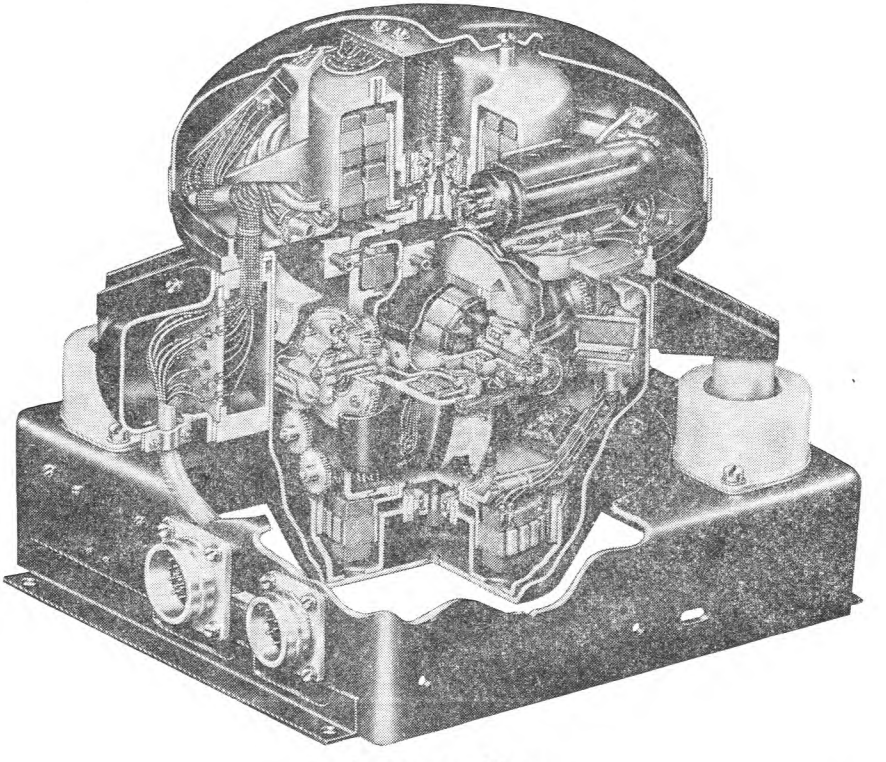


Fig 13 Directional Gyro

stabilized, and is therefore subject to gimballing error. Full details of the CL 11 gyro are given in Part 2, Sect 2, Chap 3.

42. Initially the gyro erects to indicate a random heading, since the spin axis is not aligned to any specific directional datum. The gyro is precessed subsequently, in the slaved mode, by magnetic slaving and Earth-rate correction signals, to maintain this alignment with respect to the Earth.

Master Indicator

43. The master indicator, mounted at the navigator's position, is shown in Fig 14. Heading is indicated by a rotatable heading pointer against the face of the dial, the circumference of which is graduated in increments of one degree. There are two windows in the face of the dial. The small window at the top of the face reveals either an annunciator with a dot or a cross, in the slaved mode, or a DG flag, when the DG mode is selected. The other window shows a sector of the variation-setting dial. The knob at the bottom right is for setting variation.

44. The master indicator can provide five outputs of heading simultaneously:

a. *Magnetic Heading*. There are two outputs of magnetic heading. First a synchro heading output goes to VOR and RMS (Radio Magnetic Selector) equipment. Second a synchro heading output goes via the MAG position of the MAG/TRUE switch, to the pilot's heading repeater and to the zero reader and autopilot equipment.

b. *True Heading*. Three outputs of true heading are available. The first is an M-type output to GPI equipment. The second a synchro output, via the TRUE position of the MAG/TRUE switch, to the pilot's heading repeater, zero reader and autopilot equipment. Finally there is a spare synchro output.

45. **Deviation Correction.** The compass may be corrected for small amounts of deviation coefficient A by using the variation setting knob to offset the stator of a synchro control receiver (CTB). Subsequently the variation lubber line is re-zeroed with a compass key through a keyway in the side of the master indicator. Mechanical stops limit the correction to $\pm 1°$. The effect on the compass system outputs, of this method of correction, is described in para 66.

Navigator's Controller

46. The navigator's controller is the same as that used with the GM 7 compass which is described in para 22. The desired mode of compass opera-

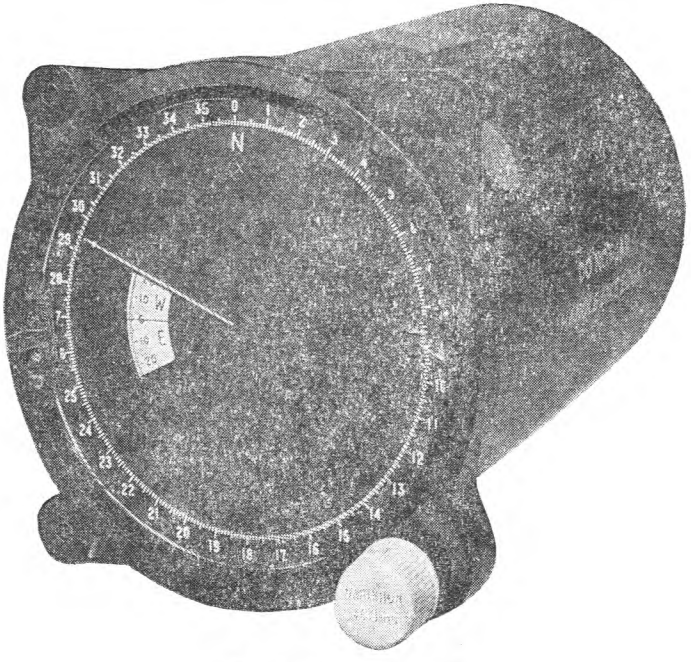


Fig 14 Master Indicator

Fig 15 Navigator's Controller

tion (DG or Slaved) is selected, and appropriate settings (hemisphere and latitude) are made to provide an appropriate torque to the directional gyro to compensate for earth-rate gyro drift.

Pilot's Indicator

47. The pilot's indicator is shown in Fig 16. Aircraft heading is read from a card which rotates against a fixed lubber-line and which is graduated in increments of two degrees. The repeater has a set-heading knob by which the pilot can turn a reference pointer over the surface of the heading card. It also has a window in the top right hand corner, which shows a DG flag when the navigator selects the DG mode. In the slaved mode the window is blank.

Pilot's Control Panel

48. The pilot's control panel is shown in Fig 17. The unit has an annunciator dial with a centre-reading needle and embodies push-button operated circuits whereby the pilot can either synchronize the compass system in the slaved mode or set any desired heading indication in the DG mode. The annunciator needle deflects to the right or left to show the direction of any de-synchronization. The pilot can quickly re-synchronize the system by pressing either of the two push buttons (L or R), irrespective of the direction of desynchronization. When the annunciator needle returns to centre, the system is synchronized, and the button can be released. In the DG mode the buttons are used solely to change the heading output to indicate any desired gyro heading. The L (left) button increases and the R (right) decreases the indicated heading.

49. **Mag/True Switch.** The MAG/TRUE switch is mounted at the pilot's position. This enables the pilot to select magnetic or true heading output for display on his heading repeater. The heading output selected is also transmitted to the aircraft zero reader and autopilot equipment. A small indicator is provided at the navigator's position to indicate which selection the pilot has made.

Mounting Rack

50. The mounting rack provides a protective environment for three elements of the compass system.

a. *Power Supply Unit.* The power supply unit provides power for the Earth-rate compensation circuits and for the manual synchronization circuit.

b. *Servo Amplifier.* The servo amplifier, as part of the heading follow-up servo loop,

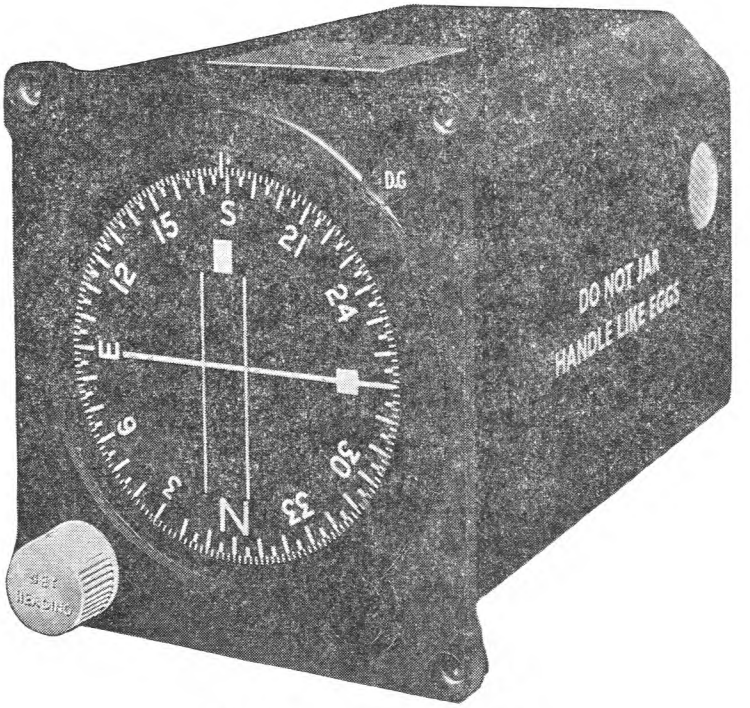


Fig 16 Pilot's Heading Repeater

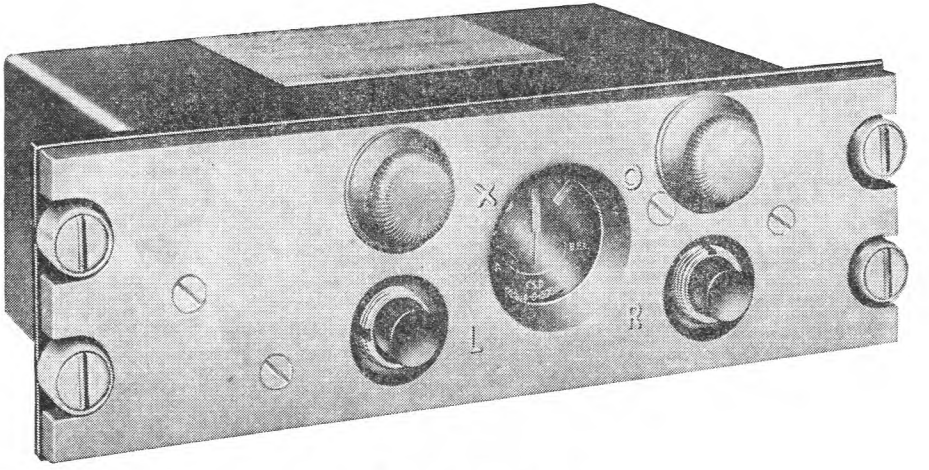


Fig 17 Pilot's Control panel

amplifies the error signal to align the master indicator and the gyro.

c. Slaving Amplifier. The slaving amplifier is part of the magnetic slaving servo loop. It amplifies the error signal to provide alignment between the detector unit and the master indicator.

51. A socket is provided on the front of the mounting rack where a DC voltmeter, *see* precise heading test set described in Chapter 1, is plugged into the compass system during compass swinging.

OPERATION OF THE SYSTEM

Introduction

52. In both modes of operation, DG and Slaved, directional gyro information is reproduced at the equipment's heading outputs. In the GM 7A the gyro can be continuously precessed to compensate for Earth-rate gyro drift. The gyro heading output is initially slewed in the master indicator to give either a desired gyro heading (DG) or magnetic heading (Slaved Mode); no attempt is made to align the gyro to any specific azimuth datum. In the slaved mode, after an initial synchronization procedure has been carried out, the gyro is subsequently precessed in azimuth by a magnetic slaving loop to maintain synchronism between the detector unit heading and the gyro heading output.

Follow-Up Servomechanism

53. The follow-up servomechanism is illustrated in Fig 18. It is a position control servo which transmits changes in the gyro reading to the equipment heading outputs. Any change, in heading for example, will result in a change of stator field being transmitted from the stators of CX 1 to the stators of CT 1. The rotor of CT 1 will no longer be in its null position so an error signal will pass via the push buttons in the pilot's control panel to the servo amplifier. The phase of this signal depends upon the direction of rotor displacement. The amplified error signal activates the servo motor (M) which turns in the direction necessary to turn the CT 1 rotor, via

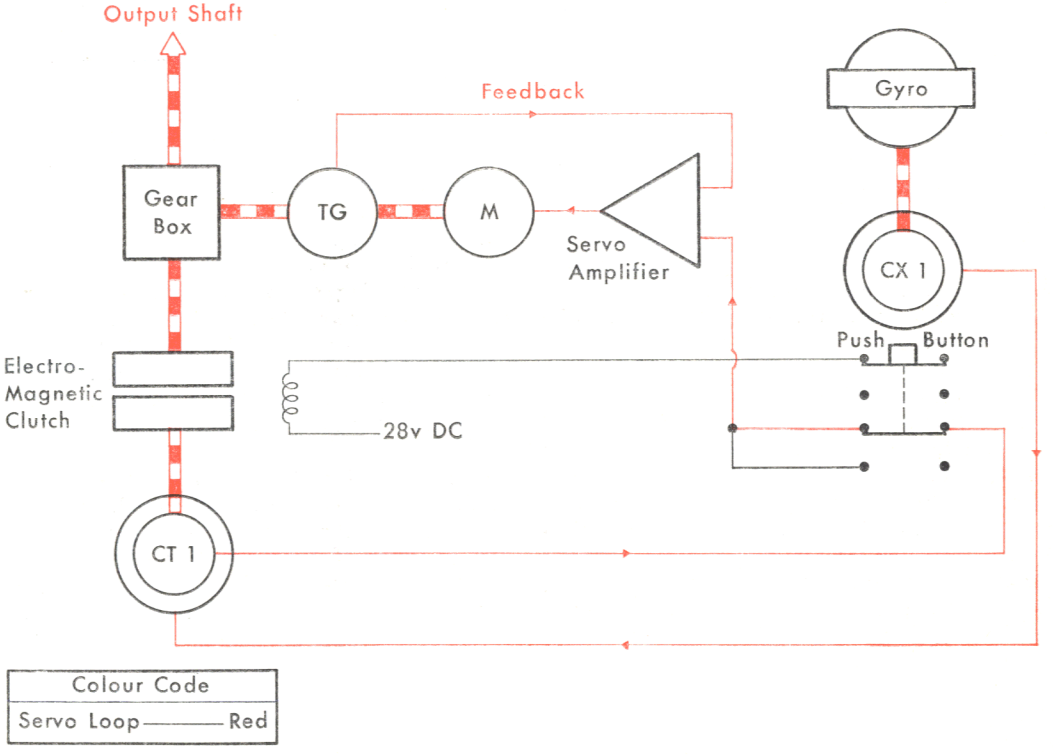


Fig 18 GM7A Follow-Up Servomechanism

the gear box and the engaged clutch, to its new null position. In so doing, the motor rotates the output shaft through an angle equal to the heading change. The output of the servo loop is damped against oscillation by velocity feedback from a tachogenerator (TG). Servomechanisms and transmissions systems are covered in Part 3, Sect 1. The follow-up rate of this servo is rapid. The rate of heading change at the output shaft in response to a change in gyro heading can reach 45° per second. The follow-up servo is shown in red on the compass signal flow diagram at Fig 19.

Magnetic Slaving Servomechanism

54. The magnetic slaving servomechanism is shown in green on the compass signal flow diagram at Fig 19. As its name implies, it is the loop through which the gyro heading output is slaved to the magnetic heading sensed by the detector unit, in the slaved mode.

55. Magnetic heading is fed electrically to the stator of the CTB from the flux-valve detector coils which themselves behave as the stator of a synchro CX. When the rotor of the CTB is not in its null position, an error signal is transmitted, via the slaving amplifier, to energise a torque motor (M) which precesses the gyro. This precession is registered as a heading change at the stator of CX 1 which is transmitted to the stators of CT 1. So the follow-up servo is activated as described in para 53, and the heading change is reproduced as a rotation of the output shaft. During this operation the follow-up servo is used as part of the magnetic slaving servo. The output shaft will continue to be turned until the rotors of the CTB and CT 1 are both nulled. When this is achieved the compass heading output will be synchronized with the heading sensed by the detector unit.

Initial Synchronization

56. When the compass system has been switched on initially and the slaved mode selected, the heading shown at the master indicator will usually be found to be desynchronized from that sensed by the detector unit. Such desynchronization is indicated on the master indicator by a steady dot or cross in the annunciator window, and on the pilot's control panel by a steady deflection of the annunciator needle. These indications show that an error signal is being transmitted from the rotor of the CTB via the amplifier to precess the gyro. If no action is taken to synchronize the compass, it will synchronize itself through the normal action of the slaving loop described in the previous paragraph. This is a slow process as the slaving rate is 1·5° per minute. Fast synchronization is necessary.

57. Two push buttons are provided on the pilot's control panel for fast synchronization. The same two buttons are used in the DG mode to select the desired initial heading. For illustrative purposes, only one of the two push buttons is shown in the diagrams. The effect on the system, of using specifically the L or R button, will be described in the text.

58. The compass system can be synchronized rapidly by pressing either of the push buttons on the pilot's control panel, irrespective of the direction of desynchronization. The CTB error signal is made to activate the motor of the follow-up servo loop. This is shown in Fig 19. The slaving error signal is fed via the slaving amplifier to the pilot's annunciator. From there it is fed via the push button, when depressed, direct to the follow up motor. The modulator converts the signal to a form suitable for application to the servo amplifier. The resulting rotation of the output shaft by the normal action of the follow-up servo continues at approximately 45° per second, until the slaving error signal at the CTB is nulled. The pilot's annunciator needle then returns to its centre zero position, indicating that synchronization is complete; the pilot should then release the push button. During the operation the electro-magnetic clutch is disengaged, as shown in Fig 19. This isolates the CT 1 rotor from the rotation of the output shaft.

59. It is of interest to note that during fast synchronization the normal operation of the magnetic-slaving servo continues, and a gyro follow-up error signal will thus build up at CT 1 until the CTB error signal is nulled. This error will be small, it is unlikely to exceed 0·1°, and will be removed automatically by normal slaving action after the push button is released.

60. In the DG mode the rapid follow-up action of the follow-up servo is again used, to enable the pilot to select any desired initial gyro heading. Fig 20 shows that the selection of DG at the navigator's controller disconnects the power supply to the slaving amplifier, which can therefore transmit no error signal. Thus the slaving servo and the fast synchronizing facility are both isolated. At the same time, the relay in the pilot's control panel is de-energized, causing an AC supply to be fed, via the relay's DG contact, to the push buttons. When either of the push buttons

is pressed this supply is fed to the servo amplifier and causes the output shaft to be turned. Operation of the L button produces an increase in the indicated heading and of the R button a decrease. During this time the electro-magnetic clutch is once more disengaged to prevent the rotor of CT 1 being driven out of its null position. The pilot should release the push button as the desired heading indication is reached.

Operation in a Turn

61. During a turn, two elements of the compass system sense the heading change independently: the gyro and the detector unit. The follow-up servo is activated by signals from the rotor of CT 1, representing the heading change transmitted from the gyro, and moves the heading output shaft and the rotor of the CTB through the amount of heading change. At the same time the CTB stator field will turn through the amount of heading change as transmitted by the detector unit. So, provided the two elements sense the same amount of heading change when the aircraft steadies at the end of the turn, the compass will still be synchronized, with the CTB rotor remaining in its null position. This is the ideal, in practice the combined presence of gyro gimballing error and magnetic turning error, *see* Chapter 1, usually causes the gyro and detector units to sense different changes of heading during a turn. There is no turn cut-out with the GM 7A compass system. The amount of residual desynchronization when the aircraft steadies is dependent on a number of factors, but it is limited by the maximum output of the slaving amplifier. The maximum possible error for this reason is limited to 1·5° for each minute of turn.

Heading Outputs and Variation

62. It was explained earlier, *see* para 44, that

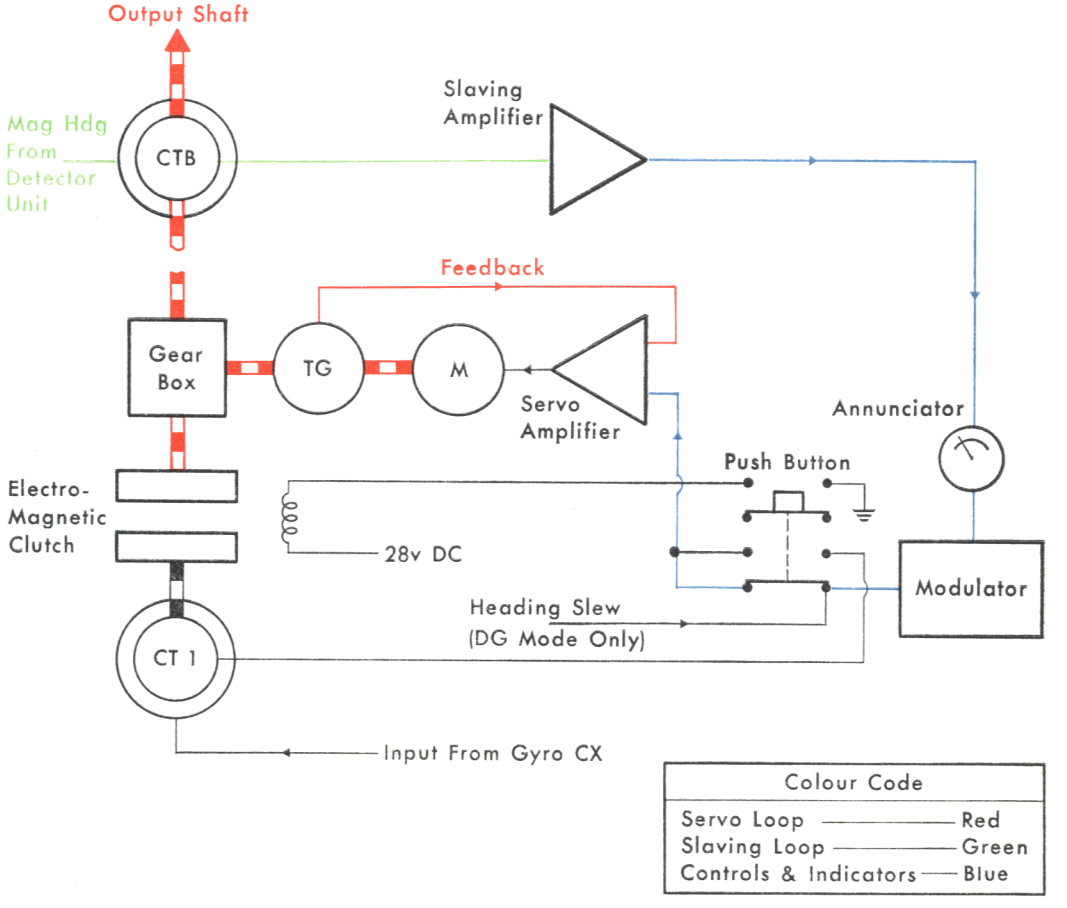


Fig 19 GM7A Fast Synchronization

displacement of the CT rotor from its null generates an error signal which, after amplification, activiates a servomotor to drive a shaft. Rotation of the shaft drives the heading card on the pilot's heading dial, and simultaneously turns the CT rotor in such a direction as to eliminate the error signal. When the error signal is zero, the heading displayed is synchronized with the selected heading output of the master indicator.

CHAPTER 6

POLAR PATH COMPASS

CONTENTS

	Paras
Introduction	1–4
Principles of Operation	5
Equipment	
Components	6–7
Directional Gyro	8–13
Fluxvalve Detector	14–17
Compass Controller	18–23
Compass Coupler	24–25
Compass Comparator	26
Operation of the System	
Introduction	27
Basic Servomechanism	28–31
Correction Servomechanism	32–34
DG Mode of Operation	35–38
Magnetically Slaved Mode	39
Transmission of Fluxvalve Information	40–46
Deviation Correction	
Introduction	47
Magnetic Compensation	48
Mechanical Compensation	49–58
Compass Swings	59–62
Operating Instructions	
Introduction	63
Before Flight	64
In Flight	65–66

Introduction

1. The Polar Path Compass is a gyromagnetic heading system providing heading outputs to ancillary equipment and to an autopilot. The gyro output is usually monitored by magnetic heading information, but can be used alone in a directional gyro (DG) mode.

2. The gyro used in the system is a low-drift instrument whose spin axis is levelled by a gravity-sensitive erection device. When the system is used in the DG mode compensation can be made both for apparent gyro drift and for systematic real drift errors.

3. Magnetic heading is derived from a pendulous fluxvalve. The heading output of the directional gyro is continuously slaved to the heading sensed by the fluxvalve in the magnetically-slaved mode. The maximum follow-up rate of the slaving system is approximately 1.75° per minute in normal slaved operation.

4. The equipment described in this chapter is duplicated to provide a twin compass installation.

Principles of Operation

5. No new principles are involved in the operation of the Polar Path Compass, although the techniques used are somewhat different from those employed in earlier gyro-magnetic systems. The theory of the fluxvalve is treated in Part 2, Sect 3, Chap 1, paras 10–22 of this volume, the principles of remote indication and control, and servo-mechanisms, are covered in Part 3, Sect 1. The reader is advised to revise his knowledge of these subjects, if necessary, before continuing.

EQUIPMENT

Components

6. The basic components of a single compass are:—

a. Directional Gyro.

b. Fluxvalve Detector.

c. Compass Controller.

d. Compass Coupler.

In a twin installation the above items are duplicated, and a Compass Comparator is provided to light warning lamps whenever a significant difference exists between the outputs of the two compasses.

7. An elementary block diagram of a single system is shown at Fig 1. Each block represents a system component. The ancillary equipments fed by the heading outputs can include:—

a. The autopilot (flight director) systems.

b. Navigation systems and displays eg CDIs, HSIs.

c. Omega

d. The VOR/ILS systems, together with inputs to Radio Magnetic Indicators (RMIs) for VOR, TACAN and ADF indication.

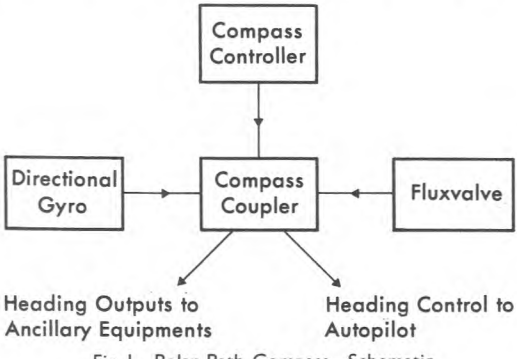


Fig 1 Polar Path Compass—Schematic

Directional Gyro

8. The directional gyro provides the basic azimuth reference for the system, since its spin axis remains fixed in azimuth as the aircraft heading changes. Aircraft heading relative to the spin axis is transmitted through the stator coils of a synchro control transmitter (CX) attached to the frame of the gyro.

9. The gyro is shown in Figs 2a and 2b, and consists of a rotating element suspended in two gimbals that are free to turn about their axes. The spinning element is the rotor of a three-phase induction motor whose stator is fixed to the inner gimbal. The outer gimbal is free to rotate 360° in azimuth, but the inner gimbal has only ±70° movement relative to the horizontal of the housing.

Fig 2a Directional Gyro

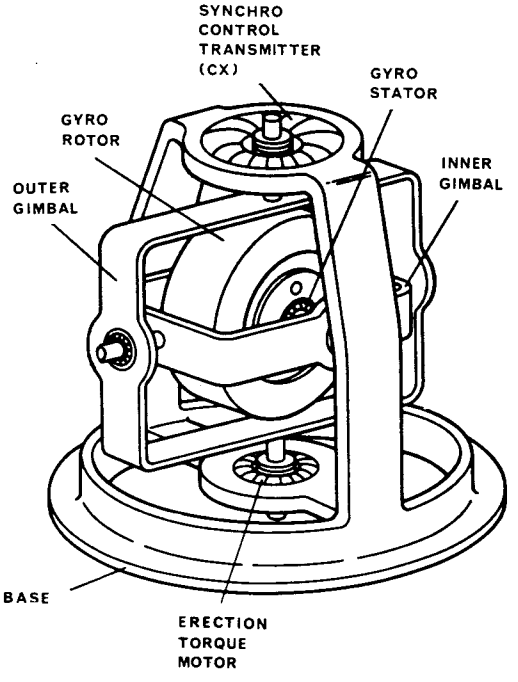


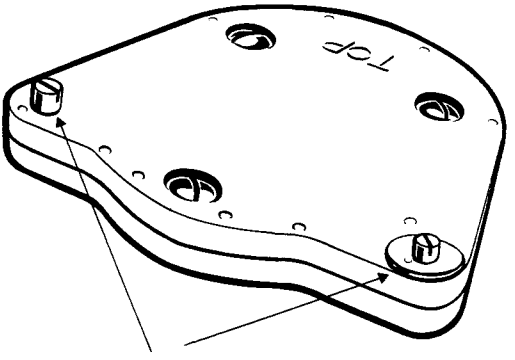
Fig 2b Directional Gyro—Simplified

10. **Erection Mechanism.** The spin axis of the gyro is levelled by a torque motor whose action is controlled by a liquid-level switch attached to the inner gimbal. The stator coils of this motor, which is a two-phase induction type, are embedded in the base of the gyro assembly, and its rotor is fixed to the outer gimbal axis. A torque about the vertical axis produced by this motor causes the gyro to precess about the inner gimbal axis.

11. **Liquid Level Switch.** The liquid level switch is mounted horizontally at the inner gimbal pivot and is a glass tube partially filled with electrolyte which provides a conductive path between three electrodes. The central electrode is connected to the compass power supply and is permanently immersed in the electrolyte. The other two electrodes, one at each end of the glass tube, are attached to opposing ends of the torque motor split control winding, and are both immersed when the gyro (and switch) is level. In this state, both halves of the control winding are energized, and the equal but opposing fields cancel, so the torque motor remains passive. However, if the gyro is tilted, one or other of the electrodes is uncovered, depending on the direction of tilt, leaving only one half of the control winding energized. The torque motor therefore becomes active; its output precesses the gyro in the appropriate direction until it is again level, and the electrolyte once more covers both electrodes. Although the switch is gravity-sensitive, it is also affected by acceleration experienced during an aircraft turn. This additional acceleration will displace the electrolyte to one end of the tube, and would thus cause the torque motor to be energized even though the gyro were not tilted. To minimize such false levelling, a cut-out switch therefore isolates the liquid level switch whenever the aircraft's rate of turn exceeds 15 degrees per minute. (After modification, this figure will become 30 degrees per minute, though the early versions of the equipment may remain un-modified.)

12. **Gyro Electrical Output.** As the aircraft heading changes the gyro case turns with it, but the spin axis remains fixed in space. The relative movement between the case and the outer gimbal of the gyro is thus equal to the change in aircraft heading. The rotor of the synchro control transmitter (CX) is fixed to the axis of the outer gimbal and parallel to the gyro spin axis. This rotor has a permanent ac supply. The stator coils are attached to the case, and therefore the voltages induced in them change as the aircraft heading changes. These voltages are then passed to the stator of a synchro control receiver (CTB), whose rotor is driven by the servo follow-up mechanism to the null position. Relative heading information is therefore available from the position of this rotor. The azimuth orientation of the gyro spin-axis itself relative to the aircraft is completely random and is not displayed, but it will be seen later that this relative heading is given significance by mechanically orientating the stator of the CTB.

13. **Gimbal Error.** Since the directional gyro is not roll-stabilized, the gyro heading output will be subject to gimbal error (*see* Chap 1, para 83 of this Section). This error occurs only when the aircraft turns, but its magnitude cannot be specified for any given turn, since it is dependent on the unknown orientation of the gyro spin-axis. However, the average value of the error will be integrated into GPI equipment to produce an error in computed position. (Maximum possible error from this particular source could result from a 90° turn, but would be unlikely to exceed 0.7 nm for an aircraft such as the VC-10 at normal cruising speed.)



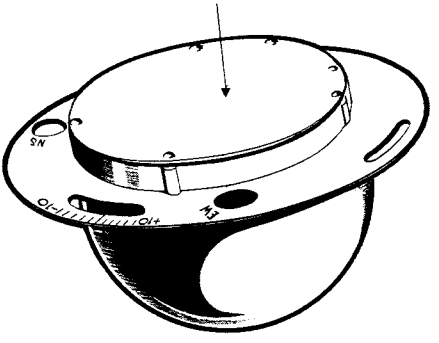


Fig 3 Detector Unit with Magnetic Compensator

Fluxvalve Detector

14. The fluxvalve detector senses aircraft heading relative to the magnetic meridian. The theory is described fully in Part 2, Sect 3, Chap 1. The principles apply equally to the Polar Path system, and the fluxvalve is similar to that used in the G 4B compass.

15. **Description.** The unit consists basically of a conventional three-spoke assembly, suspended in gimbals which permit ±25° of freedom about the pitch and roll axes, but fixed in azimuth to the aircraft fore-and-aft axis. This assembly is mounted in a hemispherical bowl which is filled with oil to damp out oscillations. The detector unit and a magnetic compensator are shown in Fig 3.

16. **Magnetic Compensator.** If the earth's magnetic field in the vicinity of the fluxvalve is likely to be distorted significantly by permanent and induced aircraft magnetism, a magnetic compensator is fitted to reduce the field distortion by introducing equal but opposite distortion. Mounted on top of the detector unit, the magnetic compensator contains two pairs of permanent magnets, one pair lying fore-and-aft, the other athwartships. The angle between the two magnets of each pair can be varied by screw adjustments at the compensator to correct for deviation coefficients B and C.

17. **Transmission of Magnetic Heading.** The three coils of the fluxvalve act as the stator coils of a synchro control transmitter, and their outputs are a function of magnetic heading. The induced voltages are passed to the stator coils of the fluxvalve follow-up synchro, a control transformer (CT), located in the Compass Coupler unit. The rotor of this CT is then driven to its null position by the associated servomechanism.

Compass Controller

18. The Compass Controller carries the controls necessary to operate the equipment, and is shown in Fig 4.

19. **Mode Selector Switch.** The mode selector is a three-position switch marked DG/SLA/SYN, from left to right. In the

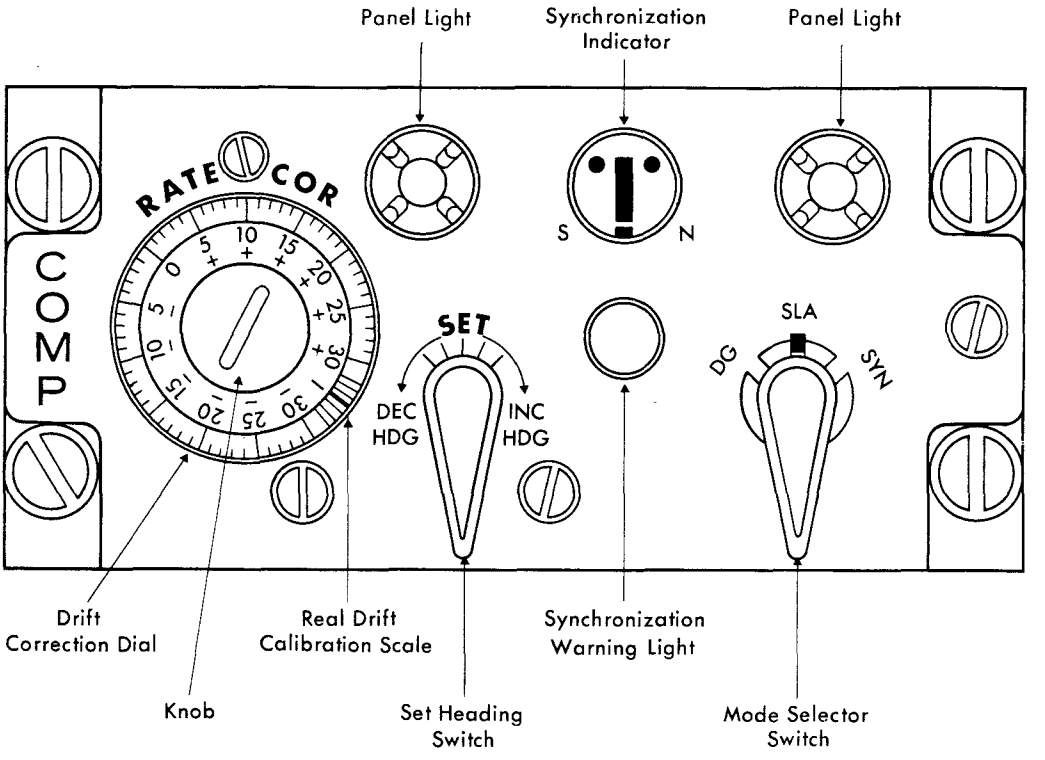


Fig 4 Compass Controller

DG position the compass operates as a directional gyro, without magnetic monitoring. In the SLA position the gyro heading output is slaved to the magnetic information, *ie* corrected slowly to agree with magnetic heading. In the SYN position rapid synchronization of gyro and magnetic data is available. The control must be held in the SYN position against the resistance of a spring. When released it will revert to the SLA position.

20. **Set-Heading Switch.** Marked SET, the set-heading switch enables the operator to slew the gyro heading output to indicate any desired initial heading when the equipment is used in the DG mode. Spring-loaded to its centre-position, the switch has two positions at either side of centre for slow and fast change of indicated heading in either direction. It is inoperative when the mode selector is at SYN.

21. **Drift Rate Correction.** The dial marked RATE COR permits correction to be made to the indicated gyro heading to compensate for earth-rate and transport-rate apparent gyro drift and, to a certain extent, for real drift of the gyro. These corrections are applied only when the system is in the DG mode. The calculation and setting of these corrections are described at the end of this chapter, in para 66. The dial is calibrated in degrees per hour, which can be set as required against the index mark at the top of the dial. Compensation for real gyro drift is described in para 38.

22. **Synchronization Indicator.** The synchronization indicator is a vertical bar viewed through a small hole. Movement of the bar to one side or the other occurs in any of the following three alternative conditions:—

a. When magnetic slaving is taking place. (The mode selector can thus be in either the SLA or SYN position.)

b. When the set-heading control is being used.

c. When gyro drift correction is taking place, *ie*, when a correction for gyro drift (either apparent or real) has been set on the RATE COR dial.

The synchronization indicator is useful in showing the state of serviceability of the compass. In the SLA mode, in level flight, the indicator should normally oscillate slowly with low amplitude about the central point, whereas a steady deflection in this mode indicates that the compass is not properly synchronized. A perfectly steady central indication occurs during turns when magnetic slaving is automatically disconnected, but in steady flight such an indication could mean that there is a power failure in the system. In the DG mode the indicator bar should not oscillate. A steady deflection shows that a Rate Correction is being applied to the gyro heading output.

23. **Synchronization Warning Light.** The synchronization warning light comes on whether the set-heading switch is used, or when the mode selector is switched to SYN. Illumination of this light shows that the compass is not in its normal condition for navigational use.

Compass Coupler

24. The Compass Coupler, shown in Fig 5, contains the servo-amplifiers, servo-mechanisms and electronic components, required to combine the information given by the other three units. In addition it houses four synchro control transmitters which each pass the compass heading output to other equipments.

25. A mechanism for the correction of residual magnetic deviations is an integral part of the Compass Coupler. The window in a hinged cover on the front panel of the unit (Figs 5 and 8) shows the sectors of two

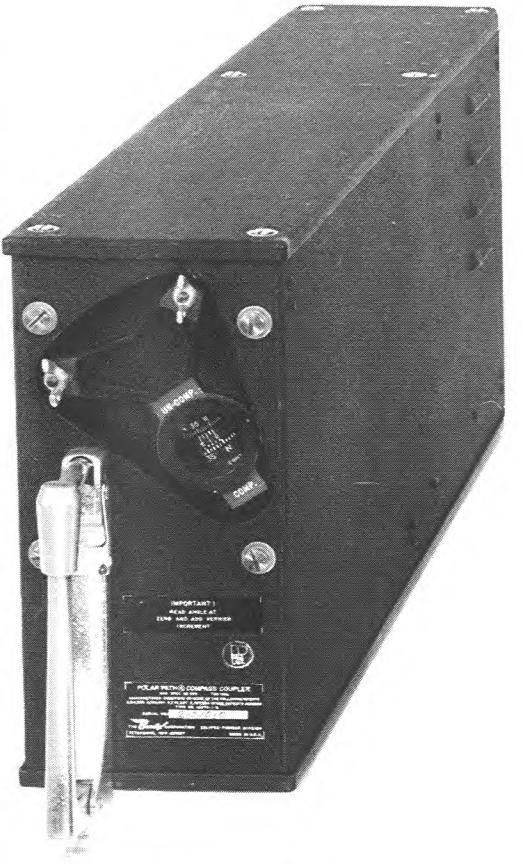


Fig 5 Compass Coupler

adjacent dials, graduated in increments of two degrees, with a vernier scale between them against which both dials are read. These dial readings can be compared to show the amount of residual deviation correction which has been inserted. Through the face of one of the dials is a hole through which a screw is visible at 30° intervals. There are 12 screws, used to make the residual deviation adjustments.

Compass Comparator

26. In a twin installation (Fig 10) the heading outputs of the two Compass Couplers are compared in a comparator circuit. With the compasses in the Slaved or DG mode, warning lamps at the pilots' positions are lit whenever a significant

difference exists between the two outputs. The value of the difference can be pre-set on the ground within the range of $2\frac{1}{2}°$ to 12°. In practice it is usually set at approximately 5°.

OPERATION OF THE SYSTEM

Introduction

27. In both modes of operation—DG and Slaved—azimuth gyro information is reproduced at the equipment heading outputs. In the DG mode the gyro output is first corrected for apparent and real gyro drift-rate, whereas in the Slaved mode it is synchronized with and slaved to the magnetic heading sensed by the fluxvalve. (It is emphasized that the magnetic slaving and corrections for gyro drift-rate are applied only to the gyro output; the azimuth alignment of the gyro itself is not corrected or controlled in any way). Transmission of gyro heading to the compass outputs by a basic servomechanism is common to both modes of operation.

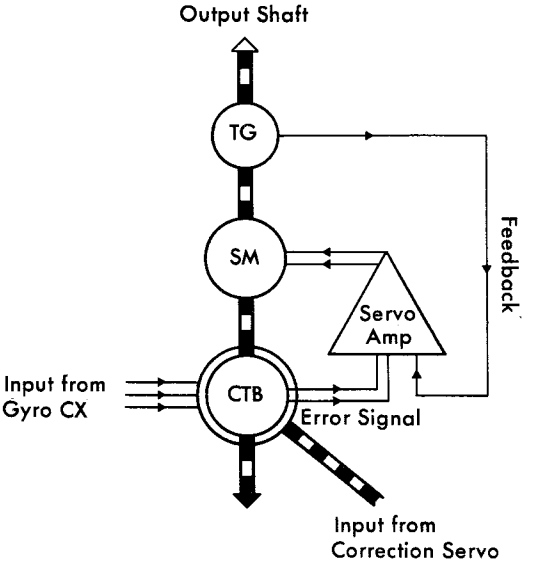


Fig 6 Basic Servomechanism

Basic Servomechanism

28. The basic servomechanism is illustrated in Fig 6. It is a position control servo, and is the means by which changes in heading are transmitted to the equipment heading outputs.

29. The error-detector is the CTB. When displaced from its electrical null position by a change of input to the stator (during a heading-change, for example), the CTB rotor transmits to the servo-amplifier a corresponding error signal whose phase depends on the direction of rotor displacement. The amplified error signal immediately activates the servomotor (SM) which turns in the direction necessary to turn the CTB rotor to its new null, and in so doing rotates the output shaft through an angle equal to the heading-change. The output of the servo loop is damped against oscillation by applying velocity feedback to the servo-amplifier from the tacho-generator (TG). (Servomechanisms and damping are discussed in detail in Part 3, Sect 1, Chap 2, of this Volume).

30. The angular movement of the output shaft represents a simultaneous combination of two inputs to the CTB stator. One of these inputs is applied electrically, the other mechanically. The electrical input always represents the actual gyro heading, *ie*, the orientation of the gyro spin axis relative to its outer gimbal. The mechanical input is used to modify this gyro heading representation by physically turning the stator, and is dependent on the selected mode of operation as follows:—

DG Mode. In the DG mode the mechanical input combines two functions: it is used to apply corrections for gyro drift-rate and is also the means by which the navigator can set the indicated gyro heading to any desired initial value.

Slaved Mode. The mechanical input to the CTB stator in the Slaved mode is magnetic heading information derived from the fluxvalve, and this modifies the gyro

heading output to make it agree with the aircraft's magnetic heading.

31. There are two servo loops in the Polar Path equipment: the output servo and the correction servo. The output servo has so far been called the basic servomechanism, and passes the modified gyro-heading as shaft rotation to the rotors of control transmitters (CXs, 2, 3, 4, 5) which are the equipment's heading outputs, shown in Fig 9 (Fig 9, which follows para 66 is used as the basis for the description of system operation and mechanical compensation for deviation.)

Correction Servomechanism

32. The correction servomechanism, as its name implies, is the loop through which the modifying corrections are applied (mechanically) to the CTB stator. In the Slaved mode the loop functions as a true closed-loop servomechanism, whose error-detector is the fluxvalve follow-up synchro CT. In the DG mode, however, no error-detector is used in the correction servo, which therefore operates purely as an open-loop system to apply gyro drift-rate correction or to enable the navigator to set his desired initial heading.

33. Between the correction servo and the CTB stator is a two-speed gear train. In the Slaved mode the low-speed gear is engaged, through which the maximum magnetic slaving rate is approximately 1.75° per minute. However, selection of SYN operates a gearchange solenoid to engage high-speed gear, causing the CTB stator to be turned rapidly (12° per second) until synchronization is complete. The high-speed gear is also engaged when in the DG or Slaved mode the Set-Heading switch is turned to any of its active positions. With the operation of this switch there are two alternative inputs to the correction servo-amplifier. The combination of amplifier input and high-speed gear enables the CTB stator to be slewed either at 12° per second, with the SET switch in either of the outer positions, or at 3° per second when either of the inner positions is selected.

34. **Damping of Correction Servomechanism.** The type of damping applied to the correction servo to prevent oscillation depends on the selected mode of compass operation:—

DG Mode. In the DG mode the correction servo transmits a continuous, steady rotation to the CTB stator, dependent on the Rate Correction dial setting, to correct for gyro drift-rate. The correction servo in this mode operates as a speed controlled servo, using the velodyne principle (*see* Part 3, Sect 1, Chap 2 of this volume) to maintain the required constant output speed of the correction servomotor.

Slaved Mode. The function of the correction servo in the Slaved mode is to maintain synchronism, at the CTB, between the compass heading output and the sensed magnetic heading. The correction-rate required will usually be small but will not necessarily be constant or continuous. Damping is therefore applied by passing the loop output through the low-speed (*ie* high-ratio) gear between the correction servomotor and the CTB stator. In this mode no velocity-feedback damping signal is fed from the tacho-generator to the correction servo-amplifier, and the correction servo thus operates as a step or "bang-bang" system. This means that the presence of an error-signal, whatever its size, causes the nulling correction to be applied at an almost constant rate. Consequently, hang-off error (*see* Chap 1, para 82 of this Section) is insignificant in this compass system.

Synchronize Function. When it is desired to synchronize the compass, the mode selector is turned to SYN, and the correction servo then provides an automatic rapid-slaving facility. A velocity-feedback damping signal is therefore fed from the tacho-generator to the correction servo-amplifier to prevent violent overswing as the synchronization point is reached.

DG Mode of Operation

35. The method by which gyro heading is picked off from the synchro CX 1 on the

gyro frame and transmitted *via* the output servo to the system heading outputs has already been described.

36. **Initial Heading.** After the gyro has been switched on it will erect to the horizontal and will indicate a random heading. In the DG mode the navigator can now use the Set-Heading control on the Compass Controller to set his desired initial heading. When the control is moved off its centre position to increase or decrease the indicated reading a correction signal is sent to the correction servo, whose motor then rotates in the appropriate direction and, through the high-speed gearing, turns the stator of the CTB. This activates the output servo which transmits the correction to the heading outputs until the desired heading is indicated. During the set-heading process the synchronization warning lamp is lit, the synchronization indicator bar on the Compass Controller deflects either to the right, for increasing heading, or to the left, for decreasing heading, and the heading output to the auto-pilot is disconnected.

37. **Apparent Drift Correction.** In the DG mode the effect of apparent gyro drift-rate can be offset by applying an equal angular movement to the CTB stator *via* the correction servo. When the navigator has calculated the necessary drift correction, he makes the appropriate setting on the RATE COR dial. This positions the wiper of a drift potentiometer which remotely controls the speed of rotation of the correction servomotor. Through the low-speed gear the motor then slowly turns the stator of the CTB at the required rate. The synchronization indicator bar deflects unless zero correction has been selected.

38. **Real Drift Correction.** Real gyro drift occurs because of physical limitations in design and manufacture, and is directly caused by such characteristics as friction and mass unbalance. The gyro in the Polar Path compass has a real-drift rate of about 1.5° per hour. However, once the value and sense of any constant component of this drift have been determined for an individual gyro, a correction can be applied from the RATE COR dial. In effect a constant correction is introduced, as in para 37 above. The centre knob of the dial has a small removable threaded cap which covers a mechanism for locking the dial on its shaft. The dial can be loosened and the shaft rotated to apply the required compensation. There are graduations on the small scale at the bottom of the dial (Fig 4); each graduation represents a drift correction of $\frac{1}{2}$ degree/hour. The dial is then replaced on its shaft in such a position that the rate indication is still correct. Once again, the effect is that the stator of the CTB is turned at the same rate and in the same sense as the gyro is drifting, thus cancelling out the effect of the drift.

Magnetically Slaved Mode

39. The fundamental difference between the DG mode and the magnetically-slaved mode is in the origin of the corrections applied to the gyro heading output at the CTB stator. In the DG mode the gyro output was aligned arbitrarily by the navigator when he set the initial heading. And alignment was maintained by applying manually-selected corrections for gyro drift-rate. In the Slaved mode the gyro output is initially synchronized with magnetic heading, and the effect of subsequent gyro drift is offset by continuously correcting the gyro output to agree with the fluxvalve heading. (No other drift correction is applied in this mode, and the RATE COR dial is inoperative.) The fluxvalve is of course subject to magnetic deviation, and mechanical compensation can be made to offset the effect of residual deviations. However, the compensation mechanism, which uses a cam compensator and two associated dials, can be temporarily ignored, since it is described later, in paras 53–58. In the meantime, the reader is asked to assume that the rotation of the drive shaft from the output servo is applied without correction to the rotor of the fluxvalve follow-up synchro CT (Fig 9).

Transmission of Fluxvalve Information

40. Fluxvalve heading information is applied to the gyro heading output *via* the correction servo which functions in the Slaved mode as a true closed-loop servo-mechanism. The error-detector is the flux-valve follow-up synchro CT.

41. The voltages induced in the fluxvalve detector coils represent magnetic heading and are reproduced in the CT stator. When displaced from its null position, the CT rotor transmits an error signal *via* amplifiers to the correction servomotor which in turn, through the low-speed gear, turns the CTB stator, thus activating the output servo. The consequent rotation of the heading-output shaft repositions the CT rotor to reduce the original error signal, driving the compass heading outputs into synchronism with the electrical input of magnetic heading to the CT stator. The correction servo and output servo are therefore interconnected, and will remain active until the error signals at the CT and CTB return to zero. At that time the compass heading output is synchronized with magnetic heading. The response of the system to change in input at the CTB is very rapid, since there is a direct drive through the output servomotor. However, the rate of response to the magnetic slaving output of the correction servo is very much slower since this drive passes through the low-speed gear before reaching the CTB stator. This gear-train determines the maximum slaving-rate of approximately 1.75° per minute.

42. **Effect of Gyro Drift.** In the Slaved mode no corrections for real or apparent drift are applied to the gyro output. Therefore when the gyro drifts the signals at the CTB stator change, an error signal is induced in the rotor and the output servo-motor turns to drive the rotor to a new null. In so doing, it also drives the rotor of the fluxvalve CT out of its null position, causing an error signal to be induced in this rotor. This signal activates the correction servo which turns, through the gearing, the stator of the CTB. The movement continues until both rotors are again nulled and gyro information is re-synchronized with magnetic heading.

43. **Operation During Aircraft Turn.** During an aircraft turn the output servo, activated by signals from the gyro CTB rotor, moves the heading output shaft and the rotor of the fluxvalve CT through the amount of the heading change. When the aircraft steadies at the end of the turn, the latter rotor will still be in its null position since it will have turned through the same angle and at the same rate as the magnetic meridian relative to the aircraft. Thus the heading outputs will still be sychronized with the magnetic heading input.

44. **Magnetic Turning Error.** During a turn, however, the detector is tilted and senses a false magnetic heading, giving rise to magnetic turning error. A cut-out switch is therefore incorporated to disconnect the magnetic slaving whenever the aircraft's rate of turn exceeds 15° per minute (30° per minute after modification). With a rate of turn less than this figure, magnetic slaving continues. Because of the turning error the gyro heading output and the sensed magnetic heading will be de-synchronized during the turn, and the consequent error signals at the fluxvalve CT will activate the correction servo which will slave the system to the false heading. The source of the turning error will be removed when the aircraft steadies on the new heading, and since the slaving rate is relatively slow, only a small amount of turning error will have reached the heading outputs during the turn. When the aircraft continues in straight and level flight, the assimilated error would be subsequently removed by the normal slaving action of the correction servo, though the system should be re-synchronized manually after a long slow turn.

45. **Synchronization.** To synchronize the system initially, the low-speed gear is by-passed by selecting SYN on the mode selector. The gyro output is then syn-

chronized with the magnetic heading at a rate of 12° per second. The processes of synchronization and slaving are similar except for the speed at which they take place. When SYN is selected the synchronization warning lamp is lit, and the heading output to the autopilot is disconnected.

46. **Synchronization Indicator.** When the system is de-synchronized the bar on the synchronization indicator will be displaced. A steady displacement will occur in the initial stage of rapid synchronization or when magnetic turning error is present during slow turns.

DEVIATION CORRECTION

Introduction

47. It has been assumed so far that the detector unit in level flight always senses the correct magnetic heading. In practice the earth's magnetic field in the vicinity of the detector unit will be distorted by permanent and induced aircraft magnetism. This distortion, together with system transmission errors, causes compass deviations. A combination of magnetic and mechanical compensation can be used to correct for all these errors.

48. **Magnetic Compensation.** If the distortion of the earth's field at the detector position is likely to be significant, a magnetic compensator is fitted above the detector unit, as a means of providing equal and opposite distortion. The compensator, described in para 16, allows corrections to be made for deviation coefficients B and C. If no magnetic compensator is fitted, all deviation corrections are applied mechanically, as described below.

49. **Mechanical Compensation.** Correction for deviation coefficient A is applied by rotating the detector unit at its mounting through the required angle.

50. After correction has been made for coefficients A, B and C (if applicable), residual deviations can be plotted as a deviation curve in the conventional way. However, before the correction of residual deviations is described, the effect of such deviations on the compass system should be clearly understood.

51. **Effect of Deviation.** When the aircraft changes heading the gyro senses correctly the heading change and transmits it through the output servo as a shaft rotation to the rotors of the heading output CXs and to the fluxvalve CT rotor. The heading change is sensed independently by the fluxvalve, but is in error by the amount of deviation at the fluxvalve on the new heading, and is passed to the CT stator. The deviation thus causes a discrepancy between the CT stator and rotor, and the consequent error signal starts to slave the system to the erroneous fluxvalve heading.

52. **Correction Principle.** The unwanted slaving can be avoided if, when the heading changes, the output servo rotation to the CT rotor is corrected for deviation, so that the rotor remains nulled, with no resulting error signal. To achieve this, an adjustable cam compensator is incorporated between the heading-output drive shaft and the CT rotor. The effect of the cam is to introduce a differential between the motion of the drive shaft and the rotor of the CT. This causes small unequal angular movements of the two at different points in their rotation. Thus when the CT rotor is in its null position, representing the deviated heading sensed by the detector unit, the system outputs nevertheless indicate correct magnetic heading.

53. **Cam Compensation Mechanism.** The heart of the compensation mechanism is a static cam. Made of flexible metal, the cam is a flat circular disc whose contour can be adjusted by 12 screws which impinge against the face of the disc at 30° intervals round its circumference. Deviation corrections can thus be made on 12 headings (030°, 060°, 090°, *etc*).

54. **Cam Follower.** A cam-follower roller, spring-loaded to maintain contact with the face-contour of the cam, is mounted on the output servo drive shaft. As the drive shaft turns, the cam follower rides round the circumference of the fixed cam. Where the cam contour has been altered by screw adjustment to compensate for deviation, the cam-follower rises or falls. This movement is added algebraically to the rotation of the output servo drive shaft, and the total rotation is then applied to the fluxvalve CT rotor.

55. **Associated Dials.** Two heading dials are associated with the mechanism: the compensated dial and the uncompensated dial. The compensated dial, driven directly from the output servo drive shaft, shows the system heading output. The uncompensated dial, mounted on the drive to the CT rotor, indicates fluxvalve heading, provided the rotor is nulled.

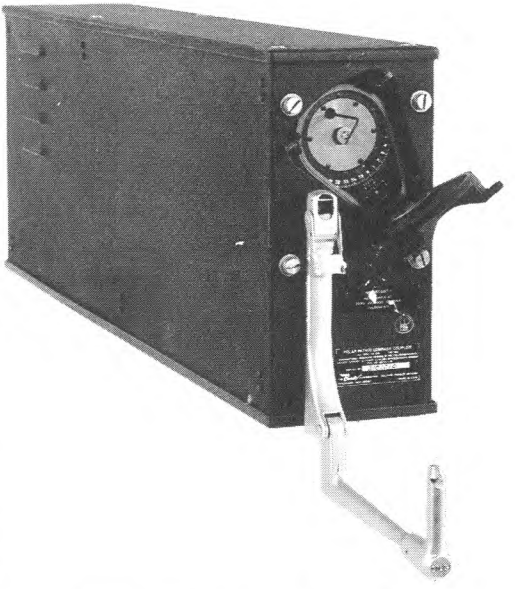


Fig 7 Compass Coupler Showing Dials

56. **Effect of Deviation Correction.** The face of each dial is a rotating card, and the dial headings are read against a common vernier scale, as shown in Fig 7. When the uncompensated dial reads any one of the 12 adjustment-headings, a hole in the face of the dial reveals the appropriate adjustable screw for that particular heading. Turning the screw with an Allen key (shown in position in Fig 7) alters the cam contour at that point and causes the cam follower to rise or fall. This movement turns both the uncompensated dial and the CT rotor, while the compensated dial remains stationary in the DG mode. The CT rotor has thus been deliberately turned from its null position, but in the Slaved mode the servo action will then operate to null the rotor again, and thus return the uncompensated dial to its original position. In so doing, it will reposition the compensated dial, by the same amount, to the correct magnetic heading.

57. **Direction of Adjustment.** The direction in which the deviation correction must be applied is explained in the following example. Assume that an aircraft with an uncorrected compass is on a heading of 180° (M), and that it is necessary to correct for a calculated deviation of +2° (*ie* 2° east). Both dials read 178° before any correction is made. To make the correction, the sign of the deviation must be reversed, and the adjustment screw turned to make the uncompensated dial read 176° temporarily. As explained earlier, subsequent servo action in the Slaved mode will drive the uncompensated dial back to its original reading of 178° (C), and in so doing will turn the compensated dial and the heading outputs on to the correct reading of 180° (M).

58. **Reading the Dials.** The graduations on both dials represent two degrees. Intermediate values can be read to better than ½° by means of the vernier scale. On either dial, heading is read as the angle indicated by the first graduation to the left of the vernier zero index. To this, add the vernier increment, which is determined by the vernier graduation mark that most closely lines up with any graduation mark on the dial. An example of the compensated (lower) and

uncompensated (upper) dial readings is given in Fig 8.

Compass Swings

59. The compass system can be calibrated by calculating coefficients A, B and C, and applying corrections for them at the detector unit and magnetic compensator. Further corrections, for residual deviations, can then be applied at the cam compensation screws in the Compass Coupler. A size 4 Allen key is required for turning the screws. (If no magnetic compensator is fitted, coefficients B and C will be included in the residual deviation corrections, and the specific references to coefficients B and C in para 60 can be ignored.)

60. **Swinging Procedure.** The general procedure for swinging is as follows:—

a. When the power supply is connected, switch on the compass system.

b. Ensure that the ac supply voltage is correct (115 v at 400 c/s).

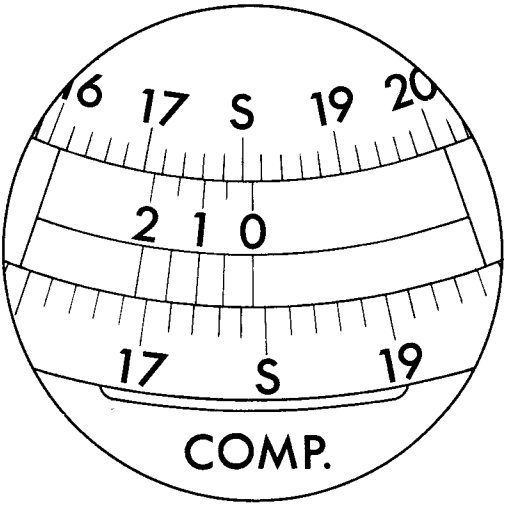
c. Carry out one or more check swings, comparing the datum heading with that of the uncompensated dial, to calculate coefficients A, B and C, and reduce them to less than 1° by correcting at the detector unit (coefficient A) and magnetic compensator (B and C coefficients).

d. Carry out a calibration-and-adjustment swing on 12 headings as follows:—

(i) On each heading, ensure that the system is synchronized, by selecting SYN until the synchronization indicator is centred.

(ii) Calculate the deviation and apply any necessary correction at the cam compensator. The method of determining and applying this correction is described in paras 61–63 below.

(iii) Re-synchronize the system, and record the readings of the compensated and uncompensated dials before proceeding to the next heading.



Uncompensated (upper) dial reading to left of vernier zero index	180°
Vernier increment (more than 1° and less than 1½°)	1° to 1½°
	181° to 181½°
Compensated dial reading to left of vernier zero index	178°
Vernier increment (more than 1° and less than 1½°)	1° to 1½°
	179° to 179½°

Fig 8 Instructions for Reading Dials

At the end of the swing the results permit two curves to be calculated. One, being a deviation curve, shows the difference between datum and uncompensated dial headings; the other, a corrected curve, compares the datum and compensated dial headings, and is thus a measure of the deviations to be expected at the compass outputs.

61. **Cam Screw Adjustments.** The method of applying residual deviation corrections at the cam screws is as follows:—

a. Ensure that the appropriate rate correction for local latitude is set on the RATE COR dial.

b. Calculate the correction to be applied. It must be remembered that when an adjustment screw is turned, this displaces

the uncompensated dial initially; the correction must therefore be calculated for application to the uncompensated dial such that during subsequent re-synchronization the compensated dial will be driven to the required magnetic heading. To determine the readings to which the uncompensated dial must be adjusted, subtract algebraically the deviation from the present readings of the compensated dial.

Example

(*Note.* In this illustration the initial difference between compensated and uncompensated dial readings is assumed to result from a correction applied during the previous compass swing.)

Hdg (M)	*Uncomp Dial*	*Deviation*	*Comp Dial Before Correction*	*Make Uncomp Dial Read*
090	092	−2	091	091 − (−2) = 093

c. Select DG.

d. Using a size 4 Allen key, turn the adjusting screw, visible through the face of the uncompensated dial, to make this dial give the desired reading. The compensated dial remains stationary in the DG mode.

REMOVE THE ALLEN KEY.

e. Select SYN, to ensure that when the synchronization indicator has centred, the uncompensated dial returns to its original reading (092 in the above example), and that the compensated dial has been driven to the desired magnetic heading.

62. **Adjustment Limitations.** On any heading the amount of screw adjustment applied to the cam must be within 2° of that made on the previous heading. This limitation must be carefully observed, since any greater adjustment can buckle the cam.

OPERATING INSTRUCTIONS

Introduction

63. The details below are a general guide for switching-on, preparing the compass for use, and operation in flight. They are not intended to form a comprehensive check-list, nor to take the place of any standard operating procedures specified by Command.

Before Flight

64. *a.* Switch on the compass and gyro power supplies, and allow one minute for warm-up.

b. Set the RATE COR dial to zero.

c. Turn the mode selector to DG.

d. Hold the SET knob to each of the Increase positions in turn, and check that:—

(i) The synchronization warning lamp is lit during each selection.

(ii) The synchronization indicator deflects to the right.

(iii) At the first selection, the navigator's RMI heading increases at 3° per second; when the second position is selected, the rate of heading-change increases to 12° per second.

e. Release the SET knob, which should automatically return to its centre-position, and check that:—

(i) The synchronization warning lamp goes out.

(ii) The synchronization indicator returns to centre.

(iii) RMI heading-change ceases.

f. Repeat steps *d* and *e* for each of the Decrease Positions of the SET knob.

g. Move the RATE COR dial to −30°. As the dial is turned, the synchronization indicator should move towards "S".

h. Repeat step *g* for "+" movement of the RATE COR dial. The synchronization indicator should deflect towards "N". Return the dial to zero.

j. Hold the mode selector to SYN, and check that:—

(i) The synchronization warning lamp is lit.

(ii) The synchronization indicator deflects, returning to centre as synchronization is achieved.

(iii) The system synchronizes at 12° per second, and RMI reads magnetic heading when synchronization is complete.

k. Release the mode selector switch, which should return automatically to SLA, and check that:—

(i) The synchronization warning lamp goes out.

(ii) RMI heading remains steady.

The synchronization indicator may start to oscillate slightly about its central position.

l. Turn the SET knob clockwise to desynchorinze the RMI heading indication by approximately 10°, then release the knob.

Check that—

(i) The synchronization indicator deflects steadily to the right.

(ii) The RMI heading should start to return slowly to the magnetic heading, at just under 2° per minute (1.75° approx).

(iii) In the twin compass installation, with the second compass still synchronized in the Slaved mode, the comparator warning lamp should light.

m. Turn the SET knob anti-clockwise to resynchronize, and continue until the RMI heading is displaced by approximately 10° in the opposite direction, then release knob. Check that the slaving action, indicated by step (1)(i), (ii) and (iii) above, is repeated in the appropriate sense.

n. Re-synchronize the compass, and select the desired mode of operation for flight. For the Slaved mode, set the appropriate variation on the Automatic Variation Computer Unit; if the DG mode is selected, set the appropriate drift correction on the RATE COR dial.

In Flight

65. *a. Slaved Mode.* In the Slaved mode the synchronization indicator should oscillate slowly about a central point when a steady-heading is flown. A steady deflection to either side means that the compass is de-synchronized, whereas a steady central indication, except during turns, could be symptomatic of a power failure. In the twin installation, the compass comparator warning lights provide a further check on compass serviceability.

b. DG Mode. In the DG mode, when the RATE COR dial is set to anything other than zero, the synchronization indicator should show a steady deflection in the appropriate sense. the RATE COR dial should be corrected periodically for changing values of earth-rate and, if applicable, transport-rate gyro drift. Any constant component of real gyro drift found during flight can also be included, if desired, in the total rate set on this dial.

66. **Corrections for Apparent Gyro Drift.** The corrections to be applied for apparent gyro drift can be calculated, or extracted from appropriate graphs. Theoretical expressions for the two drift components are:—

a. Earth-rate = 15.04° × sin latitude (degrees/hour);

b. Transport-rate = $57.3 \times \frac{U}{R} \times \tan$ lat. (degrees/hour),

where U is the easterly component of ground-speed, in knots, and R is the earth's radius, in nautical miles.

Practical approximations for this context, however, are:—

c. Earth-rate = 15° × sin latitude (degrees/hour);

d. Transport-rate = $\frac{U}{60} \times \tan$ latitude (degrees/hour).

The drift-rate correction figures on the RATE COR dial are annotated "+" or

"−". The "+" selection on the dial corrects for:

e. Earth-rate in the Northern Hemisphere.

f. Transport-rate with easterly ground-speed in the Northern Hemisphere.

g. Transport-rate with westerly ground-speed in the Southern Hemisphere.

The "−" selection corrects for:—

h. Earth-rate in the Southern Hemisphere.

j. Transport-rate with easterly ground-speed in the Southern Hemisphere.

k. Transport-rate with westerly ground-speed in the Northern Hemisphere.

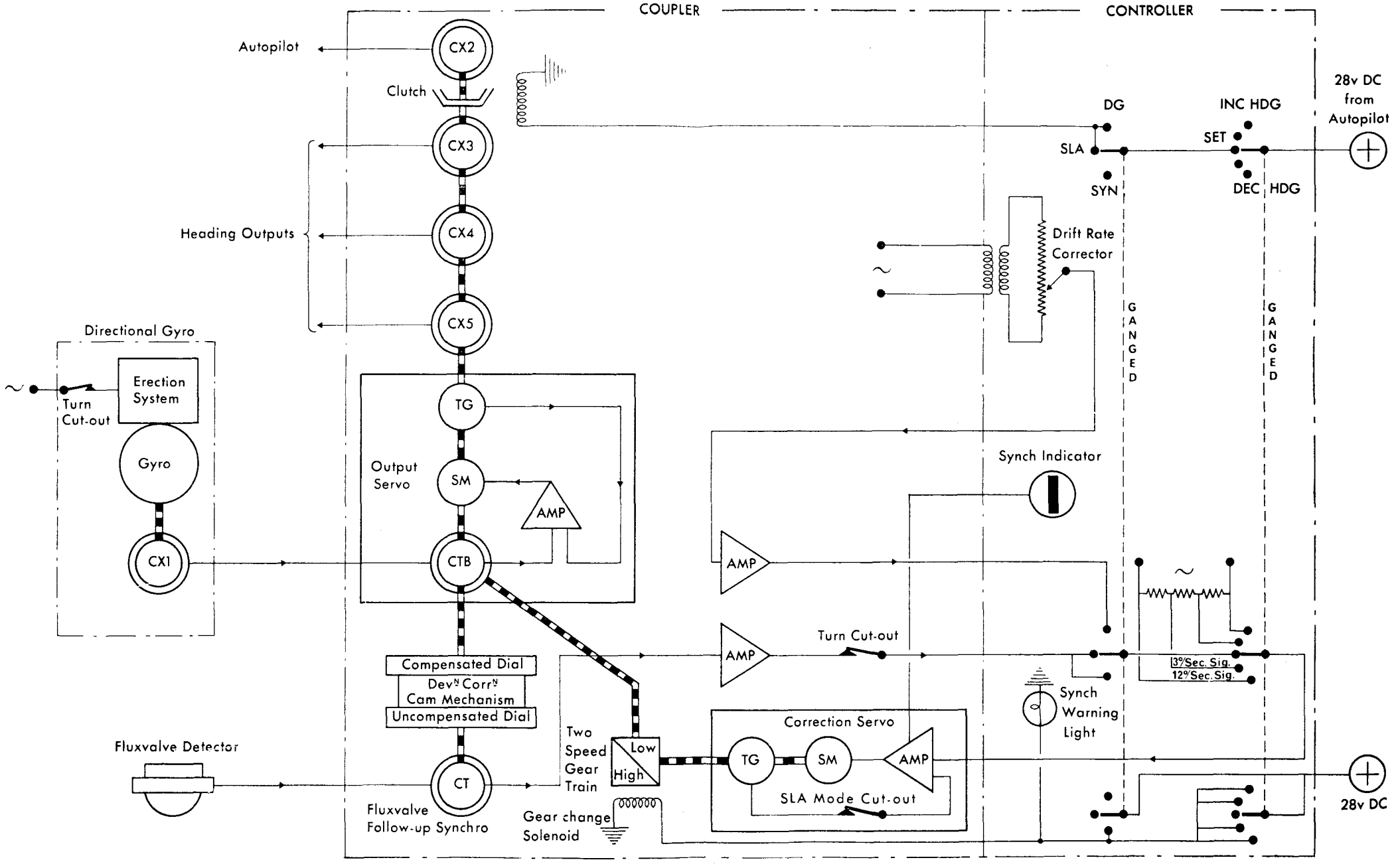


Fig 9 Functional Schematic Diagram

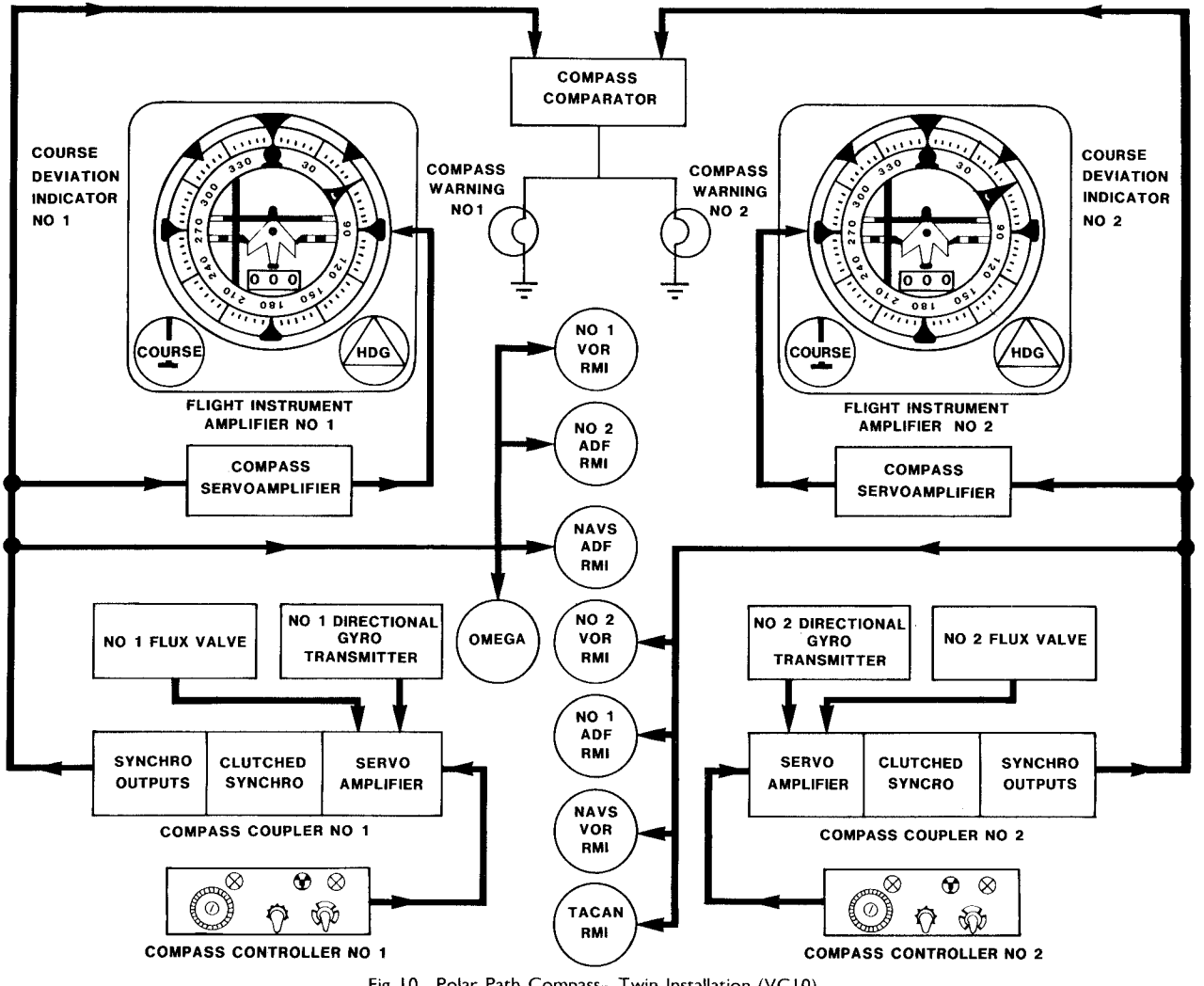


Fig 10 Polar Path Compass—Twin Installation (VC10)

CHAPTER 8

C2G AND C2J (GM8A) GYRO-MAGNETIC COMPASS SYSTEMS

CONTENTS

	Paras
Introduction	1
C2G GYRO-MAGNETIC COMPASS	
General	2
System Components	3
System Description	4-9
System Operation	10-13
C2J (GM8A) GYRO-MAGNETIC COMPASS	
General	14
System Components	15
System Description	16-18
System Operation	19

Introduction

1. The C2G and C2J (GM8A) Gyro-Magnetic Compass Systems combine the function of a directional gyro and a magnetic compass to provide a magnetically slaved gyro heading output. The two compass systems are very similar, the C2J system providing a total of four magnetic heading outputs compared with one output from the C2G system.

C2G GYRO-MAGNETIC COMPASS

General

2. The magnetic heading output from the C2G is fed to the Navigation Display and Computer (NDC) which forms part of the FE541 inertial navigation and weapon aiming system. It is used during the initial stages of the inertial platform alignment, for monitoring the inertial platform heading after alignment and for reversionary use in the event of a failure of the inertial navigation system.

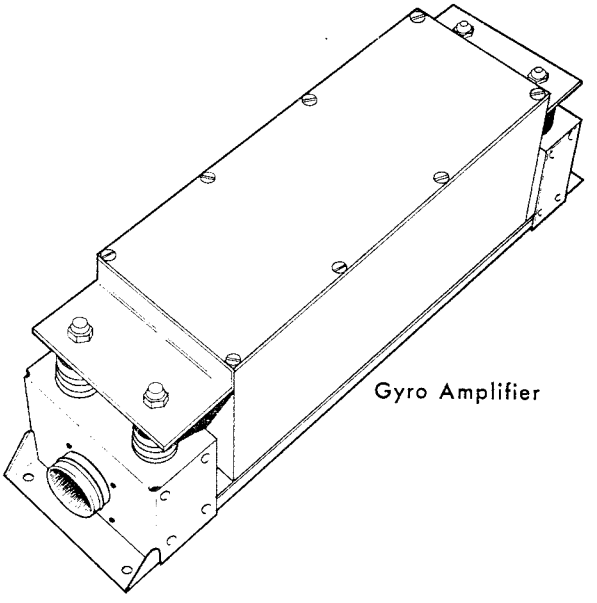
System Components

3. The C2G system consists of the following four basic units as shown in Fig 1:

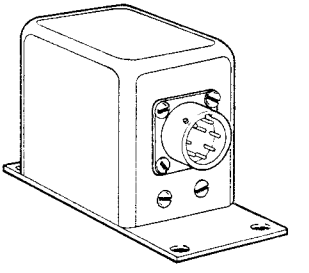
a. *Transformer Power Converter.* The transformer power converter provides the necessary conversion from the 200/115 V ± 2 V, 400 Hz ± 20 Hz 4-wire, 3-phase AC to 115 V ± 5 V, 400 Hz, 3-wire, 3-phase AC required by the gyro-amplifier unit.

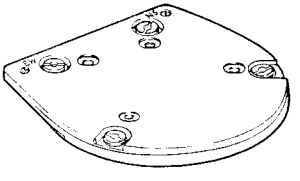
b. *Gyro-Amplifier Unit.* The gyro-amplifier unit provides the magnetically slaved gyro heading reference for the compass system. It consists of an electrically driven gyro, a slaving amplifier and two synchros. The gyro takes approximately five minutes to run up to 22,000 rpm. The system may be synchronized one minute after start-up. The gyro drift rate is ± 10° per hour exclusive of Earth rate (depending on conditions of measurement). The salving amplifier amplifies the error signal generated by the comparison of the flux valve heading and the gyro heading and applies it to the gyro precession coils to slave the gyro to magnetic North at a rate of 3° to 6° per minute exclusive of Earth rate. The time constant of the slaving amplifier is approximately 1 minute in an Earth's horizontal field strength of 0·18 oersteds.

c. *Detector Unit.* The detector unit contains a thin flux valve and a compensator unit. The flux valve consists of a

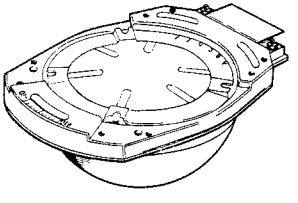


Transformer Power Converter





Compass Controller



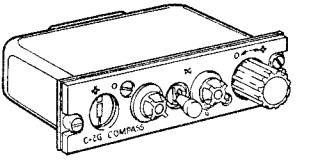


Fig 1 C2G System Components

fluid-damped, pendulous, 3-coil sensing element contained in a hemispherical case. The sensing element is pendulously suspended by a universal joint to enable it to sense the horizontal component of the Earth's magnetic field. Gimbal freedom allows ± 30° of pendulous movement prior to reaching the physical limits. There is an azimuth scale on top of the flux valve which is used as a guide to unit orientation when a correction to coefficient A is inserted. The compensator unit is mounted on top of the flux valve and is used to take out coefficient B and C errors during a compass swing.

d. *Compass Controller.* The compass controller provides the means of controlling the mode of operation and synchronization of the compass system. The

following controls and indicators are on the front panel of the controller:

(1) MAG/DG switch.

(2) Annunciator indicator.

(3) Synchronizing knob.

System Description

4. The gyro-amplifier is the principal unit of the compass system in which the gyro possesses freedom about two axes; limited freedom about a horizontal axis at right angles to the spin axis and complete freedom about a vertical axis. Spurious torques about the vertical axis of the gyro may cause the gyro rotor to precess from the horizontal plane. However this action is continuously overcome by a levelling system. This consists of a torque motor and a levelling switch. The levelling switch detects the direction of tilt and actuates the torque motor, which in turn precesses the gyro spin axis.

5. The gyro may be used to provide heading information, either as a directional gyro or slaved to magnetic North as sensed by the flux valve. The error signal developed by comparison of flux valve heading and gyro heading is amplified and phase detected by the slaving amplifier to provide a long-term reference for the gyro. Thus the compass system combines the long term stability of the Earth's magnetic field with the short-term stability of the directional gyro to give an accurate and reliable reference.

6. The synchro assembly comprises two synchros; a flux valve control transformer and a heading transmitter. The heading transmitter transmits the azimuth position of the gyro through a differential synchro contained in the compass controller to the navigation display and computer (NDC). C2G information is only displayed on the NDC's range and bearing indicator when the system is in the pre-align or in Reversionary Modes 1 or 3. At all other times unless Mag or Tac are selected, inertial heading will be displayed. The flux valve control transformer provides a signal voltage that is proportional to the degree of misalignment between the flux valve and the gyro. This error signal is fed to the slaving amplifier.

7. The slaving amplifier amplifies the error signal, performs phase detection and produces a DC output to the gyro precession coils and the annunciator contained in the compass controller. The precession coils are mounted symmetrically on either side of the vertical plane through the gyro spin axis and the current flowing in the coils creates a torque to correct system misalignment, thus precessing the gyro about the vertical axis.

8. The compass controller embodies the MAG/DG switch as shown in Fig 2. This enables the operator to select either the free gyro or slaved mode of operation. When in

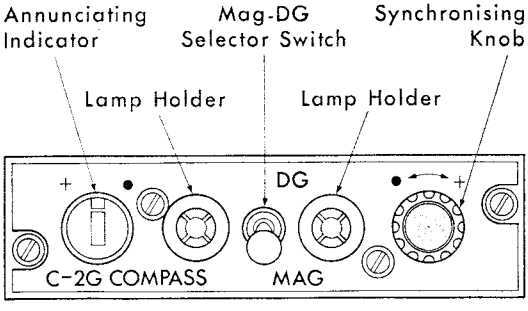


Fig 2 The Compass Controller

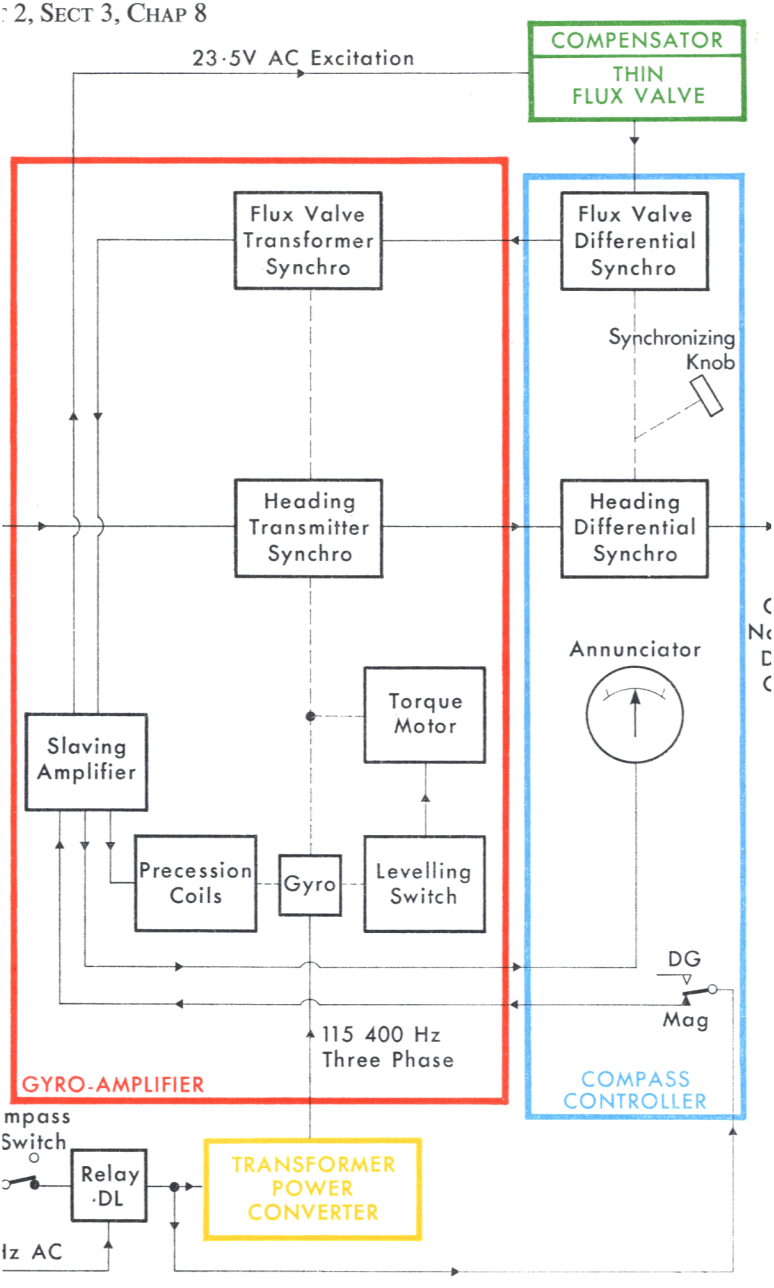


Fig 3 C2G Block Diagram

MAG mode, an annunciator, face of the compass con- s the direction and amount t for small errors between ier heading output and the signal. Misalignment is in- dicated by the annunciator pointer towards the dot (·) or cross (+) d on the sense of direction. To effect synchronization of the compass sy compass controller carries a synch control knob, which is connected

common rotor shaft of two ganged differential synchros, the flux valve differential synchro and heading differential synchro. The synchronizing control knob is turned in the direction of the arrows engraved on the panel depending on the direction of misalignment by the annunciator.

9. The flux valve detects the direction of the horizontal component of the lines of force of the Earth's magnetic field and transmits this information electrically to the flux valve differential synchro in the compass controller to an accuracy of $\pm$ 0·4°. The flux valve consists of a pendulous element which is mounted in damping fluid. The compensator, which consists of two pairs of permanent bar magnets, is mounted on top of the flux valve. One pair of magnets is located in the fore-and-aft line of the unit and the other pair in the athwartship line. Each pair of magnets is geared together and their positions relative to one another may be adjusted by two screws. The screws are designated NS and EW as applicable and are used to adjust the direction and magnitude of the correction along the relevant axis.

System Operation

10. Fig 3 shows the compass system in block diagram form. With the system in the DG mode the MAG/DG switch open circuits the power supply to the slaving amplifier in the gyro-amplifier unit, thus rendering the amplifier and the associated slaving loop inoperative. Any change in heading sensed by the gyro is transmitted by the heading transmitter synchro to the heading differential synchro in the compass controller. The synchronizing knob can be used to set any desired grid or gyro heading in the DG mode. The signal induced in the heading differential synchro rotor is fed to the NDC to provide heading information.

11. In the slaved (MAG) mode of operation the power supply to the slaving amplifier is restored and the slaving loop is able to function. The flux valve senses the aircraft's magnetic heading and transmits this information to the stator of the flux valve differential synchro in the compass controller. The loop error voltage induced in the flux valve differential synchro rotor is applied to the slaving amplifier by way of the flux valve control transformer synchro. The error signal is then amplified and its phase angle detected by the slaving amplifier, the output of which is a DC differential current. This differential current is applied to the gyro precession coils to effect a correction of the induced error. The rate of correction (*ie* precession) is determined by the difference of the currents flowing in the two precession coils. The precession rate of the gyro in azimuth is proportional to the degree of system misalignment up to saturation, and the torque produced by the currents flowing through the precession coils precesses the gyro so that the gyro heading output is synchronized with the Earth's magnetic field. Since the rotors of both the heading transmitter synchro and flux valve control transformer synchro are positioned by movement of the gyro about its vertical axis, then, as the gyro precesses in azimuth, so the rotors are turned in the required sense to reduce the error voltage input to the slaving amplifier until the flux valve control transformer synchro rotor is brought to its null position. The transmitted heading information from the heading differential synchro now corresponds with the aircraft's magnetic heading.

12. The annunciator in the compass controller also receives the DC current output from the slaving amplifier and indicates whether or not the compass system is synchronized. The system may be quickly synchronized by turning the synchronizing control knob in the direction indicated by the arrows and the annunciator pointer. Rotation of the synchronizing control knob synchronizes the heading output to the magnetic heading sensed by the flux valve.

13. A compass master switch, designated MAG COMPASS ON, energizes relay DL when selected ON. A circuit is completed allowing AC to the transformer power converter and to the MAG/DG switch in the compass controller.

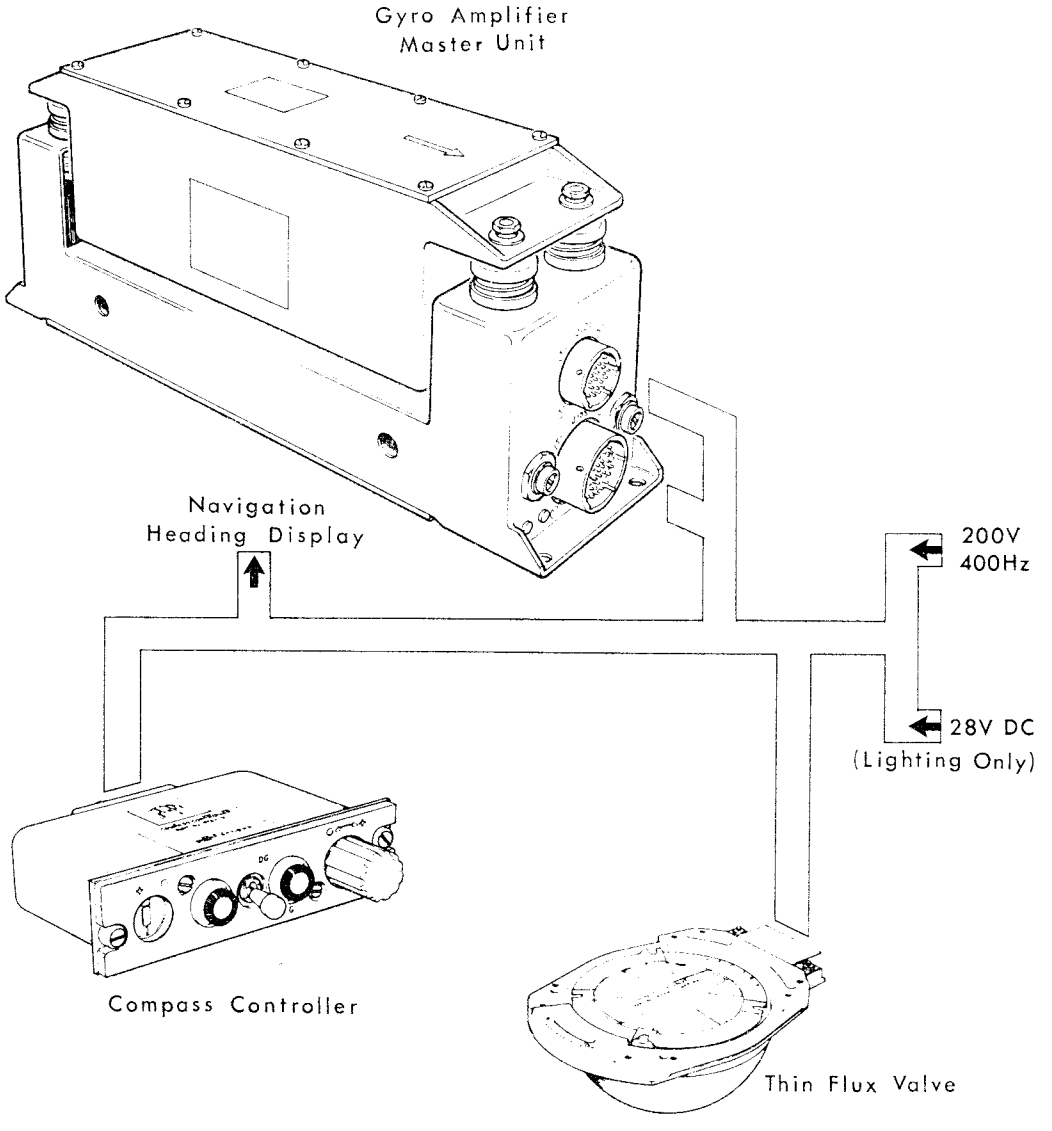


Fig 4 C2J System Components

C2J (GM8A) GYRO-MAGNETIC COMPASS

General

14. The C2J gyro-magnetic compass is basically the same as the C2G but there are two important differences. There is no compensator mounted on the flux valve of the C2J, and compensations for coefficients B and C are made via a remote corrector unit which is integral with the gyro-amplifier master unit. The gyro-amplifier master unit of the C2J employs the same gyro-amplifier as the C2G, mounted on a new cradle

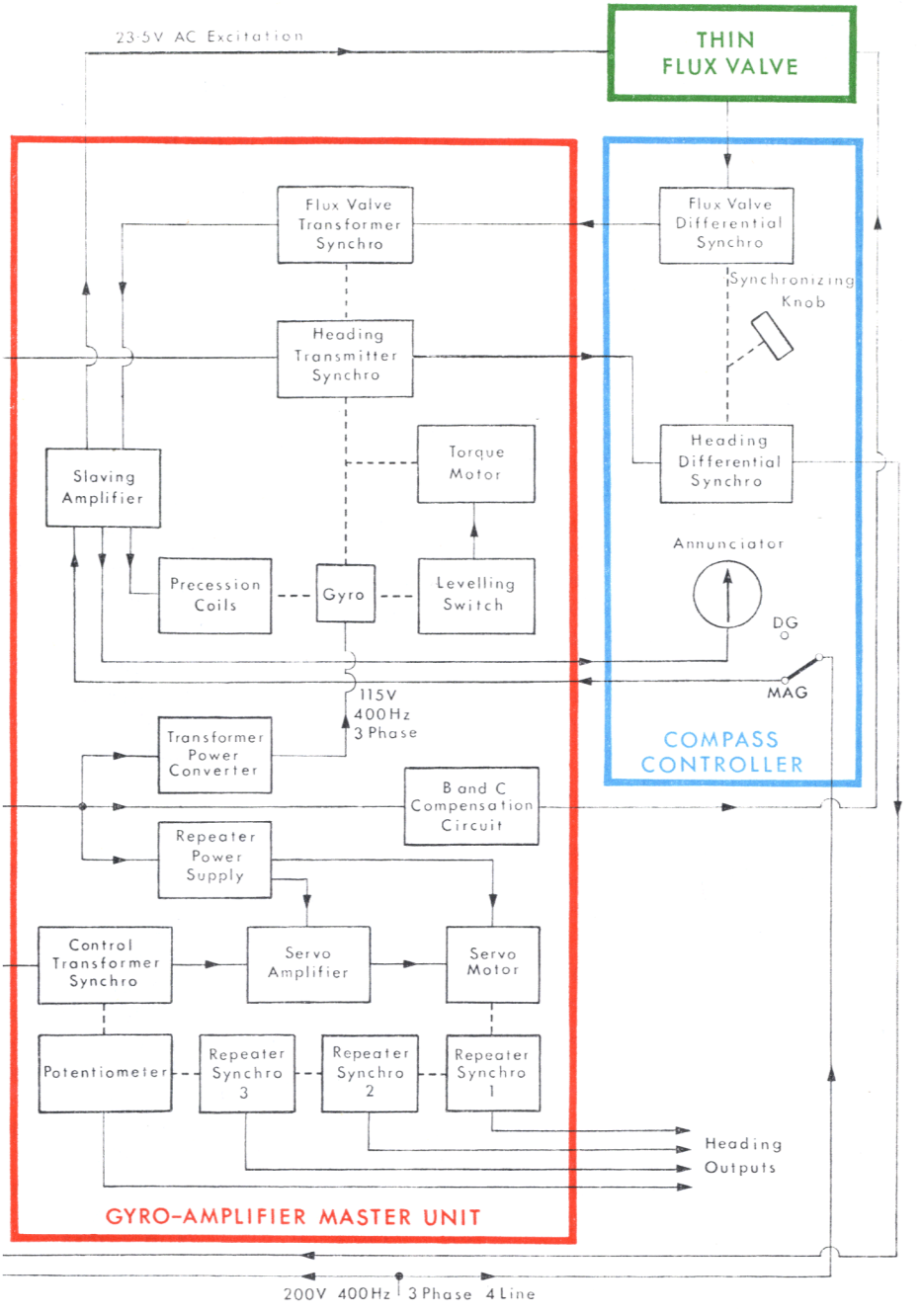


Fig 5 C2J Block Diagram

; the remote corrector unit, the
er power converter and a heading
unit. The heading repeater unit provides four separate analogue outputs of magnetic heading, three by synchronous repeaters and the fourth by a potentiometer

These heading outputs are fed to the Elliott NAVWASS (navigation and weapon aiming system), as HSI and an HUD. They are used during the initial stages of inertial platform alignment, for monitoring intertial platform heading outputs and for reversionary and should the inertial platform fail.

System Components

15. The C2J system consists of the following three basic units as shown in Fig 4:

a. *Gyro-Amplifier Master Unit.* The gyro-amplifier proper, containing the gyro, the slaving amplifier and the two synchros, is the same as that used in the C2G system. It is mounted on a cradle base containing a remote correction unit, a transformer power converter and a heading repeater unit. The base also embodies remote deviation corrector potentiometers to compensate for B and C errors in the thin flux valve output. Gyro characteristics, precession rates and slaving time constant are the same as for the C2G system.

b. *Detector Unit.* The thin flux valve (detector unit) is the same as that used for the C2G system. However, as previously mentioned, the separate compensator unit has been removed from the detector unit.

c. *Compass Controller.* The compass controller is the same for both the C2G and the C2J systems (*see* Fig 2).

System Description

16. Because of the many common units used in the two systems, this description of the C2J will be confined to the differences which are associated with the gyro-amplifier master unit. The basic gyro-amplifier already described in paras 4 to 6, houses the heading transmitter synchro which transmits the azimuth position of the gyro, through the differential synchro (housed in the compass controller), to the heading repeater unit in the base cradle. The repeater unit consists of a control transformer synchro supplying a servo amplifier which energizes a servo motor to drive the control synchro in the gyro-amplifier. This motor also drives three repeater synchros and a potentiometer to provide four outputs of magnetic heading.

17. The remote compensator mounted in the base cradle is set up by the adjustment of two locking potentiometers, one for 'B' and one for 'C' compensation, mounted at the connector end of the base assembly. Adjustment of the potentiometers during a compass swing varies minute currents flowing in the flux valve pick-off coils and thus adjusts for coefficients B and C.

18. Also housed in the base cradle is the transformer power converter which converts the 200 V 400 Hz aircraft supply into the 115 V 400 Hz supply required by the compass.

System Operation

19. Operation of the C2J system is the same as described in paras 10 to 13 except that the signal induced in the heading differential synchro motor is applied to the repeater unit in the gyro-amplifier master unit to provide the four heading outputs. These outputs are used as inputs to the NAVWASS, the HUD and the HSI for alignment of the platform, reversionary modes or when manually selected in preference to inertial heading. For full details of aircraft system operation, reference should be made to the relevant Aircrew Manual. A block diagram of the C2J gyro-magnetic compass system is shown at Fig 5.

CHAPTER 10

C12 GYRO-MAGNETIC COMPASS SYSTEM

CONTENTS

	Paras
Introduction	1
Components	2
Detector Unit	3–4
Remote Magnetic Compensator	5
Directional Gyro	6–11
Digital Controller	12
Amplifier-Power Supply	13–16

OPERATION OF THE SYSTEM

Introduction	17
Follow-up Servo-Mechanism	18–19
Magnetic Slaving Servo-Mechanism	20–21
Initial Synchronization...	22–23
Operation in a Turn	24–27
Detector Error Compensation	28–31
Gyro Error Compensation	32–35

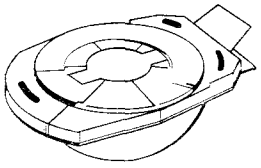
Introduction

1. The C12 gyro-magnetic compass system combines the short term accuracy of the directional gyroscope with the long-term accuracy of the magnetic azimuth detector. In lower latitudes the compass can operate as a magnetically-referenced directional gyroscope (MAG mode). In higher latitudes where the Earth's horizontal field is weak or distorted, the system can operate as an independent directional gyroscope (DG mode).

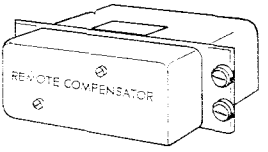
Components

2. The following components comprise the C12 system as shown in Fig 1:

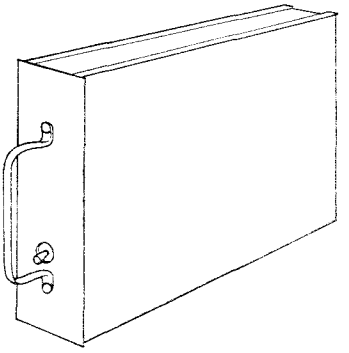
a. Detector unit.
b. Remote magnetic compensator.
c. Directional gyro.
d. Digital controller.
e. Amplifier-power supply.



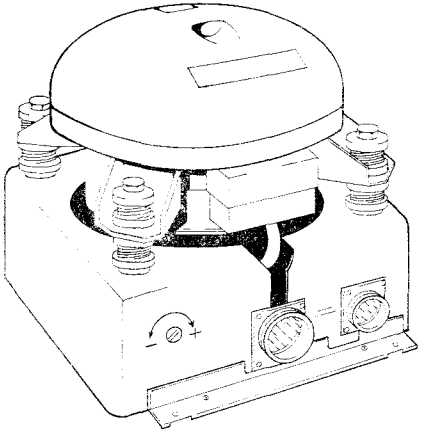
Magnetic
Azimuth Detector



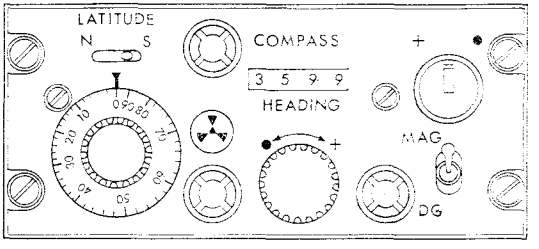
Remote Magnetic Compensator



Amplifier-Power Supply



Directional Gyroscope



Digital Controller

Fig 1 C12 Compass System Components

Detector Unit

3. The detector unit, *see* Fig 1, senses aircraft heading relative to the magnetic meridian. The detector consists of a three-spoke sensing element made up of a core assembly, excitation coil and output windings. The theory of this system is covered in Chap 1 of this section. The sensing element is pendulously suspended to lie in the horizontal plane. The assembly is mounted in a hemispherical bowl which is filled with a damping fluid to prevent excessive oscillation due to vibration and attitude changes.

4. **Turn Cut-out.** A turn cut-out stops signals from the unit whenever a bank of 15° or more is detected by the attitude reference system vertical gyro. The system then operates in the DG mode until the bank becomes less than 15°.

Remote Magnetic Compensator

5. The remote magnetic compensator is shown in Fig 2. It provides compensation for hard iron deviations (one cycle error), soft iron and transmission system deviations (two cycle errors), coriolis error and coefficient A (index error). It produces a regulated DC power supply and has provision for the connection of the MC-1M magnetic compass calibrator.

Directional Gyro

6. The gyroscope is similar to the CL-11 gyro described in Sect 2 Chap 3. The gyro is enclosed within the hermetically-sealed inner gimbal which is mounted on a double rotor race bearing system. The direction of rotation of the bearings is reversed by a switch and magnetic latch relay every two minutes. This technique is partly responsible for the low drift rate of approximately 0·8° per hour.

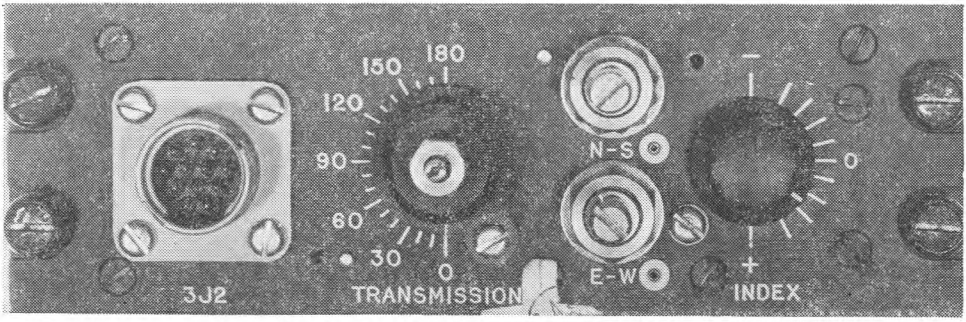


Fig 2 Remote Magnetic Compensator

7. **Magnetic Slaving.** The output stage amplifier is incorporated in the gyro unit. It receives, amplifies and supplies magnetic error signals to the slaving torque motor.

8. **Drift Bias.** An adjustment, to be used on the ground only, is provided on the outside of the gyro case which can be set with a screwdriver. This allows compensation to be made for between ± 3·25° per hour of drift bias. This bias only operates in the DG mode.

9. **Levelling.** The levelling of the gyro is governed by a liquid level switch operating a levelling torque motor. It is located between the case and the outer gimbal. This motor applies torques about the outer axis to precess the gyro to a level position within ± 0·25°.

10. **Gimbal Error Correction.** A second amplifier is incorporated in the gyro unit to supply a gimbal error correction signal. Incorrect heading is sensed when the aircraft is banked, as the gimbal axes of the gyro are not at right angles to the spin axis. A signal, from a vertical gyro, is added to the heading synchro signal in the gimbal error corrector circuit to compensate for this error.

11. **Spin-Down Brake.** A spin-down brake is provided to prevent the outer gimbal from spinning when the gyro spins down after the power is removed. Power interruptions of less than 30 seconds duration do not activate the spin-down brake.

Digital Controller

12. All the functional controls of the system are located on the digital controller, *see* Fig 3. These are as follows:

a. A North/South hemisphere switch (N/S), which determines the sense of the Earth-rate precession correction applied to the directional gyro.

b. A dial graduated in increments of two degrees of latitude, which determines the magnitude of the Earth rate and coriolis correction signals.

c. A power adequacy indicator which shows red when the power supply voltage drops below the minimum for safe and accurate operation (88V).

d. A digital readout of heading.

e. An annunciator dial, with a centre-reading needle, to provide a visual indication of synchronization in the MAG mode.

f. A synchronizing knob, to provide a rapid means of synchronizing the gyro and the flux valve in the MAG mode, or to set a desired heading in the DG mode.

g. A MAG/DG switch by which the mode of compass operation is selected.

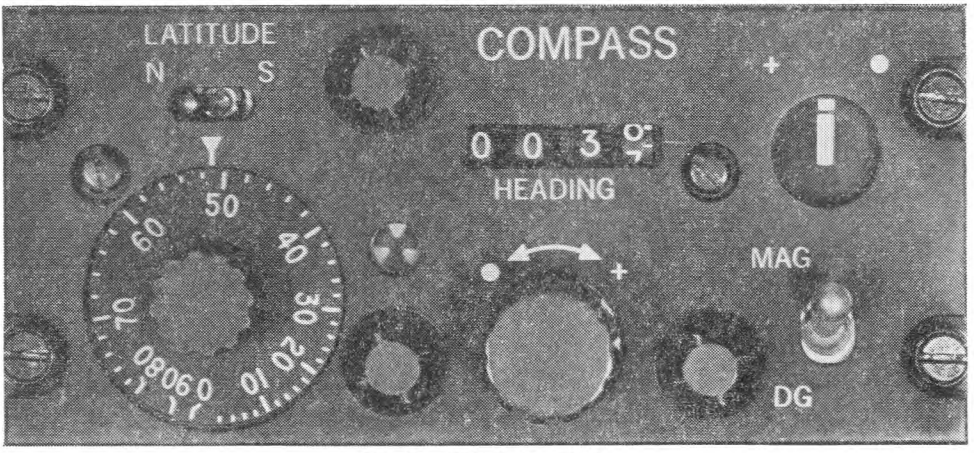


Fig 3 Digital Controller

Amplifier-Power Supply

13. The amplifier-power supply unit consists of the following components:

a. Servo-loop assembly.

b. Slaving amplifiers.

c. Compass rack.

14. **Servo-Loop Assembly.** The servo-loop assembly consists of a shaft upon which is mounted a servo-control transformer (CT), five synchro transmitters (CX), a resolver synchro (RS) and a transolver (TS). The shaft is turned by a motor generator (MG) which also provides feedback to its controlling amplifier:

a. *Meridian Convergence Resolver.* A diagram of the meridian convergence resolver is at Fig 4. A Doppler ground speed signal is supplied, via the N-S switch on the digital controller, to the resolver (RS). The rotor of the RS is positioned by the shaft (aircraft heading) and fed with the Doppler signal. The stator of the RS is held such that it has an output proportional to the East-West component of the ground speed. This component is fed to the slaving amplifier which provides an additional precession signal to the gyro to allow for meridian convergence.

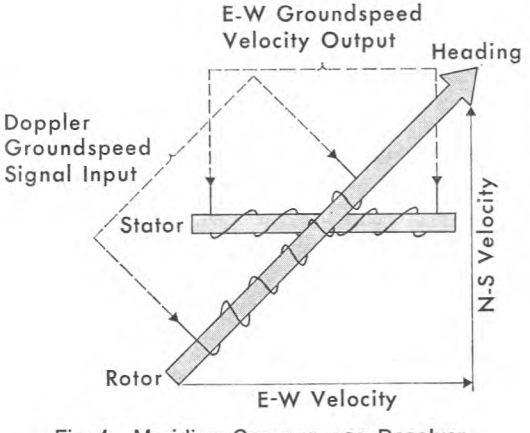


Fig 4 Meridian Convergence Resolver

b. *Transolver.* The transolver consists of a stator and two rotors with windings 90° apart, *see* Fig 5. The Earth's magnetic field, as sensed by the detector unit, is applied to the stator windings. A signal proportional to the Earth's magnetic field will be induced into the No 2 rotor winding as this rotor is parallel to the heading vector. The No 1 rotor is at right angles to the stator field and will therefore have no current induced into it. If the aircraft heading should change, the heading vector induced into the stators will rotate a corresponding amount and the No 1 rotor will now not be at right angles and will have a signal induced in it. This error signal passes round the slaving loop, described in para 20, and the shaft is turned such that the transolver rotor is driven to its new null position. The compass is then synchronized on the new heading. The No 2 rotor supplies a signal, proportional to the Earth's horizontal magnetic component, to the slaving amplifier where a meridian convergence correction is calculated.

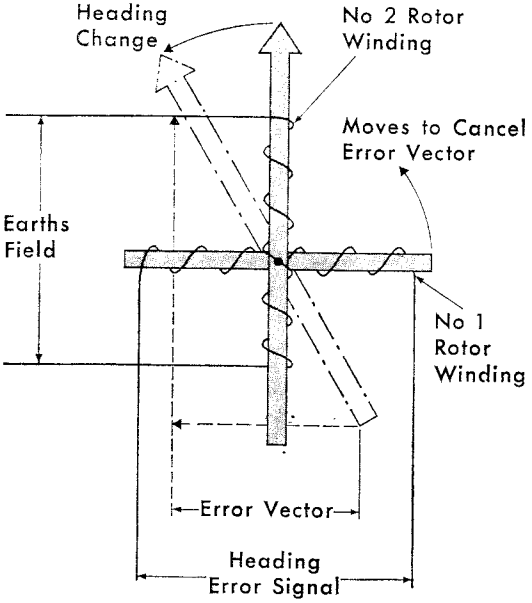


Fig 5 Transolver

15. **Slaving Amplifiers.** The slaving amplifier circuits are divided into two sections (A1 and A2). The A1 amplifier receives the heading error signals from the transolver and amplifies, demodulates and filters them to remove any inherent servo-loop noise. They are then remodulated, amplified and transmitted to precess the gyro. The second section (A2) receives signals from the transolver and the resolver, calculates the meridian convergence correction signals and sends these to the gyro.

16. **Compass Rack.** The compass rack accepts 115V, 400 Hz AC aircraft supplies and provides power to all the components of the compass system except the remote compensator. If the voltage drops below 88V, the power adequacy flag in the digital controller is activated.

OPERATION OF THE SYSTEM

Introduction

17. In both modes of operation, DG and slaved, directional gyro information is reproduced at the equipment's heading output. No attempt is made to align the gyro to any specific azimuth datum. In the slaved mode, after an initial manual synchronization has been carried out, the gyro is precessed in azimuth by a magnetic slaving loop to maintain synchronism between the detector unit heading and the gyro heading output. Transmission of gyro heading to the compass by a servo-mechanism is common to both modes of operation.

Follow-Up Servo-Mechanism

18. The follow-up servo-mechanism is a position control servo-loop which transmits changes in the gyro reading to the equipment heading outputs. It is shown in red in the compass signal flow diagram at the end of the chapter. Any change in the datum direction as defined by the gyro will be transmitted from the stators of the control transmitter (CX7) to the stators of the control transformer (CT1) in the amplifier-power supply unit. If a change occurs, the rotor of CT1 will no longer be in its null position and an error signal will be generated whose phase depends upon the direction of rotor displacement. This error signal passes to the servo amplifier which activates the motor-generator (MG). The motor turns in the direction necessary to turn the CT1 rotor back to its null position. The loop is damped against oscillation by velocity feedback from the motor generator.

19. In the amplifier-power supply unit, the shaft position, which represents the gyro heading, turns the rotors of five CXs. These are the outputs of the system. CX1 transmits the heading to the heading display control transformer (CT2) which forms part of a second servo-loop. The heading readout in the digital controller is automatically synchronized with that represented by the shaft position in the amplifier-power supply unit.

Magnetic Slaving Servo-Mechanism

20. The magnetic slaving servo is the loop by which the system heading output is slaved to the magnetic heading output sensed by the detector unit. This loop is shown in green in the signal flow diagram at the end of the chapter.

21. Magnetic heading is fed electrically from the detector unit to the stators of the transolver, whose function is described in para 14b. If its rotor is not in its null position an error signal is transmitted via the slaving amplifier (A2) and the slaving output stage to the slaving motor.

The slaving motor precesses the gyro. From this point the slaving loop is completed by the follow-up servo-mechanism. This acts as described in para 18 and turns the shaft in the amplifier-power supply unit. The rotor of the transolver (TS) is on this shaft so it will turn until it reaches its null position. When this is achieved the compass heading output will be synchronized with the heading sensed by the detector unit.

Initial Synchronization

22. **Magnetically-Slaved Mode.** When the compass is switched on, with the controller mode switch set to MAG, the compass heading will usually be found to be desynchronized. This is indicated by a left or right displacement of the annunciator pointer, showing that an error signal is being transmitted by the slaving amplifier to precess the gyro. The compass will resynchronize itself but this is a slow process as the maximum rate, *see* Fig 6, of the slaving loop is 1·5° ± 0·5° per minute. The slaving amplifier time constant is normally 80 sec over the linear portion of the graph. For fast synchronization a differential synchro (CDX) is inserted in the transmission line between the gyro control transmitter (CX7) and the control transformer (CT1) in the amplifier-power supply unit. This CDX permits any angle between 0° and 360° to be added to the gyro output. The output, which is the sum of the gyro output and that angle fed in on the synchronizing knob, is transmitted by normal follow-up servo-action to the shaft of the amplifier-power supply unit. A synchronized indication will be displayed when the rotor of the TS is turned to its null position.

23. **DG Mode.** In the DG Mode, operation of the MAG/DG switch to DG cuts the output of the slaving amplifier to the slaving motor. The error signal from the TS will now have no effect on the gyro. The synchronizing differential synchro (CDX) can be used to add to the gyro output to give any desired directional gyro heading. The resulting displacement of the TS rotor from its null will have no effect.

Operation in a Turn

24. In considering the operation of the compass

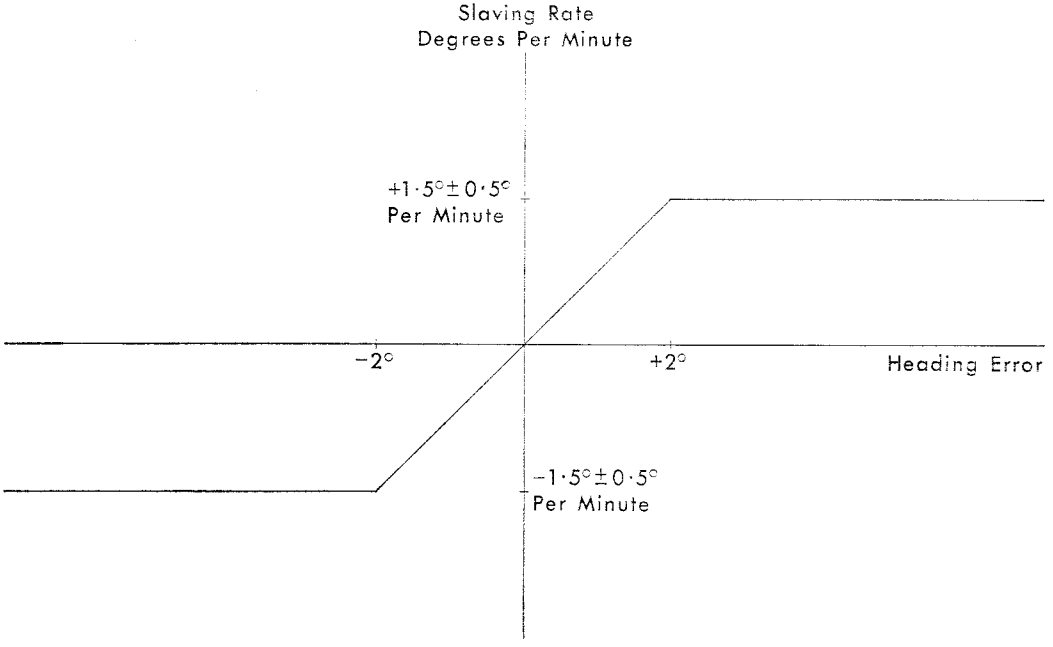


Fig 6 Slaving Amplifier Time Constant

system in a turn, the reader is asked to ignore at first the turn cut-out circuit. During the turn the follow-up servo follows the action of the gyro, transmitting the turn to the shaft in the amplifier-power supply unit. The stator of CX transmits this movement to the heading indicator. The rotor of the TS is also on the shaft and will turn with it. When the aircraft steadies at the end of the turn, the rotor of the TS will still be in its null position, for it will have been turned through the same angle and at the same rate as the stator field of the TS as transmitted from the detector unit.

25. During the turn the detector unit may be tilted and sense a false magnetic heading. A turn cut-out is therefore incorporated. A reference signal from the aircraft vertical gyro cuts off all signals from the detector unit when a bank angle of more than approximately 6° is sensed. The system then operates in the DG mode until the bank angle becomes less than 6°.

26. In a slow turn, where the angle of bank is less than 6°, the sensed magnetic heading will be incorrect and an error will be left in the system at the end of the turn. This will be removed by the normal action of the slaving servo at the slow rate.

27. In the DG mode, the action of the system in a turn is, as was described in paras 18 and 19, that of the follow-up servo-mechanism.

Detector Error Compensation

28. **Index Error.** Any mechanical misalignment between the detector unit and the aircraft, or misalignment of the system synchros, produces an error, constant for all headings, which is known as index error. A differential synchro is placed in series with the detector unit windings. The rotor can be adjusted on the front panel of the remote magnetic compensator to remove up to ± 2·5° of error. A cam device limits the rotation of the rotor shaft to approximately 5° of index error for 180° of rotation of the adjustment control.

29. **Hard Iron (One Cycle Error).** The magnetic field at the detector unit, due to the aircraft's hard iron magnetism, is compensated by a field equal in magnitude but opposite in direction, produced in the detector unit. The required current is governed by two linear potentiometers, in the remote compensator, connected across the regulated power supply of 6·8V ± 5% produced within the unit. The potentiometers are identified N-S, E-W and are adjusted by a screwdriver on the front panel, *see* Fig 2.

30. **Transmission Error (Two Cycle Error).** Transmission error is due primarily to electrical unbalance of the servo-loop synchros. The unbalance produces a sinusoidal error and a 360° rotation produces two cycles of transmission error. Hence it is sometimes referred to as two cycle error. A synchro-like device is used to introduce a signal that is equal in magnitude and opposite in direction to the two cycle error. The input and output windings of this synchro are connected in series with the three windings of the detector unit. The rotor is inserted between the input and output windings. The rotor can be moved in and out to vary the unbalance and rotated to vary the position of the error compensating vector.

31. **Coriolis Error.** Compensation is made for coriolis acceleration at the detector unit. A variable resistor, in the Doppler radar unit,

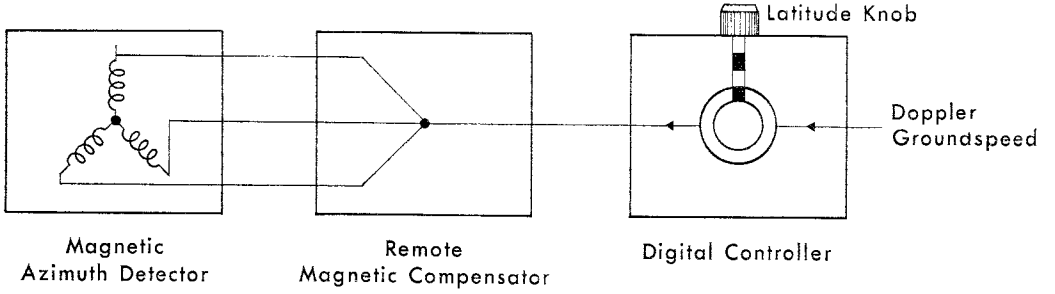


Fig 7 Coriolis Compensation Loop

provides a signal proportional to aircraft ground speed. This is fed to the digital controller where it is modified according to the latitude setting (*see* Fig 7). The resultant signal, which is proportional to latitude and ground speed, and hence to coriolis error, is applied to the remote magnetic compensator. This provides the necessary correction currents which are sent to the detector unit.

Gyro Error Compensation

32. **Drift Bias.** A drift bias circuit provides a ± 3·25° per hour adjustment for the directional gyro. The control is on the outside of the gyro unit and can be adjusted with a screwdriver. Adjustment should only be made on the ground and for best accuracy on a test bench. The drift bias is only applied in the DG mode.

33. **Gimbal Error.** When the aircraft banks, the gimbal axes of the gyro are not at right angles to the spin axis; the aircraft therefore senses incorrect headings. A signal from a vertical gyro is used in the gimbal error corrector to modify the output of the gyro to compensate for the gimbal error.

34. **Earth Rate.** The N-S hemisphere switch and the latitude setting dial, on the digital controller, determine the direction and magnitude of the Earth rate correction signals to the precession coils of the gyro. A 108V AC supply, from the gyro unit, is fed to a variable resistor controlled by the latitude knob. The output is applied to one section of the Earth rate and coriolis computer where the amount of Earth rate error correction is determined. The resulting signal is applied through the N-S hemisphere switch to the relevant gyro precession coil.

35. **Meridian Convergence.** To compensate for meridian convergence error, Doppler ground

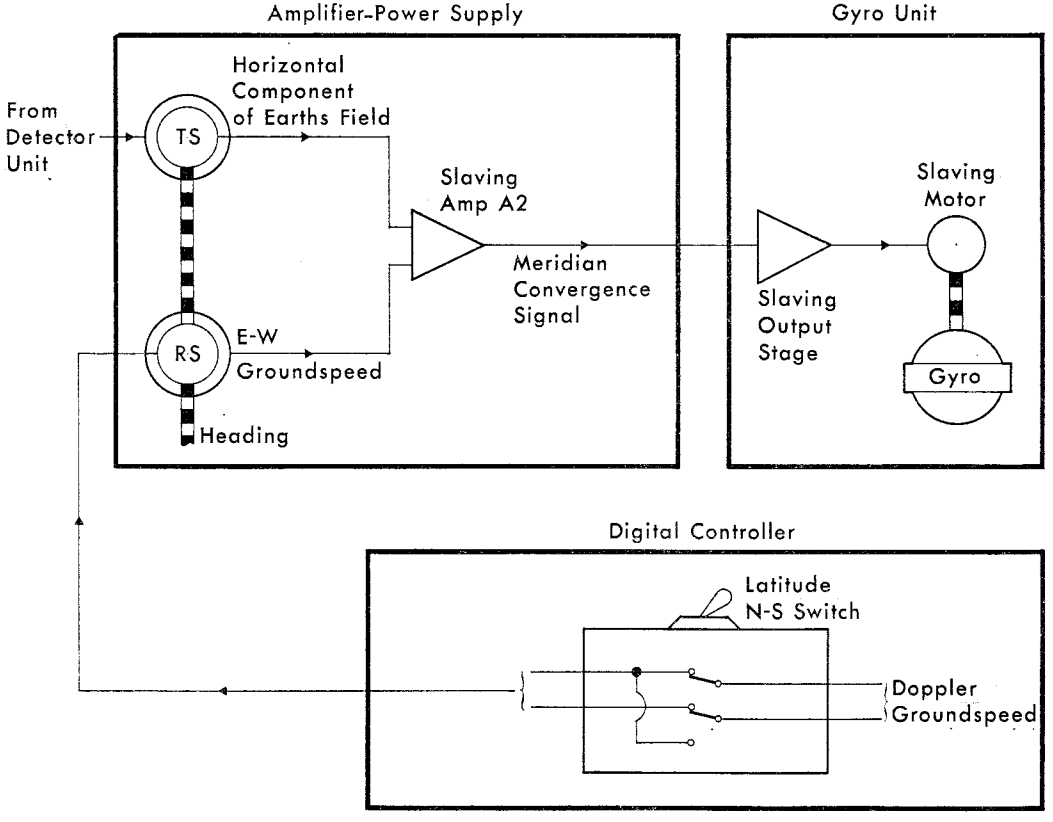


Fig 8 Meridian Convergence Loop

speed is applied through the N-S hemisphere switch, as shown in Fig 8, to the resolver in the amplifier-power supply. The resolver supplies an East-West ground speed component to the slaving amplifier. The transolver rotor winding provides a signal, proportional in magnitude to the horizontal components of the Earth's magnetic field, to the slaving amplifier. The slaving amplifier combines the two signals to form a meridian convergence signal. This is applied to the slaving amplifier output stage in the directional gyro, which drives the slaving motor to compensate for the effect of meridian convergence rate.

Note: The equation for heading correction due to geographic meridian convergence contains a latitude term. The strength of the H field in the C12 system is equated to geographic latitude which results in the convergence correction being an approximation. No correction is made for error due to change in magnetic variation.

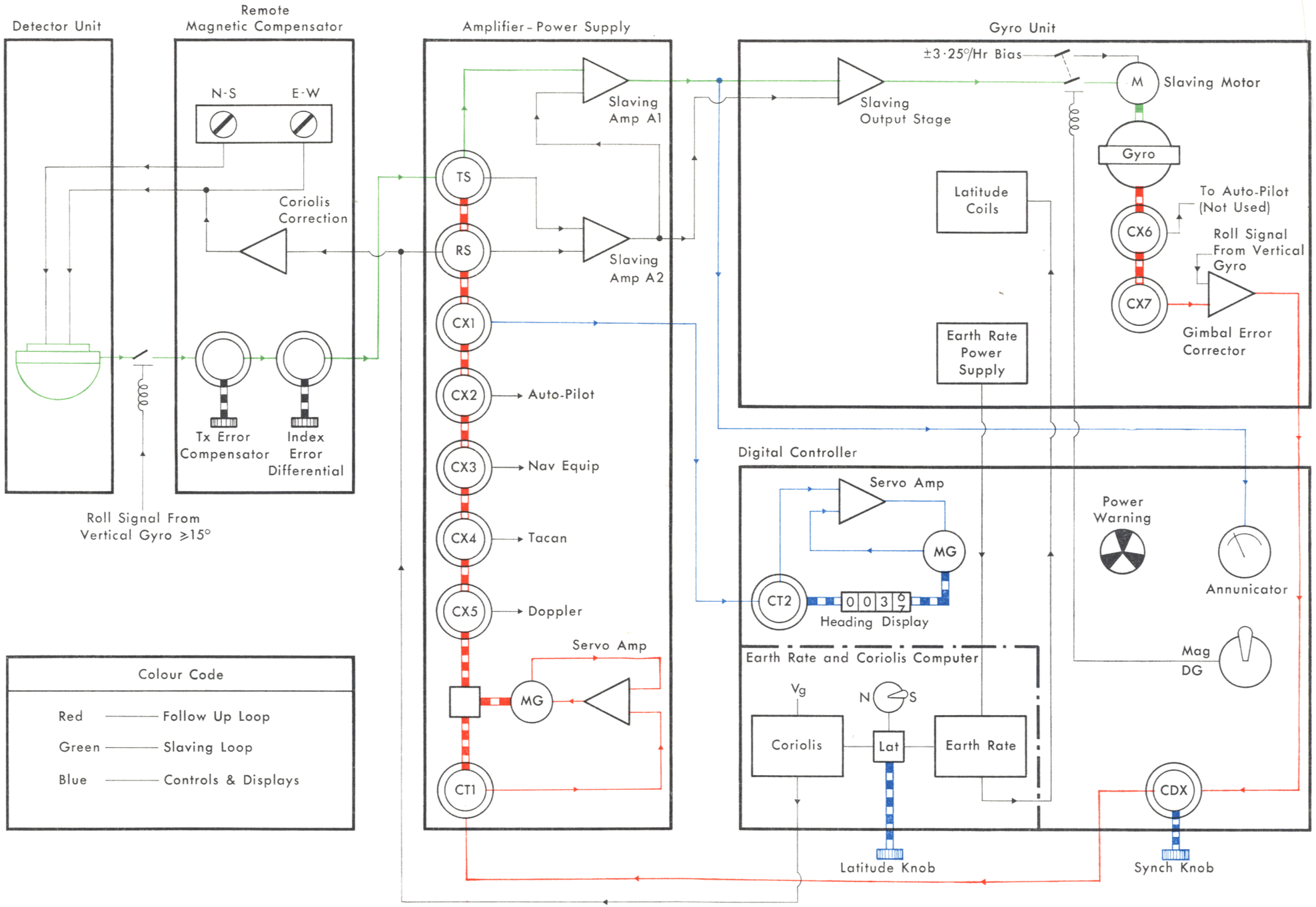


Fig 9 Signal Flow Diagram

(AL 19, Oct 71)

PART 2

SECTION 4

FLIGHT INSTRUMENT DISPLAY SYSTEMS

Chapter

1 **The Zero Reader, ZL 2**

2 **The Military Flight System Director Horizon**

3 **The Integrated Flight Instrument System**

4 **The Head Up Display**

CHAPTER 1

THE ZERO READER ZL 2

CONTENTS

	Paras
Introduction	1
The Main Units	2– 5
Operation of the Flight Computer	6– 9
THE FUNCTIONS OF THE ZERO READER	
Introduction	10
Flight Instruments	11–14
Outbound	15
Inbound	16
Glide Path	17
Test Position	18
IN-FLIGHT USE	
Introduction	19–21
General Instrument Flying	22–26
ILS Approach	27–29
Outbound	30–34
Inbound	35–39
Turn Error	40

Introduction

1. The Zero Reader is a flight director system which accepts inputs from the main flight instruments. It relates them to preset values of heading and altitude to produce demand signals in roll and pitch, which are displayed to the pilot on a parallel motion cross-pointer meter. The Zero Reader has two main functions:

a. It is used as the master flight instrument during transit flying.

b. It is used in conjunction with inputs from the ILS receiver to direct runway approaches.

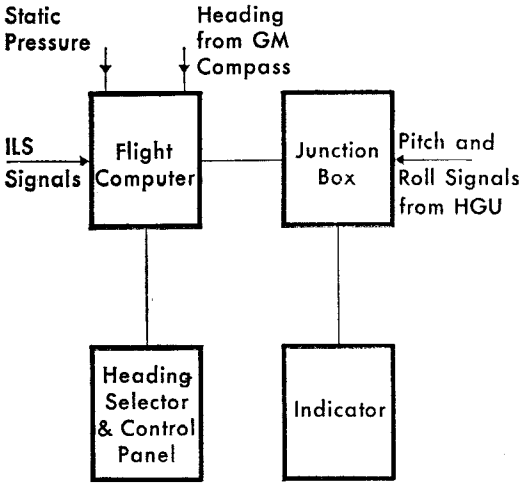


Fig 1 The Main Units of the Zero Reader

The Main Units

2. Fig 1 shows the main units of the Zero Reader which are briefly described in the following paragraphs.

3. **The Indicator.** The indicator is mounted on the flight instrument panel, the face being a section of a sphere with a zero mark in the centre (*see* Fig 2). The vertical and horizontal pointers are deflected from the centre mark by the action of moving coil meter movements. The pointers are curved to follow the surface of the sphere, thus eliminating parallax error. The pointers sweep horizontally and vertically across the face of the instrument to indicate fly right/left and up/down in accordance with inputs from the

flight computer. Graduations on the face of the instrument represent half, three quarters and full scale deflection. Two moving coil meters actuate the warning flags which appear on the following occasions:

a. Disconnection of the control panel.

b. Failure of ac power supplies or computer HT.

c. Disconnection of the HGU or failure of dc power supplies.

d. With INBOUND or OUTBOUND selected and an ILS localiser warning flag showing.

e. With GLIDE PATH selected and either of the ILS warning flags showing.

f. Selection of Test Facility.

4. **The Heading Selector and Control Panel.** The heading selector consists of a rotatable compass card which registers against a fixed lubber line. Preset values of desired heading are manually set against the lubber line by means of a knob which surrounds the TEST button at the bottom of the control panel face. The central selector is a rotary switch with five positions. FLT INST is used for transit flight, whilst INBOUND, OUTBOUND, and GLIDE PATH correspond to the relevant ILS function. The fifth position is the TEST facility. The right hand switch connects the altitude control unit to the flight computer. The left hand switch operates the pitch trim: interlocks restore the pitch knob to zero when the main selector is at GLIDE PATH or the altitude control is switched to ON or OFF.

5. **The Flight Computer.** The flight computer is an electronic analogue computer which receives signals from the horizon gyro unit, the gyro-magnetic compass, the ILS receivers and the altitude control unit, which is an integral part of the computer. The signals are compared with the values set on the heading selector and control panel. The computer then passes the required dc potential to actuate the pointers on the indicator.

Operation of the Flight Computer

6. The flight computer is made up of the following sub-units:

a. The pitch and roll channels.

b. The altitude control unit.

c. The follow-up amplifier.

d. The power pack.

A full description of these units and their operation is contained in the AP 112C Series. A brief description of the operation of the pitch and roll channels and the altitude control unit is given in the following paragraphs.

7. **The Roll Channel.** Roll signals are developed across a pick-off potentiometer in the horizon

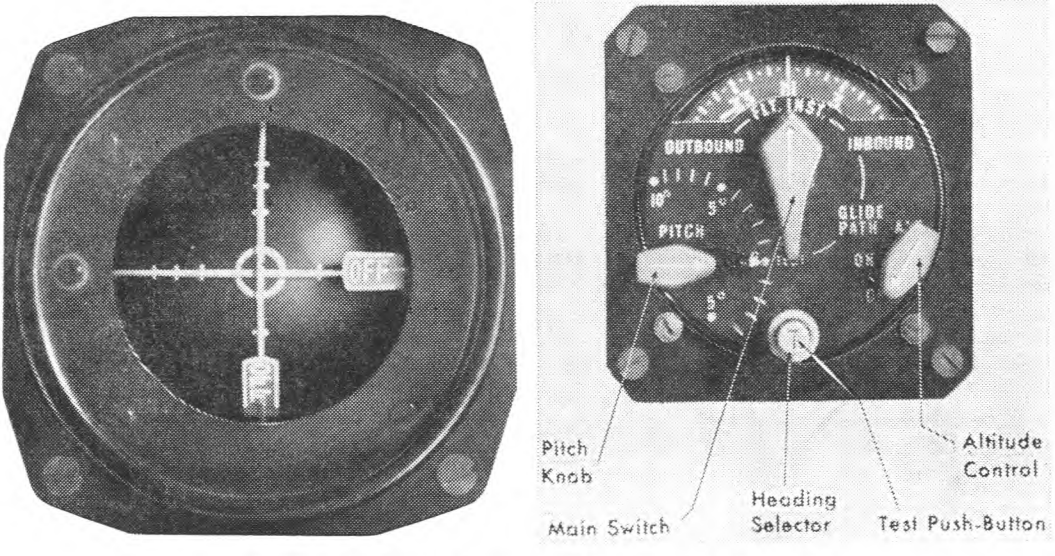


Fig 2 The Indicator and The Heading Selector and Control Panel

gyro unit. These signals are fed to the roll channel where they are compared in a bridge circuit with the roll demand signals generated by the heading selector. The output of the roll channel is limited to ensure that the preset bank angle is not exceeded.

8. **The Pitch Channel.** Pitch signals are developed in the horizon gyro unit in a similar fashion to the roll signals. They are compared in a bridge circuit with the pitch demand set in at the control panel. The output of the pitch channel is limited to prevent excessive pitch demands being fed to the horizontal pointer.

9. **The Altitude Control Unit.** The altitude control unit consists of a capsule mechanism which responds to very small changes of pressure in the aircraft static line. When the main selector switch on the control panel is set to OUTBOUND, FLT INST or INBOUND, the altitude control can be used. Switching the altitude control to ON selects the ambient pressure at that time as the altitude datum. If the aircraft subsequently deviates from the selected datum the horizontal pointer is deflected by signals from the capsule unit, through the pitch channel, to indicate the correction required to regain the datum altitude. A limiting circuit controls the extent of the pitch demand displayed, and the altitude switch is tripped to the OFF position if the altitude control unit is required to signal an altitude change of more than approximately 600 feet (20 mb). Switching the altitude control to OFF removes the altitude control signals. Note that operating the pitch knob as an attitude trim while the altitude control switch is selected ON will provide an additional input to the pitch channel.
Note. The capsule is connected to the I bar of an E and I pick-off (*see* Part 1, Sect 2, Chap 2, Fig 9). When the altitude control is ON the output of the follow-up amplifier is earthed, and the signals from the E bar pick-off are switched by relay to the pitch channel. Switching the altitude switch to OFF actuates the follow-up amplifier and motor, the motor continuously driving the I bar back to its null position.

THE FUNCTIONS OF THE ZERO READER

Introduction

10. The various functions of the Zero Reader are manually selected using the appropriate switches on the heading selector and control panel. The functions are described in the following paragraphs under headings corresponding to the positions of the main selector switch, and the signal flow is shown in Fig 3.

Flight Instruments

11. When FLT INST is selected roll and heading signals are fed through the roll channel to the vertical pointer, and pitch signals are fed through the pitch channel to the horizontal pointer. Provided that the altitude control switch is ON, altitude signals are fed through the pitch channel.

12. **Heading Signals.** The heading signals are generated by the heading selector. Sine and cosine windings on the shaft of the deviation rotor are energized according to the angle between the heading selected and the heading indicated by the GM compass. The signal from the sine winding is fed to the roll channel, via a limiter circuit which ensures that regardless of the heading error the roll demand does not exceed a predetermined level. As the aircraft approaches to within a few degrees of the selected heading the roll demand reduces progressively until the aircraft has rolled out on the correct heading.

13. **Roll Signals.** Roll signals are generated by the horizon gyro unit (*see* para 7). When the pilot banks the aircraft in response to a roll demand signal generated by the heading selector, the roll signal developed by the horizon gyro unit is fed to the bridge circuit, where it balances the roll demand signal, and the vertical needle on the indicator is centralized. The needle remains central until the roll demand signal reduces and the roll signal from the horizon gyro unit displaces it; the pilot must then reduce the angle of bank to return the needle to the centre.

14. **Pitch Signals.** Pitch signals are generated by the horizon gyro unit (*see* para 8). Pitch demand signals are provided by displacing the pitch knob from the zero position to attitude angles of up to ±20 degrees in pitch. The pitch knob is only operative when the main selector switch is at OUTBOUND, FLT INST or INBOUND: turning the switch to GLIDE PATH or TEST returns the pitch knob to zero by interlock. Similarly operation of the altitude switch either to ON or OFF zeroes the pitch knob. If the altitude switch is ON, the pitch knob can be altered to provide pitch trim for optimum attitude. The operation of the altitude control unit in relation to the pitch channel is described in para 9.

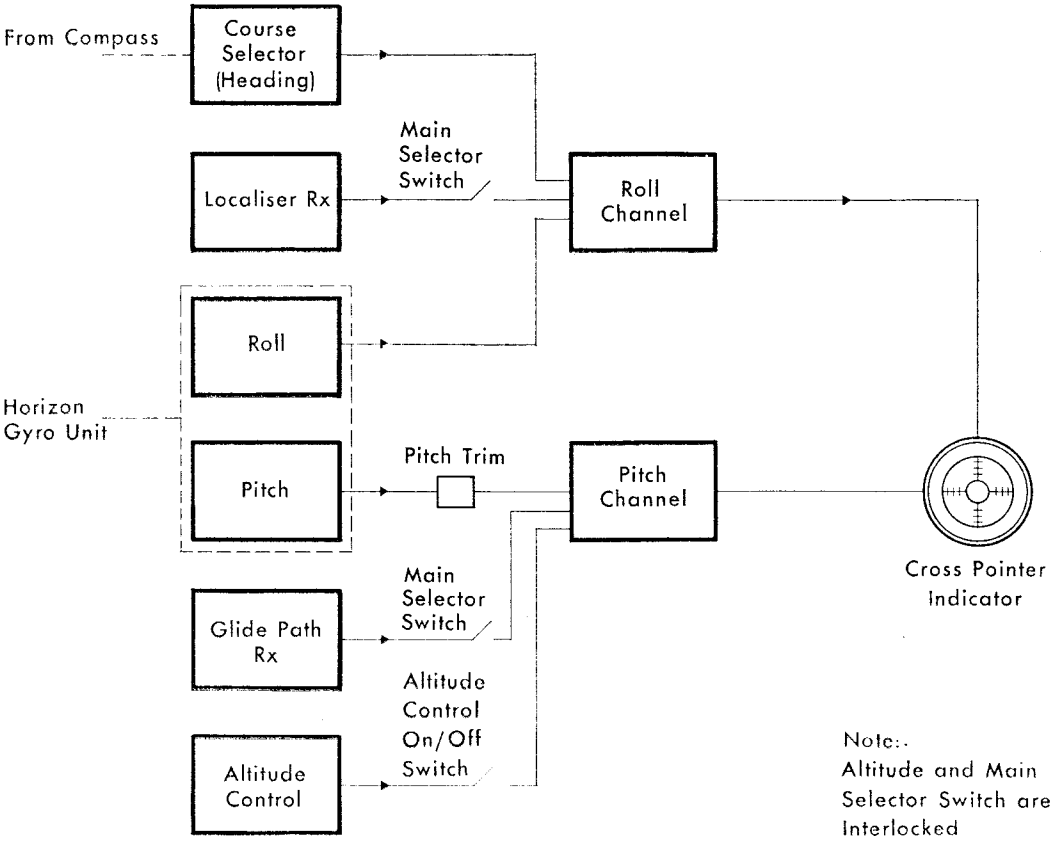


Fig 3 Block Diagram of the Zero Reader Signals

Outbound

15. OUTBOUND is selected when the aircraft is outbound on the runway QDR preparatory to making an approach along the QDM. The roll and pitch channels are connected to the indicator, and the desired heading and the altitude should be selected on the heading selector and control panel. In addition the ILS localizer signal is combined with the heading signal. Its polarity is reversed, so that it generates fly right and fly left signals in the correct sense when the aircraft is displaced from the ILS centre line. The sensitivity of the roll channel to the ILS signals is limited to reduce the closing angle with the centre line.

Inbound

16. INBOUND is selected when the aircraft is inbound on an ILS runway approach. The roll and pitch channels are connected to the indicator. In addition the localizer signal is combined with the heading signal with its polarity unchanged, as the aircraft is now flying inbound and the ILS indications are in the correct sense.

Glide Path

17. GLIDE PATH is selected as the aircraft approaches the ILS glide path. The pitch knob is switched to zero and the altitude switch is turned OFF by interlock. A preset fly down signal is fed via the pitch channel to the horizontal pointer, as is the ILS glide path signal, so that when the aircraft is on the glide path in the correct attitude for descent the horizontal pointer will be at zero. The ILS localizer signal is reduced and combined with the heading signal as a demand signal, which is then fed to the vertical pointer on the indicator. The fly down preset signal is calculated so that it is less than the fly down signal generated by the ILS glide path.

Test Position

18. Selecting TEST disconnects all normal signal

sources from the flight computer. Pressing the TEST button induces a half scale fly left and a quarter scale fly up signal. These indications appear irrespective of the aircraft attitude, providing the presets are correct and the heading selector lubber line is approximately aligned with the indicated heading. Both warning flags also drop into the OFF position.

IN-FLIGHT USE

Introduction

19. The following paragraphs describe the practical use of the Zero Reader in general terms. The appropriate volumes of the relevant Aircrew Manual and Air Staff Instructions should be checked for amplifying instructions.

20. The instrument does not provide indications of actual aircraft attitude in either roll or pitch, nor does it indicate actual displacement from the ILS centre line and glide path. Thus frequent reference should be made to the other instruments, particularly during a runway approach.

21. Once the pilot has selected the appropriate function, and set in the required values, the aircraft must be flown to bring both the indicator pointers to the centre zero position, by flying the centre circle to the intersection of the pointers. The pointers will centralize when the attitude demand has been satisfied. As the aircraft approaches the required heading and height (when using the altitude control) the demand signals change, and the attitude must be changed to match. While the aircraft is following the set conditions no further demand signals are generated and the pointers remain centralized.

General Instrument Flying

22. With the selector switch at FLT INST all normal flight manoeuvres may be performed.

23. To effect a turn, set the desired heading by the set-heading knob on the heading selector, then apply roll control to zero the vertical pointer on the indicator. Turns greater than 165 degrees should be selected in two stages to ensure that the aircraft turns in the desired direction. To maintain continuity of the turn, the second heading selection should be made before the first stage is complete.

24. Climbs or descents are made by appropriate adjustments of the pitch knob on the control panel. The horizontal pointer of the indicator, being more sensitive than the artificial horizon, enables a more accurate pitch attitude to be maintained.

25. To fly the aircraft at a selected altitude, the altitude switch on the control panel should be turned to ON at the required altitude. The pitch knob will return to its zero position, and at the same time the aneroid capsule in the computer comes into operation. By keeping the horizontal pointer of the indicator at zero the aircraft will be maintained at the selected altitude (but *see* para 26). Any variation from this altitude by more than approximately 600 ft will cause the altitude switch to automatically switch off.

26. After engaging the altitude control unit small changes in pitch setting may be required to compensate for changes in the aircraft load distribution, or changes in aircraft speed.

ILS Approach

27. It is advisable to check the correct functioning of the equipment in the air before commencing an ILS approach.

28. To do this, move the control panel selector switch to TEST and check that the set heading and heading pointers on the heading selector are aligned. Press the test button and note that the indicator should indicate fly up and fly left. This procedure will automatically disengage the altitude control unit; therefore if altitude control is still required, the altitude switch must be re-set to ON.

29. On completion of the check return the selector switch to FLT INST.

Outbound

30. When clearance has been received, the airfield is approached and a let-down commenced to the approach height. The approach height may be held by turning the altitude switch to ON when at the approach height. (*See* Fig 4.)

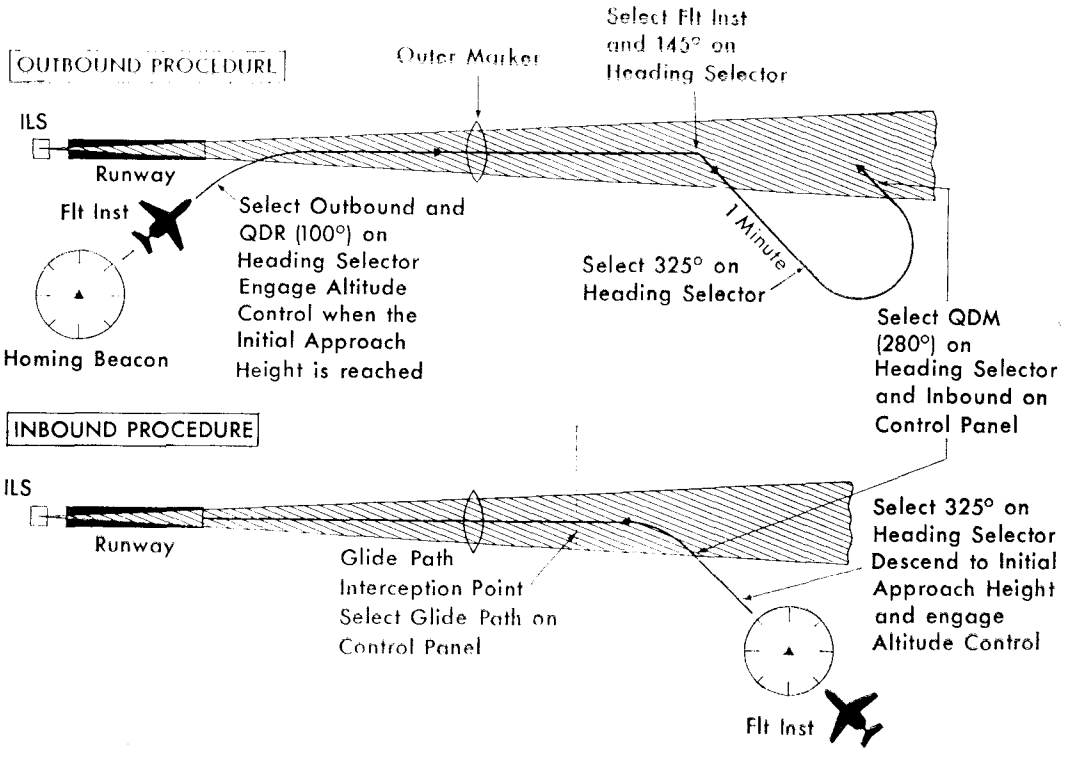


Fig 4 Diagram of ILS Procedures

31. When the ILS radio beam is being intercepted, set the selector switch to OUTBOUND and set runway QDR on the heading selector by the set heading knob. If both pointers of the indicator are kept crossed in the zero position the aircraft should enter the localizer beam at the approach height.

32. If drift exceeds 5 degrees it may be compensated for by resetting the heading, but as this is not usually of great significance during the outbound leg frequent corrections should not be made.

33. If precision is necessary, the exact heading can be calculated, making due allowance for the prevailing wind, and this heading set on the heading selector at the beginning of the approach in place of the beam QDR.

34. After flying beyond the outer marker for the stipulated period (Fig 4), usually one minute, a procedure turn should be started. Select FLT INST, set the QDR plus 45 degrees on the heading selector, and fly this heading for one minute before turning inbound. Vital actions may be completed at this stage.

Inbound

35. After homing, or during the second half of the procedure turn if the outbound approach has been flown, set the selector switch to INBOUND and set the runway QDM on the heading selector. Both pointers of the indicator are kept crossed in the zero position to bring the aircraft on to a heading which will intercept the localizer beam. Drift correction is also made during the inbound approach by adjusting the QDM to allow for calculated drift.

36. Small pitch knob adjustments may be necessary during the inbound approach to compensate for changes in the aircraft's attitude as wheels and flaps are lowered and for speed changes. This is only necessary for precise altitude holding.

37. As the glide path needle of the ILS indicator approaches the central position the selector switch should be moved to GLIDE PATH and the aircraft once again flown to maintain the indicator pointers in the zero position.

38. When the selector switch is moved to GLIDE PATH the altitude control unit is automatically disconnected and the pitch knob returned to its zero position. The ILS glide path signal together with a pre-set fly down signal will now be fed to the circuits controlling the horizontal pointer of the indicator, and by maintaining the horizontal pointer at zero the aircraft will descend the glide path. The Zero Reader is calibrated so that the aircraft will be flown slightly above the glide path beam.

39. At the same time the localizer signal sensitivity is reduced so that no violent turns will be called for as the aircraft nears the ground, and the Zero Reader is able to ride any small bends which may occur in the ILS localizer beam.

Turn Error

40. A small turn error may actually exist in the reference horizon gyro unit supplying signals to the Zero Reader, but since it is combined with the heading signal to control the vertical pointer of the indicator, the aircraft will always come out of a turn with its wings level. The only indication of an error will be the slight discrepancy between the aircraft's heading and the selected heading; this is normally less than one degree and will disappear in approximately one minute. Turn error on the reference horizon gyro unit may also be responsible for small altitude errors when turns are made with the altitude switch on. The resultant error is always positive and disappears in approximately one minute.

CHAPTER 2

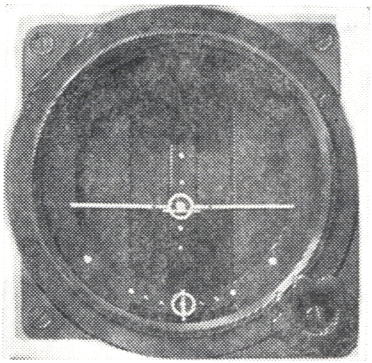
THE MILITARY FLIGHT SYSTEM DIRECTOR HORIZON

CONTENTS

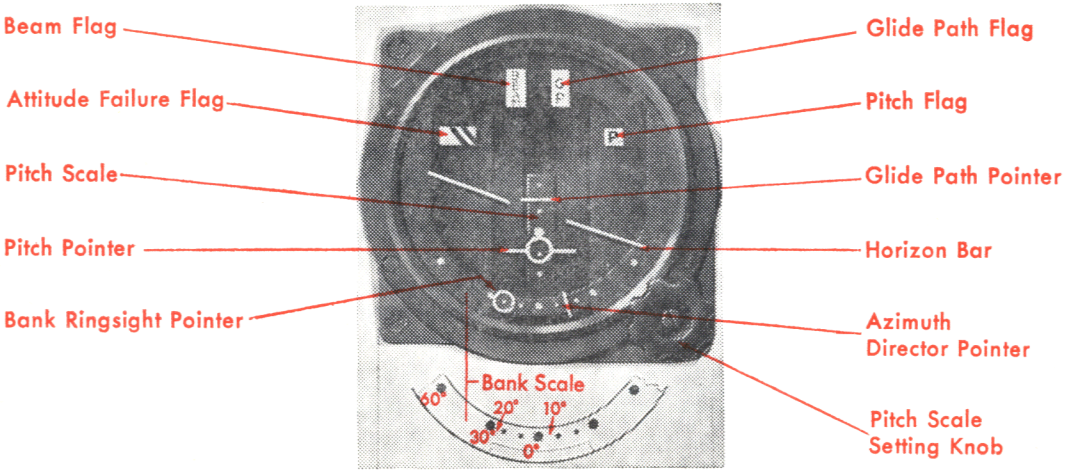
	Paras
Introduction	1
The Twin Attitude System	2–4
Director Horizons	5–8
THE MFS SELECTOR	
Introduction	9
Navigation Selector Switch	10
Pitch Selector Switch	11
Interlocks	12
SUMMARY	
Signal Flow	13
In-Flight Use	14
Conclusion	15

Introduction

1. The Military Flight System (MFS) is a system of flight instrumentation designed for bomber and transport aircraft and was introduced to simplify and rationalize the pilots' attitude displays. In the MFS several instrument facilities have been correlated to provide heading and attitude information, radio beam displacement and comprehensive flight director displays on two instruments: the director horizon (Fig 1) and the beam compass (Fig 2). This has resulted not only in a reduction in the number of instruments required but also in an improved presentation and increased facilities. This chapter describes the director horizon and the attitude functions of the flight director; the beam compass and its associated equipment are described in Part 2, Sect 3, Chap 3.



Normal Presentation



Presentation Displaced for Annotation

Fig 1 Director Horizon

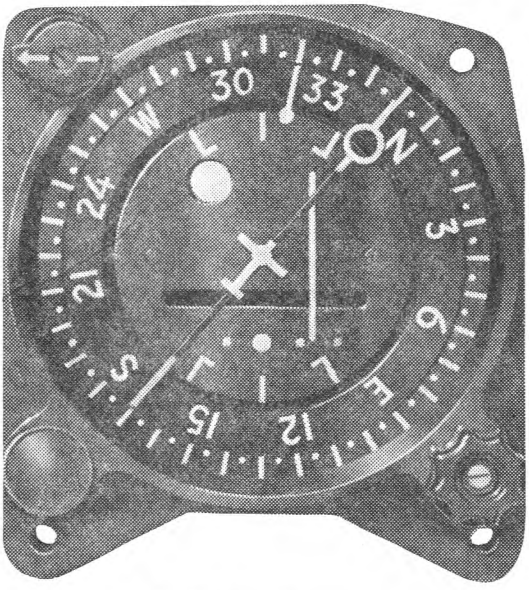


Fig 2 Beam Compass

The Twin Attitude System

2. The system consists of two independent attitude systems, one displaying information on the port director horizon and the other displaying the same information on the starboard director horizon. This duplication of systems increases the reliability of essential information, allows comparison of the two displays and enables accurate failure warnings to be given. Comparators receive signals from both systems and allow attitude failure flags to fall into view on the director horizons when a discrepency between the two systems in excess of $3\frac{1}{2}°$ in pitch or $10°$ in roll is detected, or power failure occurs.

3. **Vertical Gyro.** Each attitude system uses as its reference a remotely situated displacement gyro, the axis of which is erected and maintained to the vertical by a gravity sensing erection system. Erection is automatic on application of 115V AC and 28V DC supplies, the axis attaining the vertical within two minutes (30 seconds when the rapid erection system is incorporated). When the initial (fast) erection sequence is complete the erection rate drops to the normal rate of $2\frac{1}{2}°$ per min in pitch and $5°$ per min in roll, the lower pitch rate being used to minimize errors due to fore-and-aft accelerations. A turn switch provided in each compass amplifier (*see* Part 2, Sect 3, Chap 3) detects turns in excess of $30°$ per min and by operating a bank of contacts initiates a pitch bank erection system designed to reduce errors caused by centrifugal force. The gyros have complete freedom in roll and move through $\pm 85°$ in pitch.

4. **Transmission of Attitude Signals.** Pitch and roll signals are electrically transmitted from the vertical gyros to the director horizons by means of potentiometers and synchros. A primary and secondary pitch potentiometer are fitted to the gimbal system of each vertical gyro, the primary transmitting pitch signals to its associated director horizon. The port secondary pitch potentiometer is used to supply other equipment (*eg* terrain following radar TFR) with pitch information, while the starboard secondary pitch potentiometer feeds the flight director for control of both pitch scales (*see* para 6). The roll systems, having freedom through $360°$, use an AC synchro to transmit the primary information to the director horizons. The secondary roll pick-offs are potentiometers, the starboard information being supplied to the autopilot Mk 10B as a roll datum while the port output, although available for other equipment, is not used at present.

Director Horizons

5. Each pilot has a director horizon (Fig 1) giving a basic attitude display (Fig 3) upon which

are superimposed flight director demands. The attitude display differs from that of a conventional artificial horizon in that separate elements are used to indicate pitch and roll displacement. Roll displacement is indicated by rotation of a horizon bar about the centre of the instrument dial and by a bank ring-sight indicating precise angles of bank. Pitch information is indicated by vertical deflection of a pitch pointer, shaped as a miniature aircraft. Mk 1 series instruments show pitch displacement over a range 40° above and below the horizon bar, the movement being approximately three times more sensitive in the ± 10° range than in the remaining 30°. The Mk 2 series have a pitch indication range of ± 20°, the movement being linear throughout.

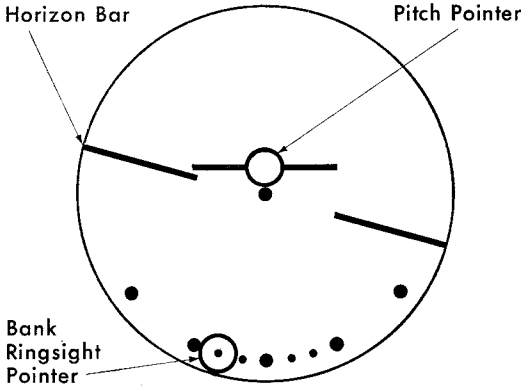


Fig 3 Attitude Display

6. **Pitch Scales.** Pitch scales over which the pitch pointers move can be displaced up or down from the central position, so that the large centre dot of the scale can serve as a datum for any pitch attitude the pilot wishes to maintain. If the pilot wishes to climb or descend at a constant pitch attitude, the movable pitch scales can be set to the required position by means of the pitch scale setting-knob on the director horizon (Fig 1). The pitch scale setting-knob enables the pitch scales to be set to the required datum position by any one of the three following modes:

a. *Mode I (Manual or Emergency Setting).* Mode 1 operation enables the scale to be set manually to any position within its full range of movement. The setting knob is pulled out from its normal position and the scale on that director horizon is then moved up or down by rotating the knob. When Mode I is selected, the pitch scale is mechanically disengaged from the remainder of the system. It is provided primarily as a safety precaution in the event of failure of the servo-mechanism.

b. *Mode II (Trimming Setting).* When the pitch scale setting-knob is in its normal position, *ie* pushed in from the Mode I setting, it has limited rotation in either direction and is spring-loaded to return to its central position when released. Clockwise rotation of either setting-knob causes the pitch scales on both director horizons to be servo-driven up at a slow fixed speed, anti-clockwise rotation driving the scales down. This enables both pitch scales to be set simultaneously to a required datum position, a small *P* flag appearing while the scales are moving. Mode II is inoperative if any director function (para 7) is selected on the pitch selector switch.

c. *Mode III (Fast Chase).* The fast chase setting enables both pitch scales to be set rapidly, so that the centre dots are in alignment with the pitch pointers, *ie* the pitch datum is set to the attitude at which the aircraft is flying when the setting is selected. It is engaged by pushing in the pitch scale setting knob against a spring pressure, the *P* flag appearing during the operation. This facility is particularly useful during an overshoot from an approach when the pilot, after putting the aircraft into a climbing attitude, sets the new pitch datum and then maintains that attitude by keeping the pitch pointer over the centre dot of the pitch scale. The use of Mode III disengages any pitch director function, and the pitch selector switch on the MFS selector (Fig 5) reverts to the central position.

7. **Flight Director Facilities.** Unlike the compass and attitude systems the flight director is not duplicated and therefore is not completely independent. It relies on signal sources from the compass and attitude systems and from external navigational and radio equipment. The signals which control the directors also control the autopilot so that automatic approaches and other turning manoeuvres are continuously displayed. The signal circuits are switched to ensure that only the desired signal is coupled to the system, and three switches forming the MFS selector (Fig 5) enable the pilot to select any director function. Director information is presented in the form of attitude commands on

the director horizons, pitch demands being displayed by the centre dot of the pitch scales or, in the particular case of following an ILS glide path beam, by the glide path pointer. Bank demands are shown by the azimuth director pointers.

8. **Azimuth Director Pointers (ADP).** All flight director demands in azimuth are fed to the azimuth director pointers (Fig 4) which move over the bank scale to a maximum of 30° of bank. The pilot answers the demand by banking the aircraft to keep the bank ring-sight over the ADP, which gradually reduces the bank demand to zero as the desired heading is approached.

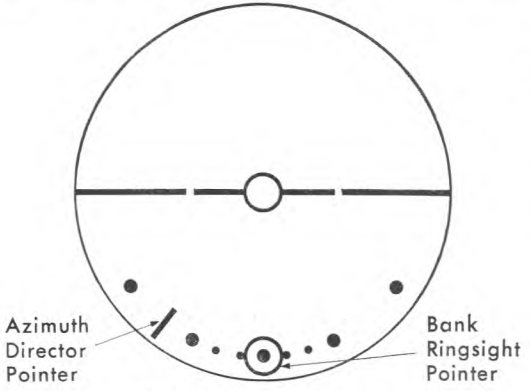


Fig 4 Azimuth Director

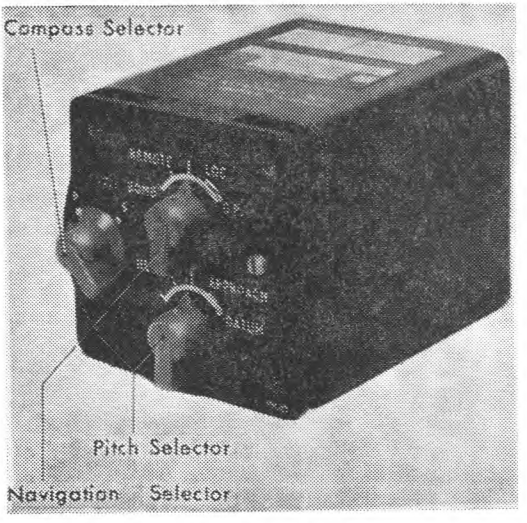


Fig 5 The MFS Selector

THE MFS SELECTOR

Introduction

9. The MFS Selector used with the Mk 1B equipment is shown in Fig 5. It is mounted centrally in the cockpit where it is accessible to both pilots. It has three selector switches: the compass selector, which selects either the port or starboard compass system; the navigation selector, whose functions are described in para 10; and the pitch selector (para 11).

Navigation Selector Switch

10. The Navigation Selector Switch (Fig 5) controls the flight director azimuth facilities by selecting the source of signals to the ADPs. It also connects ILS signals to the director horizon and beam compasses. The switch has five positions:

a. *Bomb.* When bomb is selected, steering signals from the NBS or from the bomb aimer's visual turn control are fed to the ADPs. The ratio of bank demand to turn demanded is variable from 0·5:1 to 5:1 and is controlled by the navigator.

b. *Remote.* Selection of remote allows variation and/or drift information from the Track Control Unit to affect the heading pointers of the beam compasses. The compasses therefore show magnetic heading, true heading, true track or magnetic track depending on the output of the Track Control Unit. Any deviation from the selected heading or track will appear as a bank demand on the ADPs. The ratio of bank to turn is 2:1, *ie* 2° of bank are demanded for 1° of turn.

c. *Central.* With the switch in the central position, the beam compasses show magnetic heading, and heading errors about this datum only are fed to the ADPs. If a Heading Reference System (HRS) is fitted in the aircraft, the switch must be at Central before HRS steering signals can be switched to the ADPs or autopilot.

d. *LOC.* When LOC is selected ILS localizer signals are superimposed on the magnetic heading errors supplied to the ADPs; the resultant signal enables the aircraft to couple to and follow a localizer beam. The bank demand to turn ratio is now 1:1 to reduce bank angles during the approach. When LOC signals are selected a flag marked BEAM appears on each director horizon. The flag is steady if the ILS signal is strong enough, but

d. *DATUM.* To ensure a correctly directed descent along an ILS glide path beam the centre dot of the pitch scales must provide an accurate datum attitude from which the glide path pointer can show displacement (*see* Fig 7). When the glide path beam is first intercepted the glide path pointers will move up, and will then move down as the centre-line of the beam is approached. As the glide path pointer reaches the centre dot, DATUM should be selected. This will drive the pitch scale and the centre dot to a pre-determined position representing the attitude required to maintain a descent path assuming optimum conditions for the aircraft type. As the glide path pointer is carried with the scale, the centre dot not only indicates the pitch attitude required but continues to be the datum for glide path displacement. The DATUM selection is made against a spring loading, so that when released the Pitch Selector Switch will automatically move to the APPROACH position provided that GP is already selected on the Navigation Selector Switch.

e. *APPROACH.* As the Pitch Selector Switch is released after being held at DATUM, it moves to the APPROACH position. This introduces a drift unit into the circuit which modifies the heading error signal before it is mixed with the ILS localizer signal and passed to the ADPs. It eliminates steady heading error signals, thereby automatically compensating for drift changes during the final approach. A system of pitch integration is also initiated by this selection. This adjusts the setting of the pitch scale centre dot from the DATUM setting to that required for the current approach. The adjustment also caters for variations in the head or tailwind component; it takes place automatically provided that the pilot maintains the pitch pointer over the glide path pointer.

Interlocks

12. To prevent incompatible selections being made certain functions take precedence over others:

a. With HEIGHT, APPROACH or DATUM selected, Mode II is inoperative.

b. Selection of Mode III overrides all other pitch functions and returns the Pitch Selector Switch to the Central position. (*See* para 6.)

c. As APPROACH is intended for use during ILS approaches it can remain selected only if GP is also selected.

SUMMARY

Signal Flow

13. The Mk 1B version of the MFS is used in conjunction with Terrain Following Radar (TFR), the Navigation Bombing System (NBS) and the Horizontal Reference System (HRS).

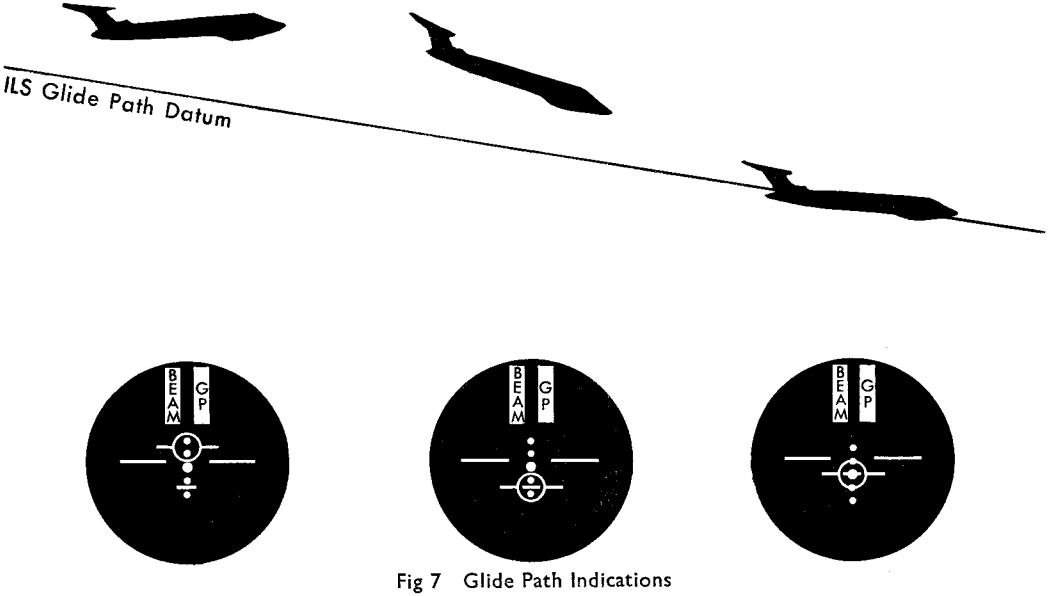


Fig 7 Glide Path Indications

The interconnections with these other systems is omitted from Fig 8 which is a simplified signal flow diagram summarizing the various MFS functions.

In-Flight Use

14. The in-flight use of the MFS is not described here as the standard procedures are specified by the user commands.

Conclusion

15. The MFS was introduced in order to simplify and rationalize the pilot's instrument display, giving all the information necessary to control aircraft attitude in all weather conditions and in all phases of flight. The pilot is able to control the flight path either manually by reference to the flight director display or by means of the automatic pilot, whose performance can be monitored by referring to the same instrument displays.

CHAPTER 3

THE INTEGRATED FLIGHT INSTRUMENT SYSTEM

CONTENTS

	Paras
Introduction	1– 3
Description	4– 6
The F4C Attitude Indicator	7
The Navigation Display	8–11

Introduction

1. Improvements in aircraft performance have led to a number of developments in the concept and design of the flight instruments used by the pilot to manoeuvre the aircraft so that it follows the desired flight path. The function of these instruments is to provide attitude and velocity information efficiently and unambiguously.

2. Autopilots, radar, and weapons systems also require attitude and velocity inputs, and in the past each system was designed with its own sensing element. This led to much duplication of such items as gyroscopes.

3. The Integrated Flight Instrument System (IFIS) is essentially a display system which derives its information from central sensors, which also provide the required inputs for other aircraft systems.

Description

4. The IFIS is provided with information by the following sensors:

a. The MRG 2 twin gyro platform (*see* Part 2, Sect 2).

b. The Air Data Computer (*see* Part 1, Sect 3).

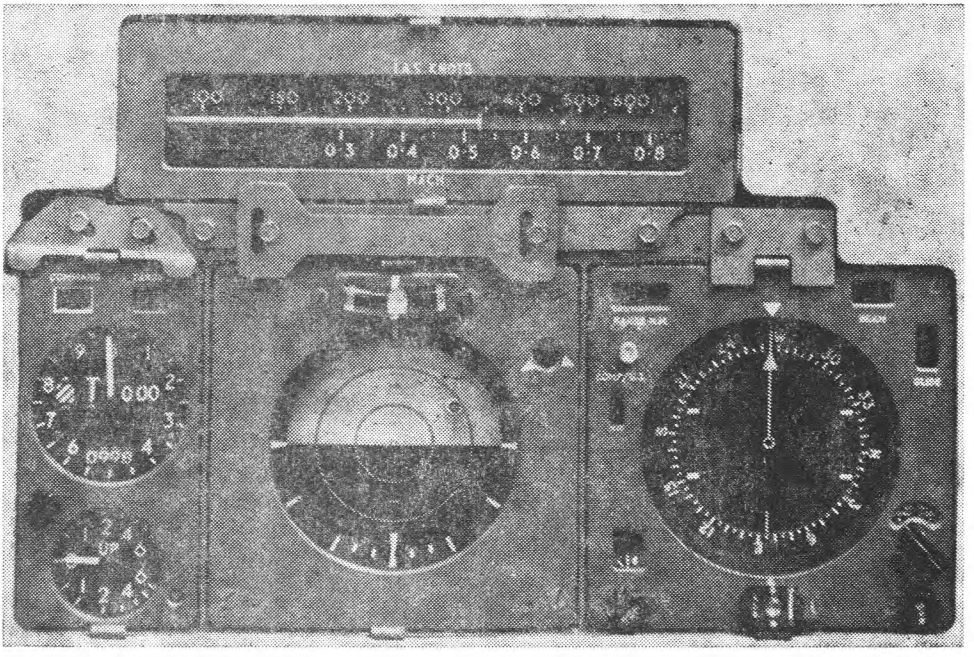


Fig 1 The IFIS Display

c. The TACAN, ILS and UHF equipments, through coupling units.

5. The IFIS display is shown in Fig 1, and consists of the following sub-units:

a. The speed display.

b. The height and rate of climb display.

c. The F4C attitude indicator.

d. The navigation display.

6. The operation of the speed, height and rate of climb displays is described in Part 1, Sect 3 in the chapter dealing with the Air Data Computer, and is therefore not repeated here.

The F4C Attitude Indicator

7. The F4C attitude indicator provides a continuous display of aircraft attitude in the pitch and roll axes relative to the horizon. The display is driven by roll and pitch servomotors controlled by the relevant outputs of the MRG 2, and consists of a roller blind which is divided into two sectors representing the earth and the sky. The centre spot on the indicator represents the aircraft, and concentric rings are calibrated in 20° increments of pitch. Roll angles are calibrated about the lower edge of the dial in 30° divisions, with 10° sub-divisions in the first 30°. A rotatable needle indicates the roll angle against the roll scale. The roller blind gives full freedom of display in both roll and pitch. Above the indicator there is a ball in tube slip indicator. A flight director bead, which is actuated by the flight control system, appears on the face of the indicator.

The Navigation Display

8. The navigation display gives a continuous indication of heading derived from the MRG 2. The gyro output can be slaved to a fluxvalve monitoring signal, or initially aligned with magnetic heading and used as a direction indicator. The heading display can be combined with any one of three functions:

a. ILS

b. Violet Picture

c. TACAN.

9. **ILS Function.** When the mode selector switch is set to ILS, glide path and localizer beam displays are shown. The glide path is represented by a horizontal bar which moves vertically up and down relative to the centre spot on the instrument to

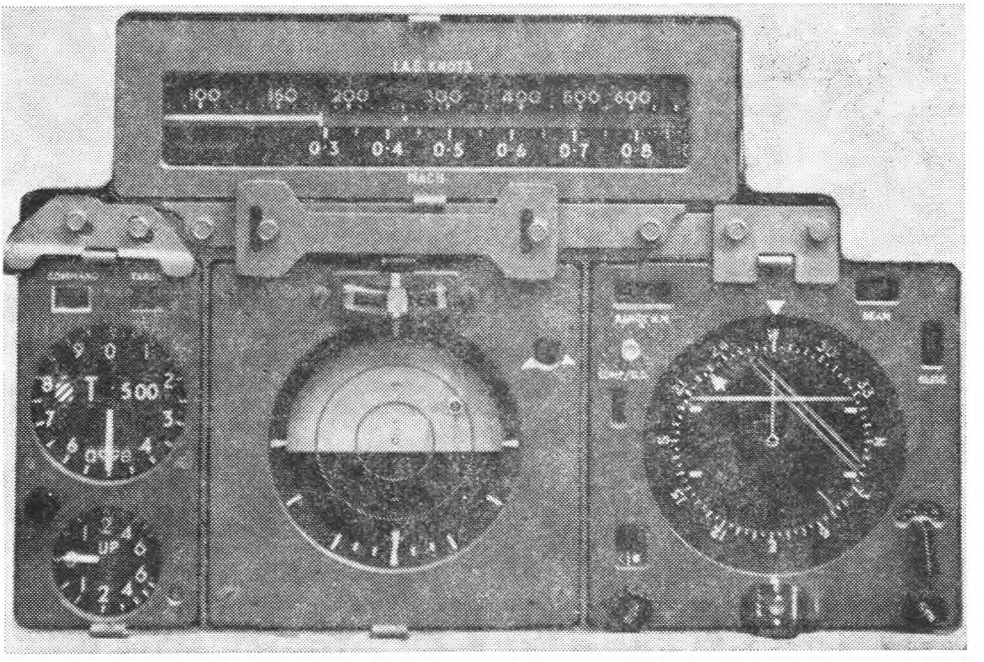


Fig 2 ILS Display—Homing to the Centre Line

indicate the position of the glide path in relation to the aircraft in elevation. The localizer beam is represented by a pair of parallel lines which can rotate as well as move laterally. The lines indicate the orientation of the glide path relative to aircraft heading, and the lateral displacement of the aircraft from the beam centre line. A QDM marker can be set on the face of the instrument by pulling out and rotating the set heading knob. Fig 2 shows the aircraft approaching the ILS localizer on a heading of 270°. Runway QDM is 226° and the desired heading arrow is also set to 226°. The aircraft is still below the glide path. Fig 3 shows the aircraft heading down the localizer centre line on the glide path. Beam and glide path amber lights are covered by shutters if the individual systems are serviceable. A blue light at the base of the display is actuated by the inner and outer ILS markers.

10. **TACAN Function.** The TACAN display appears when the mode selector switch is set to TACAN and the appropriate channel has been selected on the TACAN equipment. The display is shown in Fig 4, and consists of a series of concentric arcs calibrated in increments of 20 miles range from the beacon. In the TACAN mode range also appears in the range window. A line bisecting the range arcs indicates the magnetic bearing of the beacon when read against the compass card.

11. **Violet Picture.** UHF homing signals are displayed by the ILS localizer indicator bars when ILS mode is selected with the ILS/Violet Picture selector set to Violet Picture, provided that the appropriate UHF frequency is selected.

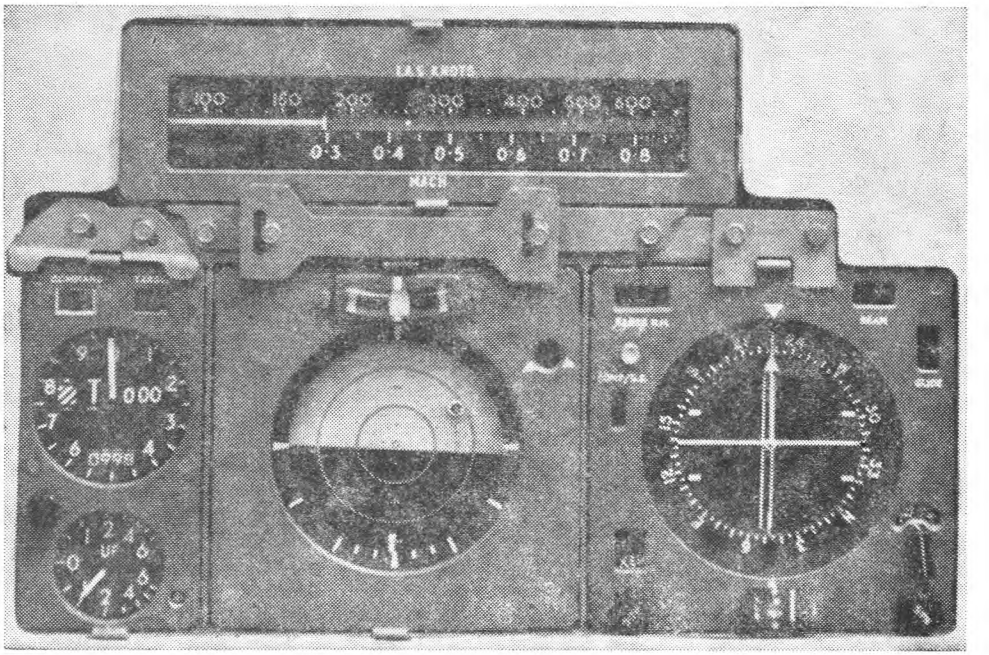


Fig 3 ILS Display—Final Approach

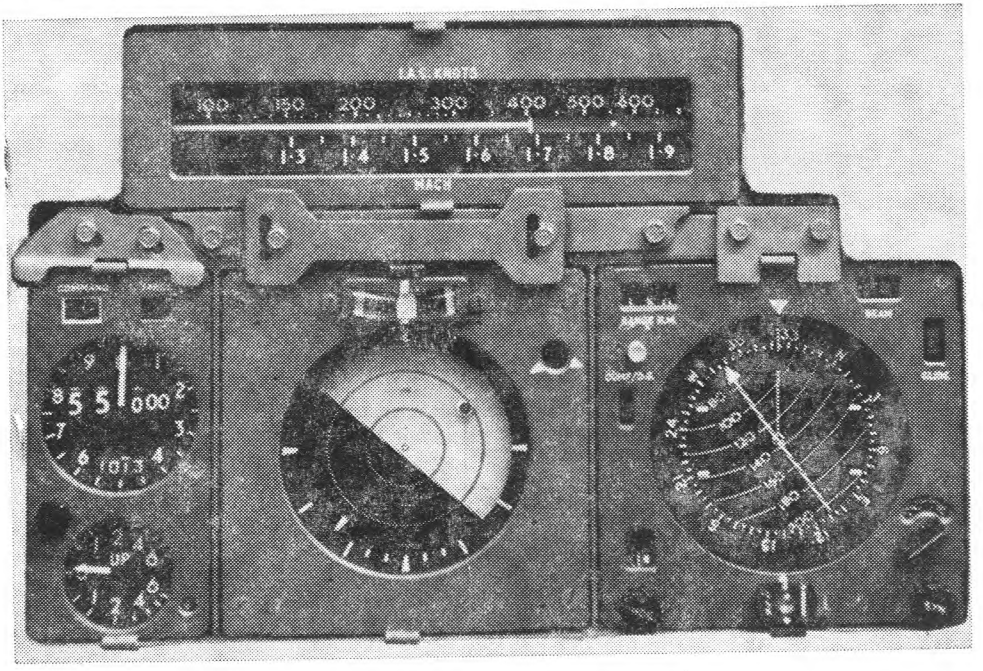


Fig 4 TACAN Display

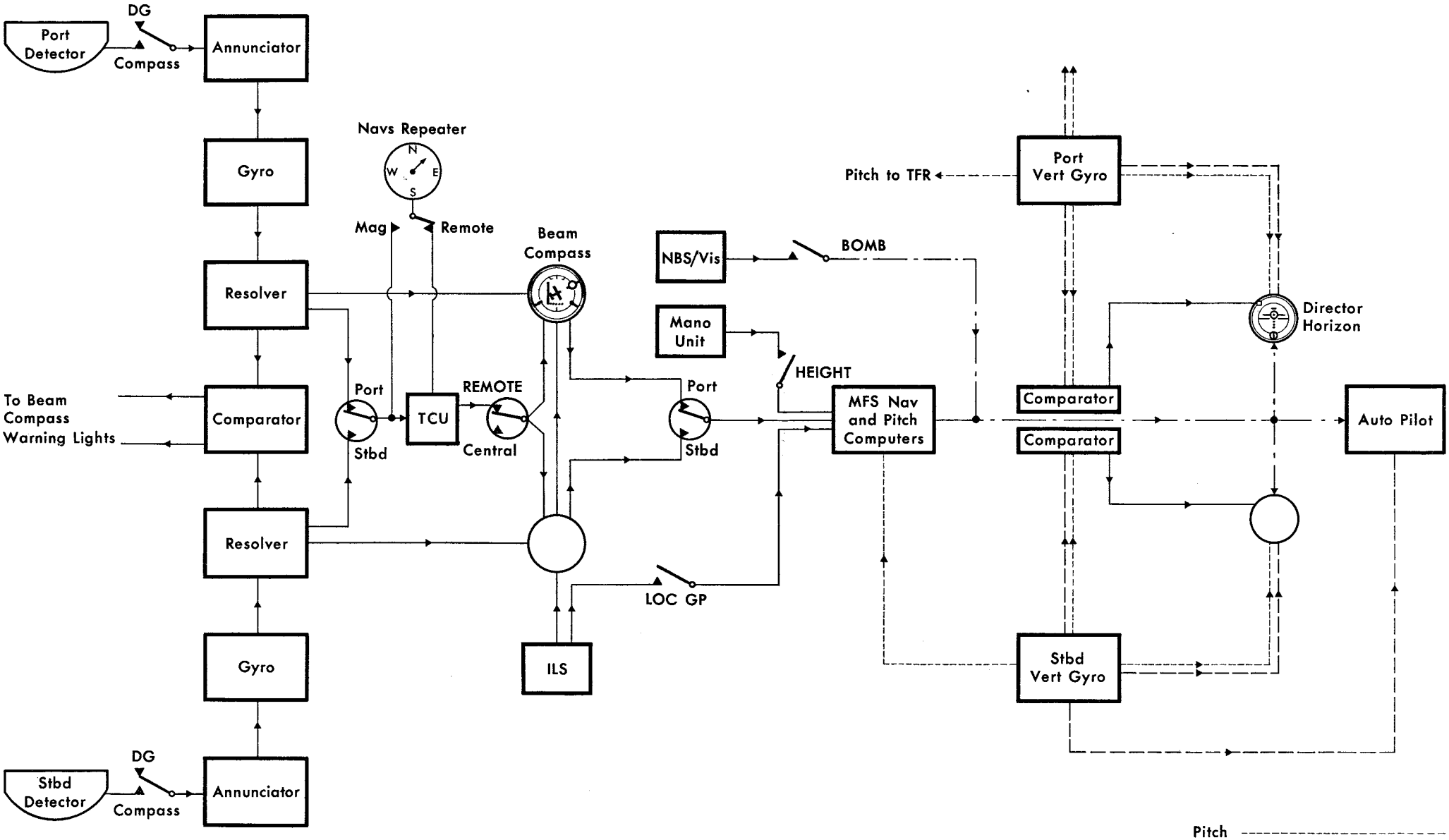


Fig 8 The MFS Signal Flow Diagram

CHAPTER 4

THE HEAD-UP DISPLAY

CONTENTS

	Paras
Introduction	1 – 4
Description	5 – 9
Current Display Format	10 – 14
In-Flight Use	15 – 17
Future Development	18

Introduction

1. The Head-Up Display (HUD) is an integrated display system which provides the pilot with the information he requires to control the aircraft during each phase of flight.

2. The information is selected by means of a control unit for display on a cathode ray tube (CRT) and is viewed by reflection using a semi-transparent glass reflector plate. The image seen by the pilot is optically focussed at infinity (collimated) and interposed in the line of sight ahead of the aircraft.

3. The HUD enables the pilot to monitor the outputs of the aircraft sensors and computers without the necessity of refocussing his eyes when transferring his attention from the outside world to the display. This confers particular advantages during weapon delivery and approach and landing phases of flight.

4. The display is compact, and although head-down instrumentation is still required, there is a considerable reduction in the demand on cockpit panel space.

Description

5. The equipment used to produce the display is shown diagrammatically in Fig 1. It should be noted that a number of different types of equipment are currently being developed but the units shown are fairly typical. The basic HUD units are:

a. The waveform generator.
b. The control unit.
c. The display unit.

6. **The Waveform Generator.** The function of the waveform generator is to provide the necessary X and Y deflection waveforms to produce the required CRT display. In order to achieve flexibility in the form and scaling of the display it is usual to hold the waveforms in a matrix store as a binary code. The selected store output can then be passed through digital to analogue convertors to produce the required voltage outputs which then form the symbol display on the CRT. Where sensor information is required in numerical form, digitizer circuits retrieve the appropriate digit codes from store and pass them to the convertors for display on the CRT. A deflector amplifier is used to shift the symbols on the CRT display in accordance with sensor and demand signals from selected inputs.

7. **The Control Unit.** The control unit is fitted in the cockpit and is used to select the display required for the particular phase of flight in progress. In most installations the manual selections are used to actuate a computer control programme in order to reduce the number of manual selections required for each mode. The main controls found on a typical unit are:

a. An ON/OFF switch.
b. A mode selector.
c. A brightness level selector.
d. A test facility.

8. **The Display Unit.** The display unit is mounted in the cockpit and consists of the following sub-units:

a. The CRT assembly with its associated extra high tension (EHT) powerpack and photo-electric brightness control.
b. The collimating lens unit.
c. The reflector.

The high intensity CRT uses magnetic deflection from two pairs of X and Y coils to display the symbols. The brightness of the displays relative to the background is automatically maintained at the manually-selected level by the photo-electric control. The reflector unit is an optically flat plate with parallel faces; one face is coated with a neutral density reflecting medium and the other is bloomed to reduced internal reflections.

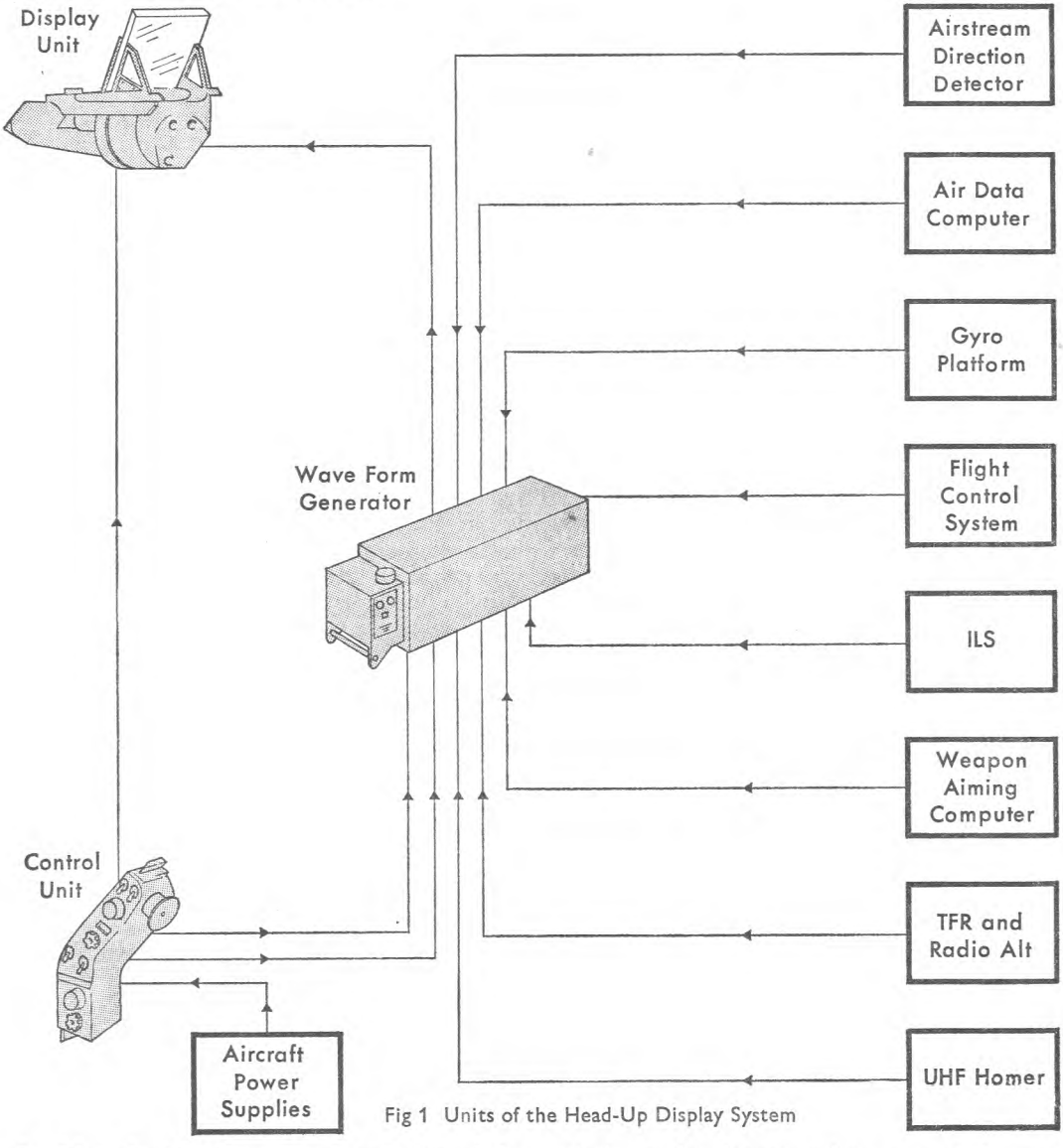


Fig 1 Units of the Head-Up Display System

9. **Method of Display.** The installation of the display unit in a single-seat aircraft is shown in Fig 2, which also shows the optical arrangement. The field of view provided by the display for a pilot's fixed head position is usually some 18° in azimuth and 21° in elevation, but with some units having two-position reflectors this can be increased to about 30° overall. The pilot must position his head so that his eyes are within the designated viewing area, otherwise all the display will not be visible to him. In installations where the entire display may move through a large angle in the pitch plane, such as in attack modes incorporating a symbol overlaying a target on the ground (*see* Fig 12), the field of view may be extended by servo-driving the reflector fore and aft. There is more freedom in the positioning of the display unit in an aircraft with a solid roof to the cockpit since it may then be positioned either below or above the windscreen. This latter form of mounting will overcome one of the problems of the low-mounted unit, that of the internal reflections of sunlight, which is only partially overcome by the use of a fine mesh graticule mounted on the collimating lens.

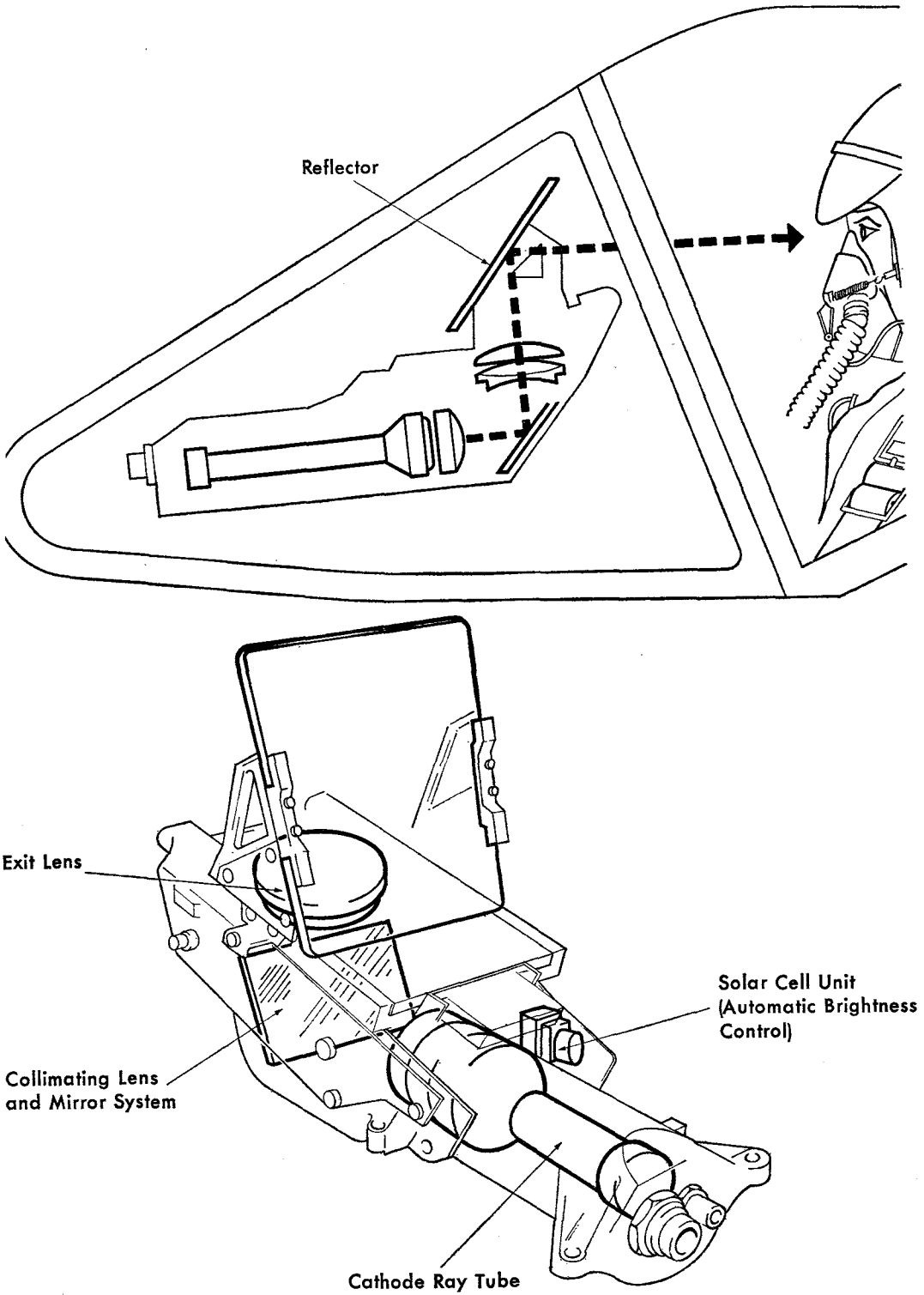


Fig 2 The Display System

Current Display Format

10. The display described in the following paragraphs is based on the Specto unit fitted in the Harrier GR Mk 1. It is emphasized that there may well be changes yet to come in HUD techniques as technology improves and experience is gained in what is still a new aspect of avionics.

11. The data displayed on the HUD are normally chosen from the following:

a. Aircraft attitude and climb/dive angle (CDA).

b. IAS and IAS error or IMN and IMN error.

c. Height.

d. Vertical speed.

e. Angle of attack (airstream direction detector (ADD) reading).

f. Heading.

g. Sideslip.

h. Director of homing information.

j. Approach information (*eg* ILS).

k. Weapon delivery information.

12. Additionally, the display as a whole, while still showing the required flight information, may be moved so that the position of the aircraft symbol viewed against the outside world shows the point in space through which the aircraft is about to pass. This is known as showing the aircraft's velocity vector and this mode may be used in the terrain following, attack or landing phases.

13. Clearly, not all the items in para 11 will be displayed at once. One of the advantages of such electronic displays is that pre-determined combinations of symbols can be selected by a simple mode switching arrangement without the need for displaying information that is only intermittently required. Thus approach, director or attack information would only appear for the appropriate phases of flight.

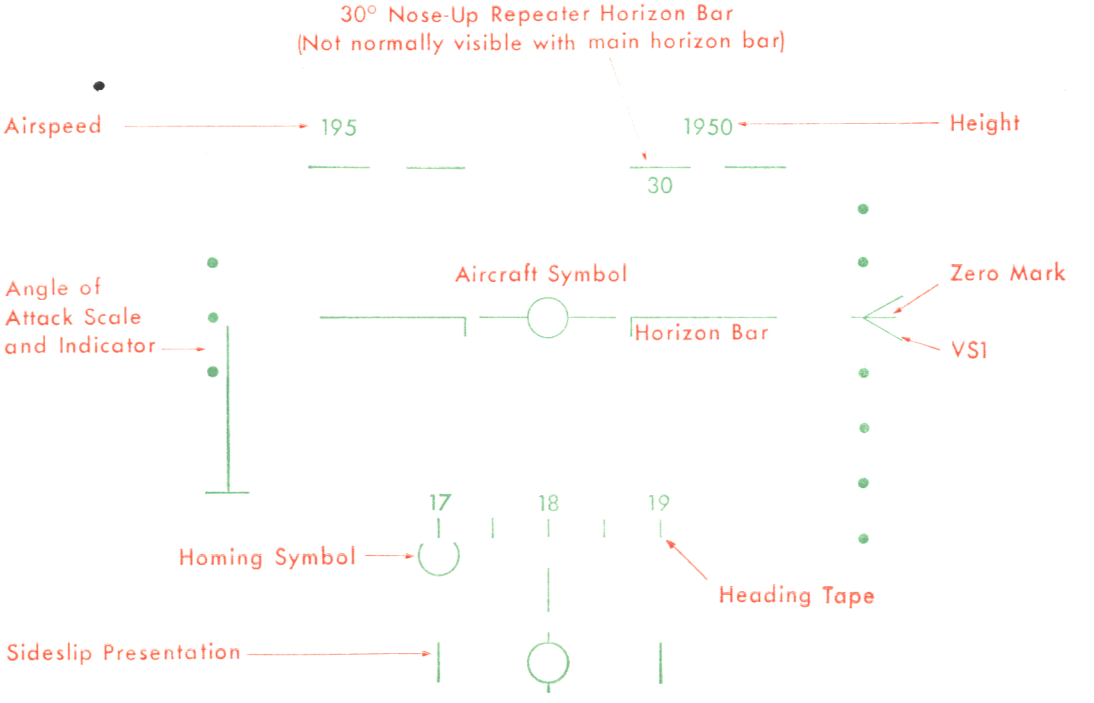


Fig 3 Typical Display

4. A typical display is shown in Fig 3. This resents all the information needed for normal onventional flight, as follows:

a. *Attitude Display*. The attitude display consists of an aircraft symbol and horizon bar; this is the same format as the display on a conventional artificial horizon. However, aircraft's longitudinal fuselage datum (LFD) and the horizon (a normal attitude display) then the aircraft symbol will be the equivalent of 5° above the horizon bar, as shown in Fig 4.

(2) *Attitude Display (CDA)*. If the display is made to show the aircraft's climb or

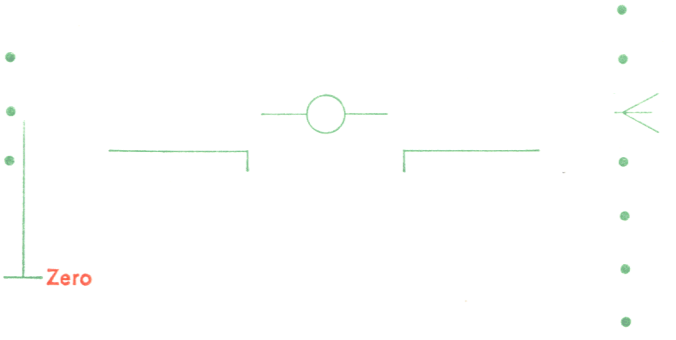


Fig 4 Attitude Display LFD

the angle between the aircraft symbol and horizon bar can equally well be made to show the climb or dive angle (CDA) of the aircraft. An aircraft flying level with, say, a 5° nose-up attitude can therefore be shown in two ways:

(1) *Attitude Display (LFD)*. If the display is made to show the angle between the dive angle (CDA) relative to the horizon then the horizon bar and the aircraft symbol are shown level. As the aircraft is in a 5° nose-up attitude but flying level, the horizon bar is deflected up by a corresponding amount, as shown in Fig 5.

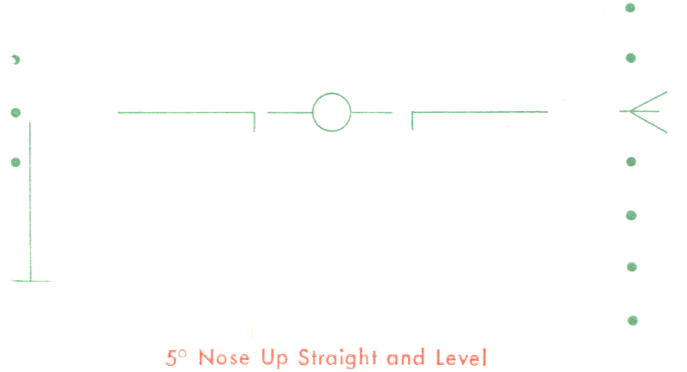


Fig 5 Attitude Display CDA

The HUD pitch indications are normally scaled down 5:1 relative to the real world for fighter-type aircraft (but for some uses where only a limited pitch range is required it may be possible to use a 1:1 ratio). This ratio, sometimes known as the gearing, means that any pitch angles of the aircraft are reduced to $\frac{1}{5}$ on the HUD. For the example given in sub-para a(1) above, the distance between the aircraft symbol and the horizon bar would subtend an angle of 1° at the pilot's eye. In conventional attitude displays, where the ratio is about 15:1, the angle seen by the pilot would be only $\frac{1}{3}$°. This increase in sensitivity of the HUD over mechanical attitude displays is an advantage which enables much more accurate changes and corrections to be made. The use of a 1:1 ratio in manoeuvreable aircraft, while it is even more accurate, would result in rapid loss of the horizon bar from the display which, as seen in para 9, normally has a range of 12° and has a maximum of 25° with pilot head movement. When the main horizon bar disappears from view due to the aircraft's manoeuvring in pitch, repeater pitch bars appear at appropriate intervals of, say, 30°, marked to show their displacement from the true horizon bar. Zenith and nadir stars indicate the vertical extremes of pitch attitude, one point of the star being extended to show the direction of the nearest horizon. The horizon bar itself may be fitted with legs of a convenient size such that when flying in the CDA mode, by maintaining the aircraft symbol at the bottom of the legs, a required angle of descent is achieved. Legs equivalent to a 3° descent angle are used on most installations (*see* Fig 6). In roll the display moves at a 1:1 ratio with the real world, as do all attitude indicators.

b. *Air Speed/Mach Number.* Either air speed or Mach number, selected by the pilot, is normally presented numerically at the top left of the display. IAS can be read to 1 kt and the IMN to 0·05M. To avoid blurring in turbulence the information is usually updated at half-second intervals. The digital ASI uses the minimum of space and is virtually impossible to misinterpret; however, it must be positively read to extract the information and it lacks the rate information that can be acquired from a rotating pointer or moving tape. To offset this deficiency a speed error scale can be selected beneath the IAS or IMN digits; this is a three dot and pointer presentation (*see* Fig 6), the centre dot representing the datum IAS or IMN selected by the pilot. If the pointer is to the right it shows IAS above datum and if to the left, IAS below datum. Thus it is possible to give speed rate information in a form that can be interpreted in a rapid scan pattern.

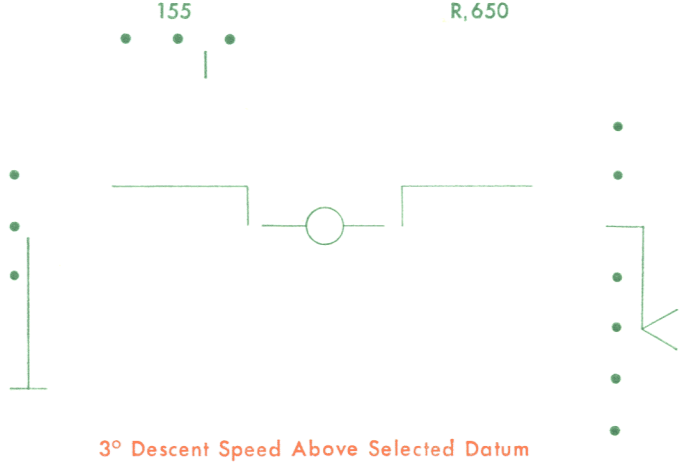


Fig 6 Attitude Display 3° Descent

c. *Height.* Height, which may be from either barometric or radar sensors, is shown numerically at the top right of the display. The millibar setting knob is on the pilot's control panel. The digits may change at either 50 or 20 ft intervals and to avoid blurring in turbulence the information is usually only updated at half-second intervals. As with the IAS display, the height digits lack obvious rate information but the VSI scale is used to provide this. A height demand facility may be provided in which the flight director is used to give pitch demands to enable the pilot to acquire and maintain his pre-selected height. During an approach the height digits or the flight director may be made to flash to remind the pilot that he is close to break-off height.

d. *Vertical Speed.* Vertical speed is shown by a "thermometer" scale to the right of the attitude presentation (*see* Fig 3). The zero datum is level with the aircraft symbol and the thermometer extends upwards to show rate of climb and down to show rate of descent. The scale is linear and each interval shown by a dot represents 500 fpm. To reduce clutter on the HUD the intervals are not numbered. The position of the end of the thermometer scale is emphasized with a horizontal arrowhead and the extending scale will usually saturate at values of 1,000 fpm greater than the scale provided.

e. *Angle of Attack.* The angle of attack is shown by a thermometer scale on the left of the attitude display with a 0° datum below the level of the aircraft symbol, so that the thermometer scale is level with the aircraft symbol when the angle of attack is at its optimum approach value (*see* Fig 6). The scale is usually graduated at 8, 12 and 16 units of angle of attack.

f. *Heading.* Heading is shown by a moving tape at the bottom of the display which is read against a central stationary mark. The tape is scaled down 5:1 in relation to the real world and the equivalent of 30° of heading is visible at any one time. Five heading markers are always visible but the number of sets of digits may be either two or three, depending upon the heading. The heading director is a O symbol (*see* Fig 7).

g. *Sideslip.* Sideslip can be shown by a suitable moving symbol read in conjunction with a fixed mark which can be the same one as used in the heading display (*see* Figs 3 and 11). Limits can also be shown and the sense of movement of the free symbol is the same as the conventional slip "ball".

h. *Director.* Director demands can be shown on a HUD by using a "pyramid" type of director (*see* Figs 8 and 9) which when used in the normal zero reader sense gives demands in pitch and roll. Desired heading can be indicated by the superimposition of a steering bug on the heading tape.

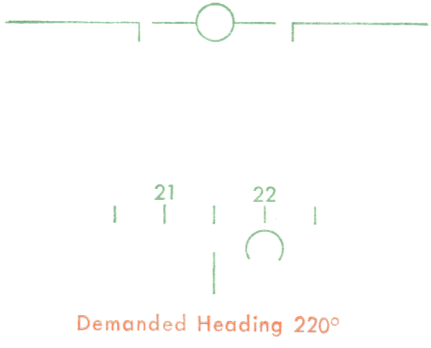


Fig 7 Heading Director

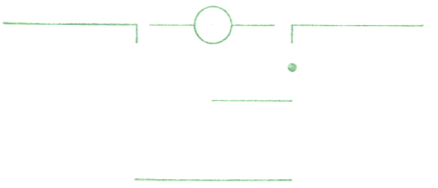


Fig 8 Pyramid Flight Director Unsatisfied Demand

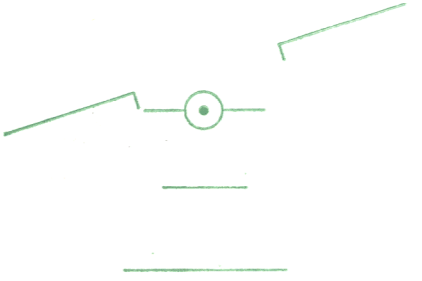


Fig 9 Pyramid Flight Director Satisfied Demand

j. *Approach Information.* ILS information can be presented "raw" by the use of an ILS cross which indicates both azimuth and glide path errors (*see* Fig 10). The ILS information can also be displayed by the flight direct (*see* Fig 11). UHF homing indications are giv by a O symbol superimposed on the headi scale (*see* Fig 3).

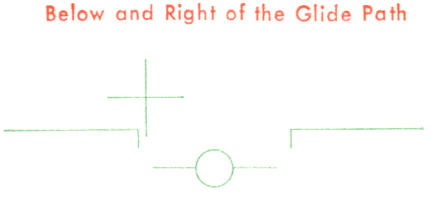


Fig 10 Raw ILS Display

Fig 11 ILS Flight Director Display

k. *Weapon Delivery*. The presentation of attack information is the most important aspect of a HUD in a strike or combat aircraft. The form of presentation will differ widely depending upon the weapon and delivery technique used. The flexibility of a HUD allows many different displays to be used and each one can be rapidly selected. For details of a specific attack system reference should be made to the appropriate Aircrew Manual.

In-Flight Use

15. The use of the HUD by the pilot is rapidly learned. Despite the small area of the display it it necessary to scan each individual piece of flight information to absorb it; it is impossible to assimilate the total flight information with just a glance. It is particularly necessary to read the digital presentations of air speed and height and to convert the figures read into meaningful quantities.

16. The weakness of the impact of the information presented at the periphery of the HUD, when compared with conventional electro-mechanical instruments, is compensated for by the increased sensitivity of the attitude display. In particular, when used in the CDA mode it allows extremely precise control of the aircraft's flight path to standards comparable to those obtainable with conventional head-down instruments.

17. Despite its novelty, by making it easier to fly precisely and because of its great flexibility, the HUD soon becomes readily acceptable to the pilot.

Future Development

18. Future development of the displays may well include:

a. Improvement of presentation beyond the current use of lines, numbers and letters which have evolved after fairly limited flight experience in an area where technology has lagged behind the designer. These improvements may include the use of colour. It may be found necessary to display speed and height in an analgue form which, although creating greater clutter in the HUD, may be more acceptable than is at present thought.

b. Integration of HUD with electronic head-down instruments. These can be displaced on a directly-viewed CRT using the same principle as a HUD, that is, a moving spot which actually writes each symbol. They can also use the TV principle which displays the image on a raster scan. Such a principle allows cloud warning radar, navigation information, flight information, combinations of these or any other desired display to be selected at will.

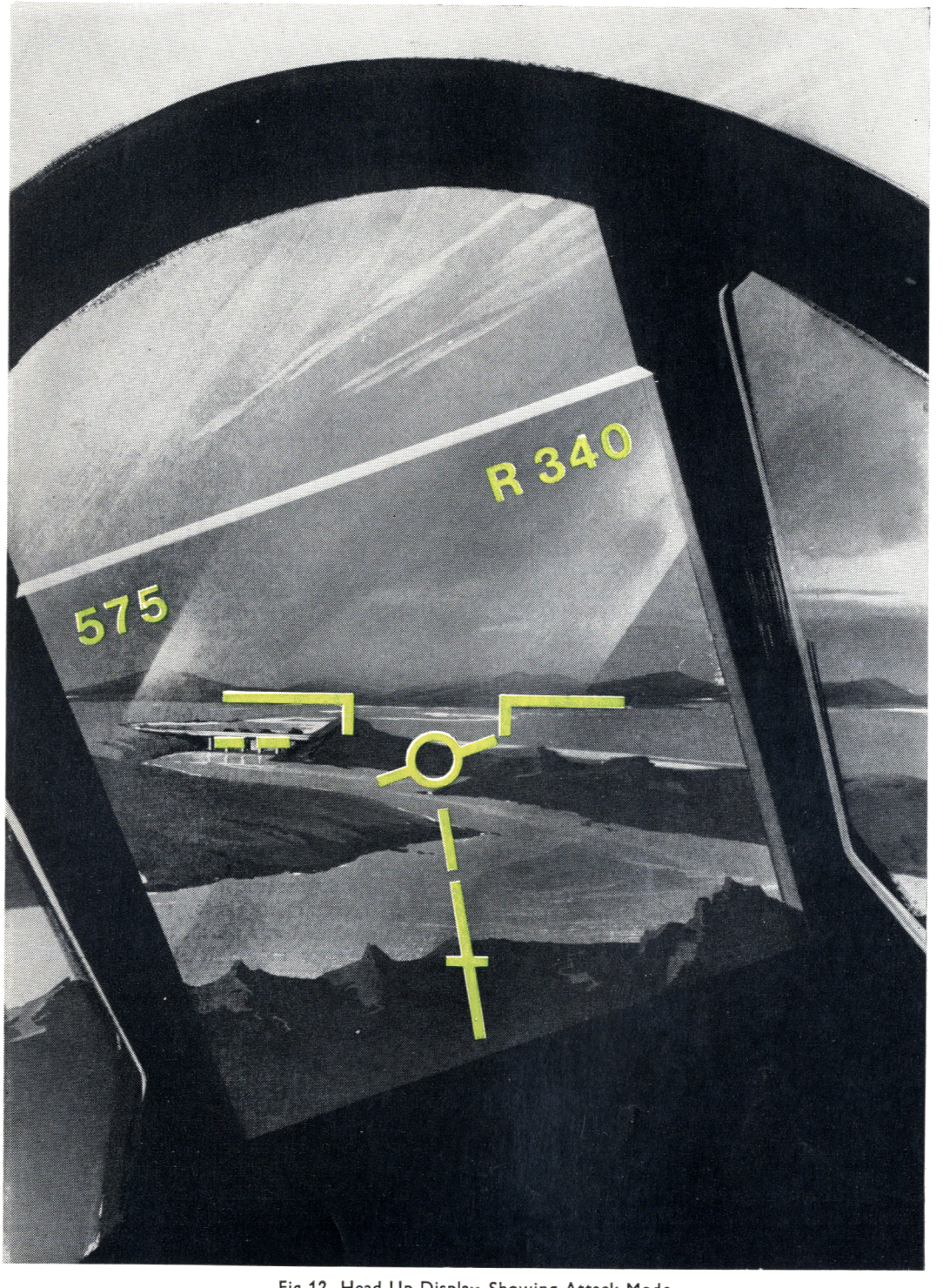


Fig 12 Head-Up Display Showing Attack Mode

PART 2

SECTION 5

ALIGNMENT INSTRUMENTS

Chapter

1	**Medium Landing Compass**
2	**Pattern 2 Prismatic Compass**
3	**Watts Datum Compass, Marks 1 and 2**
4	**MC-1M Magnetic Compass Calibrator**
5	**The Precise Heading Test Set**

CHAPTER 1

MEDIUM LANDING COMPASS

CONTENTS

	Paras.
Introduction	1
Description...	2– 6
Operation	7
Accuracy	8

Introduction

1. The Medium Landing Compass is a small portable bearing compass, having a high degree of accuracy, which can be used for ground calibration of aircraft compasses. It is designed for use on a tripod and is fitted with a bubble level. The horizontal compass card is read through a prism which forms part of the sighting head.

Description

2. The construction of the compass, whilst following the general principles of aircraft magnetic compasses, differs from them owing to its being designed solely for use on the ground. The simple design does not incorporate a container for the compass bowl and no anti-vibration suspension is provided.

3. A metal compass bowl, which forms the body of the instrument, is filled with liquid. The magnet system, which consists of a float carrying two parallel bar magnets and the compass card, is supported on a pivot attached to a bridge in the compass bowl, by a jewelled bearing attached to the float. The bowl is sealed and the magnet system retained in position by the verge glass. Changes in the volume of the compass liquid due to variation in temperature are accommodated by a diaphragm.

4. The sighting head consists of a rotatable ring carrying a folding frame with a vertical foresight wire and a prism mounting with a vertical backsight slot. The wire and slot are used to sight the object and the bearing of the line of sight is then read through the prism. The compass card is graduated every degree with 10 degree intervals marked in mirror image. Reading the card through the prism reverses the figures, which then appear normal.

5. The tripod head is a brass casting with a protrusion which fits into a socket on the base of the compass. The two are then locked together

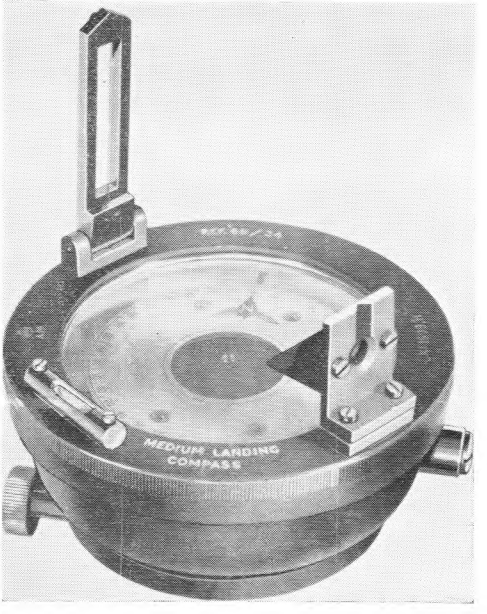


Fig. 1. Medium Landing Compass

by tightening the knurled screw on the side of the compass. The tripod has pivoted wooden legs which can be folded up and fastened in a compact form. The ends of the legs are pointed to give a firm grip in the ground when in use.

6. Each medium landing compass is stored in its own wooden carrying case. There are two types of case, one in which the compass is stored in an upright position, and the other in which the compass is stored upside down. The second type of case is designed to keep the weight of the magnet system off the pivot during storage, thus minimizing wear.

Operation

7. When taking the bearing of a line of sight with the medium landing compass the procedure

is as follows:—

(*a*) Extend tripod legs and attach compass firmly to tripod head.

(*b*) Set up tripod on extended datum line of aircraft with one leg of tripod pointing away from operator.

(*c*) Level the compass. This is done in three steps:—

(i) Turn the sighting ring until the bubble level is parallel to the fore-and-aft line. Level the tripod by moving the forward leg of the tripod until the bubble is central in the level.

(ii) Move the sighting ring through 90° and level by moving rear legs to centralize bubble.

(iii) Realign with fore-and-aft axis and check level.

(*d*) Sight object required by looking through slot and prism. Adjust sighting ring until object, foresight wire and backsight slot are in line.

(*e*) Read compass card scale through prism where foresight wire appears to touch compass card.

Accuracy

8. The medium landing compass is tested to an accuracy of half a degree. To ensure that this is maintained, the compass must always be kept in its box when not in use and the compass must be periodically inspected as detailed in the chapter on Pilot Type Compasses. In the medium landing compass an error of up to 1 degree due to pivot friction is permitted.

CHAPTER 2

PATTERN 2 PRISMATIC COMPASS

CONTENTS

	Paras
Introduction	1
Description	2—5
Operation	6—8

Introduction

1. The Pattern 2 Prismatic Compass is a small portable bearing compass suitable for use in aircraft compass calibration. Each compass is calibrated and a calibration chart showing the errors is carried in the lid of the storage box.

Description

2. The compass is similar in construction to the medium landing compass described in Chapter 1. The compass bowl is filled with liquid and contains the bridge-supported pivot on which rests the magnet and float assembly. Two parallel bar magnets are pendulously suspended from the float and the artificial sapphire bearing in the centre of the float rests on the osmium-iridium pivot. A bellows in the bottom of the bowl allows for variation in the volume of the liquid due to temperature changes. The base plate covers the bellows and provides a socket for tripod mounting.

3. The compass card, graduated in half degrees, is mounted on pillars attached to the magnets. A verge glass and rubber sealing ring seal the bowl and are held in position by the verge ring which is fastened to the bowl by screws.

4. A foresight, consisting of a nylon sighting line fitted in a folding sighting frame, and a prism and slot backsight are fitted to the verge ring. A circular spirit level, which enables the compass to be set level in all directions, is also fitted to the verge ring.

5. The compass fits on a tripod which has a ball and socket assembly for quick levelling. The base plate fits on the spigot of the ball which is secured in its socket by a clamping ring. The compass is then free to rotate on the spigot although secured by the compass clamping screw. A plumb line and bob is supplied for centralizing the compass over any selected spot, and this is attached to a hook at the centre of the tripod.

Operation

6. The operating procedure when using the compass is as follows:

a. Set the tripod central over the selected spot using the plumb line and bob suspended from the hook on the tripod.

b. Ensure that the mating faces of the compass base plate and tripod spigot flange are free from burrs so that the compass will sit truly on the flange.

c. Secure the compass to the tripod by the clamping screw and check that the compass rotates freely on the spigot.

d. Level the compass by adjusting the ball and socket joint and referring to the circular spirit level.

e. Raise the sighting frame vertically and, sighting through the hole in the backsight, line up the sighting line with the aircraft datum marks.

f. Settle the compass card by gently tapping the verge glass

g. Read the bearing through the prism at the point where the sighting wire appears to cut the scale.

7. **Inspection.** The Pattern 2 Prismatic Compass should be inspected periodically as laid down for magnetic compasses in Part 2, Sect 1, Chap 3, paras 16-18.

8. **Servicing.** The compass is to be returned by hand to the Admiralty Compass Observatory (ACO) every 12 months for recalibration (AP 3158, Vol 2, Leaflet L refers).

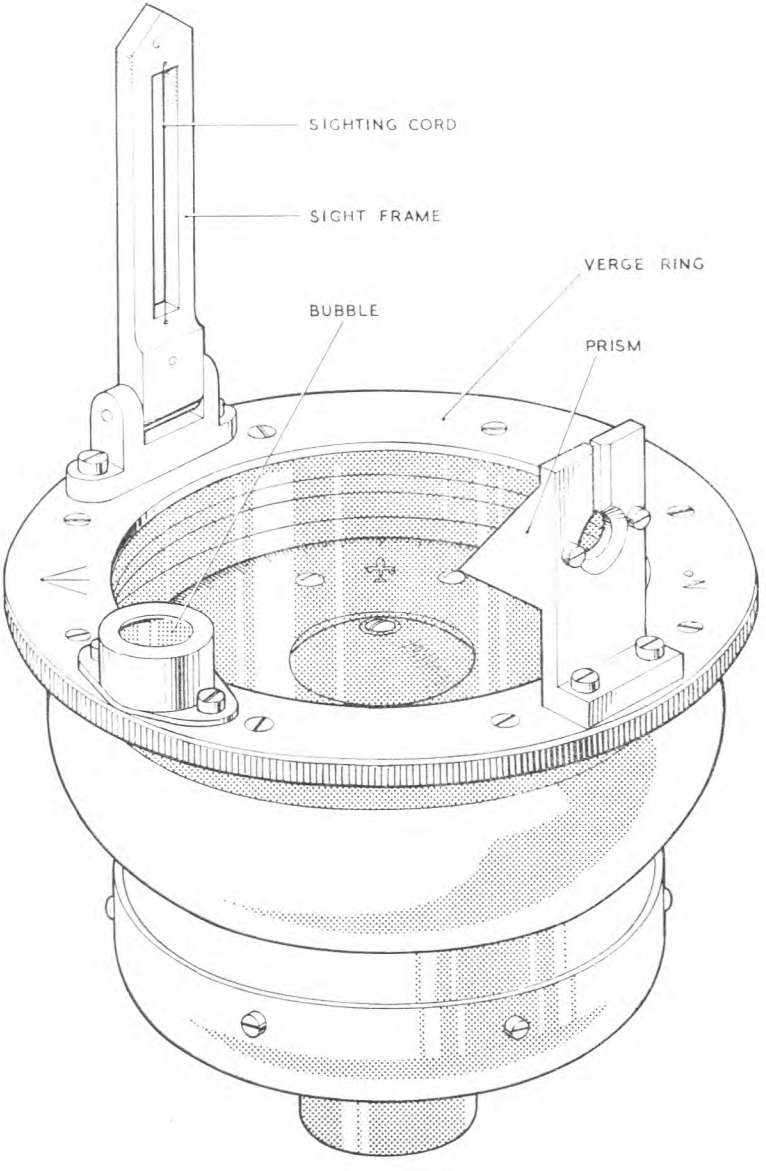


Fig 1 Pattern 2 Prismatic Compass

CHAPTER 3

WATTS DATUM COMPASS, MARKS 1 AND 2

CONTENTS

	Paras
Introduction	1–2
Principle	3
WATTS DATUM COMPASS, MARK 1	
General	4
Compass	5–8
Sighting Telescope	9–10
Azimuth Circle	11–13
Base	14
Instrument Case	15–17
Tripod	18–19
WATTS DATUM COMPASS, MARK 2	
General	20–22
Compass	23–24
Sighting Telescope	25
Azimuth Circle, Upper and Lower Casing	26–28
Instrument Case	29
Tripod	30
ADJUSTMENTS	
General	31
Periodic Collimation	32
Adjustment of Spirit Levels	33
Adjustment of Levelling Screws	34
Adjustment of Compass for Dip	35
OPERATION	
General	36–38
Setting Up	39
Take a Bearing	40
Casing the Instrument	41–42
ACCURACY	
Alignment	43
Accuracy of Compass Deviation	44–45

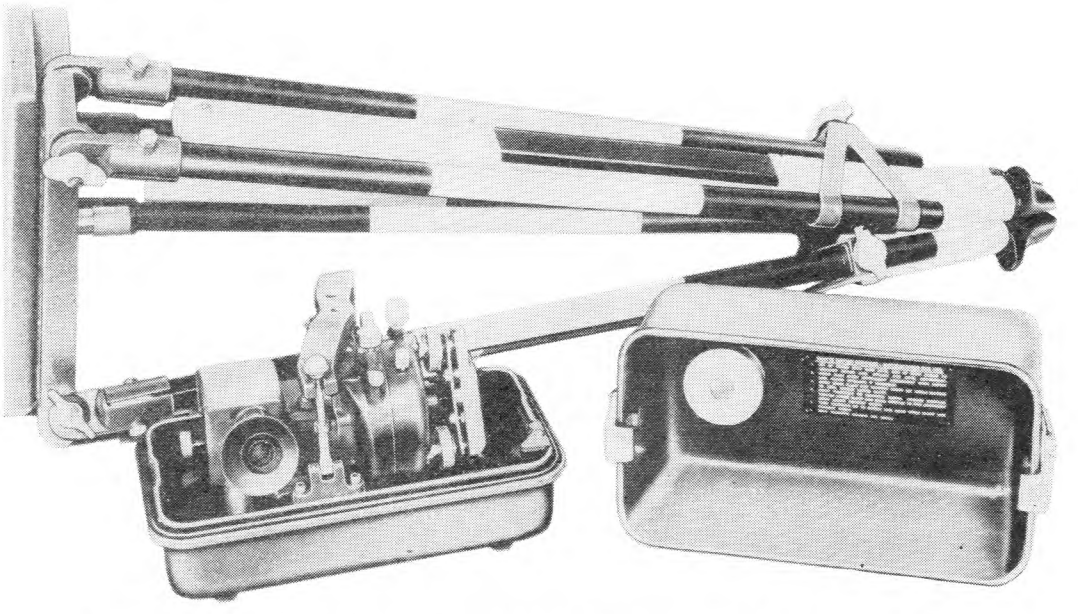


Fig 1 Mk 1 Instrument Clamped in Case (Cover Removed) and Tripod with Protective Cap

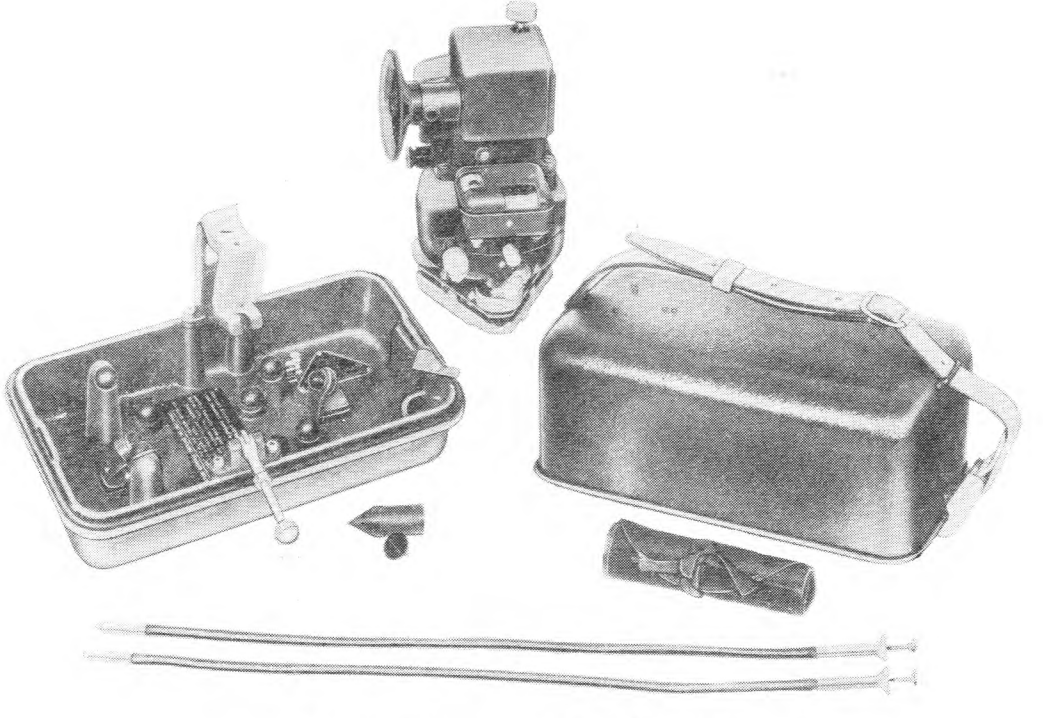


Fig 2 Mk 1 Instrument, Case and Accessories

Introduction

1. The Watts Datum Compass is a high precision instrument for use where extreme accuracy of alignment is required. It is used for the ground calibration of aircraft compass system, where accurate heading information is required to be fed to navigation equipment, and for the alignment of the aerial system of certain airborne radio installations. There are two marks of instrument; the Mark 2 instrument is very similar to the Mark 1, but incorporates internal lighting in the form of luminescent "Betalights" and sealing and shielding of the instrument to make it showerproof. These improvements allow the Mark 2 instrument to be used in poor light conditions and moderate rain.

2. Both marks of equipment consist of the following units:

a. Instrument
b. Instrument case and accessories
c. Tripod
d. Tripod head protective cap

The total weight of the equipment is 30 lb. The Mark 1 equipment is shown in Figs 1 and 2.

Principle

3. The instrument consists essentially of a compass system, a bearing plate, and a sighting telescope. The compass system accurately defines the magnetic meridian and the bearing plate is then aligned with, and locked to this meridan.

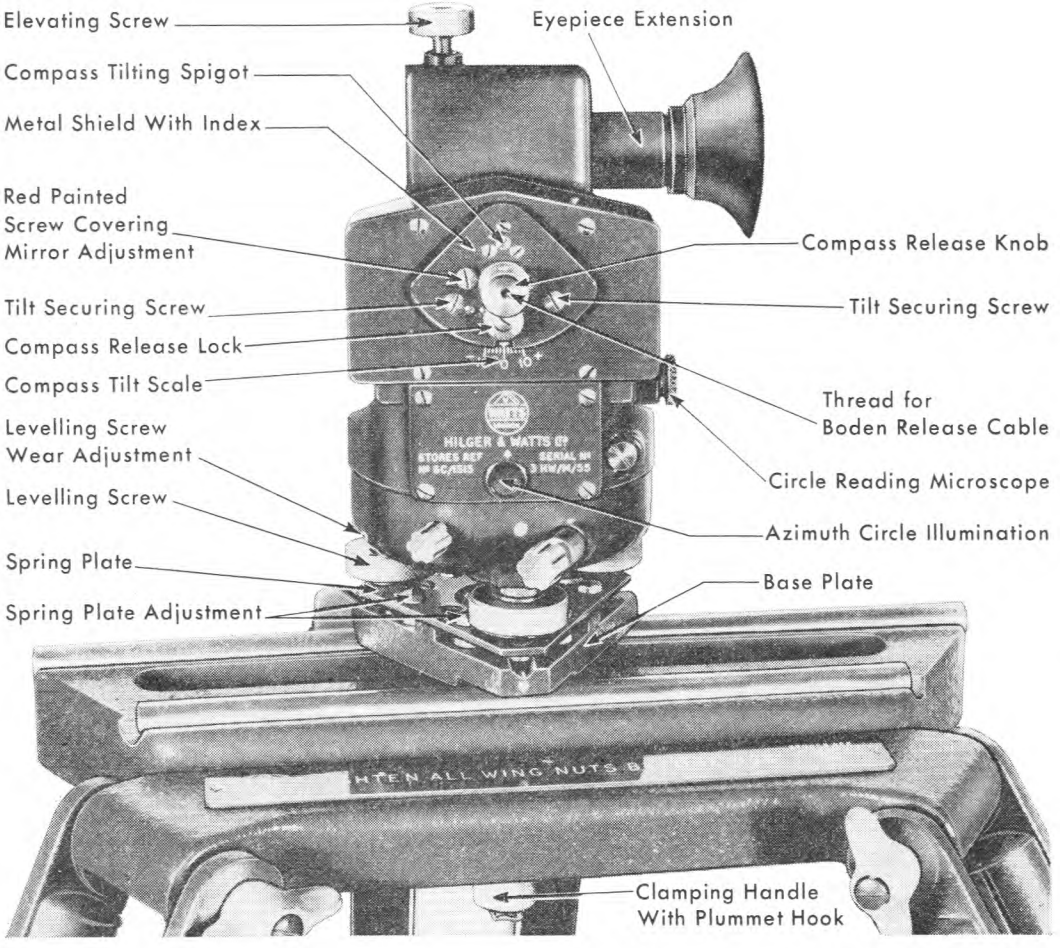


Fig 3 Mk I Instrument, Side View

The sighting telescope is sighted along the datum line and the magnetic bearing of the line of sight is read off the bearing plate through a microscope.

WATTS DATUM COMPASS, MARK 1

General

4. An aluminium body houses the compass, azimuth circle (the bearing plate) and sighting telescope. The three parts are totally enclosed within the body of the instrument, the necessary controls being mounted externally. A three-screw levelling base supports the body and provides the tripod mounting point for the instrument (*see* Figs 3, 5 and 7).

Compass

5. The compass consists of a magnet, fitted with an artificial sapphire jewel bearing, in a containing box (Fig 4). The pivot stem, protruding from the base of the box, is tipped with osmium-iridium. A leaf spring normally keeps the magnet lifted off its pivot and presses its centre boss against a forked bracket above the pivot. The magnet, which when lowered on its pivot swings between two copper damping blocks, carries an aluminium vane with vertical fine wire filaments at each end. The compass box is closed at its North end by a ground glass window, and at its South end by a convex lens. A mirror above the magnet pivot faces the lens and can, for collimation purposes, be moved about its vertical axis by a small adjustment screw (*see* para 32).

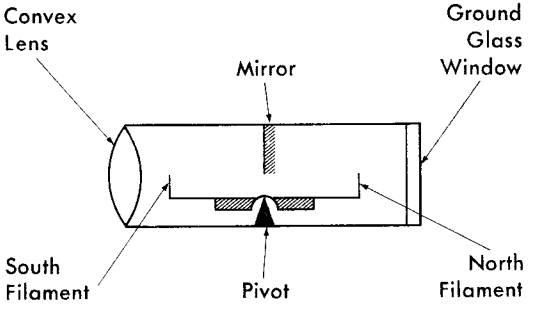


Fig 4 Mk I Compass Box, Schematic

6. The casing which houses the compass box is provided with two oval glass windows permitting illumination and viewing of the compass. The compass box is mounted on a horizontal spigot so that it may be tilted to allow for dip. On the external face of the compass casing is a metal shield through the centre of which the compass caging control passes. The shield (Fig 3), with an index mark on its lower edge, is mounted on the same spigot as the compass box and is tilted with it. The compass can be tilted up to 10° either side of the horizontal position and locked in position by two screws in the shield. The amount of tilt is indicated on a scale engraved on the casing below the shield. A red-painted screw on the shield conceals the access to the mirror adjustment screw (para 5).

7. The compass can be uncaged either by pressing the knob in the centre of the shield or by screwing a bowden cable release into the centre of this knob and then operating the cable release (Fig 5). In either instance, the pressure operates a lever which depresses the leaf spring and lowers the magnet on to its pivot. A safety lock on the finger release knob is provided in the form of a small knurled disc under the release knob. When the disc is turned anti-clockwise to its stop the release knob is prevented from being pressed in. The lock does not prevent the use of the bowden cable release.

8. The compass is aligned with the magnetic meridian when, with the magnet on its pivot, the North filament and the image of the South filament reflected in the mirror form one continuous vertical line seen through the convex lens. The South filament is seen only through the mirror since the filament itself is invisible to the observer, being out of the focus of the convex lens which focuses on the North filament.

Sighting Telescope

9. A fixed focus prismatic telescope enclosed in the main portion of the upper casing is used to define the line of sight. The telescope optical system (Fig 6) consists of an objective lens looking vertically downwards, two prisms and an eye-piece. Beneath the objective is an oblique mirror which deflects upwards the rays of light entering the telescope window. The mirror can be tilted by the elevating screw on the top of the casing and will allow the line of sight to be varied

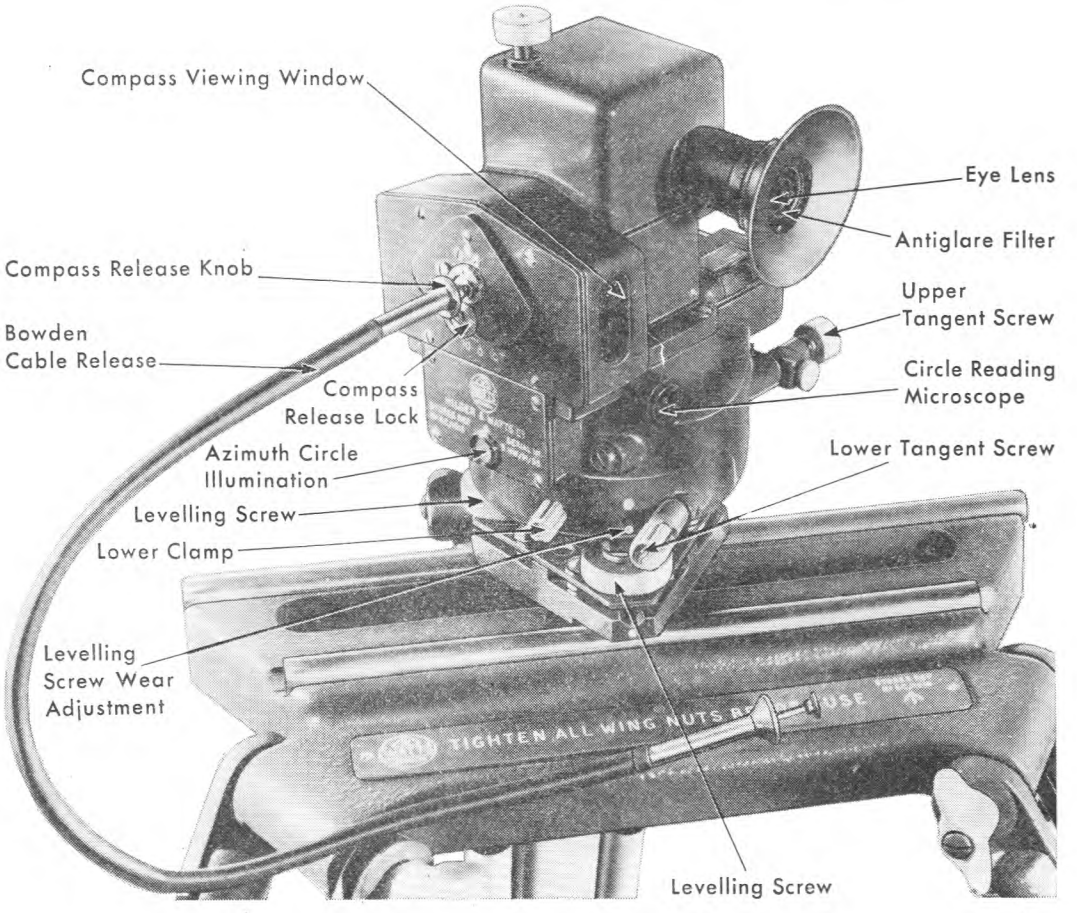


Fig 5 Mk I Instrument, Eye-End View

by 25° in a vertical plane, from 5° depression to 20° elevation.

10. A sighting graticule, consisting of a vertical line with a short crossline in the centre, is provided for accurate alignment. A green clear glass anti-glare filter may be swung across the eye lens when required (Fig 5). The telescope gives an erect image with × 6 magnification and a field of view of 8°.

Azimuth Circle

11. The azimuth circle is made of glass and is graduated at intervals of 0·1° with every degree mark numbered. It is illuminated through a ground-glass window situated below the compass system (Fig 3) and is read against a fixed index line through a variable focus microscope (Fig 5).

12. The azimuth circle and the upper casing (which covers the compass box, the telescope and the reading microscope with its index) are mounted independently on the vertical axis of the instrument and each of them is provided with a clamp and a tangent screw. With the upper clamp loose and the lower clamp tightened the azimuth circle will be fixed to the base of the instrument and the upper casing can be rotated in relation to the circle. Loosening the lower clamp and tightening the upper clamp will lock the azimuth circle to the upper casing, enabling both to be rotated in relation to the base of the

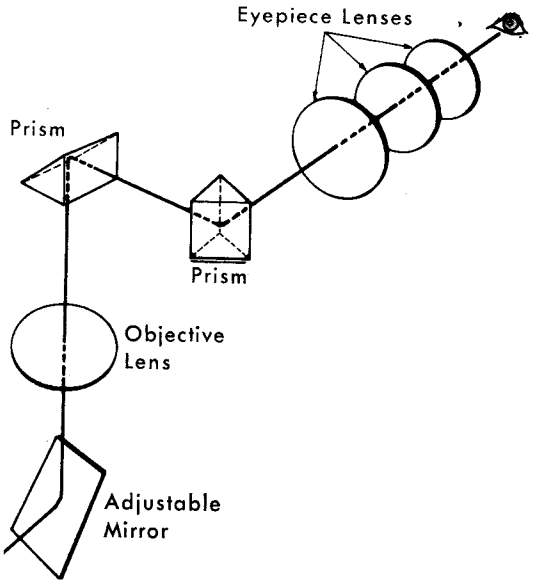


Fig 6 Mk I Telescope Optical System

instrument. The tangent screws enable fine adjustment to be made to the locked position after their respective clamps have been tightened. To differentiate between the two sets of clamps and tangent screws, the lower set has fluted screw-heads coloured yellow and the upper set has milled screw-heads silver in colour.

13. The levelling of the instrument is indicated by two tubular spirit levels mounted on the upper casing to the right of the sighting telescope. Graduations on the levels are provided at intervals corresponding to 2 minutes of arc. The levels, one parallel to the line of sight and the other at right angles to it, are protected by a perspex cover and can be adjusted by means of the lock-nut screws under the level housing (Fig 7).

Base

14. The lower casing of the instrument is supported by three levelling screws, the ball end of each resting in a corresponding groove in the triangular base plate. A triangular spring plate retains the base plate to the instrument. The under surface of the base plate has a machined flat and groove to register with, and slide along, the rods of the tripod head. A hole in the centre of the base plate is threaded to accept the tripod head bolt.

Instrument Case

15. The instrument case consists of an aluminium base tray and cover. A rubber seal ensures a dustproof fit when the two are fitted together and fastened by the two lockable toggle clasps. A compass corrector key will open the locks. The instrument rests in the tray on its side with circle clamps and levels uppermost. The base tray is fitted with rubber buffers to cushion the instrument, a clamping arm to hold it in place, and houses the following accessories:

a. 2 Bowden release cables

b. Plumb line

c. Tool roll, containing:
   2 spanners
   1 screwdriver
   1 camel hair brush
   1 capstan pin.

16. Three yellow marks are painted on the instrument, one on the upper casing, one on the lower casing and one on the base plate. These indicate the position in which the instrument should be placed in its case. The marks are to be in line, uppermost, and in line with the single yellow mark in the instrument case. Abbreviated instructions for the use of the instrument are engraved on a plate inside the lid of the instrument case.

17. The cylindrical dessicator, clipped into the lid of the case, affords some protection during initial storage and transit. Once the compass is brought into regular use, however, this dessicator can act as a water reservoir and create the conditions it is intended to prevent. It may be reactivated by being heated in a warm oven for about three hours and should subsequently be put back in the lid of the case before the instrument is put into long storage or returned for repair.

Tripod

18. The tripod head provides the platform upon which the instrument is mounted. It is a cast aluminium plate fitted with a circular spirit level for initial setting-up and has two machined rods to register with the base plate. A longitudinal slot through the head carries the captive

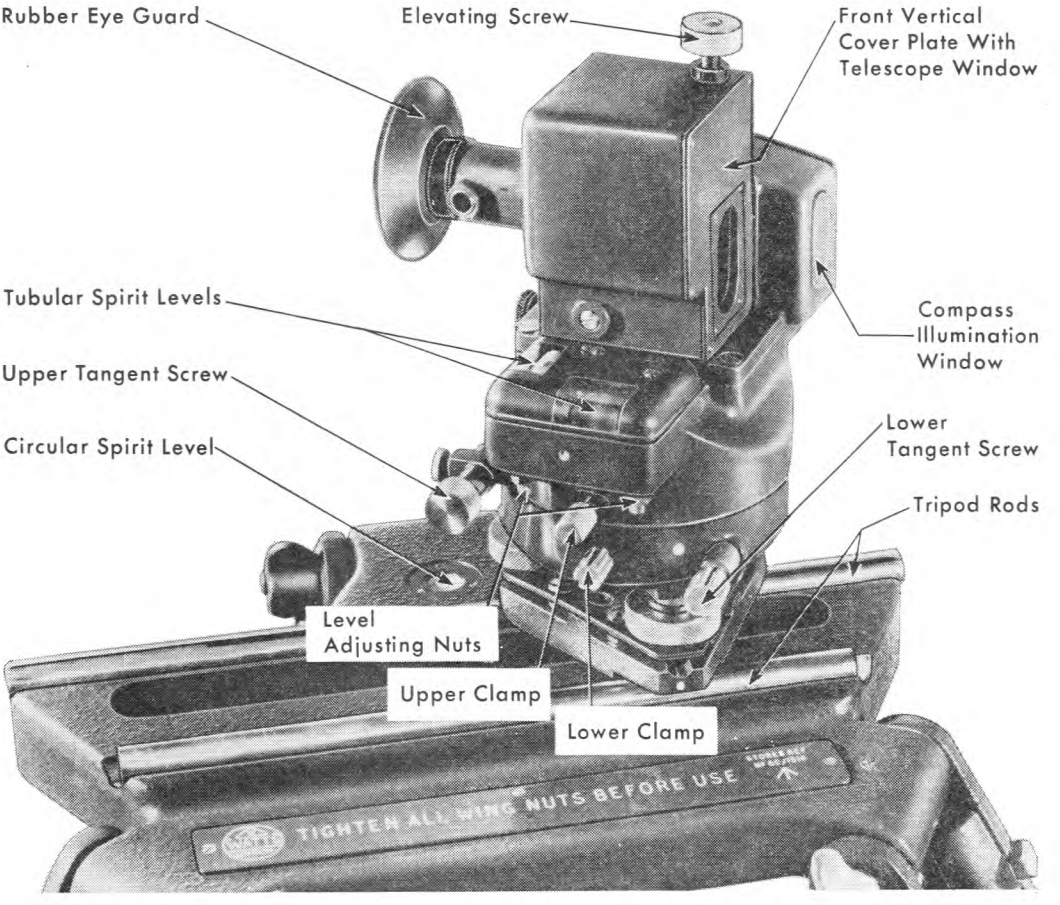


Fig 7 Mk I Instrument, Objective-End View

attachment bolt for the instrument and permits nine inches of lateral adjustment. A hook is suspended in the hollow handle of the bolt for the attachment of the plumb line. When the tripod is not in use, a metal protective cap for the head is secured by the bolt.

19. The tripod legs are attached to the head by swivel attachments, one at each end and one in the centre of one side. The legs are of wood and are painted with conspicuous black and yellow bands. The height of the tripod is approximately 5 ft when fully extended, but the legs are adjustable and may be locked in any position between 3 and 5 ft by wing nuts. Leather straps and handles are provided for carrying the tripod in the closed position.

WATTS DATUM COMPASS, MARK 2

General

20. The Watts Datum Compass, Mark 2 is a development of the Mark 1. Functionally it is the same, but changes have been made to the body of the instrument to limit the ingress of rain water, and internal lighting has been introduced so that magnetic bearings may be taken in poor light (or even total darkness) conditions. As the Mark 2 instrument has so many similar features to the Mark 1, this description will be confined to the changes between the two marks.

21. **Showerproofing.** The Mark 2 instrument is not completely waterproof. However the introduction of a skirt around the base of the upper casing,

seals around the entrances of the controls to the main body and shields over the compass box, lens and telescope window effectively prevent ingress of moderate rain and allow the instrument to be used under such conditions.

22. **Internal Lighting.** A self-contained internal lighting system is introduced into the Mark 2 instrument using luminescent "Betalights". Betalights are small glass tubes containing radioactive material which emits harmless Beta radiation. The Beta radiation causes the emission of visible light from the internal coating of the tube and all excess radiation is absorbed in the tube wall. They are used to illuminate internally the compass filaments, the telescopic graticule, the azimuth circle and the spirit levels. The advantage of Betalights is that they require no electrical power and are non-magnetic. The radioactive material has a half life of approximately seven years. A warning label referring to the Beta lights is mounted on the side of the main casing.

Compass

23. The compass system is identical to that used for the Mark 1 instrument. However, four Betalights are mounted inside the compass box, two, one each side of the North filament, and two, one each side of the South filament, to provide illumination. The positions of the Betalights are shown in Fig 8a and b. In conjunction with the Betalights and to ensure maximum illumination of the filaments, grooves are fitted either side of the ground glass window into which is fitted a removeable metal slide. This slide obstructs light entering the compass box and ensures that the illumination from the Betalights is not dissipated. The slide is not necessary for daytime use. (*See* Fig 9).

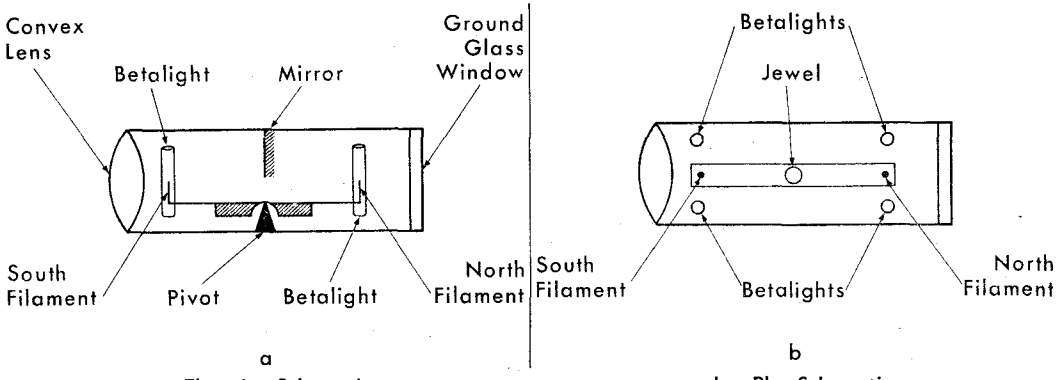
24. Other changes to the compass box have been introduced as part of the showerproofing improvements. These are the fitting of a small bracket to the South end lens, on which can be mounted a soft rubber eye-piece, and a new type of caging knob which has a sealing ring around its shaft. The safety lock on this caging knob is operated by turning the knob anti-clockwise. A Bowden cable can be used as with the Mark 1 instrument. (*See* Fig. 10).

Sighting Telescope

25. The optical system of the Mk 2 telescope is identical to the Mk 1, however, a Betalight is introduced between the eye-piece lenses and the first prism, to illuminate the graticule (*see* Fig 11). In addition the Mk 2 has a new type of soft rubber eyepiece which must be removed when the instrument is stored in its case, and a cylindrical shield around the telescope window (Fig 9). This shield prevents rain falling on the window and obscuring the view. The elevating screw on the top of the casing has an additional seal around its shaft.

Azimuth Circle, Upper and Lower Casing

26. The only change connected with the azimuth circle is the addition of a moveable Betalight mounted in a holder and fitted to a shaft which



a. Elevation Schematic

b. Plan Schematic

Fig 8 Mk 2 Compass Box

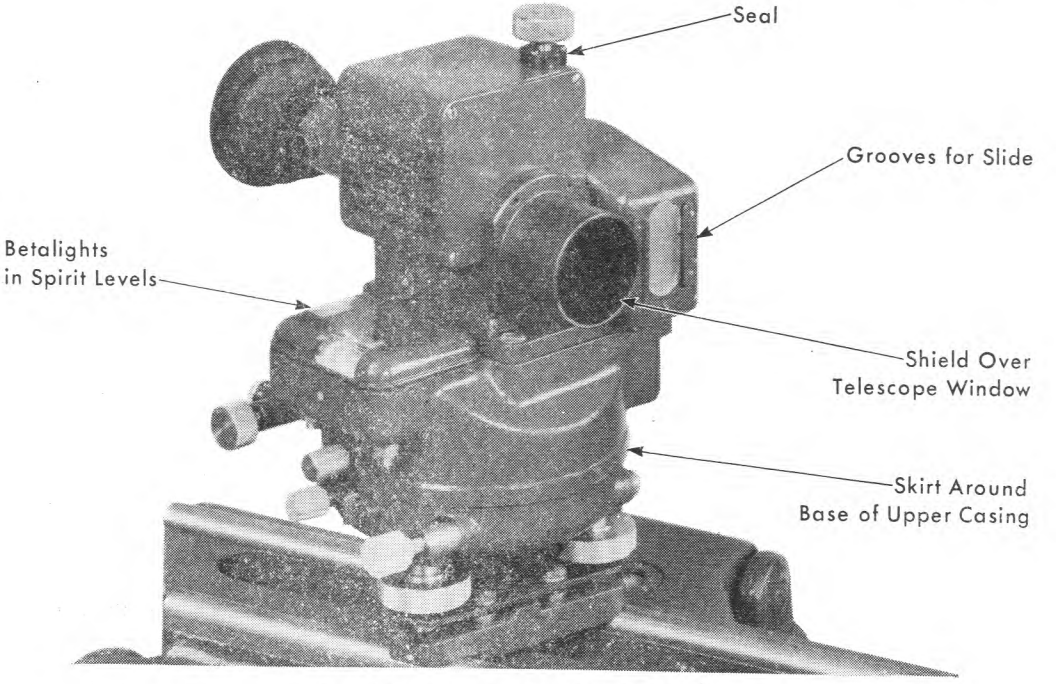


Fig 9 Mk 2 Instrument, Objective-End View

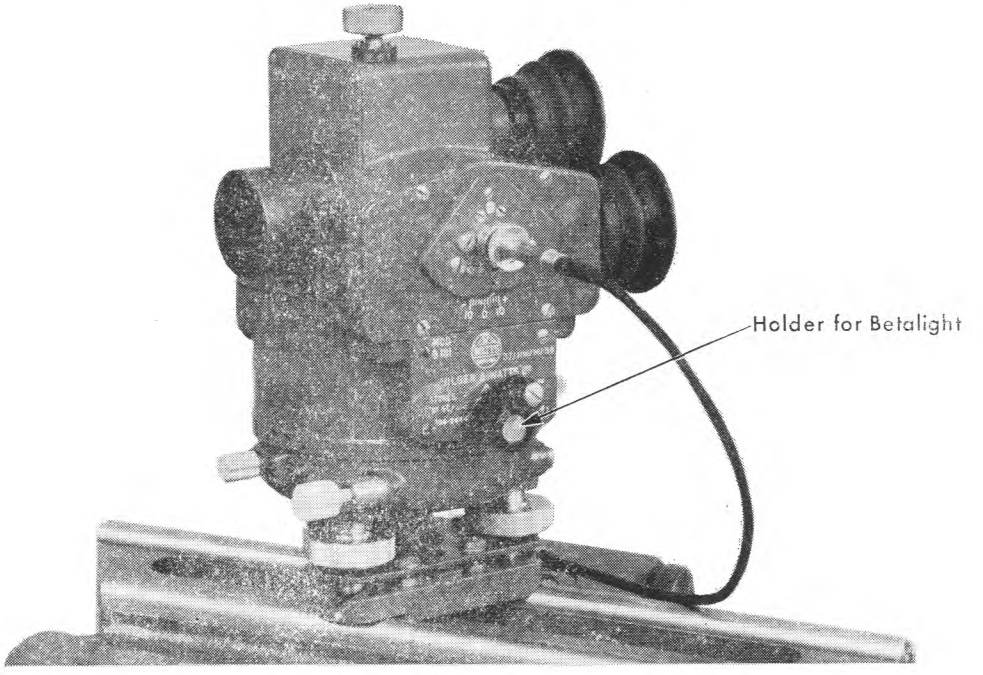


Fig 10 Mk 2 Instrument Showing Bowden Cable Release

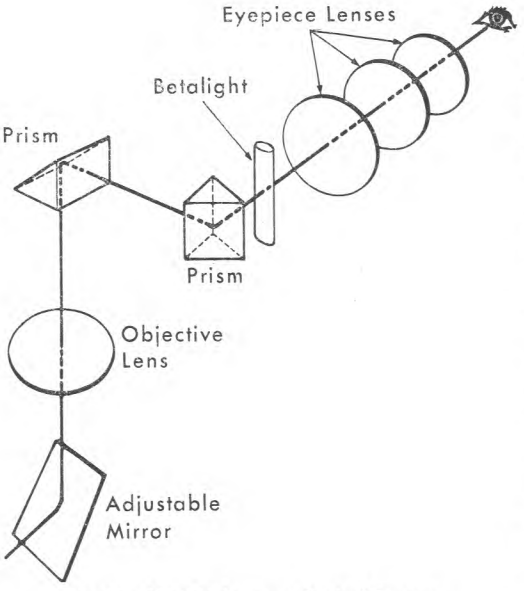


Fig 11 Mk 2 Telescope Optical System

allows it to be rotated into a position over the ground glass window situated below the compass system. The Betalight illuminates the azimuth circle and should be used in poor light conditions. In bright light conditions the Betalight should be rotated away from the window and normal daylight illumination used. The Betalight is shown in Figs 10 and 12.

27. Other changes to the upper casing include a skirt to the base of the casing and shields and seals around the shafts of the upper and lower clamps and tangent screws to protect against the ingress of water.

28. The Mk 2 instrument is fitted with a new type of spirit levels which are sealed against rain and incorporate Betalights for illumination. Otherwise the base is identical to the Mk 1 instrument and adjustment of the levels is the same.

Instrument Case

29. The same instrument case is used for both marks of instrument but, because the Mk 2 has

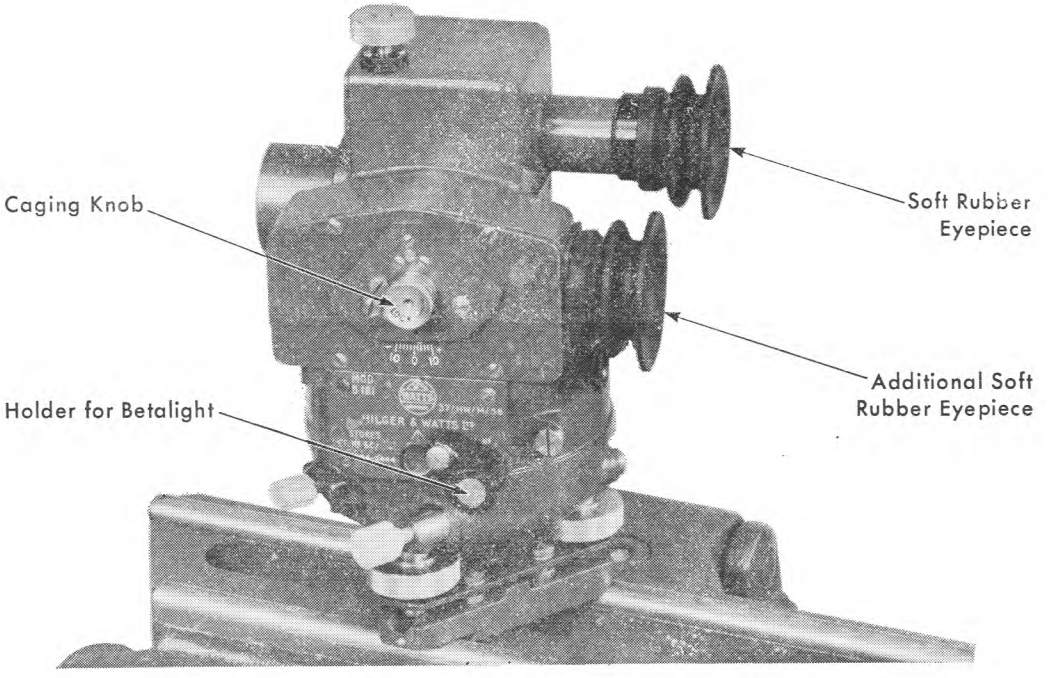


Fig 12 Mk 2 Instrument, Side View

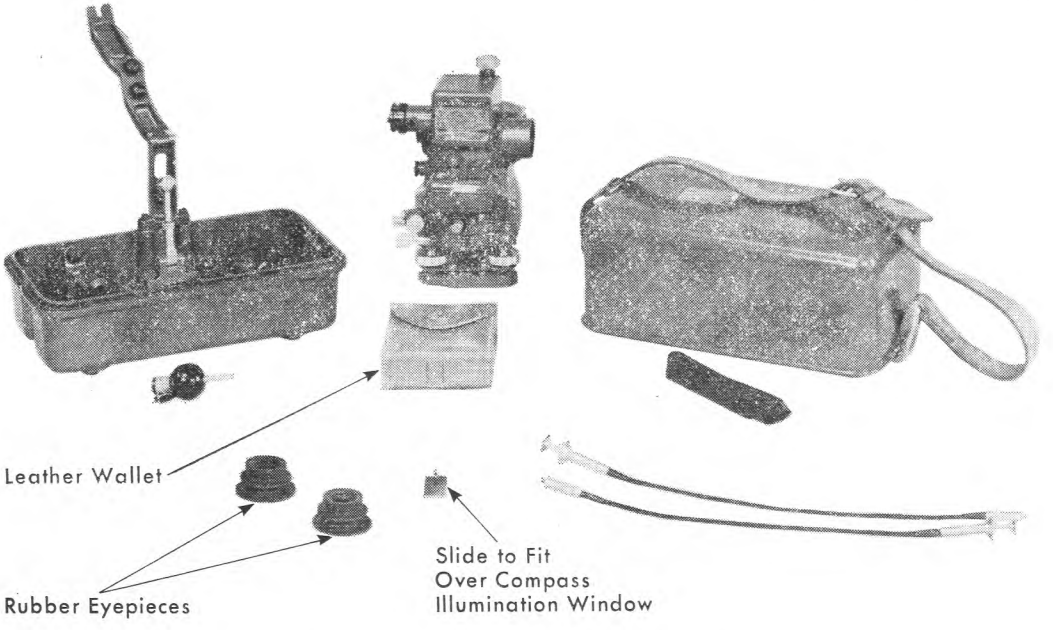


Fig 13 Mk 2 Instrument, Case and Accessories

larger soft rubber eye-pieces, these must be removed before the instrument is stored in the case. In addition to the accessories listed in para 15, the Mk 2 instrument case has a leather wallet in which are stored both the eye pieces and two slides (one a spare) for fitting over the compass box ground glass window during poor light conditions. This is shown in Fig. 13. The Mark 2 case carries a radioactive material warning label referring to the Betalights.

Tripod

30. The same tripod is used with both marks of instrument.

ADJUSTMENTS

General

31. The continued correct and accurate functioning of both marks of Watts Datum Compass depends upon optical, mechanical and magnetic adjustments. These which apply to both the Mk 1 and the Mk 2, are as follows:

a. Periodic collimation
b. Adjustment of the spirit levels
c. Adjustment of the levelling screws
d. Adjustment of the compass for dip.

Periodic Collimation

32. The instrument must be checked for accuracy every six months or earlier if the accuracy is in doubt. The special equipment required for this check is not available at user units and it is therefore necessary to return the instrument, with the tripod, to the Admiralty Compass Observatory (ACO) by hand (AP 3158, Vol 2, Leaflet L3 refers), where this equipment is available. The alignment of the vertical filaments is checked when the axis of the magnet is parallel to the line of sight. Minor adjustments may then be made by moving the compass mirror (paras 5-6).

Adjustment of Spirit Levels

33. Each of the two spirit levels on the upper casing has a lock-nut adjusting screw which, when turned, will cause a vertical movement of one end of the level in relation to the other end. The

levels can be checked and, if necessary, adjusted by the following method:

a. Rotate the upper casing of the instrument until one of the levels is parallel to a line joining any two of the levelling screws on the base plate. Turn these two screws simultaneously by equal amounts in opposite directions to bring the bubble to the centre of the level.

b. Centralize the bubble of the other spirit level by turning the third levelling screw only.
c. With the upper casing in the same position repeat a. and b. until both bubbles are central.

d. Rotate the upper casing through 180°.

e. The spirit levels are in correct adjustment if in this new position the bubbles remain central. If the bubbles are off-centre, note the position of each bubble and correct one half of the error using the appropriate levelling screws, and the remaining half using the spirit level adjusting screw.

f. Repeat d. and e. until the bubbles remain central in both positions of the upper casing.

g. After adjustment ensure that the lock-nuts are tight.

Adjustment of Levelling Screws

34. Two adjustments are provided for the levelling screws as follows:

a. To ensure the correct fit of each levelling screw in its bush, and to take up slack due to wear, there is an adjustment screw just above each levelling screw (Fig 3). When these screws are tightened a spring leaf is closed in the bush, thus ensuring a good fit on the screw.

b. The pressure applied by the triangular spring plate on the ball ends of the levelling screws can be varied by adjusting the six bolts securing the spring plate to the base plate (Fig 3). This pressure should be adequate but not excessive.

Adjustment of Compass for Dip

35. The magnetic element is balanced for zero vertical magnetic field, as at the magnetic equator, but it will swing freely for a small departure from the magnetic equator. However, when the vertical component of the Earth's magnetic field becomes greater, the magnet will tilt on its pivot to such an extent that it will touch one of the damping blocks and will cease to swing freely. The compass box must then be adjusted in tilt to restore the free swing. This adjustment is carried out as follows:

a. Set up the instrument as detailed in para 39.

b. By rotating the upper casing, align the compass with the magnetic meridian as detailed in para 40 g and h.

c. Loosen the two securing screws on the shield. This will permit tilting of the shield and the compass box.

d. Tilt the compass to obtain a free swing of the magnet, turning the instrument in azimuth if necessary.

e. Continue tilting slowly until the magnet is just touching the damping block, resulting in the magnet sticking. Note the reading on the scale engraved on the upper casing.

f. Tilt in the reverse direction until the magnet sticks again and note the scale reading.

g. Set the shield to the mid-position between these two readings and tighten the two securing screws to lock the shield and the compass box in position.

OPERATION

General

36. The Mark 1 instrument can only be used in good light conditions and should not be used in rain. On the other hand the Mark 2 instrument can be used in all light conditions ranging from bright daylight to complete darkness. It may also be used in conditions of moderate rain if this is necessary. Operating procedures for the two marks in daylight conditions are the same. To use the Mk 2 instrument in poor light conditions the following extra two actions are required:

a. Fit the metal slide in the grooves each side

of the compass box ground glass window.

b. Rotate the Betalight holder over the azimuth circle illuminating window.

These actions ensure that illumination is adequate for the compass filaments and the azimuth circle scale.

37. It must be remembered that although the Mk 2 instrument can be used for bearing measurements even in complete darkness it is still limited by the amount of illumination of the image on which the bearing is being taken. Thus, if a compass calibration is being carried out at night, some form of artificial illumination will be required to light the aircraft datum points.

38. The following operating procedures apply equally to both marks of instrument.

Setting Up

39. The instrument is set up as follows (*see* Fig 14):

a. Set up the tripod in the desired position. The centre of the head should be approximately on the line of sight, the slot in the tripod head at right-angles to the line of sight, and the centre leg of the tripod pointing in the direction of the line of sight.

b. By manipulating and adjusting the legs, set the tripod at a convenient height and level it by reference to the circular spirit level.

c. Attach the instrument by locating the groove on the lower face of the base plate with one of the tripod head rods and then screwing the tripod head bolt into the threaded hole in the centre of the base plate.

d. By adjustment of the levelling screws, level the instrument by reference to the spirit levels on the upper casing.

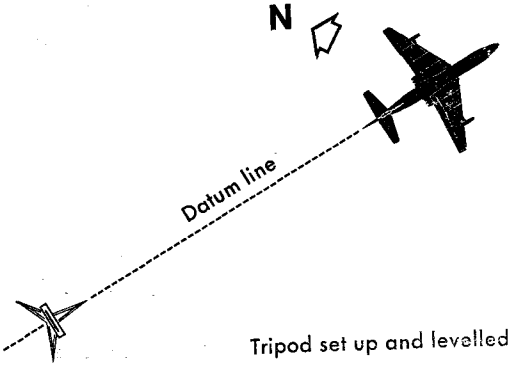


Fig 14 Tripod Set-Up

Note: It is essential at this stage that all magnetic and paramagnetic materials are moved clear of the instrument. In particular the instrument case must be at least 12 ft away, a trolley accumulator 15 ft away, and a small tractor or petrol-electric generator 25 ft away from the instrument. The operator must ensure that there are no magnetic materials on his person; a watch, for example, can have a marked effect on the compass.

Taking a Bearing

40. It was stated in para 3 that the Watts Datum Compass is essentially a bearing plate which can be orientated in relation to the magnetic meridian. Once this fact is appreciated by the operator, and he is familiar with the various controls, the procedure for taking a bearing is comparatively straightforward. Having set up the tripod and instrument as detailed in para 39, proceed as follows (*see* Fig 15 a, b, c and d):

a. Loosen the lower clamp and tighten the upper clamp.

b. Turn the upper casing to align the sighting telescope with the aircraft's datum points. If necessary, the line of sight can be adjusted vertically by rotation of the elevation screw at the top of the instrument, and laterally by sliding the instrument on the transverse rods.

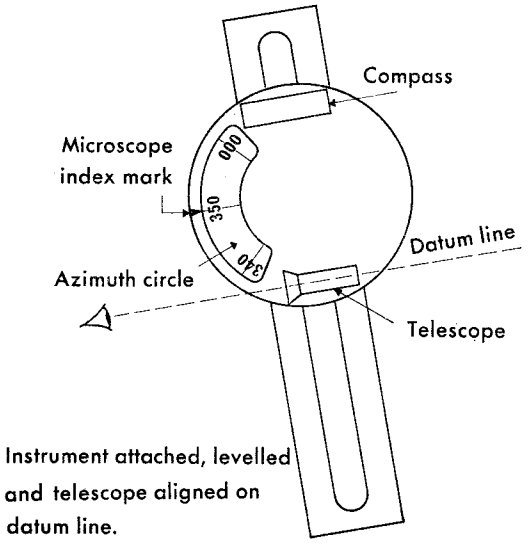
c. Check the instrument levelling and adjust if necessary.

d. Tighten the lower clamp and loosen the upper clamp.

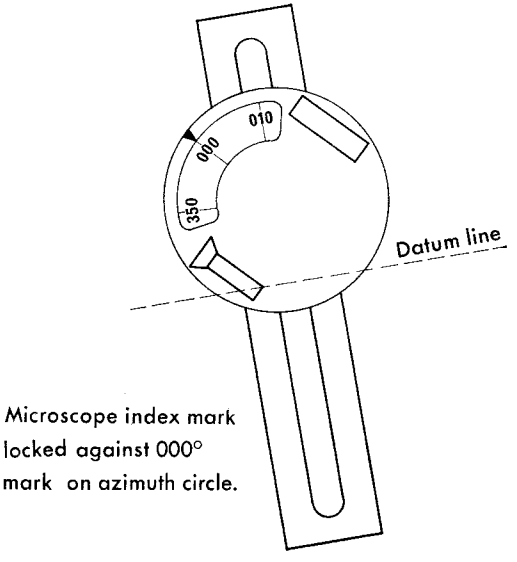
e. Adjust the scale microscope as required to read the azimuth circle.

f. Rotate the upper casing until the index seen through the microscope reads zero (or 90° or 180° as required). Tighten the upper clamp and make fine adjustment with the upper tangent screw.

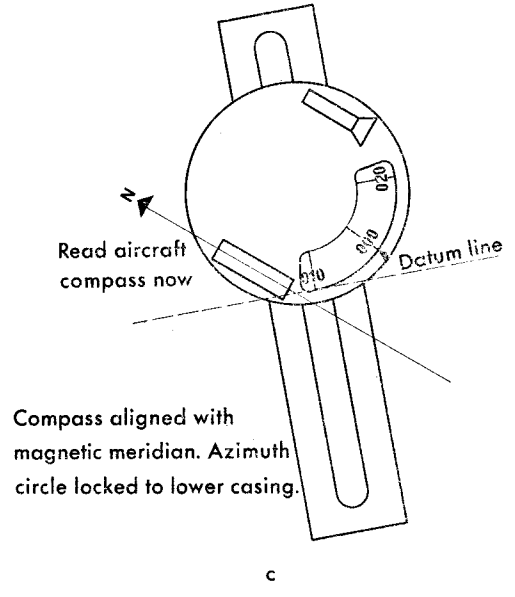
g. Loosen the lower clamp and rotate the upper casing to align the compass approximately with the magnetic meridian. Uncage the compass.



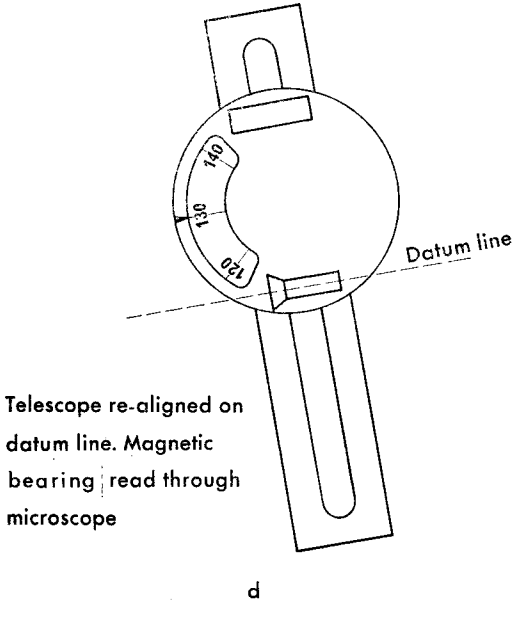
a. Telescope Aligned on Datum Line



b. Azimuth Circle Indexed



c. Compass Aligned on Magnetic Meridian



d. Telescope Re-Aligned on Datum Line

Fig 15 Taking a Bearing

h. When the magnet settles down, tighten the lower clamp and align the compass filaments accurately, using the lower tangent screw. If the magnet does not swing freely, adjust for dip (para 35).

j. Record the readings of the aircraft compasses which are being calibrated.

k. Cage the compass and check that the azimuth circle still reads zero (or 90° or 180° as required).

l. Loosen the upper clamp and rotate the upper casing to align the telescope with the aircraft's datum points. Tighten the upper clamp and make fine adjustment with the upper tangent screw. Take care not to disturb the lower clamp or tangent screw during this operation.

m. Through the microscope read the magnetic bearing from the azimuth circle.

Casing the Instrument

41. The instrument and tripod should be dismantled and cased as follows:

a. Loosen upper and lower clamps. Cage the magnet system and unscrew bowden cable release, if fitted.

b. Remove the instrument from the tripod head by unscrewing the clamping handle to disengage the tripod head bolt.

c. If a Mark 2 instrument, remove the rubber eyepieces and the slide (if it has been used), and stow them in the leather wallet.

d. Stow the instrument in the tray of the case, ensuring that the yellow index marks on the upper casing, lower casing, and base plate are in line with each other and with the line on the tray.

e. Clamp the instrument to the tray with the clamping arm.

f. Stow all auxiliary equipment in the tray.

g. Fasten the lid to the tray with the clasps.

h. Collapse tripod legs and attach socket of tripod protective cap to tripod head bolt. Fasten leather straps.

42. Misting of the optical system may occur, even with the Mark 2, if the instrument is used under damp conditions, and it is therefore important that the instrument should, in these circumstances, be dried out following its use. This may be achieved by keeping it for a few hours in a warm room with the case open.

ACCURACY

Alignment

43. When all systematic errors have been eliminated by the adjustments provided, the Watts Datum Compass can be aligned to the magnetic meridian to an accuracy of $\pm$ 0·02° (50% error). This accuracy will deteriorate in unfavourable wind conditions since a surface wind speed in excess of about 15 knots will cause vibration of the uncaged compass system, and affect the operator by causing his eyes to water.

Accuracy of Compass Deviation

44. The accuracy of compass deviation measurement using the Watts Datum Compass depends not only on the instrument accuracy of the datum instrument, but also upon the precision of the instrument alignment with the aircraft's datum points and the accuracy with which the aircraft compass can be read. This last factor is independent of the datum equipment performance and is, at the moment, likely to cause the largest error. With practice and care however, reasonable estimates to 0·1 degree can be made in reading remote indicating compasses. It has been found that wind speeds above 15 knots cause the aircraft to rock and it is therefore inadvisable to attempt an accurate swing under these conditions.

45. Although the Mk 2 instrument has been modified so that it can be used in rain or poor light conditions it should be remembered that the accuracy of bearings depends to a great extent on the operator and the conditions under which he works. It would, therefore, not be advisable to attempt to carry out a compass swing in heavy rain. Similarly with the lighting modification; this was provided so that a compass swing could be completed in light conditions which would preclude the use of a Mk 1 instrument. It is suggested that, only in an emergency, should one plan to carry out a compass swing entirely in darkness.

Printed for Her Majesty's Stationery Office by Wilfred Edmunds Ltd, Station Road, Chesterfield.

CHAPTER 4

MC-1M MAGNETIC COMPASS CALIBRATOR

CONTENTS

	Paras
Introduction	1
Principle	2
Advantages	3
Disadvantages	4
DESCRIPTION	
General	5
Monitor	6—8
Turntable	9—11
Control Console	12
Optical Alignment Equipment	13
Static Inverter	14
Tripods	15
Connecting Cables	16
OPERATION AND ACCURACY	
Operation	17
Accuracy	18

Introduction

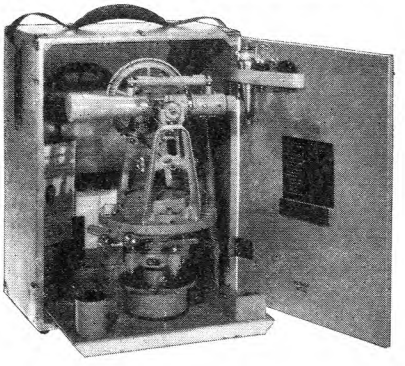
1. The MC-1M Magnetic Compass Calibrator Set provides magnetic heading inputs to a magnetic compass system enabling a compass swing to be carried out electrically without the need to rotate the aircraft on a compass base. The MC-1M may also be used to carry out magnetic surveys of areas to be used for MC-1M compass swings.

Principle

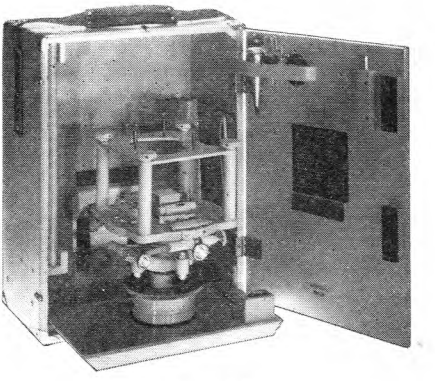
2. If the strength and direction of the Earth's magnetic field at any location is known then the magnetic field corresponding to any specific heading may be simulated electrically in the azimuth detector unit (detector unit). A compass swing may therefore be carried out by creating magnetic fields in the detector unit corresponding to specific headings from 0° to 345° and comparing these headings with the actual compass headings produced. Any errors will be caused by coefficients A, B, C, D and E. Coefficient A may be removed by aligning the detector unit accurately with magnetic north before mounting it in the aircraft, and by maintaining this alignment during the installation of the detector unit in the aircraft. When the pre-aligned detector unit is installed in the aircraft and connected to the aircraft compass system, a full electrical compass swing may be carried out to determine and remove coefficients B, C, D and E. It should be noted that coefficients D and E due to the aircraft structure cannot be determined correctly or removed, and if present will cause errors in an MC-1M compass swing. Coefficients D and E caused by the compass system will be measured and may be removed.

Advantages

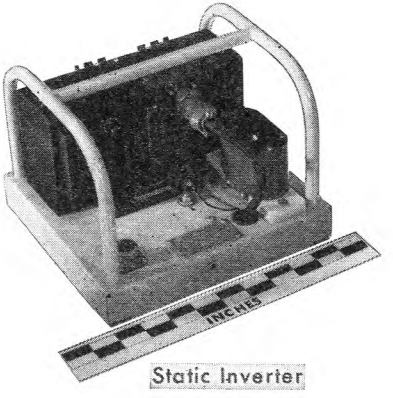
3. Both the MC-1M and the Watts Datum methods of refined compass swinging require an accurately surveyed base on which no magnetic disturbances greater than 0·1° are present. However, the MC-1M compass swing does not necessarily require the normal compass swinging



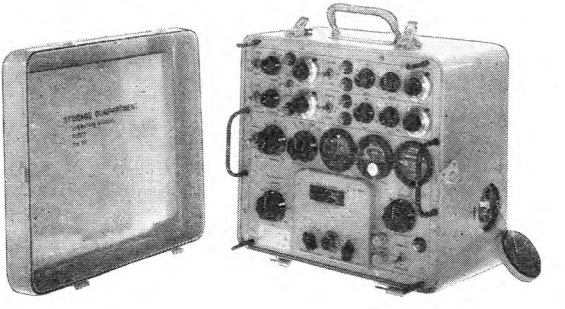
Monitor



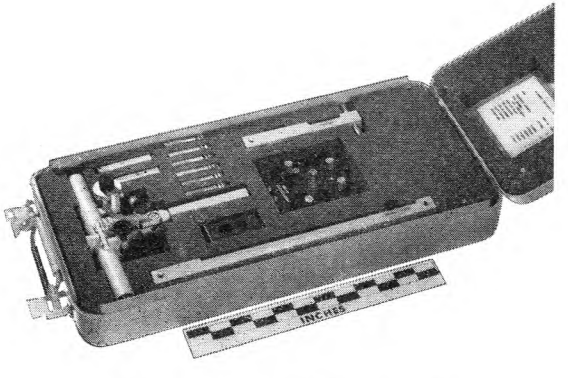
Turntable



Static Inverter



Control Console



Optical Alignment Equipment

Tripod and Connecting Cable

Fig 1 MC-1M Magnetic Compass Calibrator Set

base. Any area free from magnetic disturbances will suffice; *eg* an MC-1M swing may be carried out in a suitable area adjacent to a hangar. The aircraft is positioned facing north with its fore/aft axis coincident with a pre-marked north/south line. No further movement of the aircraft is required during the swing. The MC-1M method of compass swinging has the following advantages:

a. A convenient base close to the servicing hangars may be used, allowing aircraft to be moved on and off the base without disrupting other airfield activity.

b. Once the aircraft is positioned on the base it remains stationary while the swing is carried out, releasing the towing vehicles and associated personnel for other tasks.

c. All measurements and corrections are made inside the aircraft. The MC-1M swing may therefore be completed in the rain, and also in strong winds when the wind direction is close to the north/south axis.

d. An MC-1M swing can be completed in $1\frac{1}{2}$ hours, which is quicker than the time required for a conventional swing. Aircraft are therefore released for flying sooner.

e. The calibrator may be used to establish the serviceability of the overall compass system prior to the swing. Faults in individual components of the compass system are quickly diagnosed using the MC-1M.

f. The aircraft is not sterile during the MC-1M swing. Rectification and preparation of aircraft components and systems which do not affect the compass system can be carried out while the MC-1M swing is in progress.

g. Detector units can be pre-swung and stored ready for fitting to aircraft when required. This further reduces the time the aircraft is required on the compass base.

Note: The MC-1M Compass Calibrator is suitable for use with American compass systems. It is incompatible with British systems, and its use in the RAF is confined to the C12 compass system in the Hercules.

Disadvantages

4. Standby compasses cannot be swung using the MC-1M equipment. This disadvantage is overcome by taxying the aircraft to the normal compass swinging base on completion of the MC-1M swing, and using the C12 compass in the aircraft to provide the heading datum when swinging the standby compass.

DESCRIPTION

General

5. The MC-1M Magnetic Compass Calibrator consists of the following units which are shown in Fig 1:

a. Field Monitor Unit (Monitor).
b. Turntable.
c. Control Console.
d. Optical Alignment Equipment.
e. 400-cycle Power Supply (Static Inverter).
f. Two Tripods.

Monitor

6. The Monitor (Fig 2) is a theodolite with a 22-power telescope. The unit is mounted on a slideboard, and the complete assembly fits into a

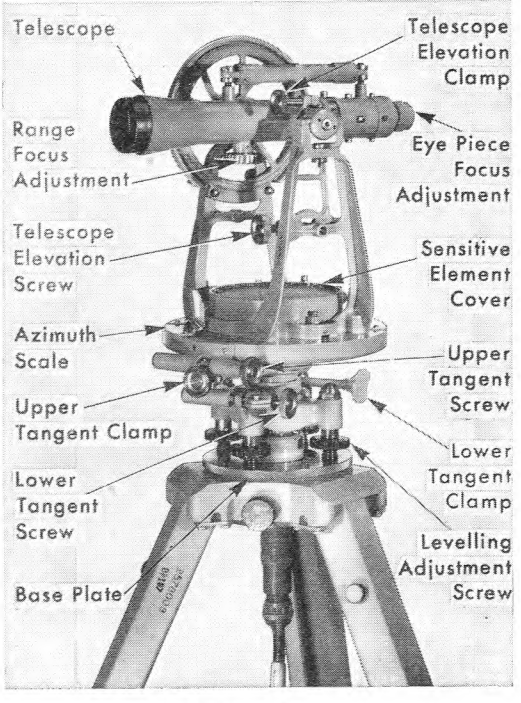


Fig 2 The Monitor

wooden carrying case which also contains certain accessories.

7. Liquid levels, levelling adjustment screws, azimuth scales, and upper and lower tangent screws are provided to level the unit and rotate it to various headings. The telescope is levelled, elevated and depressed by means of an elevation screw and clamp. Two telescope adjustments are provided, one on top of the eyepiece for range focus, and one on the eyepiece to bring the horizontal and vertical cross hairs into view. The two azimuth scales are calibrated in degrees and half degrees, and verniers are provided for readings in minutes. The direction viewed through the telescope is read off the A scale, and the reciprocal from the B scale.

8. A non-pendulous sensitive element has been substituted for the compass needle on the Monitor. This element is energized through a connector located beneath the centre of the baseplate. A plumb-bob adapter may be attached to this connector when required. The Monitor, in conjunction with the Console, measures the strength and determines the direction of the horizontal component of the Earth's magnetic field, and is used for the magnetic survey of areas to be used for MC-1M compass swings and also to monitor changes in magnetic conditions during a compass swing. The Monitor is pre-calibrated at the factory to sense magnetic north when the telescope is pointing in that direction.

Turntable

9. The Turntable (Fig 3) is another theodolite with the compass, vertical circle and telescope removed and replaced by a detector unit mounting frame. The Turntable is mounted on a slideboard, and the complete assembly fits into a carrying case similar to that of the Monitor. As with the Monitor, certain accessories are included with the Turntable.

10. The Turntable is made of non-magnetic materials and is identical to the Monitor in construction, method of levelling and azimuth adjustment, and is mounted on an identical tripod. Power is supplied through a connector located beneath the baseplate. The Turntable and the detector unit are linked by a connector located on top of the Turntable. The top plate may be supported by four 4-in legs or alternatively by four 12-in extension legs which attach to the top of the Turntable. The detector unit mounting flange is attached to the top plate with three thumb screws, and the detector unit may be top or bottom mounted depending on the type of mounting used in the aircraft.

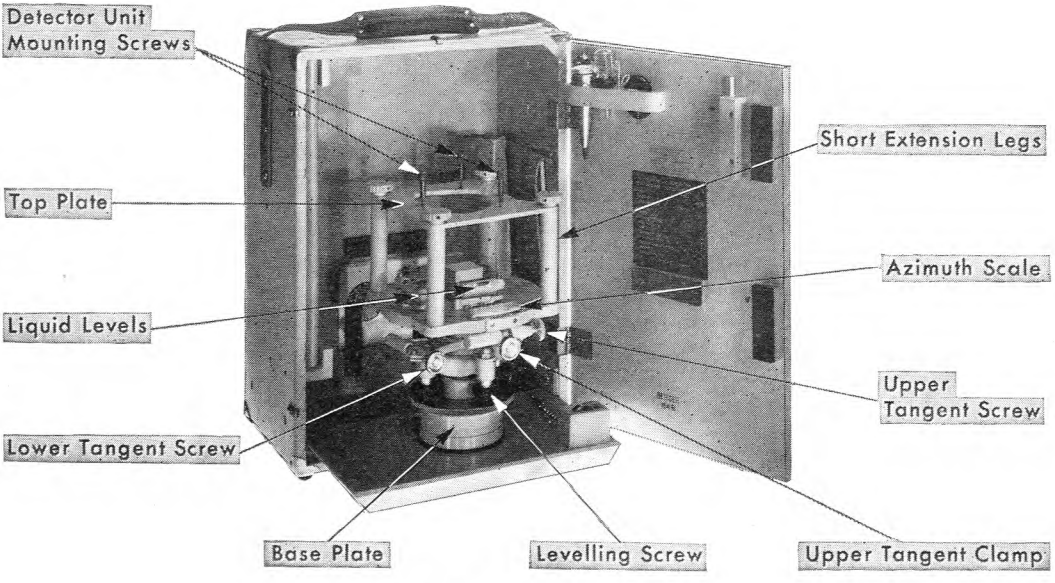


Fig 3 The Turntable

11. The Turntable is used to align the detector unit with magnetic north before the detector unit is installed in the aircraft. The Turntable horizontal circle serves as the azimuth reference for positioning the detector unit on the desired compass headings.

Control Console

12. The Control Console (Fig 4) is the electronic heart of the calibrator, and contains the controls, switches and associated electronic sub-assemblies. The front panel controls and indicators are accessible after the cover is removed. The connectors at the side of the Console are marked *Power*, *Mon* and *Trans*, and provide interconnection of the 400-cycle power supply, the remote Monitor and the Turntable. The operating controls, indicator lights, meters and fuses appearing on the front panel of the Console are identified in Fig 4. *Note:* The heading and error read-out is in degrees and minutes of arc.

Optical Alignment Equipment

13. The Optical Alignment Equipment (Fig 5) consists of a fixed focus telescope with Hooke's joint, two coupling shafts, a support plate and

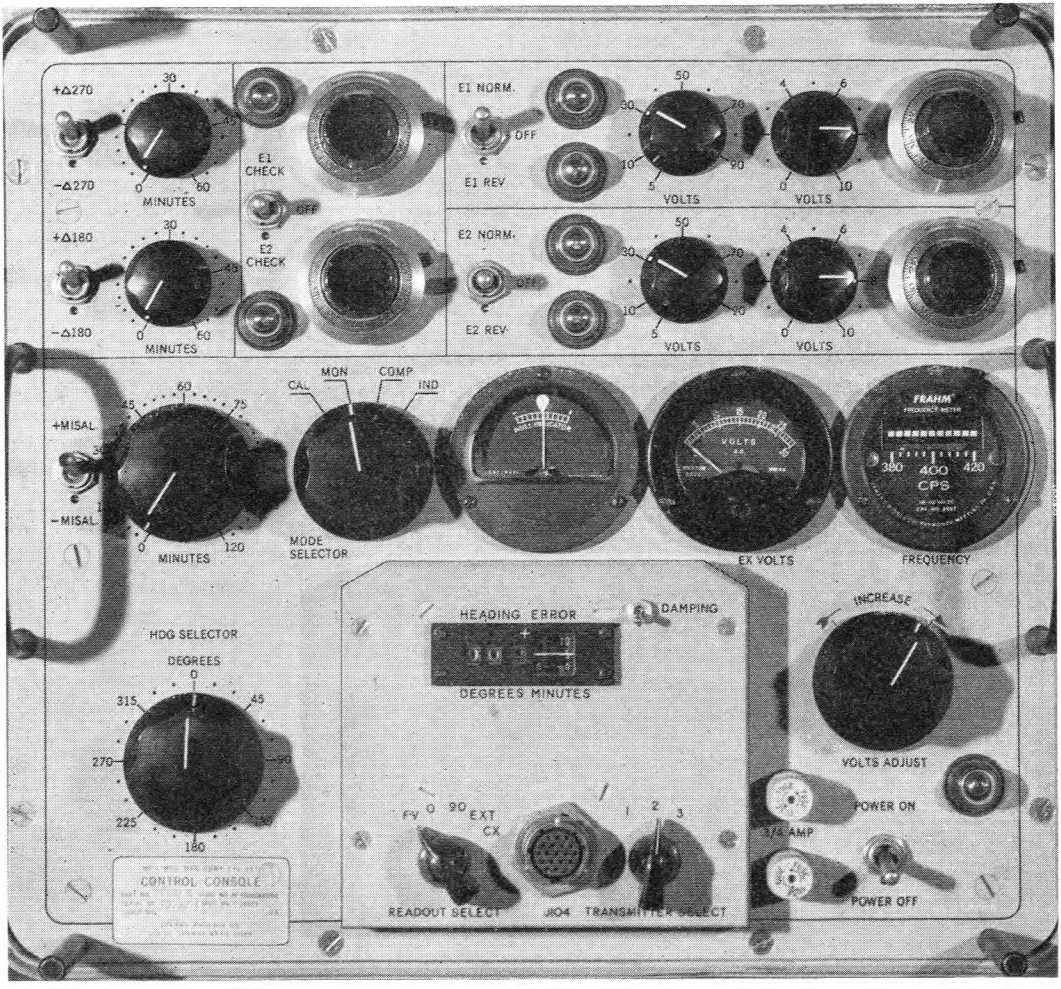


Fig 4 Control Console

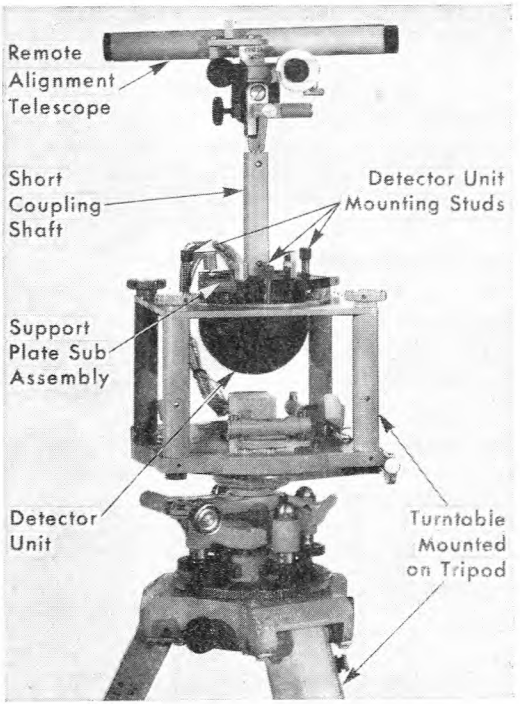


Fig 5 Optical Alignment Equipment

miscellaneous mounting bolts. The parts required depend on the aircraft type and the type of detector unit being used. The Optical Alignment Equipment is used to align the detector unit in the aircraft during the transfer from the Turntable. The detector unit is aligned with magnetic north and the telescope is moved in azimuth and aligned with a distant target and locked to the detector unit. The detector unit, with the telescope still attached, is then transferred to the aircraft. Re-aligning the telescope with the target ensures that the detector unit alignment is maintained during the transfer to the aircraft. Parallax errors are eliminated if targets at ranges greater than $\frac{1}{2}$ nm are chosen. Detector units can be pre-swung and stored with the Optical Alignment Equipment attached until they are required for fitting in an aircraft.

Static Inverter

14. The Static Inverter is solid state, with its own connecting provisions and power cable. It converts a 27·5 volt DC supply to 115 volts, 400 cycles, single phase.

Tripods

15. Two identical tripods are provided for mounting the Monitor and the Turntable.

Connecting Cables

16. Connecting Cables are provided to interconnect the units of the compass calibrator. The power cable is 100 ft long, the Monitor cable is 200 ft long, and the Turntable cable is 50 ft long.

OPERATION AND ACCURACY

Operation

17. The full setting-up procedures and operating instructions for the MC-1M Compass Calibrator are lengthy and are not within the scope of this chapter. The relevant information may be obtained from TO 5N3-3-7-1.

Accuracy

18. The MC-1M is capable of providing heading inputs in 15° increments from 0° to 345° with a 0·1° accuracy (1σ).

Printed in England for Her Majesty's Stationery Office by Bemrose & Sons Ltd., Derby 3840 Dd. 898798. 5/73. 12500. Gp. 528

CHAPTER 5

THE PRECISE HEADING TEST SET

CONTENTS

	Paras
Introduction	1-3
Description	4-9
Method of Use	10
Calibration	11

Introduction

1. In carrying out a refined compass calibration, apart from a knowledge of the actual magnetic heading of the aircraft which is determined by the Watts Datum Compass, it is necessary to know the magnetic heading indicated by the aircraft gyro-magnetic compass system to a high degree of accuracy. It is also essential that the compass system be synchronized before any readings are recorded.

2. Most gyro-magnetic compass systems fitted to aircraft that require a refined compass calibration have a display which is graduated to 1° of heading. However, in order to achieve an accuracy comparable with the Watts Datum Compass, it is necessary to record the compass reading to ± 0·1°. In addition, in most compasses the indication that the compass is synchronized is either by a (·/+) annunciator or rudimentary centre-reading voltmeter, neither of which is sufficiently accurate.

3. The Precise Heading Test Set (PHTS) was designed to satisfy these two requirements. It incorporates a display of compass heading by means of veeder counters which can be read to an accuracy of 0·05° and an accurate centre-reading voltmeter.

Description

4. The Precise Heading Test Set is illustrated at Fig 1. It is a rectangular box which has two hinged halves. Fig 2 shows it in the opened position.

5. **Heading Counters.** The left-hand half has two windows displaying a veeder counter indication of compass heading. The right-hand window indicates tenths of a degree and can be read to an accuracy of at least 0·05°. The left-hand window indicates whole degrees of compass heading from 000° to 359°. There is also a calibration certificate and a calibration graph which allows corrections for instrument error to be made to the heading counter readings.

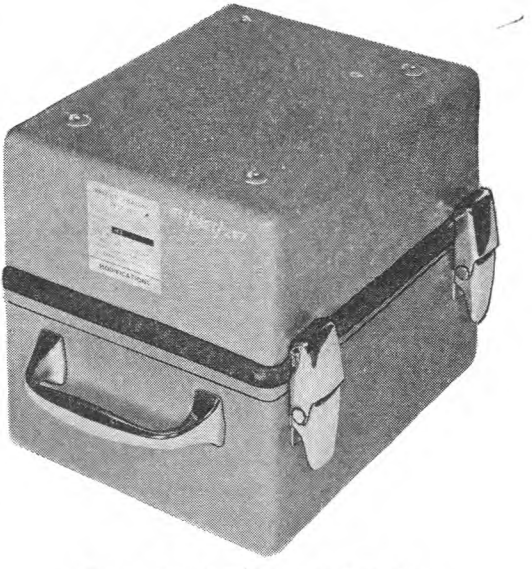


Fig 1 Precise Heading Test Set

6. **Centre-Reading Voltmeter.** The right-hand half of the PHTS contains a centre-reading voltmeter whose scale is graduated 3-0-3. The voltmeter is used to read the voltages present at the slaving amplifier annunciator output (*ie* the state of synchronization) and, on some compass systems,

the voltages present at the adjustable potentiometers in the remote correction unit (*ie* the deviation correction voltages for Coefficients B and C being fed to the flux valve detector coils). The voltmeter can be centred by turning a zero-adjuster screw.

7. **Function Switch.** Mounted above the voltmeter is a five position selector switch marked SYNC, B-×3-C. B-×1-C. Selection of these positions gives the following facilities:

a. *SYNC.* When SYNC is selected the voltmeter shows the DC voltage output from the slaving amplifier *ie* when the needle is central, the compass is synchronized.

b. *B-×3-C.* Two switch positions are shown against the B-×3-C marking. Use of these switch positions depends on the type of compass system being calibrated. On some compass systems, the left-hand position allows the DC voltage correction to the flux valve detector coils set in at the B potentiometer of the remote correction unit to be displayed on the voltmeter. The right-hand position allows the voltage set in at the C potentiometer to be displayed. On other compass systems, only the left-hand position is used and selection between B and C voltages is made by inserting the red and white probes, which are part of the test harness, into the sockets adjacent to the B and C potentiometer correction dials as appropriate. The particular method of using these switch positions must be decided from knowledge of the compass system being calibrated and the particular test cable and test sockets being used. The voltage indicated with the switch in either of these positions is one third of the actual voltage at the correction potentiometers *ie* an indication of 2 volts represents an actual measured voltage of 6 volts.

c. *B-×1-C.* Use of these two switch positions is exactly the same as in b. above. However, in this case the voltage indication on the voltmeter equals the correction voltage set in at the potentiometers.

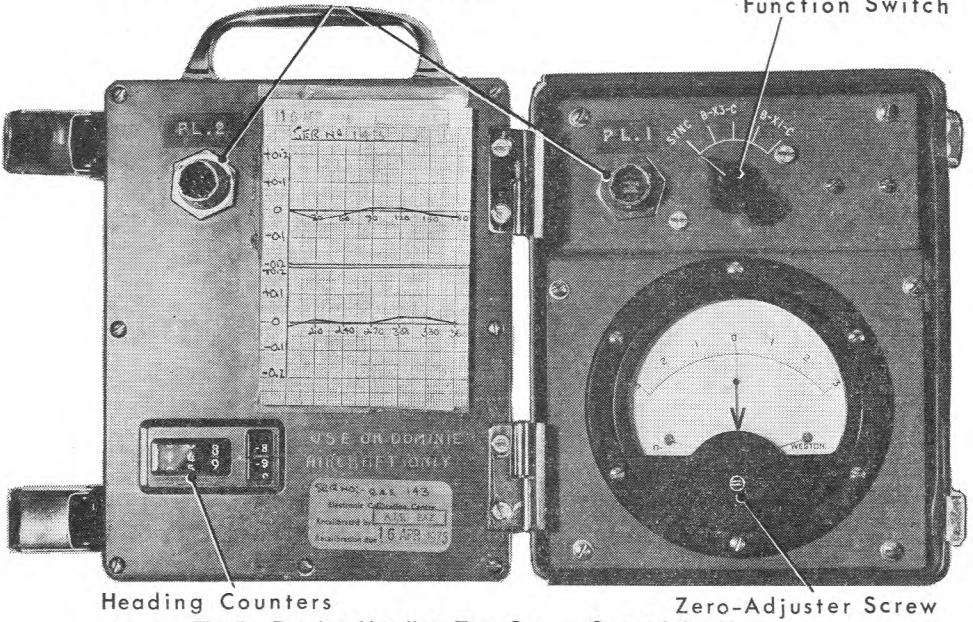


Fig 2 Precise Heading Test Set — Opened for Use

8. **Change-Over Switch.** On some compass systems, because of the design of the test socket outputs, the heading readouts on the PHTS was 180° removed from the actual heading. A modification has been introduced (which at the time of writing, 1974, has not been incorporated in all PHTSs) to add a two-position switch on the face of the test set which allows the heading readout to show the correct heading from the compass. This change-over switch is not shown in Fig 2.

9. **Test Cable Harness.** Two sockets, one on the left-hand half and one on the right-hand half of the PHTS allow the test set to be connected to the compass system by means of a cable harness. Because of the variation in the position and type of test sockets on the various compass systems, a different cable harness is required for each type of compass. Reference should be made to the compass swinging procedures for the particular aircraft/compass system to ensure that the correct cable harness is used.

Method of Use

10. The actual method of use of the PHTS during a compass calibration will depend on the type of aircraft/compass system combination being swung. However the following general points apply to all systems:

a. Ensure that the correct cable harness is used and connect the PHTS, via the harness, to the appropriate test sockets on the compass system.

b. Having switched on, synchronize the compass by reference to the voltmeter needle with the function switch set to SYNC. Allow sufficient time for any remaining error to be removed by the slaving system; depending upon the compass this can be as long as 10 minutes. It will often be found in practice that the voltmeter needle may appear to steady after a short time, but slaving will subsequently continue after this initial steady indication. It may be found that the ultimate steady position of the needle is not necessarily at the centre point of the dial, though the needle will usually steady at approximately the same point on each successive aircraft heading. If this point is determined at the start of a particular swing, it will simplify subsequent readings if, at this stage, the zero-adjuster screw is used to move the needle to the zero position. Compare the readings of the main compass indicator and the PHTS. If necessary, operate the change-over switch (if fitted) to make the PHTS read the same as the compass. If the change-over switch is not fitted and the PHTS reads 180° removed from the compass, 180° must be added to all PHTS readings. PHTS headings are used throughout the compass swing.

c. During the compass swing, the function switch is set to SYNC and the centre-reading voltmeter is used to indicate that the compass is accurately synchronized before a heading reading is taken.

d. On some compass systems, the voltages set in by the B and C potentiometers on the remote correction unit may be measured and recorded. This is normally carried out after the calibration swing when the B and C potentiometers have been adjusted. Depending upon the compass system, this is achieved either by selecting the function switch to B-×3, ×3-C, B-×1 or ×1-C as appropriate, or selecting B-×3 or B-×1 and inserting the red and white probes on the cable harness into the sockets adjacent to the B and C potentiometers. Reference should be made to the compass swinging procedures for the aircraft/compass system to determine the particular method to be used. A record of the voltages allows a replacement correction unit to be set up temporarily to the existing correction values, until such time as a compass swing can be carried out.

Note: It is imperative that, before the correction voltages are measured, the

needle of the voltmeter is zeroed, particularly if an adjustment to the zero-adjuster screw has been made during the calibration swing.

Calibration

11. Like all items of test equipment, the PHTS must be calibrated at regular intervals. Results of the calibration are recorded in the form of a graph on the front of the left-hand half (*see* Fig 2). Corrections to be applied to the PHTS heading counter readings should be extracted from this graph and applied to each reading.

PART 2

SECTION 6

COMPASS CORRECTION AND CALIBRATION

Chapter

1 **Magnetic Compass Deviations**

2 **Compass Swinging Procedures**
Annex—The Air Swing

3 **The Analysis of the Compass Swing**

4 **The Magnetic Survey of Compass Bases**

CHAPTER 1

MAGNETIC COMPASS DEVIATIONS

CONTENTS

	Paras
Introduction	1
THE EARTH'S MAGNETIC FIELD	
Components of the Earth's Field	2–4
THE AIRCRAFT'S MAGNETIC FIELD	
General	5–7
Descriptive Convention	8
THE HARD IRON FIELD AT THE SENSOR	
Resolution of the Hard Iron Field	9
Component P	10–11
Component Q	12
Component R	13
THE SOFT IRON FIELD AT THE SENSOR	
The Soft Iron Field Induced by the Hard Iron	14
The Soft Iron Field Induced by the Earth's Field ...	15–18
TOTAL DEVIATING FORCES	
General	19
Grouping of Components	20–23
Coefficients	24
THE DERIVATION OF DEVIATION CURVES AND COEFFICIENTS	
Coefficient B, due to P and cZ	25–28
Coefficient C, due to Q and fZ	29–32
Coefficient D, due to aX and eY	33–36
Coefficient E and A, due to bY and dX	37–42
Other Sources of Deviation	43–46
Combined Effects of the Deviation Components	47–48
Derivation of Coefficients from Total Deviation	49–52
THE PRINCIPLE OF COMPASS CORRECTION	
The Aim of Compass Correction	53
Correction Procedure	54
CHANGES IN COMPASS DEVIATION	
Introduction	55
Change of Magnetic Latitude	56–58
Changes in Aircraft Permanent Magnetism	59–63

Introduction

1. A magnetic sensor influenced only by the Earth's magnetic field will detect the direction of that field at its position. If installed in an aircraft, the sensor will also be influenced by the numerous magnetic fields associated with the magnetic material of the aircraft. It will indicate the direction of the resultant of all the magnetic fields, including that of the Earth, at that particular position in the aircraft. The difference between the direction of the horizontal component of the Earth's field, and the direction of the horizontal component of the resultant of all the fields at the sensor position, is the deviation angle, named easterly (positive) or westerly (negative), depending whether the resultant field direction is to the East or West of the Earth's field (*see* AP 3456G, Part 1, Sect 1 Chap 1). It will be shown that this angle can vary with the position of the sensor in the aircraft, with aircraft heading, with change of geographical position of the aircraft, and with the passage of time. In this chapter the magnetic fields which cause deviation are examined, and later chapters describe methods of calibrating—or swinging—the aircraft compass.

THE EARTH'S MAGNETIC FIELD

Components of the Earth's Field

2. Except at the magnetic equator the Earth's magnetic field is inclined to the Earth's surface, the angle of inclination being known as the angle of dip. The total field (T) can be resolved into two components, a horizontal component (H) and a vertical component (Z), as in Fig 1. The modern fluxvalve unit uses only the H component to sense the direction of the local magnetic meridian, and H can therefore be considered the directive force acting upon the sensor. Other horizontal magnetic fields will increase or decrease the directive force, or act as deviating forces. The Z component is significant in considering deviation only in so far as it contributes to the magnetism induced in the magnetic material of the aircraft.

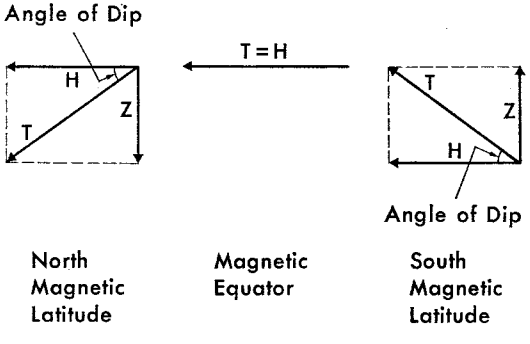


Fig 1 The Earth's Magnetic Field Resolved

3. In considering aircraft magnetism it is useful to further resolve the horizontal component H into two components, one (X) acting along the fore-and-aft axis of the aircraft, and the other (Y) acting athwartships. In this chapter the aircraft will be considered only in a level attitude, therefore the three components X, Y and Z will correspond to the three major aircraft axes, as shown in Fig 2.

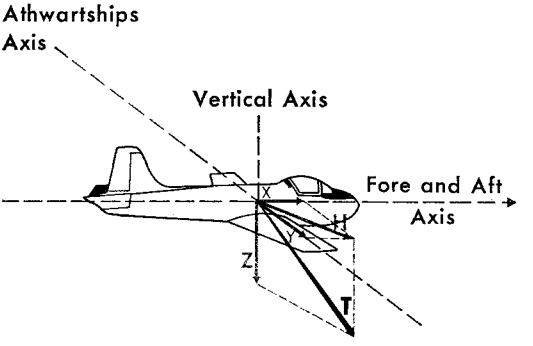


Fig 2 X, Y, Z Components of the Earth's Field

4. The values of H and Z will vary with change in magnetic latitude, and it should be noted that for a given geographical location the X and Y components of H will change with a change in aircraft heading, as shown in Fig 3, although the Z component will remain constant. A sign convention is adopted to indicate the direction in which the components act, positive components X, Y and Z acting forward, starboard and downward respectively.

THE AIRCRAFT'S MAGNETIC FIELD

General

5. Magnetic fields, other than the Earth's field, will be present in an aircraft due to:

a. *Hard Iron.* Magnetic material of the aircraft structure or aircraft components which has acquired permanent magnetism is described as hard iron. This magnetism may have been acquired during manufacture, or during the flying, servicing, or structural testing of the aircraft. The slow change of this permanent magnetism with time and

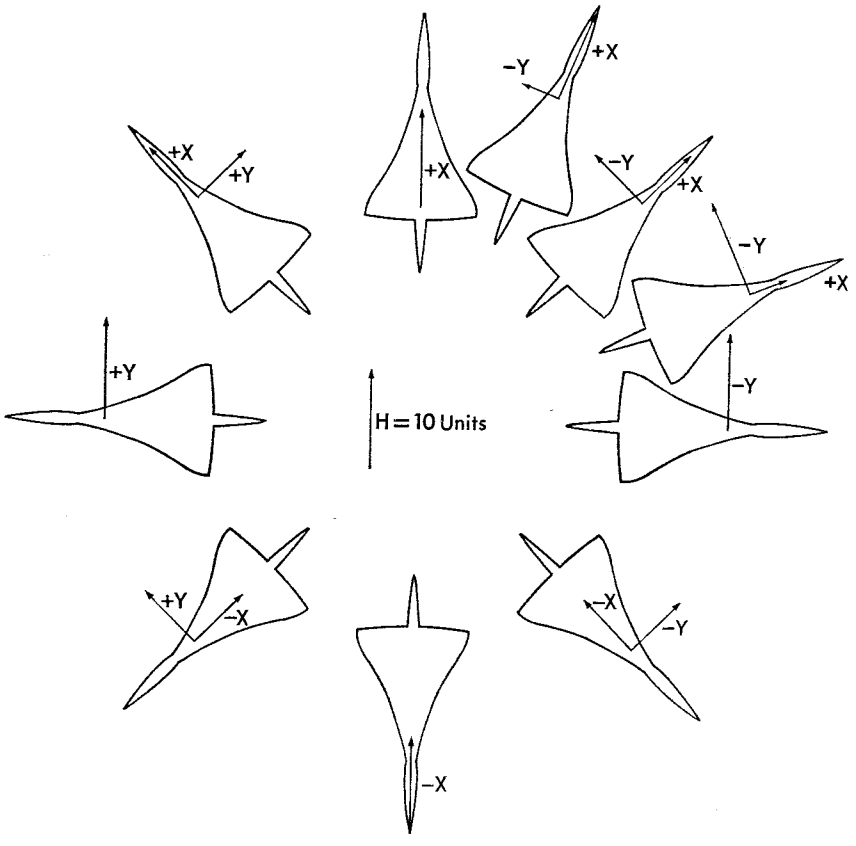


Fig 3 Change in the Magnitudes of the X and Y Components with Change of Heading

rapid changes sometimes associated with lightning strikes are mentioned later, but are ignored in the general consideration of compass deviation. Magnetic components of instruments permanently installed in the aircraft are included in the general designation hard iron.

b. *Soft Iron.* Magnetic material in which temporary magnetism is induced while in the presence of external fields is described as soft iron. The temporary magnetism may be induced by:

(1) The Earth's field.

(2) The hard iron.

(3) Electrical currents.

(4) Weapons or cargo, which may contribute to either soft or hard iron effects, or both.

6. If the effects of electrical currents were appreciable the calibration of the compass would have to be done with all systems operative, so that the direct and induced effects were corrected as part of the hard and soft iron fields. However, careful selection of the sensor position can reduce the effects to negligible proportions. Normally the sensor position is such that stores or payload also will have negligible magnetic effect, and this chapter deals therefore mainly with magnetic fields due to the hard and soft iron of the aircraft itself.

7. It will be appreciated that the division of magnetic material into hard and soft iron is quite arbitrary; the aircraft's magnetic fields are derived from innumerable pieces of magnetic material, whose intensity of magnetization differs, as does their capacity to retain magnetism. However, this treatment as two distinct sources of aircraft magnetism enable a

realistic analysis of the overall effect of aircraft magnetism on the Earth's field to be carried out.

Descriptive Convention

8. The aircraft magnetic field will be analysed by breaking it down into a series of elements each having a different source. Each element makes a contribution to the direction and strength of the magnetic field at the sensor position. The system used in this chapter to portray the effect of the constituent elements is to:

a. Represent H, the directive force, by a horizontal vector aligned with the magnetic meridian at the aircraft position.

b. Break down the aircraft magnetism into a series of elements according to source, each element being considered to act in isolation, and being represented by a vector aligned with the magnetic field which would be produced by that element at the sensor position, the vector length being proportional to field strength.

c. Resolve each element into component vectors along the principal axes of the aircraft, the fore-and-aft and athwartships components being considered as deviating forces.

d. Apply the vector representing each horizontal deviating force as a vector acting from the north end of the H vector, to determine the deviation it would produce on various aircraft headings.

THE HARD IRON FIELD AT THE SENSOR

Resolution of the Hard Iron Field

9. The many elements of hard iron will together form a permanent magnetic field, irregularly shaped, but whose orientation with respect to the aircraft axes will not change with heading; they can in short be treated as a permanent magnet fixed to the aircraft. The hard iron field at the fixed sensor position will therefore be constant in strength and, in relation to the aircraft axes, direction. In Fig 4 this field is represented by a vector, which is resolved into three components P, Q and R, acting along the fore-and-aft, athwartships, and vertical axes of the aircraft. As with X, Y and Z, components acting forward, starboard and downward are designated positive.

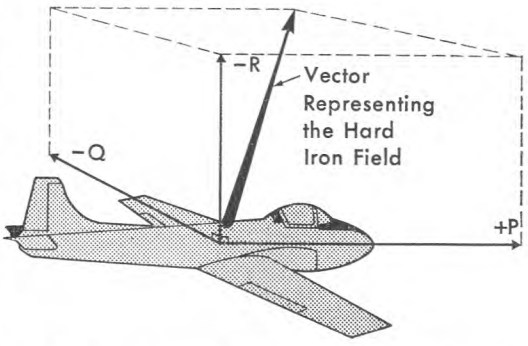


Fig 4 Resolution of the Hard Iron Field

Component P

10. Fig 5 shows the effect of a +P vector when the aircraft is heading on the cardinal and quadrantal points as established by the aircraft compass. The deviating force will cause most deviation when acting at right angles to the directive force H, that is, on compass headings East and West. On compass North and South the effect of the +P vector is to increase and decrease respectively the directive force.

11. At Fig 6 are graphs of the deviations produced by +P and −P plotted against compass headings. It will be noted that the graphs are in the form of sine curves, and it can be deduced that the deviation δ produced by P on any compass heading will be given by the formula:

$$\delta\theta = \delta_{max} \times \sin \theta, \quad \text{where}$$

$\delta\theta$ = Deviation on heading θ, and
δ_{max} = Deviation on headings 090° or 270°.

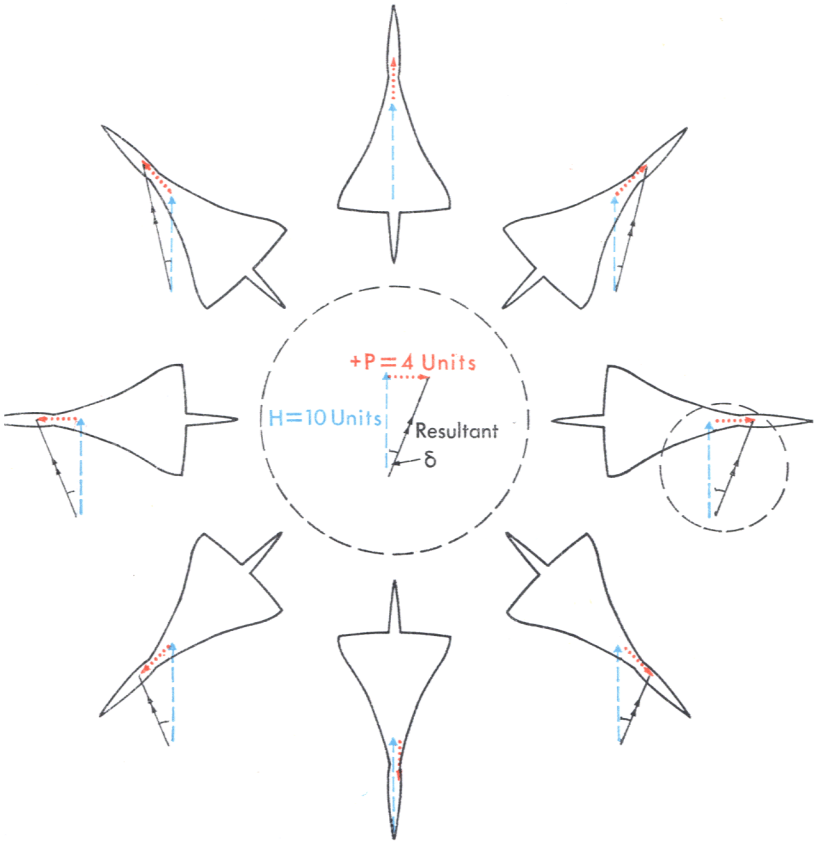


Fig 5 Deviating Effect of +P

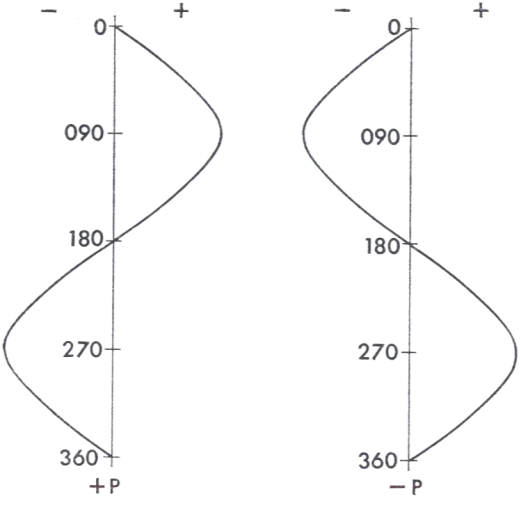


Fig 6 Graphs of Deviation due to P

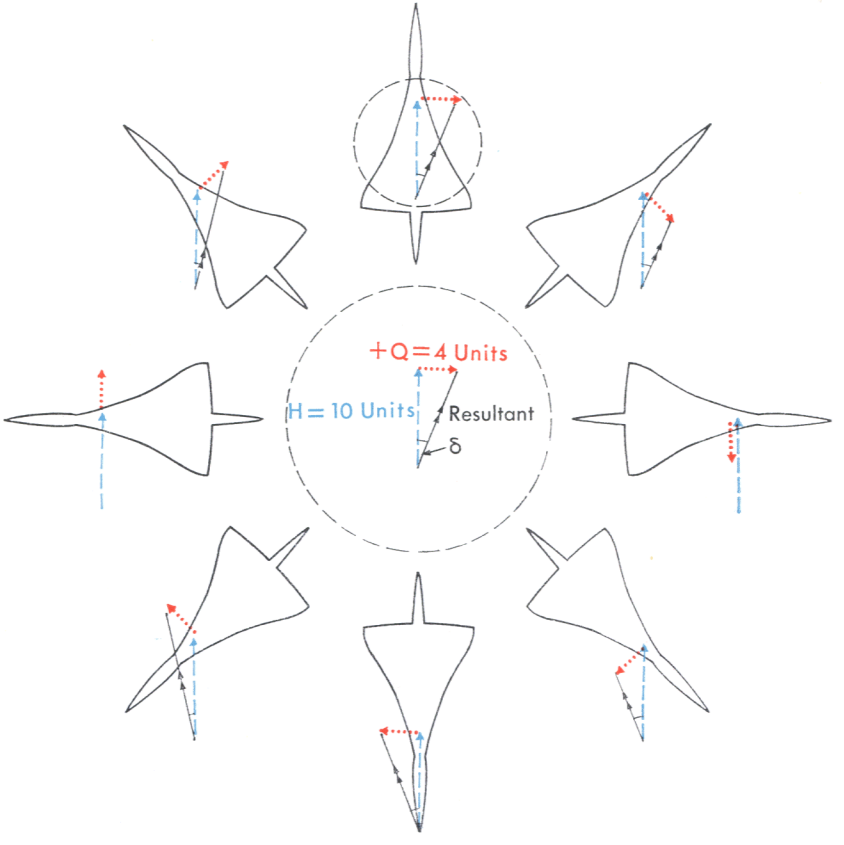


Fig 7 Deviating Effect of +Q

Component Q

12. Fig 7 is a similar illustration of the effects of +Q, and at Fig 8 are graphs of the deviations produced by +Q and −Q. It will be seen that the graphs are in the form of cosine curves, and the deviation produced by Q on any compass heading θ will be given by:

$$\delta\theta = \delta_{max} \times \cos\theta, \quad \text{where}$$

$\delta\theta$ = Deviation on heading θ, and
δ_{max} = Deviation on heading 000° or 180°.

Component R

13. Component R, the vertical component of the hard iron field at the sensor, will exercise no deviating effect when the aircraft is in a level attitude.

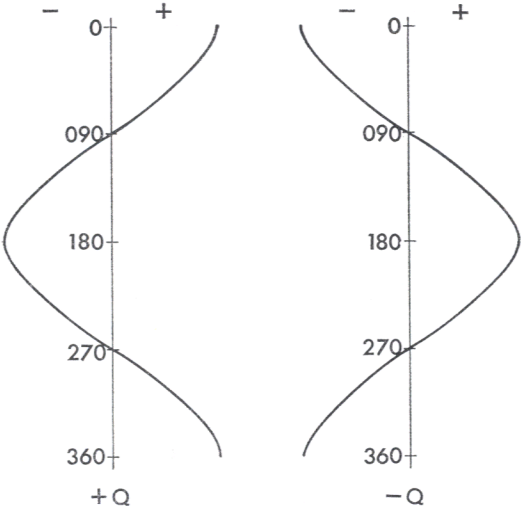


Fig 8 Graphs of Deviation due to Q

THE SOFT IRON FIELD AT THE SENSOR

The Soft Iron Field Induced by the Hard Iron

14. Magnetism will be induced in the aircraft's soft iron by two fields: the Earth's field, which is the dominant one, and the aircraft's hard iron field. The hard iron field is constant in relation to the airframe, the soft iron position is fixed, and therefore the field induced in the soft iron by the hard iron will also be constant. The effect of the soft iron magnetism induced by the hard iron will be to distort the hard iron field; the two sources of deviation are inseperable, and the earlier consideration of hard iron magnetism has therefore taken into account the total effect of the hard iron magnetism and magnetism induced by it in the soft iron.

The Soft Iron Field Induced by the Earth's Field

15. Magnetism will be induced in the aircraft's soft iron by the total Earth's field (T). It is however simpler to consider separately the effects of the three elements X, Y and Z (Fig 2) of the Earth's field. Each of these elements will induce a three-dimensional field in the soft iron, and the horizontal components of the induced fields will have a deviating effect at the sensor position. The amount of deviation will depend on:

a. The amount, the permeability, and the location in relation to the sensor, of the soft iron. For a given aircraft these are constant.

b. The geographical location of the aircraft. The inclination and field strength of T vary with change of position, therefore elements X, Y and Z will vary.

c. For elements X and Y, the heading of the aircraft (Fig 3).

16. As with hard iron, the aircraft soft iron is composed of numerous metal elements in the aircraft, but as their positions are fixed (with minor exceptions), the soft iron can be considered as a single block fixed in relation to the airframe. In Fig 9 the soft iron is represented by a rectangular block, and two possible sensor positions by L_1 and L_2. The block is considered as being influenced only by the X element of the Earth's field. The lines of force shown indicate the horizontal field that might result from magnetism induced in the block if the induced field could be considered in isolation.

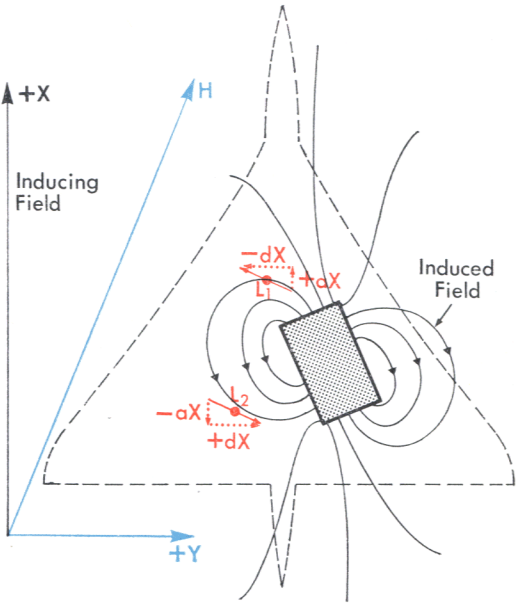


Fig 9 Horizontal Components aX and dX of a Three-Dimensional Field Induced by X

Although X acts along the fore-and-aft axis of the aircraft the induced field would be three-dimensional, and its vector representation could be resolved along all three aircraft axes. Only the components (shown in Fig 9 and labelled aX and dX) along the horizontal axes would have a deviating effect. It will also be noted from Fig 9 that:

a. The further the sensor is removed from the effective soft iron mass the weaker will be the deviating effect of the induced field (wider spacing of the lines of force).

b. At different sensor positions within the induced field the deviating effect will vary. In fact, once the sensor position has been decided by the manufacturer, the sensor will be fixed in relation to the soft iron.

17. **Components of Soft Iron Magnetism.** As the three components X, Y and Z of the Earth's field are each considered to induce soft iron fields, and as the vector representing each of these fields can be resolved into three vectors along the aircraft axes, there will be a total of nine soft iron components. A two-letter designator is assigned to each component, the first letter of which can be considered as representing a constant factor, independent of heading

or magnetic latitude, of the inducing component indicated by the second letter. Thus whilst the inducing field will vary with heading and change of magnetic latitude, the field induced by it in the soft iron is assumed to be a constant proportion of the inducing field. This assumption is sufficiently accurate for practical purposes. The designators assigned to the components of the soft iron magnetism are listed in Table 1.

Inducing Field	Soft Iron Field Components		
	Fore-and-Aft	Athwartships	Vertical
X	aX	dX	gX
Y	bY	eY	hY
Z	cZ	fZ	kZ

Table 1 Soft Iron Components

18. **Vector Sign Convention.** The components of fields induced in the soft iron are represented by vectors similarly named. It has been shown (Fig 9) that a +X field can produce an aX component acting either forward or backward, depending on the position of the sensor in the induced field. It will be shown that some components act along reciprocal directions on different headings. As it is necessary to identify the directions in which the soft iron deviating fields act, a sign convention is adopted whereby the component is named positive if it acts forward or starboard on aircraft headings in the North West quadrant.

TOTAL DEVIATING FORCES

General

19. The total deviating forces acting at the sensor are summarized in Table 2. As the magnetic sensor detects only the resultant of the horizontal component of the Earth's field and the horizontal components of the deviating fields, it is unnecessary to consider further the vertical components R, gX, hY, and kZ of the deviating fields.

Grouping of Components

20. **Components P and cZ.** Z does not change with alteration of the aircraft heading, therefore neither will cZ. The effects of P and cZ, both of which act fore and aft, can be considered together, taking due account of their signs and relative magnitudes.

21. **Components Q and fZ.** Component fZ is also independent of heading, and can be considered with the other constant athwartships component Q.

22. **Components aX and eY.** These two components are later shown to produce similar patterns of deviation, and they are examined together.

23. **Components bY and dX.** These components are also similar in effect, and are examined together.

Coefficients

24. The effects of each pair of deviating field components will be shown to produce deviations which vary as sine or cosine functions of the aircraft's heading. The size of the deviation for a particular pair of deviating components will be a maximum on headings for which the appropriate trigonometrical function is a maximum, which, in the case of sines and cosines, is when it equals ± 1. The product of this maximum deviation in degrees and the appropriate trigonometrical function of any heading will give the deviation produced by that pair on that heading. The maximum deviation is termed a coefficient, and is assigned a letter to indicate the pair of components to which it refers. The coefficients are given positive and negative signs, the conventions used being explained later.

Axis along which the force acts	Deviating Force	
	Hard iron component	Soft iron components
Fore and-Aft	P	aX, bY and cZ
Athwartships	Q	dX, eY and fZ
Vertical	R	gX, hY and kZ

Table 2 Total Deviating Forces

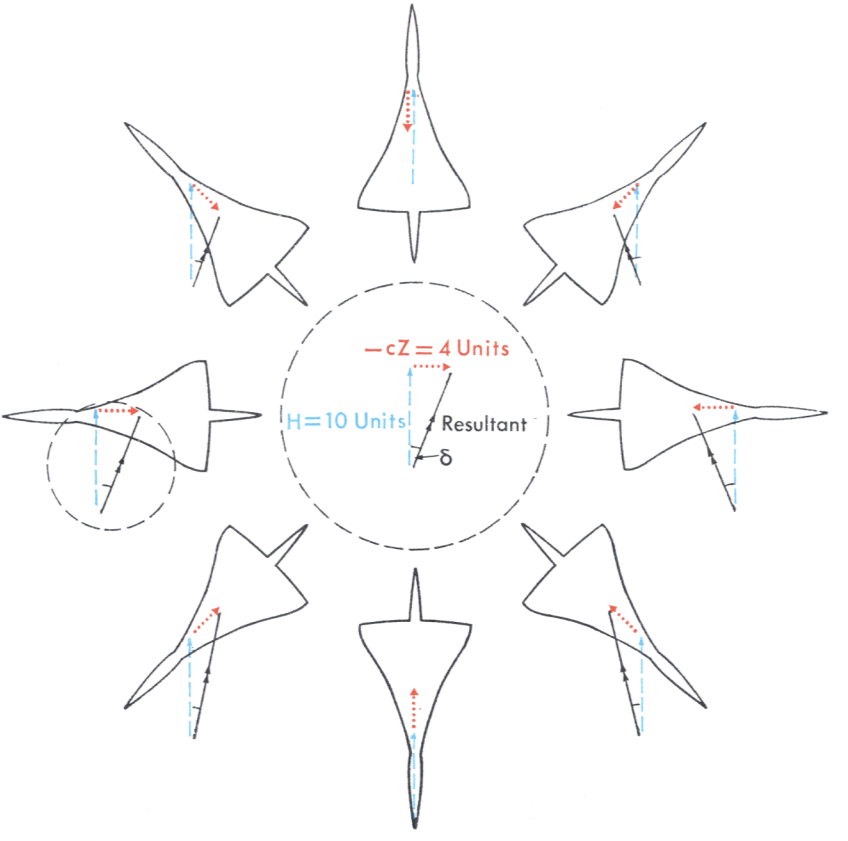


Fig 10 The Effect of a −cZ Component

THE DERIVATION OF DEVIATION CURVES AND COEFFICIENTS

Coefficient B, due to P and cZ

25. **P.** The deviating effect of hard iron component P, acting fore-and-aft, has been shown to have the form of a sine curve (Fig 6).

26. **cZ.** Component cZ is the fore-and-aft component of the aircraft's soft iron temporary magnetism induced by the vertical component Z of the Earth's field at the sensor. Fig 10 shows the effect of a −cZ component on the cardinal and quadrantal headings. The maximum deviations occur on the East and West compass headings; there is no deviation on compass North and South; on North the directive force is decreased and it is increased on compass South (as in the case of a −P component). Fig 11 shows graphs of the deviations caused by +cZ and −cZ. As with P, the graphs have the form of sine curves, either negative or positive according to the sign of cZ.

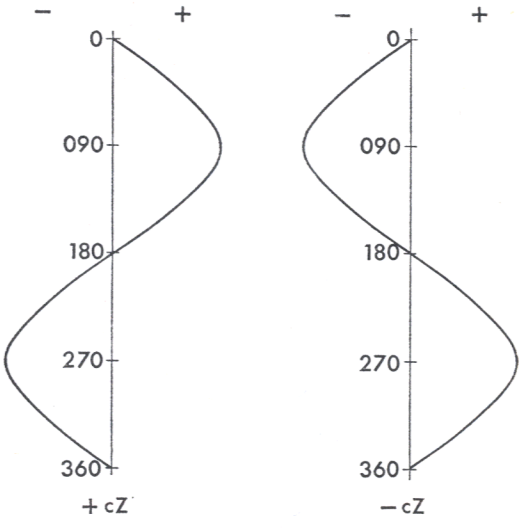


Fig 11 Graphs of Deviation due to cZ

27. **P and cZ.** Since both P and cZ have the same effect on the compass they can be grouped together, irrespective of sign. When both components have the same sign the deviations caused by each independently are added together making a larger resulting deviation. When they are of opposite sign their effects tend to cancel out, and the net result is a smaller deviation. This is illustrated in Fig 12. If the components were of opposite sign and exactly equal in effect, there would be no resultant deviation.

28. **Coefficient B.** The maximum values of deviation occur on headings of 090°(C) and 270°(C). Thus if the deviations δ_E and δ_W due to P and cZ are measured on these headings, the value of coefficient B can be found from the formula:

$$\text{Coefficient B} = \frac{\delta_E - \delta_W}{2}$$

The deviations must be given their correct signs, and subtraction is necessary because the deviations on East and West are of opposite signs (Fig 12). Coefficient B is positive when the deviation due to P and cZ on compass East is easterly. Once coefficient B has been determined, the deviation due to P and cZ on any compass heading θ can be obtained from the equation:

$$\delta\theta = B \sin \theta$$

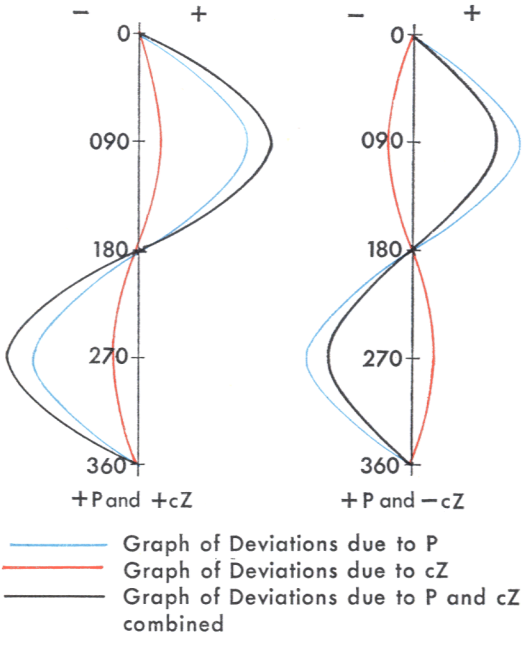


Fig 12 Combined Graphs of Deviation due to P and cZ

Coefficient C, due to Q and fZ

29. **Q.** The hard iron component Q, acting athwartships, produces deviations in the form of a cosine curve (Fig 8).

30. **fZ.** Component fZ is the athwartships component of the soft iron field induced by Z. The effects are similar to those for Q of the same sign.

31. **Q and fZ.** Since both Q and fZ are represented by similar vectors they can be considered together. Fig 13 shows the effect of a +Q component acting in combination with a +fZ component. The maximum deviations occur on compass headings North and South. There is no deviation on East and West, but in the example illustrated there will be an increase in the directive force on West and a decrease on East. Fig 14 shows graphs of the deviations caused by Q and fZ of the same sign. If the components are of different signs the effect will be to decrease the total deviation; the resultant curves will be flatter, and will be negative or positive cosine curves depending on the sign of the larger vector.

32. **Coefficient C.** The maximum values of deviation occur on headings of 360°(C) and 180°(C). Thus if the deviations δ_N and δ_S due to Q and fZ are measured on these headings, the value of coefficient C can be found from the equation:

$$\text{Coefficient C} = \frac{\delta_N - \delta_S}{2}$$

The deviations must be given their correct signs, and as with coefficient B subtraction is necessary as the deviations on North and South are of opposite signs (Fig 14). Coefficient C is positive when the deviation due to Q and fZ on compass North is easterly. When coefficient C has been determined, the deviation due to Q and fZ on any compass heading θ can be obtained from the equation:

$$\delta\theta = C \cos \theta$$

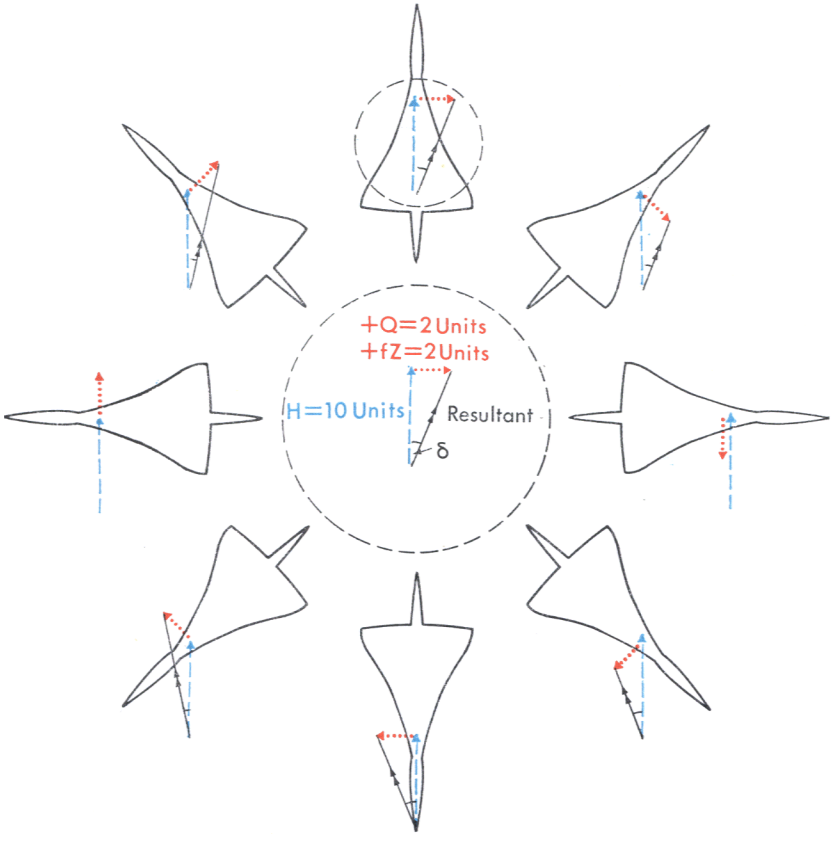


Fig 13 The Effect of +Q and +fZ Components

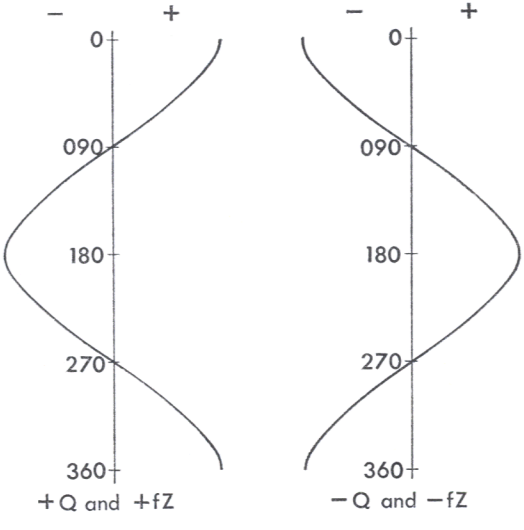


Fig 14 Combined Graphs of Deviation due to Q and fZ

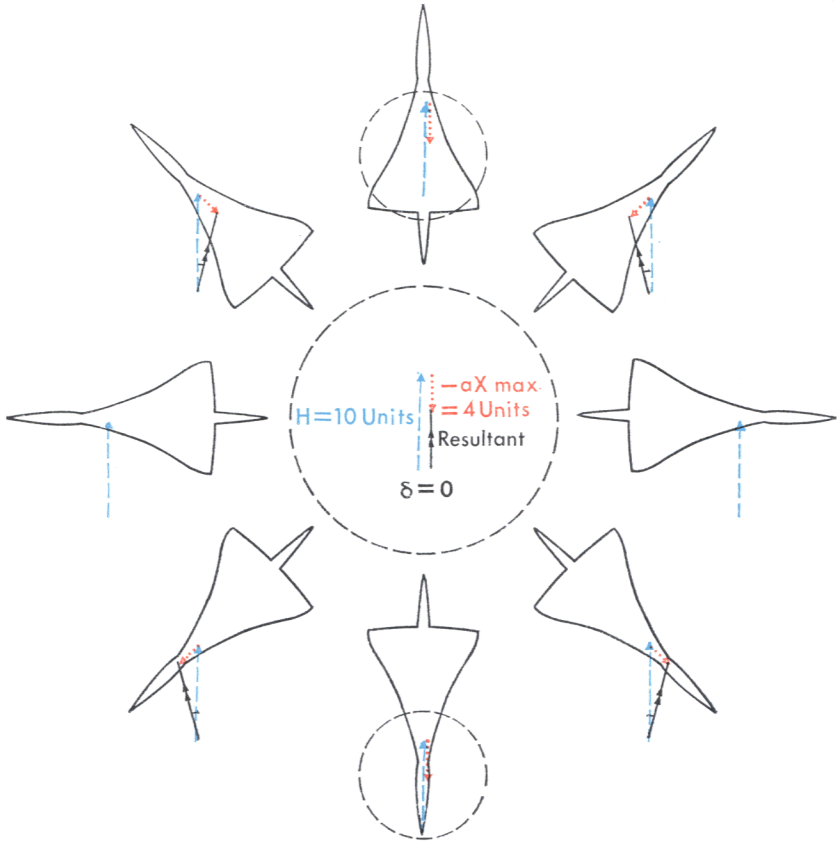


Fig 15 The Effect of a −aX Component

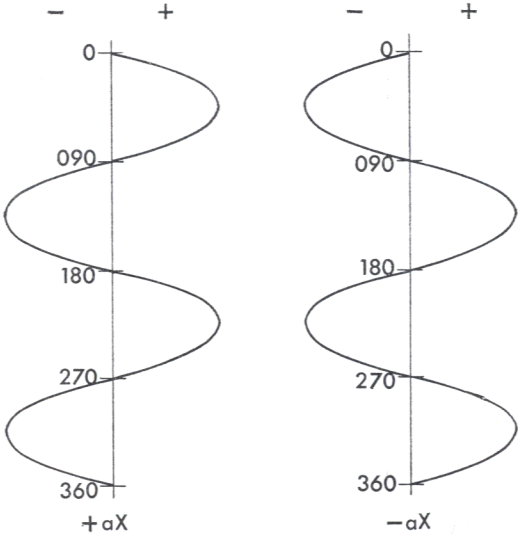


Fig 16 Graphs of Deviation due to aX

Coefficient D, due to aX and eY

33. **aX.** aX is the component acting fore-and-aft of the soft iron magnetism induced by the X (fore-and-aft) component of the Earth's field. On headings North and South, X is at its greatest (Fig 3) and the induced field aX will also be at its greatest. As both H and the aX vector are in line there will be no deviating effect, but merely a loss or gain of directive force. On headings East and West, X is zero (Fig 3), no aX field will be created and therefore there will be neither a change in directive force nor any deviation due to aX. Between the cardinal headings there will be an aX field, not aligned with H, which will therefore produce deviations, greatest on quadrantal headings. Fig 15 shows the effect of a −aX component, and Fig 16, graphs of deviations caused by +aX and −aX. The graphs take the form of double sine curves, *ie* their shapes are governed by the values of the sines of twice

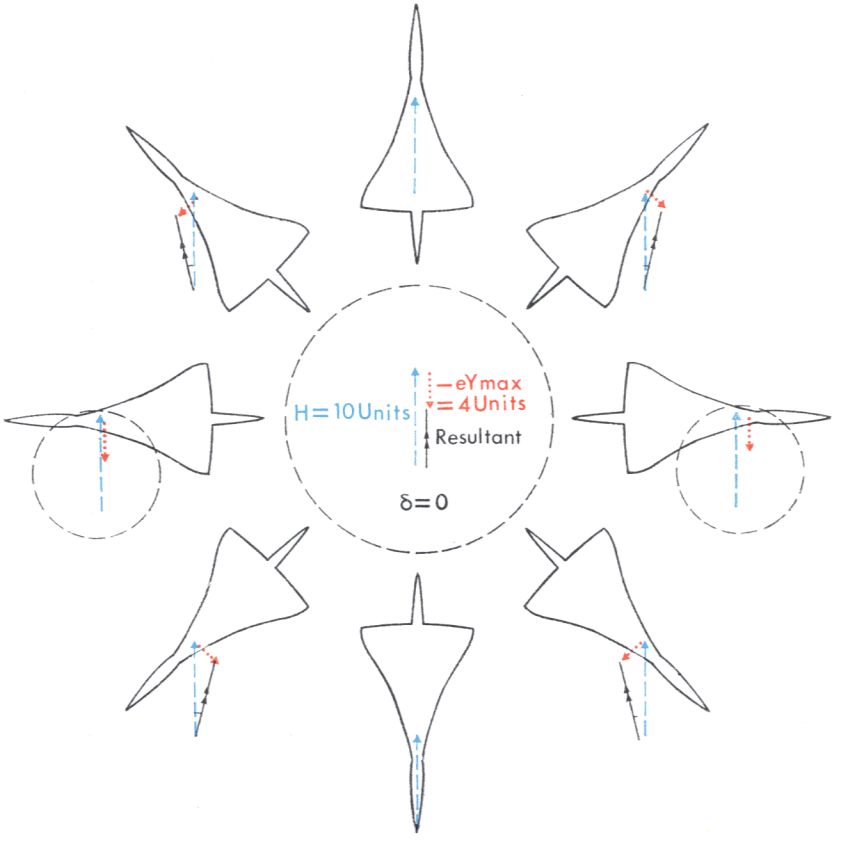


Fig 17 The Effect of a −eY Component

the compass headings, the deviations being maximum on the four quadrantal headings.

34. **eY.** eY is the athwartships component of the soft iron magnetism induced by the Y (athwartships) component of the Earth's field. On headings East and West, Y is at its maximum (Fig 3) and the induced field eY will also be at its maximum. As the vector is in line with H there will be no deviating effect, but only a loss or gain of directive force. On headings North and South, Y is zero (Fig 3) and therefore no eY field will be created. The maximum deviations will occur on the quadrantal headings, where there is an appreciable component Y and the induced eY field is not aligned with H. Fig 17 shows the effect of a −eY component and Fig 18, graphs of the deviations caused by +eY and −eY. As with aX, the graphs take the form of double sine curves.

35. **aX and eY.** As the deviations caused by the aX and eY components are similar, they

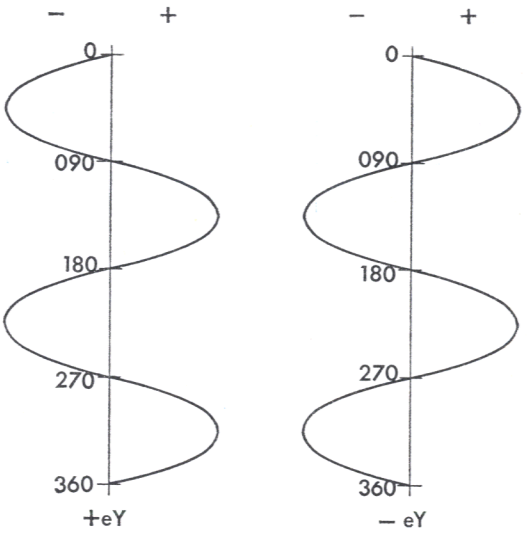


Fig 18 Graphs of Deviation due to eY

can be considered together. It is apparent from Figs 16 and 18 that aX and eY of the same sign will tend to cancel each other, whereas aX and eY of different signs will be additive. Unless the two vectors are of the same sign and equal, when there will be no deviation, the resultant curve will, like those of its components, be a positive or negative double sine curve, with maximum deviations on the quadrantal headings.

36. **Coefficient D.** If the deviations δ_{NE}, δ_{SE}, δ_{SW}, and δ_{NW} due to aX and eY on the quadrantal headings are measured, the value of coefficient D can be found from the equation:

$$\text{Coefficient D} = \frac{(\delta_{NE} + \delta_{SW}) - (\delta_{NW} + \delta_{SE})}{4}$$

The deviations must be given their correct signs when applying the formula. Coefficient D will be positive when the deviations due to aX and eY on compass headings North-East and South-West are easterly. Knowing the value of coefficient D, the deviation due to aX and eY on any other compass heading θ can be obtained from the equation:

$$\delta\theta = \text{D} \sin 2\theta$$

Coefficients E and A due to bY and dX

37. **bY.** bY is the fore-and-aft component of temporary magnetism induced by the Y (athwartships) component of the Earth's magnetic field. On compass headings North and South there is no deviation, since on these headings the Y component of the Earth's field is zero, and no magnetism will be induced by it. The maximum deviations occur on compass headings East and West, where the maximum induction takes place and the deviating forces are at right angles to the H field. On the quadrantal headings the deviations have intermediate

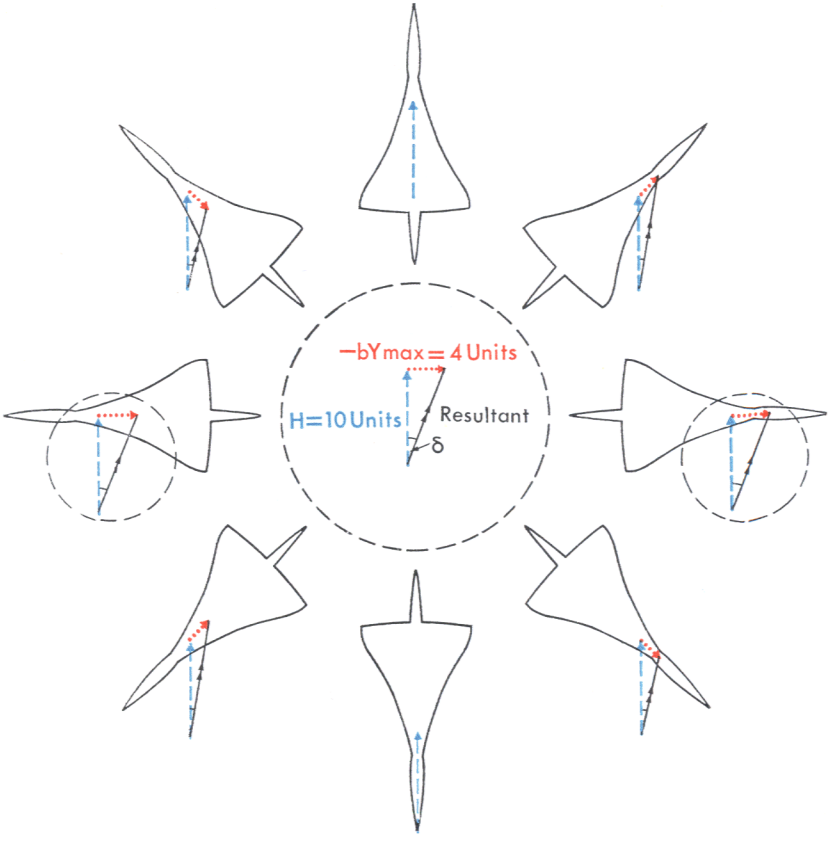


Fig 19 The Effect of a −bY Component

values. The deviations have the same sign on all headings; in the example illustrated at Fig 19, that of −bY, they are all positive, or easterly. On North and South the directive force is unchanged. The graphs of deviations due to +bY and −bY are shown in Fig 20. The shape of the graphs is that of a double cosine curve, *ie* a curve of values which vary as the cosines of twice the compass headings.

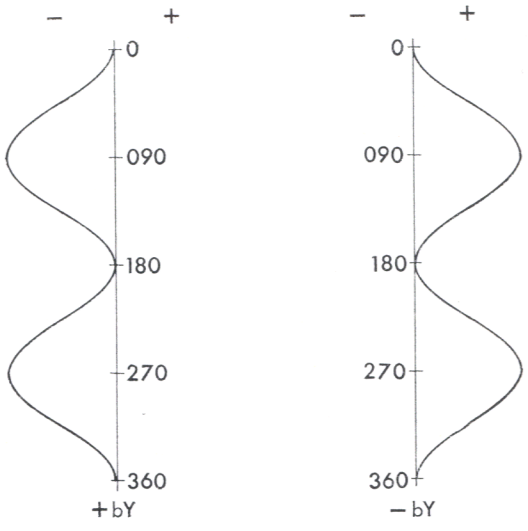


Fig 20 Graphs of Deviation due to bY

38. **dX.** dX is the athwartships component of the aircraft's temporary magnetism induced by the X (fore-and-aft) component of the Earth's field. Fig 21 shows the effect of a +dX component. There is no deviation on compass East and West since X is zero on these headings. Maximum deviations occur on compass North and South, and on the quadrantal headings the deviations have intermediate values. All the deviations have the same sign; in the case illustrated, that of +dX, they are positive. The directive force is unchanged on East and West.

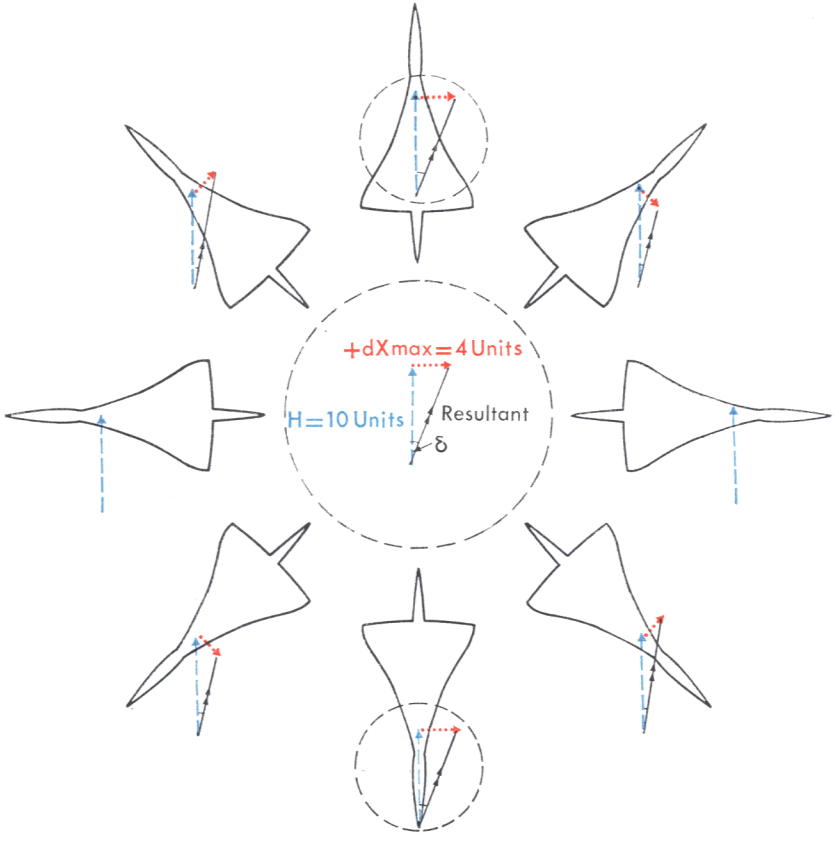


Fig 21 The Effect of a +dX Component

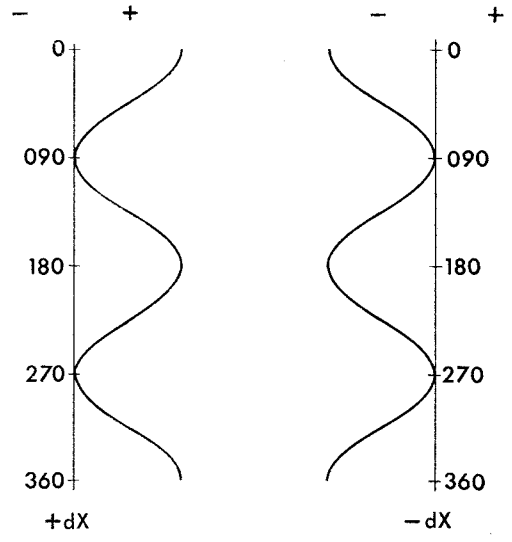


Fig 22 Graphs of Deviation due to dX

Graphs of the deviations caused by +dX and −dX are shown at Fig 22. The graphs show that, in common with those due to bY, the deviations due to dX are of the form of a double cosine curve deplaced to one side or other of the zero axis.

39. **bY and dX.** Since the deviations due to bY and dX take similar forms, they can be considered together, for various combinations of bY and dX:

a. *bY and dX Equal and of Like Sign.* The combination for positive components is shown in Fig 23. The resultant deviation curve takes the form of a double cosine curve, symmetrically disposed about the zero axis, with amplitude twice that of either component. It gives maximum deviations on the four cardinal headings, and not just on two as did the individual components.

b. *bY and dX Equal and of Unlike Sign.* Fig 24 illustrates the result of combining equal +bY and −dX components: a negative deviation constant on all headings. Similar results are obtained by combining equal −bY and +dX components, the constant deviation in this case being positive.

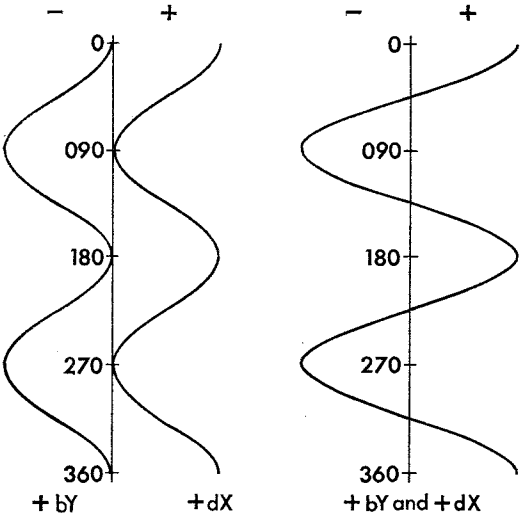


Fig 23 Combination of Deviation due to Equal Components of +bY and +dX

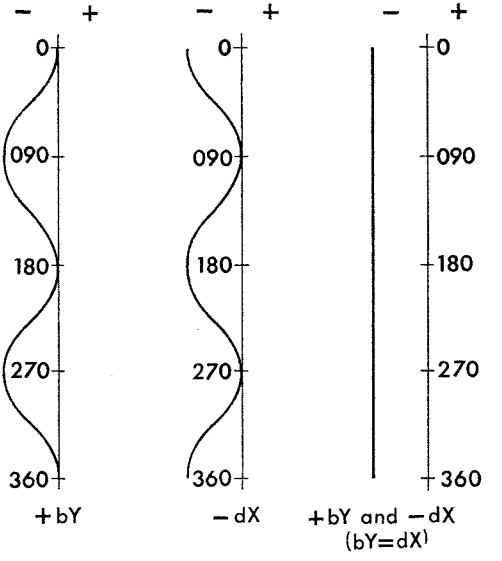


Fig 24 Combination of Deviation due to Equal Components of +bY and −dX

c. *bY and dX Unequal and of Like Sign.* Fig 25 shows the results of combining unequal bY and dX components of like sign, dX being the greater. The resultant graph has the shape of a double cosine curve of large amplitude, with its axis displaced from the central axis of the graph. Similar results will be obtained when the value of bY is the greater, or when both signs are negative and bY and dX are unequal.

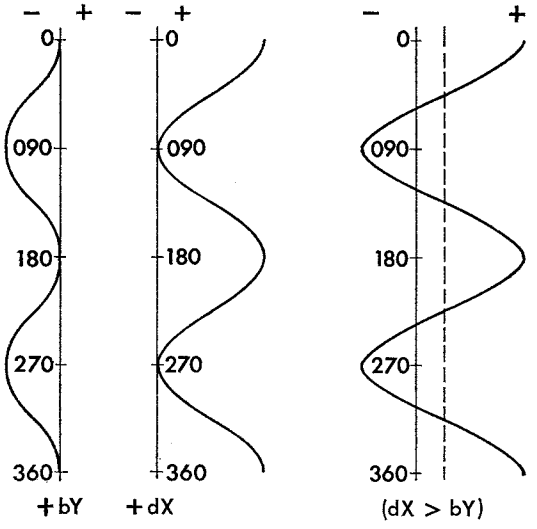


Fig 25 Combination of Deviation due to Unequal Components of +bY and +dX

d. *bY and dX Unequal and of Unlike Sign.* Fig 26 shows that combining a +bY with a smaller −dX component results in a double cosine shaped curve of small amplitude, displaced from the zero axis. Part of the +bY component can be considered as having combined with the −dX component of equal strength to produce a constant deviation, the remainder of the +bY component superimposing a small variable deviation of the double cosine type on this constant value. Similar results are obtained with other combinations of unequal bY and dX components of unlike sign.

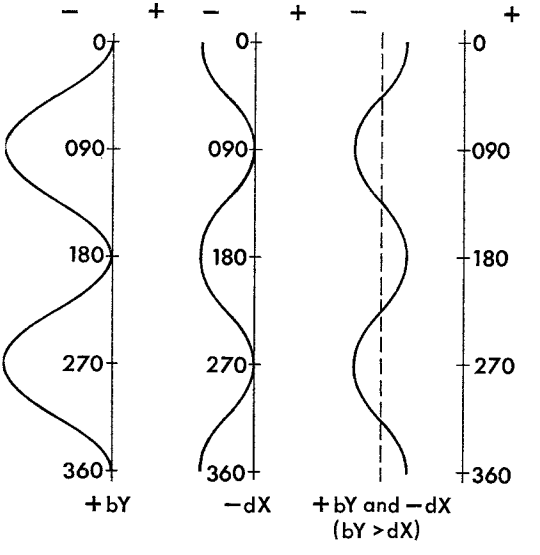


Fig 26 Combination of Deviations due to Unequal Components of +bY and −dX

40. **Coefficients Associated with bY and dX.** It has been shown that bY and dX will produce:

a. If equal, a constant deviation, or one which varies as the cosine of twice the heading.

b. If unequal, a constant deviation and one which varies as the cosine of twice the heading.

It is therefore convenient to consider the deviation due to bY and dX in two parts, the variable deviation being represented by coefficient E (para 41), and the constant deviation by coefficient A (para 42).

41. **Coefficient E.** The maximum values of deviation occur on the cardinal headings. If the deviations δ_N, δ_E, δ_S and δ_W on the cardinal headings are measured, the value of coefficient E is given by:

$$\text{Coefficient E} = \frac{(\delta_N + \delta_S) - (\delta_E + \delta_W)}{4}$$

The deviations must be given their correct signs when substituting in the equation. Coefficient E is positive when the deviation on compass North is easterly. The variable deviation due to bY and dX on any compass heading θ can be found from:

$$\delta\theta = E \cos 2\theta$$

42. **Coefficient A.** Coefficient A represents the constant deviation due to the vectors bY and dX. It is determined by taking the average of the deviations measured on any number of equally spaced headings. The more headings used the greater the accuracy, and whilst for the initial correction of a compass before calibration it is customary to calculate coefficient A from only the four quadrantal headings, it is normally calculated from observations on eight headings for deviation analysis, thus:

$$\text{Coefficient A} = \frac{1}{8}(\delta_N + \delta_{NE} + \delta_E + \delta_{SE} + \delta_S + \delta_{SW} + \delta_W + \delta_{NW})$$

Coefficient A is positive or negative, depending on whether the constant deviation which it represents is easterly or westerly.

Other Sources of Deviation

43. So far only deviations due to permanent and induced magnetism of the aircraft have been considered. Other causes of incorrect indication of the magnetic meridian are mentioned briefly below.

44. **Index or Alignment Error.** If the sensor is not correctly aligned with the axis of the aircraft, or if the transmission synchros are out of alignment, an error constant for all headings will be present. The effect will be similar to that of coefficient A, and to permit differentiation in discussion, the error due to bY and dX is termed Real A, and the error due to other causes Apparent A. In practice it is not necessary to distinguish them, and they are both understood to be included in the term coefficient A.

45. **Electrical Fields.** Direct currents will create fields which have similar effects to hard iron magnetism. The effects can be determined by calibrating the aircraft with and without the appropriate circuits operating, but provided that the sensor is in a remote part of the aircraft the effect of the fields will be negligible.

46. **Transmission Errors.** With remote indicating compasses, impedance and voltage imbalances in the flux valve and synchros can cause errors of the sin 2 θ or cos 2 θ form. These are in fact usually greater than those due simply to induced magnetism, but again it is unnecessary for calibration purposes to differentiate the source of the errors, and both are covered in the examination of coefficients D and E.

Combined Effects of the Deviation Components

47. The total magnetism due to the aircraft structure which affects the magnetic sensor in the horizontal plane has been split into two hard iron components P and Q, and six soft iron components, aX, bY, cZ, dX, eY and fZ. These are grouped according to the similarity of their effects on the compass in the following way:

a. P and cZ giving a deviation of the form B sin θ.

b. Q and fZ giving a deviation of the form C cos θ.

c. aX and eY giving a deviation of the form D sin 2 θ.

d. bY and dX (equal amplitudes, like signs) giving a deviation of the form E cos 2 θ equally disposed about the zero axis.

e. bY and dX (unequal amplitudes, like signs) giving a deviation of the form E Cos 2 θ displaced from the zero axis.

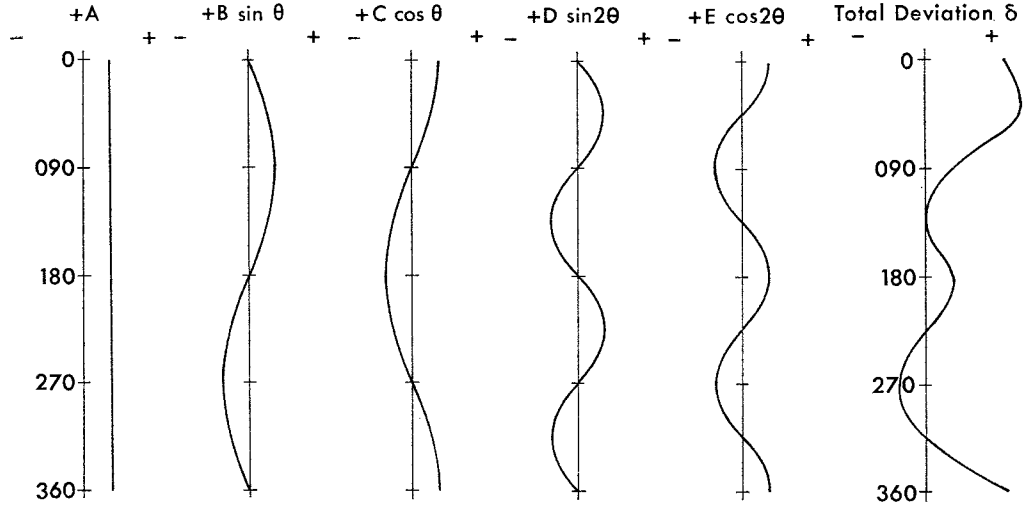


Fig 27 Graphs of Component Deviations and Total Deviation

f. bY and dX (equal amplitudes, unlike signs) giving a deviation of A.

g. bY and dX (unequal amplitudes, unlike signs) giving a deviation of E cos 2 θ displaced from the zero axis.

A, B, C, D and E are the coefficients representing the maximum deviations caused by the individual sets of components, and θ is the compass heading of the aircraft.

48. Since the expressions B sin θ *etc* represent the values of deviations due to a given set of components, the sum of the expressions will give the deviation due to the total aircraft magnetism acting in the horizontal plane. Fig 27 illustrates the graphs of individual deviations added to give a combined graph representing the total deviation. The total deviation δ for any compass heading θ is given by:

$$\delta = A + B \sin\theta + C \cos\theta + D \sin 2\theta + E \cos 2\theta$$

Derivation of Coefficients from Total Deviation

49. In the foregoing paragraphs it has been shown how the values of individual coefficients can be determined from maximum deviations caused by the corresponding component of the aircraft's magnetic field. All the previously discussed components act simultaneously and there are no practical means of separating their effects. It will be shown, however, that if the total deviation due to the simultaneous effect of all components is measured on the eight compass headings at which the individual maxima occur, the values of all coefficients can be obtained by a simple analysis of the deviation equation given above.

50. By substitution in the deviation equation:

$$\delta = A + B\sin\theta + C\cos\theta + D\sin 2\theta + E\cos 2\theta$$

the total deviations measured on compass cardinal and quadrantal headings will be expressed by the following equations:

$$\delta_N = A + C + E \quad (1)$$

$$\delta_{NE} = A + B\sin 45° + C\cos 45° + D \quad (2)$$

$$\delta_E = A + B - E \quad (3)$$

$$\delta_{SE} = A + B\sin 45° - C\cos 45° - D \quad (4)$$

$$\delta_S = A - C + E \quad (5)$$

$$\delta_{SW} = A - B\sin 45° - C\cos 45° + D \quad (6)$$

$$\delta_W = A - B - E \quad (7)$$

$$\delta_{NW} = A - B\sin 45° + C\cos 45° - D \quad (8)$$

Thus there are eight independent equations from which to determine five unknown coefficients. The sum of all equations will give an expression:

$$\Sigma\,\delta = 8A$$

as all other coefficients cancel. From this the value of Coefficient A can be determined as:

$$A = \tfrac{1}{8}\Sigma\,\delta$$

where $\Sigma\,\delta$ denotes the algebraic sum of total deviations measured on eight equally-spaced compass headings. It will be observed that Coefficient A can be evaluated by taking the sum of only four equations for symmetrically disposed headings, *eg* equations (1), (3), (5) and (7), or equations (2), (4), (6) and (8).

51. By forming the difference of equations (3) and (7), the relation

$$\delta_E - \delta_W = 2B$$

is obtained as, again, all other coefficients contributing to the total deviations on East and West cancel. From these two deviations Coefficient B is found:

$$B = \tfrac{1}{2}(\delta_E - \delta_W)$$

Similarly, from the difference of equations (1) and (5)

$$\delta_N - \delta_S = 2C \text{ and}$$

$$C = \tfrac{1}{2}(\delta_N - \delta_S)$$

Taking now the four equations in which Coefficient D appears, *ie* equations (2), (4), (6) and (8), two differences can be formed, between (2) and (4), and (6) and (8). This will eliminate Coefficients A and B giving the following equations:

$$\delta_{NE} - \delta_{SE} = 2C\cos 45° + 2D$$

$$\delta_{SW} - \delta_{NW} = -2C\cos 45° + 2D, \text{ from which}$$

$$D = \tfrac{1}{4}[(\delta_{NE} - \delta_{SE}) + (\delta_{SW} - \delta_{NW})]$$

$$E = \tfrac{1}{4}[(\delta_N - \delta_E) + (\delta_S - \delta_W)]$$

Finally, from equations (1), (3), (5) and (7), which include Coefficient E:

$$\delta_N - \delta_E = -B + C + 2E$$

$$\delta_S - \delta_W = B - C + 2E$$

52. The five coefficients of the deviation equation can thus be evaluated from the total compass deviations measured on eight suitable headings. It is now possible, using the deviation equation, to calculate the total deviation for any compass heading.

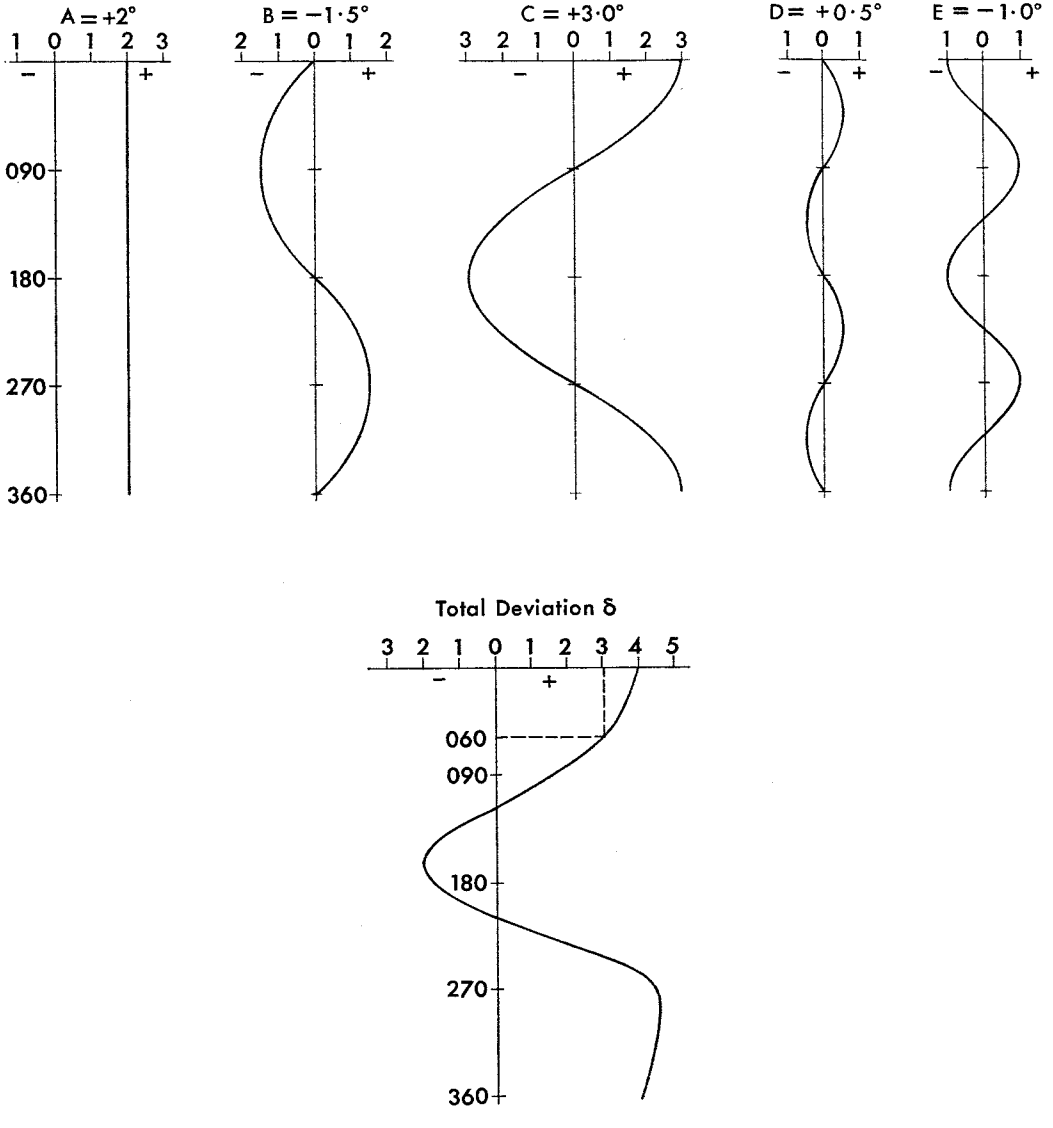


Fig 28 Graph of Total Deviation

Example. The coefficients calculated for a particular compass fitted in a certain aircraft were A = + 2·0°, B = − 1·5°, C = + 3·0°, D = + 0·5° and E = − 1·0°. What is the total deviation when the compass heading of the aircraft is 060°(C)?

$$\delta_{60} = A + B\sin\theta + C\cos\theta + D\sin 2\theta + E\cos 2\theta$$

$$= +2{\cdot}0 - 1{\cdot}5\sin 60° + 3{\cdot}0\cos 60° + 0{\cdot}5\sin 120° - 1{\cdot}0\cos 120°$$

$$= +2{\cdot}0 - 1{\cdot}5\times 0{\cdot}87 + 3{\cdot}0\times 0{\cdot}5 + 0{\cdot}5\times 0{\cdot}87 - 1{\cdot}0\times(-0{\cdot}5)$$

$$= +2{\cdot}0 - 1{\cdot}3 + 1{\cdot}5 + 0{\cdot}4 + 0{\cdot}5$$

$$= +3{\cdot}1$$

Thus on heading 060°(C) the total deviation is + 3·1. The magnetic heading of the aircraft will be 063·1°(M). If the graphs of the individual coefficients in this example are plotted, and combined to give the total deviation curve (*see* Fig 28), it will be seen that on 060°(C) the plotted deviation is + 3·1°.

THE PRINCIPLE OF COMPASS CORRECTION

The Aim of Compass Correction

53. It has been shown that the value of the coefficient determines the amplitude of the deviation curve for the components represented by that coefficient. If the coefficients are reduced in size, the graphs will be flattened and more nearly approach the central axis of the graph. If all the coefficients can be reduced the total deviation curve will also be reduced in amplitude, until, when all the coefficients are zero, the deviation curve will become a straight line coincident with the central axis, *ie,* there will be no deviation at all. It is the purpose of compass correction to approach this condition of zero deviation as closely as possible by reducing the values of the coefficients. This is achieved by setting up, by means of a corrector device, magnetic fields which are equal in strength but opposite in direction to those caused by the components of the aircraft magnetism. In practice it is impossible to reduce the coefficients entirely.

Correction Procedure

54. The procedure for compass correction and calibration, described in detail in Chap 2, is as follows:

a. Measure the deviation on those headings where the coefficients have their maximum values.

b. Calculate the coefficients.

c. Reduce the coefficients by the use of a corrector device.

d. Measure the remaining deviations (called residual deviations) on a number of equally spaced headings.

e. Record the residual deviations so that they will be available for correcting the indicated compass readings.

CHANGES IN COMPASS DEVIATION

Introduction

55. In the foregoing examination of aircraft magnetism a constant Earth field and a constant hard iron component of aircraft magnetism have been assumed. If the magnetic latitude of the aircraft is changed the directive force H will change. Over a long period of time, or for reasons described in paras 58-61, the hard iron component will change. In either case the ratio of the hard iron deviating force to the Earth's directive force will alter, and there will be an increase or decrease in the resulting deviation angle. As regards deviation caused by the soft iron components, both the deviating and directive components will be increased or decreased when the magnetic latitude is changed. The effects of changes in the Earth's field and changes in the aircraft's permanent magnetism are summarized below.

Change of Magnetic Latitude

56. **Hard Iron Components.** Deviation due to P and Q varies inversely as the directive component (H) of the Earth's field.

57. **Soft Iron Components.** Deviation changes are as follows:

a. *cZ and fZ.* The magnitude of the components varies in direct proportion to the inducing component Z, and the resultant deviation will therefore be proportional to Z/H.

b. *aX, eY, bY and dX.* The deviations caused by these components would vary in

inverse proportion to H, but for the fact that the vector magnitude is directly proportional to H. The deviations due to these components therefore remain constant with changes in magnetic latitude.

58. **Effect of Change of Magnetic Latitude after Correction.** If P and Q alone were completely compensated at one position using magnetic correctors, the compensating deviation set up by the magnets of the corrector would also vary inversely as H. Compensation would remain correct at all places over the Earth. In practice, however, (P − cZ) and (Q − fZ) are neutralized by the magnetic corrector, and although there will be no change in deviation due to the P and Q components, coefficients B and C will change with position due to the changes in the Z-induced components cZ and fZ.

Changes in Aircraft Permanent Magnetism

59. If an aircraft is left on one heading for any length of time the Earth's field will induce magnetism in the magnetic material of the aircraft. This magnetism will alter the permanent magnetism already in the aircraft, hence the instruction that a compass will be recalibrated after the aircraft has been standing on the same heading for more than four weeks.

60. On long flights on one heading the deviations caused by the permanent magnetism will alter. This is because the aircraft is being vibrated within the Earth's field and will thus adopt the induced magnetism more quickly. It is accepted that the changes are relatively insignificant.

61. The carriage of stores may affect the permanent and temporary magnetism of the aircraft. Chap 3 shows a way of finding whether a particular load is likely to affect the aircraft deviations. It may occasionally be desirable to fit two deviation cards, one for the aircraft with the load, and the other without it.

62. Modern aircraft with good bonding, more efficient electric dischargers and remote indicating compasses are less affected by lightning strikes than older types of aircraft. But it should be borne in mind that a lightning strike causes an intense electrical discharge through the aircraft's frame that must alter its magnetic characteristics. AP 3158 (Technical Services Manual) Vol 2 Leaflet B22 and DCIs state the actions to be taken in the event of a lightning strike. The main requirements are that the compass be swung before the next flight; if there is a marked change in the coefficients A, B, or C, the aircraft must be de-magnetized; the compass system is then to be regarded as suspect for two months and re-swung once a fortnight until the deviations have settled down.

63. If the permanent magnetism of an aircraft does change, the resultant deviation changes are directly proportional to the changes in magnitude of the hard iron vectors P and Q.

CHAPTER 2

COMPASS SWINGING PROCEDURES

CONTENTS

	Paras
Introduction	1—3
Occasions for a Compass Swing	4—5
THE STANDARD SWING	
Introduction	6
Checks Before Swinging the Compass	7
Accuracy Limits of the Swing	8
Recording the Swing	9—10
The Correcting Swing	11
The Calibration Swing	12
Correction for Coefficient A	13—14
THE REFINED SWING	
Introduction	15—16
General Procedure	17—18
Additional Corrections	19—20
THE ELECTRICAL SWING	
Introduction	21—22
The General Procedure	23—30
Corrections	31
Annex: The Air Swing	

Introduction

1. In the previous chapter the causes of magnetic compass deviations were described and the component deviating fields illustrated. In this chapter practical methods of measuring aircraft compass deviations and correcting the compass indications are described. The procedures can be adapted to suit nearly all compass systems in use today. Only ground swings are described; methods of carrying out air swings are dealt with in the Annex.

2. The accuracy with which deviations are measured and corrected depends on two main factors:

a. The accuracy to which it is possible to read both the compass and the datum instrument during the swing.

b. The accuracy requirements stipulated by the user, which will usually depend on how important the magnetic compass is to the aircraft primary navigation system.

3. Compass swinging procedures are specified by the user commands; most military aircraft are swung to high standards of accuracy using the Refined Swing, although for aircraft which are not fitted with Doppler equipment, the Standard Swing procedure may be acceptable. The electrical Compass Swing provides accurate compass calibration without turning the aircraft.

Occasions for a Compass Swing

4. The occasions on which an aircraft compass is to be swung are listed in DCIs. Provided that the aircraft is in regular use, the maximum period between compass swings is to be six months for aircraft fitted with weapon or supply aiming systems requiring high order accuracies and nine months for all other aircraft. Both these periods may be extended by a maximum of six weeks to coincide with aircraft servicing cycles.

5. In addition, aircraft compasses are to be calibrated and adjusted on the following occasions:

a. On acceptance by a user unit, if the aircraft has been delivered direct from a contractor to the unit.

b. Before delivery of the aircraft from a storage unit to a user unit. The standard swing is normally to be carried out regardless of aircraft type.

c. After an aircraft has been repaired or subjected to conditions likely to affect the compasses. Examples of such repairs and conditions are as follows:

(1) A change of component within the compass system likely to create a significant change of deviation.

(2) On change of position, replacement, addition or permanent removal of any magnetic material, or alteration to any electrical circuit in the vicinity of a direct reading compass or compass detector unit of a remote reading compass.

(3) When an aircraft has been standing, heading in one direction, for four weeks or more.

(4) When the aircraft has been subjected to conditions of severe static electricity or magnetic crack detection examination.

(5) On transfer of the aircraft from one theatre of operations to another, if this entails a large change of magnetic latitude. This need not apply to aircraft on detachments of less than four weeks, unless higher order accuracy is operationally required.

(6) At any time when the accuracy of the compass or deviation card is in doubt.

THE STANDARD SWING

Introduction

6. Usually the Standard Swing is for aircraft not fitted with Doppler equipment. The swing may be broken down into three phases:

a. The Correcting Swing, which is the calculation of and adjustment for coefficients B and C.

b. The Calibration Swing, which is the calculation and recording of the deviations.

c. The calculation and correction for coefficient A. On initial installation the A correction should be done first. When a large coefficient A (2° or more) has been removed, the swing for coefficients B and C must be repeated.

Checks Before Swinging the Compass

7. Before starting a compass swing the following general points should be checked to prevent embarrassment and delay:

a. Check the compass for serviceability.

b. Collect all the items required for the calibration of the compass, *eg* Prismatic Compass (Medium Landing), centre reading Voltmeter, corrector keys and compass log book.

c. Ensure that the weather is suitable and that the compass swinging base is available for use.

d. Ensure that non-magnetic tools are available.

e. Remove any item of moveable equipment which may affect the magnetic sensor, *eg* tool-kits or spares in the vicinity of the magnetic sensor.

f. Obtain permission from ATC to tow the aircraft to the compass base.

g. Check that the appropriate power source is available and that both tractor and generator have adequate fuel.

h. Brief the compass swing team on the procedures to be followed. Ensure that personnel remove from their person all metallic objects which are likely to cause deviation. For instance, SD hats should not be worn by datum observers. If the swing is conducted with the aircraft engines running, non-magnetic ear defenders should be worn by the datum observer rather than a headset.

j. Before starting the swing, switch on aircraft electrical circuits which are likely to cause deviation in flight, but observe any restrictions on ground operation of equipments.

Accuracy Limits of the Swing

8. Individual commands state the limits of accuracy of the Standard Swing.

Recording the Swing

9. Results of a Standard Swing are to be entered in the RAF Form 343 (RN FA66), the Aircraft Compass Log Book. Fig 1 is a completed sheet of F343 on which the residual deviations have been plotted to show the critical headings and their associated deviations. The point at which the residual deviation curve crosses from one value of deviation to another is called the critical heading.

10. **The Deviation Cards.** There are various deviation cards in Service use (RAF Forms 316 C, D, E and F). The reverse of the cards show how the deviation should be entered and extracted. The example shown at Fig 2 is the F316C which is for use at the navigator's position. When the F316 has been completed from the F343 it is to be fitted in the holder in the aircraft and the old deviation cards destroyed. The following entry is to be made in the F700: "Compass swung, deviation cards placed in the aircraft".

The Correcting Swing

11. During the correcting swing coefficients B and C are calculated and corrected. All compass headings must be within 5° of those required. The procedure is as follows:

a. Turn the aircraft on to North and record the exact readings of the aircraft compass and datum compass.

b. Turn the aircraft on to East and record the readings.

c. Turn the aircraft on to South and record the readings.

d. Calculate coefficient C using the formula:

$$\text{Coeff C} = \frac{\delta N - \delta S}{2}$$

e. Correct for coefficient C as follows:

(1) Change the sign of coefficient C and add it algebraically to the aircraft compass reading.

(2) Activate the correction device to make the compass show the required reading. The solenoid type adjuster marked "C", or the athwartships micro-adjuster should be used.

A/C TYPE & No JAVELIN 8 XH988 DATE 12 8 61 PLACE LEUCHARS
MAIN COMPASS TYPE GM4B D.U. No L 13199 G.U No 676/57 M.I. No L4779
AMP No L 18364 S/BY COMPASS TYPE & No E2B 10164A/55 ADJUSTER J. Jones Fg. Off

CORRECTING SWING

		MAIN COMPASS			STANDBY COMPASS		
APPROX HDG	DATUM HDG (a)	COMPASS HDG (b)	DEV (a)-(b)	COEFFICIENTS	COMPASS HDG (c)	DEV (a)-(c)	COEFFICIENTS
N	001	359 1/4	+ 1 3/4		005	-4	
E	090 3/4	092 1/4	- 1 1/2		097	-6 1/4	
S	176	178 1/2	- 2 1/2	C = (DEV N - DEV S)/2	178	-2	C = (DEV N - DEV S)/2
COEFF C (SIGN CHANGED)		-2			+1		
MAKE COMPASS READ		176 1/2	= (+1 3/4 + 2 1/2)/2 = +2 1/8		179	= (-4 + 2)/2 = -1	
W	272	270	+ 2	B = (DEV E - DEV W)/2	272	0	B = (DEV E - DEV W)/2
COEFF B (SIGN CHANGED)		+1 3/4			+3		
MAKE COMPASS READ		271 3/4	= (-1 1/2 - 2)/2 = -1 3/4		275	= (-6 1/4)/2 = -3 1/8	

CALIBRATION SWING

APPROX HDG	DATUM HDG (d)	COMPASS HDG (e)	DEV (d)-(e)	RESIDUAL DEV (DEV - A)	COMPASS HDG (f)	DEV. (d)-(f)	RESIDUAL DEV (DEV - A)
N	000 1/2	000 3/4	- 1/4	0	003	- 2 1/2	+ 1/2
NE	045 1/2	046 1/4	- 3/4	- 1/2	047	- 1 1/2	+ 1 1/2
E	089 1/2	089 3/4	- 1/4	0	094	- 4 1/2	- 1 1/2
SE	134 1/4	135	- 3/4	- 1/2	140	- 5 3/4	- 2 3/4
S	178 3/4	179	- 1/4	0	183	- 4 1/4	- 1 1/4
SW	224 1/2	224 1/2	0	+ 1/4	226	- 1 1/2	+ 1 1/2
W	272	271 3/4	+ 1/4	+ 1/2	275	- 3	0
NW	315 1/4	315	+ 1/4	+ 1/2	319	- 3 3/4	- 3/4
		TOTAL = -1 3/4 ÷ 8 = A			TOTAL = -26 3/4 ÷ 8 = A		
		A = - 1/4			A = - 3		
		RESIDUAL COEFFICIENTS B = -1/4 C = 0			RESIDUAL COEFFICIENTS B = -3/4 C = +7/8		

TO REMOVE +A, TURN LUBBER LINE OR DETECTOR UNIT CLOCKWISE.
TO REMOVE -A, TURN LUBBER LINE OR DETECTOR UNIT ANTI-CLOCKWISE

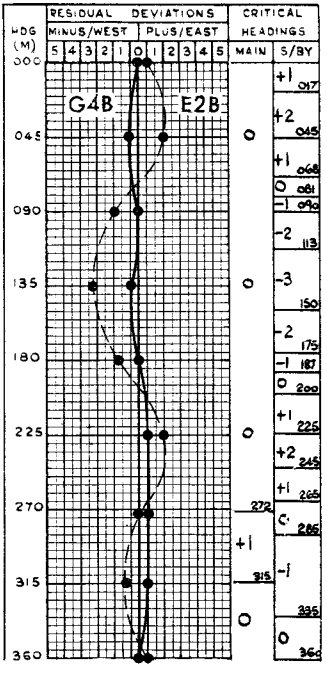


Fig 1 The Form 343

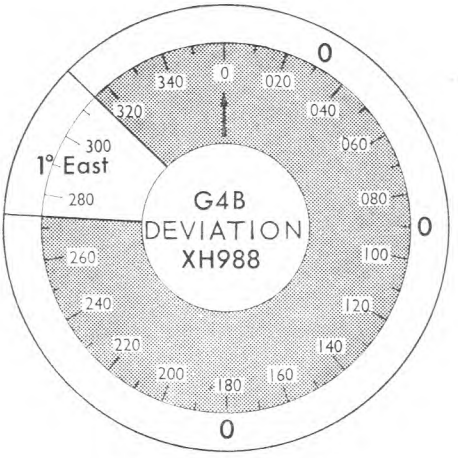


Fig 2 The Deviation Card

f. Turn the aircraft on to West and note the readings.

g. Calculate coefficient B using the formula:

$$\text{Coeff B} = \frac{\delta E - \delta W}{2}$$

h. Correct for coefficient B as follows:

(1) Change the sign of coefficient B and add it algebraically to the aircraft compass reading.

(2) Activate the correction device (the fore-and-aft micro-adjusters) to make the compass show the required reading.

The Calibration Swing

12. The calibration swing deviations are used to calculate coefficient A and to determine the residual deviations. The procedure is to turn the aircraft on to NW, N, NE, E, SE, S, SW and W in turn, recording the aircraft and datum compass readings. Enter these on the F343 and calculate the deviations.

Correction for Coefficient A

13. Coefficient A may be calculated from the deviations found in the calibration swing by adding them algebraically and dividing by eight:

$$\text{Coeff A} = \frac{\delta NW + \delta N + \delta NE + \delta E + \delta SE + \delta S + \delta SW + \delta W}{8}$$

14. Coefficient A may be corrected for as follows:

a. *Pivoted Compasses.* Apply coefficient A algebraically to the compass reading (on any heading). Set the required heading against the lubber line, loosen the compass bowl securing screws and turn the bowl until the compass needle is parallel to the grid wires. Tighten the securing screws.

b. *Remote Indicating Compasses.* Apply coefficient A algebraically to the compass reading. The method to make the compass show the required reading varies from one system to another. In general, coefficient A should be removed at the detector unit, although in some systems the correction, if small, can be made at the variation setting control. The method to be used for a particular compass system is described in the chapter of this volume dealing with that specific compass, or in the appropriate technical publication.

THE REFINED SWING

Introduction

15. The Refined Swing is used to calibrate the compass systems of aircraft fitted with Doppler equipment. The checks to be performed before the swing are as details in para 7 except that a Watts Datum Compass, a Precise Reading Test Set and sighting rods are used. Great care must be taken by the compass swing team to ensure that both datum instrument and aircraft compass are read to the limits of accuracy simultaneously. The compass system must be allowed to settle down after every change of heading before the actual reading is taken, since even one inaccurate reading may necessitate the whole swing being repeated. After the corrections have been made to the compass system, the results of the calibration swing are used as the basis of a Fourier analysis of the swing.

16. User commands stipulate the accuracy and limits to which their aircraft must be swung. In general, coefficients should be reduced to less than 0·5°, and the 50% error of an observed deviation to less than ±0·30°. There are many types of compass system in use in the Service, and they may differ from each other in the way in which the swing should be done. The following procedures are general, and reference should be made to descriptions of specific compass systems for detailed adjustment procedures.

The General Procedure

17. **The Correcting Swing.** The purpose of the correcting swing is to reduce all the correctable coefficients to within the limits set. The swing may have to be repeated several times to achieve the required accuracy. Each correcting swing is to be entered in the appropriate page of the

F343B (Naval Form A66B). The procedure is as follows (all headings to be within 5° of those stated):

a. Head the aircraft on South, record the aircraft and datum compass readings.
b. Head the aircraft on West, record the readings.
c. Head the aircraft on North, record the readings.
d. Head the aircraft on East, record the readings.
e. Calculate the deviations.
f. Sum the deviations algebraically and divide by four to find coefficient A.
g. Apply coefficient A to the compass reading (sign unchanged) and correct the compass.
h. Calculate coefficient B $\left(\frac{\delta E - \delta W}{2}\right)$
j. Apply coefficient B (sign unchanged) to the resultant compass reading after correcting for coefficient A, and with the aircraft still on East correct the compass.
k. Calculate coefficient C $\left(\frac{\delta N - \delta S}{2}\right)$
1. Turn the aircraft on South, apply coefficient C (sign changed) to the compass reading and correct the compass.

When there are no coefficients to be corrected the calibration swing may start.

18. **The Calibration Swing.** The aircraft is moved through a twelve-point swing and the datum and aircraft compass readings are recorded every 30°. The deviations obtained from this swing form the basis of the Fourier and accuracy analyses described in Chapter 3.

Additional Corrections

19. **Coefficients D and E.** Although coefficients D and E are to be calculated, most compass systems do not incorporate any correction device for the deviations due to these coefficients. An exception is the C-12 compass fitted to the Hercules aircraft. The C-12 system includes both D and E correctors, which are used to compensate for D and E values greater than 0·4°. On other aircraft, if it is suspected that the D and E errors are caused by misalignment of transmission components, the coefficients can sometimes be removed by changing the main compass units.

20. **Deviating Fields in the Vertical Axis.** Although deviating fields in the vertical axis were not described in any detail in Chapter 1, it is nevertheless obvious that aircraft fitted with tail-wheels should be swung in a flying attitude, otherwise part of the deviations measured will be due to the vertical deviating fields. Since there are very few aircraft remaining in Service which do not have tricycle undercarriages the procedures for removing deviation errors caused by the vertical components of aircraft magnetism are not described here.

THE ELECTRICAL SWING

Introduction

21. In Standard and Refined compass-swinging, the aircraft (and therefore the compass detector unit) is rotated in the Earth's magnetic field. At each required heading the compass readout is compared with the heading of the aircraft as measured by the datum compass. The aircraft is positioned on a North-South line and not moved in an electrical swing. The area used for calibration need not be free from magnetic disturbances, provided magnetic stability exists.

22. Rotation of the aircraft within the Earth's magnetic field is simulated by connecting the MC-1M Compass Calibrator (*see* Sect 5, Chap 4) to the compass detector unit and varying the current flow within the detector unit. Either the compass error is read directly from the calibrator set, or the compass readout is compared with the heading selected on the MC-1M, depending on the particular installation.

General Procedure

23. The general swing procedure consists of the following operations which must be performed in the order given:

a. Carry out a magnetic survey of the swing site.
b. Mark a magnetic North-South line at the site.
c. Establish the MC-1M turntable and monitor positions.
d. Check the direction of magnetic North.
e. Magnetically align the detector unit.
f. Determine the E1 and E2 voltages.
g. Determine the detector unit crosstalk errors.
h. Optical transfer of the detector unit into the aircraft location.
j. Compass swing.

Stages a, b and c must be carried out when the proposed swing site is surveyed, and then again annually (*see* Chapter 4).

24. **Check the Direction of Magnetic North.** The North-South line indicates the direction

of the magnetic meridian at the time of the area magnetic survey. To allow for any change in the direction of the magnetic meridian, the direction of magnetic North at the monitor location is determined, and adjustment for a change is made in the misalignment calculation.

25. **Align the Detector Unit Magnetically.** The detector unit is installed on the turntable location. The detector unit is aligned to magnetic North by measuring the index (A) error of a four-point manual swing and off-setting as required. For a manual swing the turntable is physically rotated on to each of the cardinal headings. A further four-point manual swing is then carried out to obtain the manual swing values, corrected for index error.

26. **Determine the E1 and E2 Voltages.** The E1 and E2 voltages are used to produce the required DC currents for the electrical swing. The DC currents are fed to the secondary coil of the detector unit to generate the magnetic field appropriate to the heading selected. The E1 current is applied to the A-leg coil, and the E2 current to the B and C-leg coils. The E1 and E2 voltages are required to cancel the field in the detector unit when the appropriate leg is aligned with magnetic North. It is necessary to allow for any changes that may occur in the earth's field between the time the E1 and E2 voltages are measured and the time of the compass swing. It is assumed that a change of the field in the Monitor will be the same as the change at the turn-table location. The field in the Monitor is measured and noted.

27. **Determine the Detector Unit Crosstalk Errors.** Crosstalk errors are caused by differing coil sensitivities and unequal air-gaps separating the collector horns. The effect is to offset the field from that originally intended. The errors are measured at 90°, 180° and 270° headings, and are the differences between the electrical swing errors and the manual swing errors corrected for index error. Crosstalk error does not occur at 0° heading as no currents are applied to the detector unit to simulate this heading.

28. **Optical Transfer of the Detector Unit into the Aircraft Location.** The aircraft is towed into position with the longitudinal axis located as near as possible over the marked North-South line, with the nose of the aircraft pointing North. The aircraft is positioned so that the detector unit access is directly over the turntable location. The detector unit is then transferred from the turntable to the aircraft. The optical alignment equipment is used to maintain the alignment of the detector unit during the transfer. (A target at least ½ nm distant should be used.) The vernier scale permits adjustment of the telescope azimuth to allow for misalignment of the aircraft from the North-South line.

29. **Compass Swing.** The compass swing can now be carried out. If the compass system readout is used, the console could be set up in the aircraft. Final adjustment of the E1 and E2 voltages is made just prior to the swing. The swing procedure adopted depends on the type of compensation available. Desired headings at intervals of 15° can be selected with the flick of a switch. The time interval between readings is determined by the rate of synchronization available in the compass system.

30. **Detailed Procedures.** A detailed step-by-step explanation of the electrical compass swing is contained in TO 5N3-3-7-1.

Corrections

31. The corrections for compass deviation are the same as in conventional compass swinging. Coefficients A, B and C measured by the refined swing and the electrical swing should, in theory, be the same. Further, the two-cycle errors which are due to defects in compass system manufacture, commonly known as transmission error, should be identical. However, the deviation caused by horizontal soft-iron is twin-cyclic, but is not measured in an electrical compass-swing, except on a reading of magnetic North. Because the aircraft is stationary in the electrical swing, the deviating force due to horizontal soft iron is of constant magnitude and direction, and may cause incorrect measurement of coefficients B and C. This effect is likely to be much more significant in small aircraft than in large aircraft, since the detector unit will be much closer to the deviating force.

THE AIR SWING

CONTENTS

	Paras
Introduction	1– 2
Methods of Determining Heading in the Air	3– 8
Swing Procedures	9–11

Introduction

1. The standard method of swinging an aircraft compass is to tow the aircraft in a circle around a surveyed compass base and to measure the compass deviations with an extremely accurate datum instrument. Usually the power supply for the aircraft compass system is from an external source, as it is impracticable to run the engines for the time required to take accurate observations. Although the compass system is not subjected to the forces encountered in flight, and the aircraft undercarriage is down, any differences in deviation due to these limitations are far outweighed by the advantages of accurate observation in a stable environment.

2. Providing an accurate datum for determining heading whilst airborne is available, and the local values of variation are known, it is possible to swing an aircraft compass in the air, although the accuracy of the swing is subject to the following limitations:

a. Error in measuring coefficient C due to coriolis acceleration (*see* Part 2, Sect 3, Chap 1, Appendix). Note that although this may result in inaccurate correction of coefficient C on the ground, it may be an advantage to have local coriolis acceleration taken into account in the deviation card resulting from the check swing.

b. Settling time after turns, which is a function of hang-off error (*see* Part 2, Sect 3, Chap 1, Appendix).

Methods of Determining Heading in the Air

3. **Use of a Celestial Datum Bearing.** Bearings may be obtained from a celestial body by using the precomputed true or magnetic azimuth in flight to find the aircraft's true or magnetic heading, or by measuring the relative bearing of the body and doing the calculations later. The latter method has the advantage that it is not restricted to a pre-determined time of operation, and the observer can take the bearings under as near ideal conditions as possible.

4. **Use of a Gyro Datum Bearing.** The use of a low-drift gyro not already fitted in the aircraft would necessitate some minor modifications to the aircraft. Electrical supplies and mounting brackets would be needed. If an independent gyro is already fitted, *eg* the CL-11 in the Britannia and the inertial platform in the Nimrod, the procedure is simplified.

5. **Taking the Datum Bearings.** A set of five readings should be taken on each heading and averaged. It is essential that the readings of the datum and the compass are taken simultaneously, and the following procedure is recommended:

a. The datum bearing observer (*ie* the sextant operator or the gyro observer) takes a bearing every 15 seconds for one minute, indicating to the crew when he does so, and averages the readings.

b. At each indication the compass observer records the compass readings every 15 seconds for one minute, and averages the readings.

c. At the third reading an accurate fix and the time are recorded.

6. **Magnetic Bearing from the Celestial Datum.** Extreme accuracy is required in all calculations, and because of this AP 3270 is not sufficiently accurate: a 10 nm DR error at 60°N, altitude 40°, and azimuth 090°, gives an azimuth error of 1.1°. Sets of azimuth tables produced by Mr. Scott of the Royal Observatory which give more accurate azimuths are available. Using the azimuth extracted and the variation interpolated to the

nearest tenth of a degree for the position and time recorded at sub-para 5c the magnetic datum heading may be found from:

Hdg(M) = Azimuth(T)—Relative Bearing + West (or—East) Variation.

7. **Magnetic Bearing from the Gyro Datum.** At the compass base the magnetic heading of the aircraft is found before the flight, using the Watt's Datum Compass. At the same time the gyro reading is recorded. The difference, Watt's Datum Compass reading minus gyro reading, is the correction to be applied to the gyro reading to obtain the datum magnetic bearings. The correction cannot be applied directly because of gyro drift, and the gyro drift rate must be assessed. This is done quite simply by returning the aircraft to the compass base after the airborne recordings, and once again finding the difference between the Watt's Datum Compass and the gyro readings. The difference in corrections is due to gyro drift. The gyro drift is applied proportionally to the airborne gyro readings. An example of the calculations follows in Tables 1 and 2.

	Before Take-off	After Landing
Time	1000 hours	1030 hours
Datum Compass	030.45°	073.92°
Gyro	302.96°	346.62°
Correction	+087.49°	+087.30°
Gyro drift = —0.19°, and gyro drift rate = —0.38° per hour		

Table 1—Calculation of Gyro Drift Rate

Time	Gyro	Corrn for drift rate	Corrn to gyro	Datum Gyro Brg.	Corrn for Varn. diff	Final Datum
1010	273.25°	—0.06°	+087.43°	000.68°	+0.4°	001.08°
1015	001.85°	—0.10°	+087.39°	089.24°	0	089.24°
1018	094.00°	—0.12°	+087.37°	181.37°	—0.3°	181.07°
1022	182.20°	—0.14°	+087.35°	269.55°	—0.1°	269.45°

Table 2—Application of Gyro Drift Rate

The correction for difference in variation arises from the difference in position between the compass base and the aircraft at the time of the observations. It can be found by plotting the aircraft's position on a chart which shows variation and extracting the difference between the compass base variation and the in-flight variation. Assessment of gyro drift rate should be made before and after the correcting swing and before and after the calibration swing.

8. **The Accuracy of the Datum Bearings.** Some factors affecting the accuracy of each type of datum bearing are mentioned below:

a. *Celestial Relative Bearing.* To convert a true bearing to magnetic a value of variation is applied. This value, taken from a chart of variation for ground level, may be inaccurate, and variation may change with height, The accuracy of the periscopic sextant in azimuth, assuming no error in the mount alignment, is $\pm 0.75°$ at the 95% probability level. Ignoring the variation factor, and assuming the use of Scott's tables and a vernier scale sextant, the probable error (50%) of a celestial relative bearing should be $\pm 0.3°$.

b. *Gyro Readings.* The calculations shown in para 7 assume that the gyro drift rate is constant, which may not be the case. Since the Watt's Datum Compass is used to give the magnetic correction, the ground level variation inaccuracy is no longer a significant factor, although the possibility of errors due to change of variation with height has not been eliminated. Ignoring the variation factor and assuming a high-grade gyro of good repeatability the probable error should be $\pm 0.25°$. Because the celestial bearing method uses a value of variation extracted from a chart, the gyro bearing will be more accurate as it is corrected for observed variation values. The correction for variation difference should be reasonably accurate, even though the spot variations may be considerably in error.

Swing Procedures

9. The preliminaries to the ground compass swing apply, in general, to the air swing. In particular, a digital read-out of compass and gyro readings, if not built in, should be fitted if available. Additional points are to advise ATC that an air swing will be carried out, and to seek advice on the best height to fly, bearing in mind the avoidance of turbulence and the possible need to see a celestial body.

10. **The Gyro Datum Swing.** The procedure for a gyro datum swing is as follows:

a. Carry out the preliminary checks.

b. Taxi the aircraft to the compass base, measure the magnetic heading of the aircraft using a Watt's Datum Compass and record the heading, the gyro reading and the time.

c. Take-off and climb to the recommended altitude.

d. Head the aircraft successively on to North, East, South and West. Record the compass headings, gyro readings, the fixes and times on each cardinal heading.

e. Plot the fixes and extract the corrections for variation difference.

f. At the compass base obtain the new gyro correction and calculate the drift rate.

g. Apply all corrections as shown in para 7 to obtain the datum bearings. Enter the datum and compass headings in the appropriate columns of F343B.

h. Calculate and correct for coefficients A, B and C as for the ground Refined Swing.

j. Obtain and record the gyro reading correction before take-off for the second part of the swing.

To save time, if any coefficients were corrected, the aircraft should be flown first on the four cardinal headings. By applying the predicted drift rate and re-calculating the coefficients it can be seen whether they are less than 0.5°. If they are not the aircraft should be landed and the coefficients removed as in sub-paras e to h, but if they are less than 0.5° the calibration swing may proceed as follows:

k. Obtain the readings every 30°. The aircraft may be flown on chords to a circle roughly centred on the airfield.

l. After landing re-assess the gyro drift rate.

m. Complete the F343B and the appropriate deviation card.

11. **The Celestial Bearing Swing.** The celestial bearing swing differs from that described in para 10 only in that it is not necessary to spend time on the compass base to determine gyro corrections and drift, and the swing requires only one flight if the coefficients are less than 0.5°.

CHAPTER 3

THE ANALYSIS OF THE COMPASS SWING

CONTENTS

THE FOURIER ANALYSIS

	Paras
The Derivation of the Coefficients	1–5
The Calculated Deviations	6–7
Summary of the Fourier Analysis	8

THE ACCURACY ANALYSIS

Introduction	9–10
Statistical Analysis of the Swing	11–12
Further Applications of Statistics	13–14

THE FOURIER ANALYSIS

Derivation of the Coefficients

1. The purpose of the Fourier Analysis is to extract from a set of observations the most accurate assessment of the deviation coefficients and residual deviations. Chapter 2 described how the coefficients can be found, but in two cases, B and C, only two readings were used. A more accurate method is needed.

2. In Chapter 1 it was shown that the deviation caused by coefficient B is a function of the sine of the heading. The observed deviation on each heading is multiplied by the sine of that heading, and the results algebraically summed. It can be shown that division of this sum by $\frac{n}{2}$, where n is the number of headings, gives coefficient B. Similar calculations may be done to find coefficients C, D and E. Coefficient A is derived from the sum of the deviations and the number of readings. The results can be summarized by the equations:

$$A = \frac{\Sigma\delta}{n} \qquad B = \frac{2\ \Sigma\delta \sin\theta}{n}$$

$$C = \frac{2\ \Sigma\ \delta\cos\theta}{n} \qquad D = \frac{2\ \Sigma\ \delta\sin 2\theta}{n}$$

$$E = \frac{2\ \Sigma\ \delta\cos 2\theta,}{n}$$

where δ is the observed deviation on heading θ, and n is the number of observations.

3. The greater the number of readings used the greater will be the accuracy of the derived coefficients. As the band of error only decreases as the inverse square root of n, twelve readings have been accepted as the practical figure, *ie* n=12. As an aid to calculation a table of values of sin θ, cos θ, sin 2θ and cos 2θ, at 30° intervals, is incorporated in RAF Form 343B, which is used for the Fourier Analysis. For convenience these values have been extracted and are listed at Table 1.

Hdg (θ) a	sin θ b	cos θ c	sin 2θ d	cos 2θ e
0	0	+1.0	0	+1.0
30	+0.5	+0.87	+0.87	+0.5
60	+0.87	+0.5	+0.87	—0.5
90	+1.0	0	0	— 1.0
120	+0.87	—0.5	—0.87	—0.5
150	+0.5	—0.87	—0.87	±0.5
180	0	—1.0	0	+1.0
210	—0.5	—0.87	+0.87	+0.5
240	—0.87	—0.5	+0.87	—0.5
270	—1.0	0	-0	—1.0
300	—0.87	+0.5	—0.87	—0.5
330	—0.5	+0.87	—0.87	+0.5

Table 1—Values of Functions of θ

4. **Observed Deviations.** At Fig 1b is the total deviation curve derived from the component curves at Fig 1a. From Fig 1b, the observed deviations every 30°, starting at 0°, are: —0.5°, +1.1°, +1.0°, +0.5°, +0.3°, +1.0°, +1.5°, +0.6°, —1.4°, —3.5°, —3.9° and —2.6°. These deviations are used in the Fourier Analysis.

5. **To Calculate the Coefficients.** Table 2 is an extract of those colums of RAF Form 343B used for the calculations. The observed deviations are entered in column 2, and multiplied by the values shown in columns b, c, d and e of Table 1. These results are entered in Form 343B, and the columns are then totalled to obtain $\Sigma\delta$, $\Sigma\delta\sin 0$, $\Sigma\delta\cos 0$, $\Sigma\delta\sin 2\theta$, and $\Sigma\delta\cos 2\theta$. Dividing column 2 by n, and columns 7, 10, 13 and 16 by n, gives the calculated coefficients: $A=-0.49$, $B=+1.97$, $C=-0.93$, $D=+0.94$, $E=+1.01$.

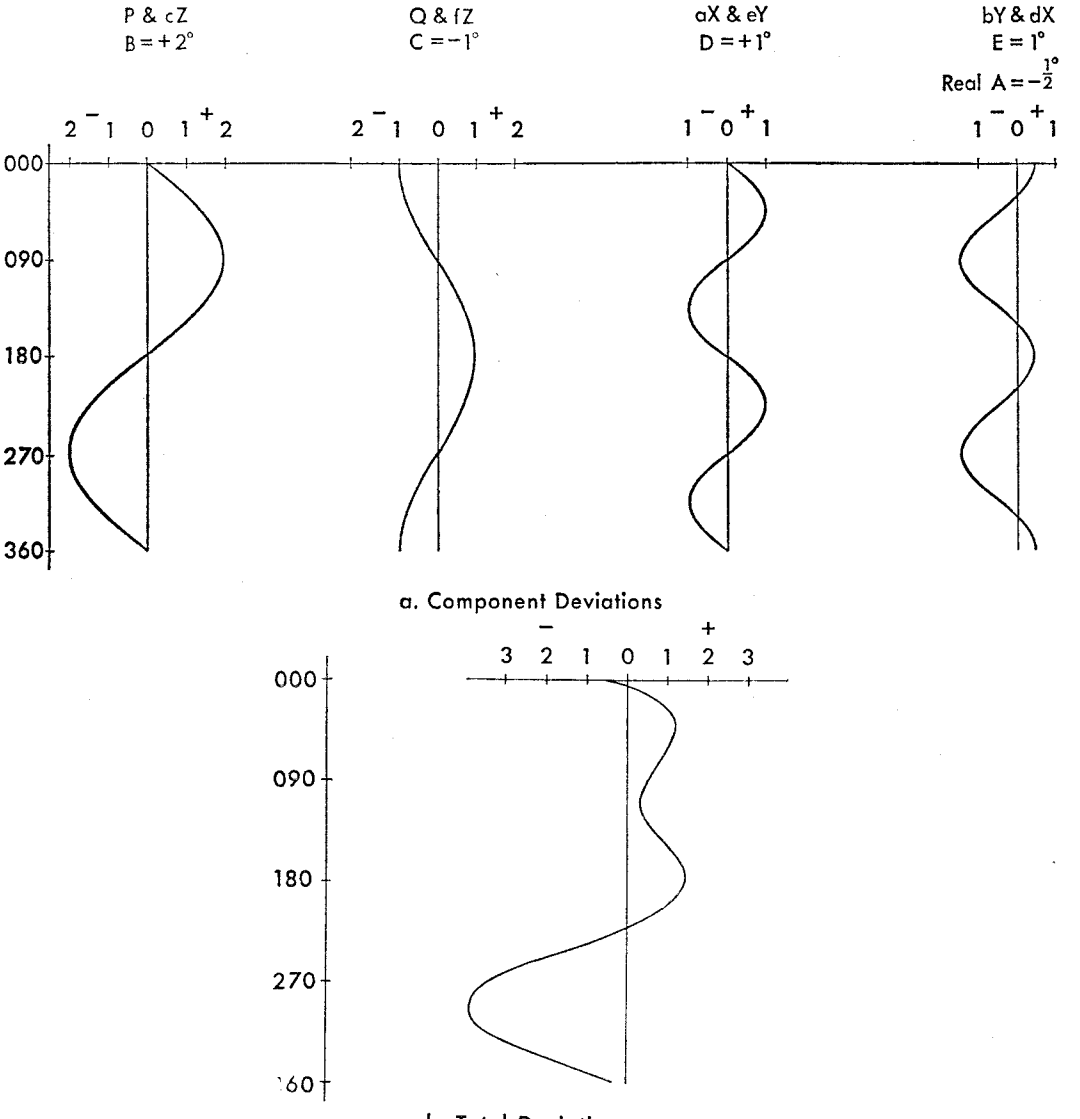


Fig 1 Deviation Graphs

Hdg (θ) 1	Observed Devi'tion (δ) 2	δ sin θ 7	δ cos θ 10	δ sin 2θ 13	δ cos 2θ 16
0	—0.5	0	—0.50	0	—0.50
30	+1.1	+0.55	+0.96	+0.96	+0.55
60	+1.0	+0.87	+0.50	+0.87	—0.50
90	+0.5	+0.50	0	0	—0.50
120	+0.3	+0.26	—0.15	—0.26	—0.15
150	+1.0	+0.50	—0.87	—0.87	+0.50
180	+1.5	0	—1.50	0	+1.50
210	+0.6	—0.30	—0.52	+0.52	+0.30
240	—1.4	+1.22	+0.70	—1.22	+0.70
270	—3.5	+3.50	0	0	+3.50
300	—3.9	+3.39	—1.95	+3.39	+1.95
330	—2.6	+1.30	—2.26	+2.26	—1.30
Sums	—5.9	+11.79	—5.59	+5.65	+6.05
Divisors	12	6	6	6	6
Coeffs	—0.49	+1.97	—0.93	+0.94	+1.01

Table 2—The Derived Coefficients

The Calculated Deviations

6. The second part of the Fourier Analysis is to find the calculated deviations. In effect, this is the reverse of the first process: having made the most accurate assessments of the coefficients they are used to determine the most accurate deviation curve. In Chapter 1 the composite curve was found by visually adding together the coefficient curves as in Fig 1. The Fourier Analysis uses a similar process, but by calculation.

7. **The Calculated Deviation Curve.** Table 3 is an extract of the columns of RAF F343B used for the process of finding the calculated deviation curves and the composite curve. The coefficients are multiplied by their associated trigonometrical functions from Table 1. When columns 6, 8, 11, 14 and 17 are complete, each line is summed and the totals entered in column 3. These totals are the end result of the Fourier Analysis—the calculated deviations which are used to plot the deviation curve and to complete the aircraft deviation card.

Hdg (θ) 1	Calculated Deviation 3	A 6	B sin θ 8	C cos θ 11	D sin 2θ 14	E cos 2θ 17
0	—0.41	—0.49	0	—0.93	0	+1.01
30	+1.01	—0.49	+0.99	—0.81	+0.82	+0.50
60	+1.07	—0.49	+1.71	—0.47	+0.82	—0.50
90	+0.47	—0.49	+1.97	0	0	—1.01
120	+0.37	—0.49	+1.71	+0.47	—0.82	—0.50
150	+0.99	—0.49	+0.99	+0.81	—0.82	+0.50
180	+1.45	—0.49	0	+0.93	0	+1.01
210	+0.65	—0.49	—0.99	+0.81	+0.82	+0.50
240	—1.41	—0.49	—1.71	+0.47	+0.82	—0.50
270	—3.47	—0.49	—1.97	0	0	—1.01
300	—3.99	—0.49	—1.71	—0.47	—0.82	—0.50
330	—2.61	—0.49	—0.99	—0.81	—0.82	+0.50
Sums	—5.88	—5.88	0	0	0	0

Table 3—The Calculated Deviations

Summary of the Fourier Analysis

8. Any periodic function (the compass swing period is 2 π) can be broken down into sinusoids of different amplitudes (the coefficients) and phases (sin, cos, *etc*). If sufficient readings are available, the derived parts of the original can be built up again to give the most accurate assessment of the function. A convenient form for the breaking down and building up processes is RAF F343B, Aircraft Compass Log Book (Refined Swing).

THE ACCURACY ANALYSIS

Introduction

9. The accuracy analysis gives a statistical assessment of the reliance that can be placed on the results of the swing, and enables one swing to be compared with another. The analysis is based on the differences between the observed and calculated deviations, differences which arise because the aircraft and datum instruments are being used at or beyond their accuracy limits.

10. It will be useful to summarize the following terms which are used in a Fourier Analysis. The probable error (ε) is the difference between the mean of a series of observations and any single observation which will not be exceeded on 50% of occasions. Probable error equals 0.674 σ, where σ (sigma) is the standard deviation. Normally the standard deviation is found from:

$$\sigma = \pm \sqrt{\frac{\Sigma (X-\bar{X})^2}{n}}$$, where X is the particular reading, and $\bar{X}$ is the mean of all the readings. As the compass calibration method does not provide a mean, the calculated deviation is used instead. The probable error formula then becomes:

$$\text{Single reading } \varepsilon = \pm 0.674 \sqrt{\frac{\Sigma D^2}{n-s}}$$, where D is the difference between the corresponding observed and calculated deviations and s is the number of unknowns (*ie* the coefficients). To find the greatest probable error of coefficient A use is made of the formula:

$$\varepsilon_A = \frac{\varepsilon}{\sqrt{n}},$$

and for coefficients B, C, D and E of the formula:

$$\varepsilon_{B \text{ to } E} = \varepsilon \cdot \frac{\sqrt{2}}{\sqrt{n}}, \text{ or } 1.4\ \varepsilon_A.$$

Statistical Analysis of the Swing

11. Fig 2 shows a completed F343B for the swing used in the Fourier Analysis. Column 4 is D, column 5 is D^2. Thus for the figures used:

$\varepsilon = \pm 0.05°$.

$\varepsilon_A = \pm 0.014°$, *ie* A = —0.49 $\pm 0.014°$.

$\varepsilon_{B \text{ to } E} = \pm 0.02°$, *ie* B, C, D and E are within $\pm 0.02°$ ε of their stated figures.

12. **The Meaning of the Probable Errors.** The figure for ε of $\pm 0.05°$ means that any single observed deviation has an evens chance of being within .05° of the calculated deviation, and one would therefore expect half of the differences to be within $\pm 0.05°$. Column 4 confirms this, The coefficient's probable errors provide a means of comparing one compass swing with another, a form of correlation test.

Further Applications of Statistics

13. Fig 3 shows a completed F343B. No observed deviation differs from the next by more than 1°—at first sight a good swing. But examination of the ε values shows that the swing gives coefficients and calculated deviations that are meaningless: the coefficients all stand an evens chance of equalling zero. Fig 4 shows another set of observed deviations where consecutive readings change by as much as 1.5°—at first sight a bad swing. But examination shows that the rapid changes are due to large coefficients D and E. The probable accuracy of the single reading is better than the accepted maximum of $\varepsilon = \pm 0.20$, and coefficients which can be corrected are less than the accepted maximum of 0.5°.

14. **The Effect of Carriage of Stores.** To show how statistics can be used to compare one swing with another the effect of a load of bombs will be considered. A further statistical limit must be explained—a result is only considered as being significant when it is at the 95% probability level, 2σ or 3ε. The following two sets of figures may be compared.

Coefficients	*With Bombs*	*Without Bombs*	*Difference*
A	—0.06	—0.11	+0.05
B	+0.29	—0.14	+0.43
C	+0.16	—0.24	+0.40
D	—0.81	—0.45	—0.36
E	+0.67	+0.24	+0.43
Probable Errors			
ε	= ± 0.28	± 0.25	
$\varepsilon_{B \text{ to } E}$	= ± 0.114	± 0.102	

At first sight there are large differences in the values of the coefficients B to E. But first the probable error (since all the figures are at the 50% level) of the differences must be found. This is done by finding the square root of the sum of the squares of the probable errors of the coefficients:

$$\varepsilon_D = \sqrt{\varepsilon_1^2 + \varepsilon_2^2}$$

To use the figures shown:

$$\varepsilon_D = \pm\sqrt{0.114^2 + 0.102^2}$$
$$= \pm 0.153°.$$

This figure becomes significant at the 3ε level, *ie* 0.459°. Therefore it can be said that the bombs have no effect on the aircraft's magnetism because no difference exceeds 0.459°. And there is a better than 19 to 1 chance of being right.

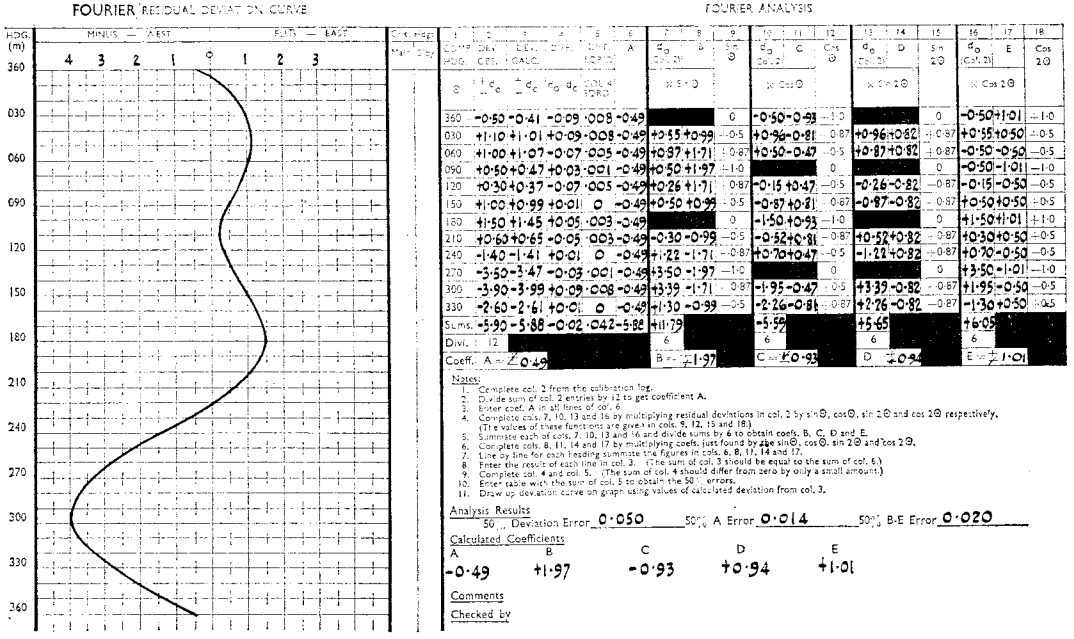


Fig 2 Form 343 B—Aircraft A

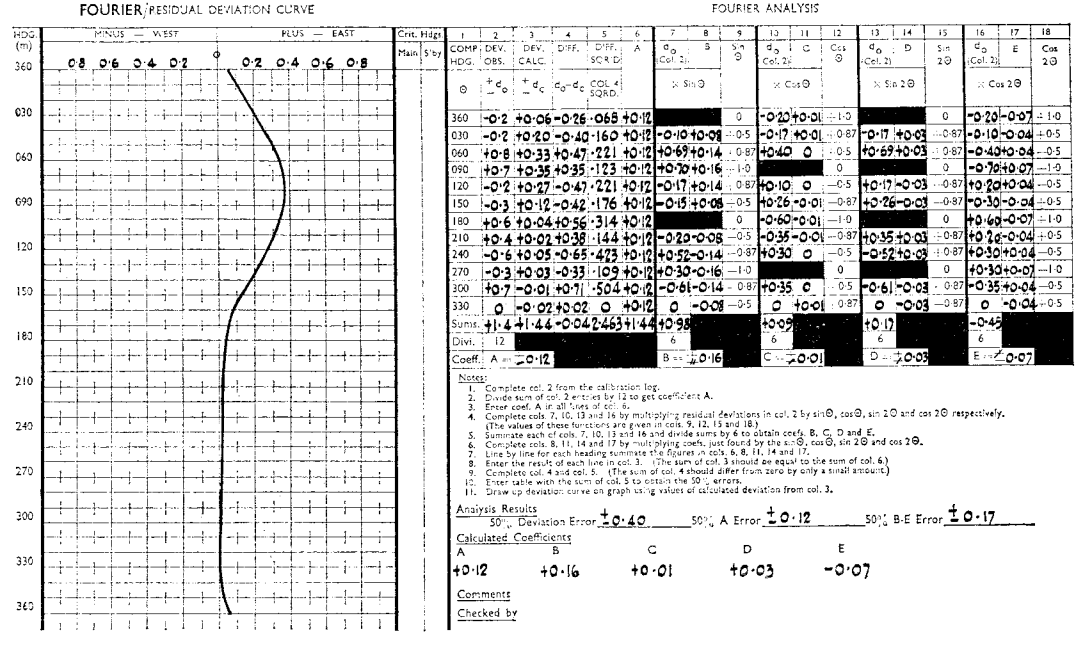
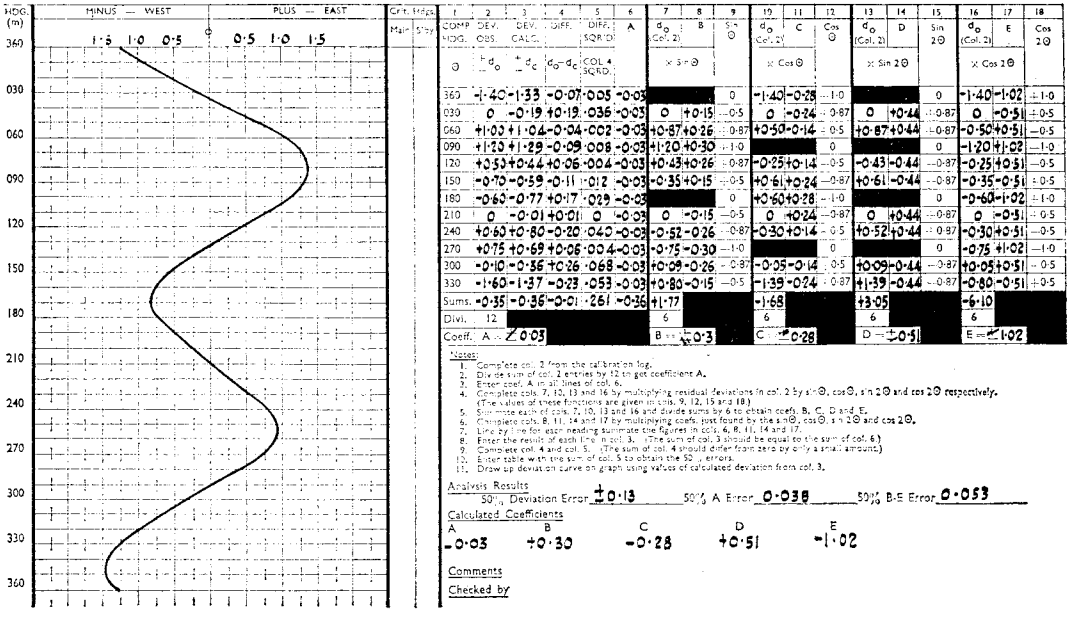


Fig 3 Form 343 B—Aircraft B



FOURIER/RESIDUAL DEVIATION CURVE

MINUS — WEST PLUS — EAST

1·5 1·0 0·5 0·5 1·0 1·5

HDG. (m): 360, 030, 060, 090, 120, 150, 180, 210, 240, 270, 300, 330, 360

Corr. Index: Main / Stby

FOURIER ANALYSIS

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18
COMP HDG. Θ	DEV. OBS. $^{E}_{W}d_o$	DEV. CALC. $^{E}_{W}d_c$	DIFF. d_o-d_c	DIFF. SQR'D COL 4 SQR'D	A	d_o (Col. 2) × Sin Θ	B	Sin Θ	d_o (Col. 2) × Cos Θ	C	Cos Θ	d_o (Col. 2) × Sin 2Θ	D	Sin 2Θ	d_o (Col. 2) × Cos 2Θ	E	Cos 2Θ
360	-1·40	-1·33	-0·07	·005	-0·03			0	-1·40	-0·28	-1·0			0	-1·40	-1·02	+1·0
030	0	-0·19	+0·19	·036	-0·03	0	+0·15	-0·5	0	-0·24	-0·87	0	+0·44	-0·87	0	-0·51	+0·5
060	+1·03	+1·04	-0·04	·002	-0·03	+0·87	+0·26	-0·87	+0·50	-0·14	-0·5	+0·87	+0·44	-0·87	-0·50	+0·51	-0·5
090	+1·20	+1·29	-0·09	·008	-0·03	+1·20	+0·30	-1·0			0			0	-1·20	+1·02	-1·0
120	+0·50	+0·44	+0·06	·004	-0·03	+0·43	+0·26	-0·87	-0·25	+0·14	-0·5	-0·43	-0·44	-0·87	-0·25	+0·51	-0·5
150	-0·70	-0·59	-0·11	·012	-0·03	-0·35	+0·15	-0·5	+0·61	+0·24	-0·87	+0·61	-0·44	-0·87	-0·35	-0·51	+0·5
180	-0·60	-0·77	+0·17	·029	-0·03			0	+0·60	+0·28	-1·0			0	-0·60	-1·02	+1·0
210	0	-0·01	+0·01	0	-0·03	0	-0·15	-0·5	0	+0·24	-0·87	0	+0·44	-0·87	0	-0·51	+0·5
240	+0·60	+0·80	-0·20	·040	-0·03	-0·52	-0·26	-0·87	-0·30	+0·14	-0·5	+0·52	+0·44	-0·87	-0·30	+0·51	-0·5
270	+0·75	+0·69	+0·06	·004	-0·03	-0·75	-0·30	-1·0			0			0	-0·75	+1·02	-1·0
300	-0·10	-0·36	+0·26	·068	-0·03	+0·09	-0·26	-0·87	-0·05	-0·14	-0·5	+0·09	+0·44	-0·87	+0·05	+0·51	+0·5
330	-1·60	-1·37	-0·23	·053	-0·03	+0·80	-0·15	-0·5	-1·39	-0·24	-0·87	+1·39	-0·44	-0·87	-0·80	-0·51	+0·5
Sums:	-0·35	-0·36	+0·01	·261	-0·36	+1·77			-1·68			+3·05			-6·10		
Divi.	12					6			6			6			6		
Coeff.	A = -0·03					B = +0·30			C = -0·28			D = +0·51			E = -1·02		

Notes:
1. Complete col. 2 from the calibrat on log.
2. Divide sum of col. 2 entries by 12 to get coefficient A.
3. Enter coef. A in all lines of col. 6.
4. Complete cols. 7, 10, 13 and 16 by multiplying residual deviations in col. 2 by sinΘ, cosΘ, sin 2Θ and cos 2Θ respectively. (The values of these functions are given in cols. 9, 12, 15 and 18.)
5. Summate each of cols. 7, 10, 13 and 16 and divide sums by 6 to obtain coefs. B, C, D and E.
6. Complete cols. 8, 11, 14 and 17 by multiplying coefs. just found by the sinΘ, cosΘ, sin 2Θ and cos 2Θ.
7. Line by line for each reading summate the figures in cols. 6, 8, 11, 14 and 17.
8. Enter the result of each line in col. 3. (The sum of col. 3 should be equal to the sum of col. 6.)
9. Complete col. 4 and col. 5. (The sum of col. 4 should differ from zero by only a small amount.)
10. Enter table with the sum of col. 5 to obtain the 50% errors.
11. Draw up deviation curve on graph using values of calculated deviation from col. 3.

Analysis Results

50% Deviation Error ±0·13 50% A Error 0·038 50% B-E Error 0·053

Calculated Coefficients

A	B	C	D	E
-0·03	+0·30	-0·28	+0·51	-1·02

Comments

Checked by

Fig 4 Form 343 B—Aircraft C

CHAPTER 4

THE MAGNETIC SURVEY OF COMPASS BASES

CONTENTS

	Paras
Introduction	1—2
Physical Features	3—5
Magnetic Considerations	6—11
Survey Instruments	12—15
Survey Aspects	16—18
Survey Methods	19—21
Survey Procedures	22—24

Introduction

1. To ensure that the deviations derived from a compass swing are caused only by aircraft magnetism, the swing must be carried out in an area free from magnetic fields other than that of the Earth. (This does not apply to the Electrical Compass Swing—*see* Chapter 2.) The purpose of this chapter is to describe how a suitable area is selected and how magnetic surveys are carried out to ensure that any local magnetic influences and anomalies are insignificant.

2. The Admiralty Compass Observatory (ACO), Ditton Park, Slough, Bucks, is the official Ministry of Defence department with overall responsibility for surveying compass bases abroad and at home. Requests by RAF commands for survey assistance should be addressed to MOD, Ops Nav 1a (RAF), Main Building, Whitehall, SW1. Naval requests for survey assistance are addressed directly to the ACO.

Physical Features

3. **Basic Requirements.** The compass base should meet the following requirements:

a. It must be accessible.

b. Its use should not interfere with normal aircraft movements on the airfield.

c. It must be large enough to take all types of aircraft likely to use it.

d. Its surface should not preclude its use in wet weather.

e. It must be free from magnetic fields, other than the Earth's, which might affect aircraft compass deviations. It is unlikely that runways or taxiways would be suitable for refined swings.

4. **The Size of the Base.** The size of the base will depend on the types of aircraft operated from the airfield. For example, V-bombers require a base of 190 ft radius, and helicopters one of 80 ft radius. Additional specifications are given in DCIs. Factors to be considered are:

a. Whether the aircraft will be towed or taxied during the swing.

b. The radii of the aircraft's turning circles.

c. The position of the sighting rods on the aircraft and their likely path during the swing.

d. The position of the aircraft's magnetic sensor.

e. Modern aircraft sighting rods are designed to be viewed 80 ft away from the nearer rod.

5. **Base Markings.** The compass base must be clearly and permanently marked to show:

a. The base centre.

b. The central area in which the aircraft's sensor should remain during the swing.

c. The datum compass circle. This is the

circle around the central area which shows where the datum compass should be placed.
d. Areas of magnetic anomalies.
e. Nose wheel turning circles where applicable.

Paint is the best medium for marking concrete. The datum compass circle, which may well be on grass, must be marked permanently with a narrow continuous path of non-magnetic material such as tarmac, gravel *etc.* Temporary markings can be made with such agents as acid, caustic soda or old engine oil.

Magnetic Considerations

6. **Classes of Bases.** There are two types of compass base:

a. *Class One.* These are bases possessing no known magnetic anomaly in excess of ± 0·1°. Bases of this accuracy are needed for refined swings.
b. *Class Two.* These are bases possessing no known magnetic anomaly in excess of ± 0·25°. Such bases are suitable for standard swings.

7. **Ferro-Magnetic Interference.** Natural ferrous deposits of sufficient size to cause deviations rarely occur in the middle of airfields. The most likely causes of ferro-magnetic interference are:

a. Buried scrap metal such as pierced steel planking.
b. Reinforced concrete.
c. Drainage systems.
d. Wire fences.
e. Conduit for electrical cabling.
f. Odd pieces of metal concealed near the surface.

If such items are found at the selected site they should be removed if possible. Even though the area may be within the limits stated in para 6, ferro-magnetic material should still be removed, as its magnetic effect may change with time.

8. **Electro-Magnetic Interference.** Usually the base site is chosen so as to be clear of electrical cabling. Where such cables cannot be avoided, their effect, with and without current flowing, must be checked at intervals along their length. Generally, direct current will cause deviations unless the positive and negative lines are run close together.

9. **Anomalous Areas.** When readings over ± 0·06° for a Class 1 and ± 0·15° for a Class 2 base survey are found, the area around is regarded as anomalous. Anomalous areas should be checked thoroughly immediately they are found as their presence may cause the site to be abandoned. The position of maximum deviation may indicate the source. If the source is buried the ACO should be consulted before trying to excavate. An anomolous area will usually be due to one of the causes mentioned in paras 7 and 8.

10. **Height of Aircraft Sensors.** Usually the survey is done at the height of the datum compass tripod (about 5 ft). Most aircraft sensors are above this. If a site is to be used for aircraft whose sensors are below 5 ft, assessment closer to the ground may be needed.

11. **Changes in Variation.** Changes in variation may occur through diurnal changes and magnetic storms.

a. *Diurnal Changes.* Diurnal changes in variation may vary from a few arc minutes close to the magnetic equator to many degrees close to the magnetic poles. In southern England the diurnal change varies from about 0·25° ~~4 ft~~ in the summer to about ~~15~~ ft in the winter. The change is most westerly at about 1330 GMT and most easterly at about 0800 GMT.
b. *Magnetic Storms.* Magnetic storms are usually associated with sun spot activity. Large storms can occur which alter the variation in the UK by up to 0·5°, and they may last several hours or even days. Fortunately, the frequency of occurrence is only about once per year. In the vicinity of the sub-point of the Aurora Borealis variation can change for short periods by up to 5°.

Survey Instruments

12. The effect of other magnetic fields is to cause distortion of the direction and intensity of the Earth's field. The effect on the horizontal intensity is very small and may be ignored. Only the effect on the direction need be examined, and this is done using accurate magnetic bearing compasses.

13. **The Types of Instruments.** Part 2, Sect 5 describes the types of instruments used. They are the Medium Landing, the Pattern 2 Prismatic, the Watt's Datum and the MC-1M. The first two types are only sufficiently accurate to be used for the detection of gross errors, such as in the initial survey (para 22). The Watt's Datum Compass (WDC) must be used for any detailed survey (para 23). The MC-1M is used to survey an area prior to an electrical compass swing.

14. **Instrument Checks.** Every instrument used should be given the full serviceability checks as

laid down in Sect 5. Particular attention should be paid to the tests which affect the repeatability of the readings, pivot frinction. As the WDC is subject to index shift it should be rechecked every hour. The presence of index shift errors in the datum is due to diurnal variation changes.

15. **Correction of Datum Instruments.** When the reciprocal bearing method (para 19) is used, the datum instruments must first be aligned to correct for instrument and operator error. Usually two WDCs are used and one of them is designated as the master compass. The other is called the mobile compass. The correction is found as follows:

a. Erect a tripod as near to the anticipated centre of the base as is convenient.
b. Mount the mobile compass on the tripod.
c. Align the compass with the magnetic meridian.
d. Take a set of six bearings on a distant object.
e. Obtain and note the average bearing.
Note that b, c and d should be done by the person who will operate the mobile compass during the survey.
f. Replace the mobile compass by the master compass without disturbing the tripod.
g. With the bearing plate set to the bearing obtained in sub-para e, the master compass operator should align the sighting telescope on the distant object, then tighten the magnetic clamp to ensure that both compasses will be using a common magnetic bearing datum.
h. Repeat actions a to g every hour or when a change of variation is suspected.

Survey Aspects

16. **Preparation for Surveys.** The site chosen for a compass base should be agreed with all the unit sections concerned. Para 3 describes the basic requirements; the engineering, ATC and DOE staffs should confirm that the requirements are met. Future building programmes should be examined to make sure that the site is not scheduled for other work. People with long service at the unit may be able to recall whether the site was ever used in such a way (*eg* as a rubbish tip) as to preclude its use as a calibration area.

17. **Scope of the Survey.** So that the surveyor can state the accuracy limits of the site chosen, the area must be examined at close, regular intervals, say every 20 ft. The interval should be smaller when an anomaly is found. A survey of a large base may require some 300 readings and may take well over seven hours to complete.

18. **Types of Survey.** There are five types of survey:

a. *The Initial Survey.* The first assessment of a site to determine gross errors is called the Initial Survey. It should be done by command or unit personnel before asking the ACO to do a detailed survey.
b. *Establishment Survey.* The Establishment Survey is the detailed survey, generally done by the ACO staff, to establish a compass base.
c. *Periodic Re-Survey.* Once a base has been established it must be given a detailed re-survey at the intervals shown below:
(1) Class 1 bases every five years, by the ACO staff.
(2) Class 2 bases every two years, by command or unit personnel, with ACO help if needed. In addition, Class 2 bases should be surveyed by ACO every six years.
d. *Annual Check.* Once per year, the officer i/c the compass base should visit the base and check that:
(1) The base is clearly defined.
(2) No work has been done to the base which might alter its magnetic properties. If in doubt, the suspected area should be given a detailed magnetic survey.
(3) No obvious magnetic objects have been placed on the site since the last survey. Examples of likely objects are metal chocks and fire extinguishers.
e. *Area Survey.* An area survey uses the MC-1M compass calibrator.

Survey Methods

19. **Reciprocal Bearing.** The reciprocal bearing method uses two or more WDCs and is the most accurate method in use. It is to be used for all detailed surveys. The setting-up and operating procedures for the WDC are given in Sect 5, Chap 3. The master compass is not re-aligned for every observation, but the magnetic bearing of the distant object should be checked frequently. The two compasses are aligned on each other's sighting telescope mirrors, which at a distance appear as circles of light; when close, the sighting graticules can be seen clearly. The difference between the readings of the two compasses, less 180°, is the magnetic deviation between the two compass positions. The sign convention used for the errors is that, if the reading of the mobile compass is greater than that of the master compass, the deviation is negative.

20. **Distant Bearing.** The procedure to be

followed when using the distant bearing method is:

a. Select a distant object at least 2 nm away.

b. Locate the positions of the compass base and the distant object accurately on a large scale map and measure the distance between them.

c. Mark the line of sight from the centre of the compass base to the distant object (the sight line).

d. Calculate the angular correction to be applied to bearings taken away from the site centre using the formula:

$$\text{Correction (degrees)} = \frac{\text{Lateral distance from the sight line} \times 180}{\text{distance to the object} \times \pi}$$

e. Ensure by visual inspection that the base centre is magnetically clean.

f. Measure the bearing of the distant object from the base centre using the bearing compass. This will be the datum bearing.

g. From selected points around the site, take bearings of the distant object.

h. Apply the calculated corrections from and compare the bearings with those from the base centre. The differences between the bearings are due to the deviation present, assuming that the base centre is free from deviation.

The distant bearing method may be used only for initial surveys, and for gross error checks of Class 2 bases resulting from doubts raised during the annual checks referred to at sub-para 18d. Its use at any time can only be justified by a second datum compass being unavailable.

21. **Area Survey.** A compass-swing area must be free of nearby traffic and have sufficient space for an aircraft to be towed onto a North heading. In addition, the Earth's magnetic field in the vicinity of the detector unit should be uniform both in magnitude and direction. The area survey is conducted to determine if the proposed site meets this requirement. When the conditions of the survey are satisfied, it is not necessary to re-survey for another year, unless there are any changes in the physical features of the area that might result in a magnetic disturbance. The method to be followed is as follows:

a. *Vertical Magnetic Gradient Check.* The vertical magnetic gradient check is carried out at the proposed detector unit position (*see* Fig 2). The direction and strength of the Earth's magnetic field with the MC-1M monitor-tripod lowered to minimum height are compared with the direction and strength measured with the tripod raised to its maximum height. The readings should be taken within 30 minutes to lessen the possibility of a change in the Earth's field.

b. *Horizontal Magnetic Gradient Check.* The horizontal magnetic gradient check is carried out at the proposed detector unit location. This check determines the uniformity of the Earth's magnetic field over a circle of 5 ft radius with its centre at the proposed detector unit location. The readings are made with the monitor-tripod at normal operating height. The direction and strength of the Earth's magnetic field at the centre of the circle are compared with the values measured at four points equi-spaced around the perimeter of the circle.

c. *Monitor Location Check.* During the compass swing procedure the monitor will be used to measure the Earth's field. The position selected should be far enough from the calibration site to prevent the aircraft from having any magnetic effect on the monitor readings. At the selected location the survey requires that the Earth's magnetic field direction and strength are determined and noted. The bearing of a reference target from this location is obtained. This target should be at least ½ nm away.

d. *Detector Unit Location Check.* The monitor is re-positioned at the proposed detector unit location, and the direction and strength of the Earth's magnetic field are measured. These readings should be taken within 30 minutes of those taken at the monitor location, and they must agree within the limits specified. This position is now accurately marked and referred to as the turntable location.

e. *Marking the North-South Line.* The North-South line is offset from the turntable location by a distance equal to the distance from the aircraft detector unit location to the aircraft centre-line (for a wing-mounted detector unit). If the detector unit is mounted in the aircraft tail-fin, the turntable location is a point on the North-South line. The line may be chalk-marked at this stage, and painted on the ground later.

The monitor and turntable locations should be permanently marked on the ground. Other turntable locations may be surveyed and marked when required.

Survey Procedures

22. **Initial Survey.** The requirements of paras 3, 4 and 16 should be met in selecting a site. For the initial survey, either type of bearing method may be used, but the reciprocal method using two WDCs will give the most accurate

MAGNETIC SURVEY RECORD

SURVEYORS: Flt Lt Allen
Flt Lt Brown
Fg Off Smith
DATE: 20 Nov 1972

BASE : RAF.....
RADIUS : 120 ft
CATEGORY : 1

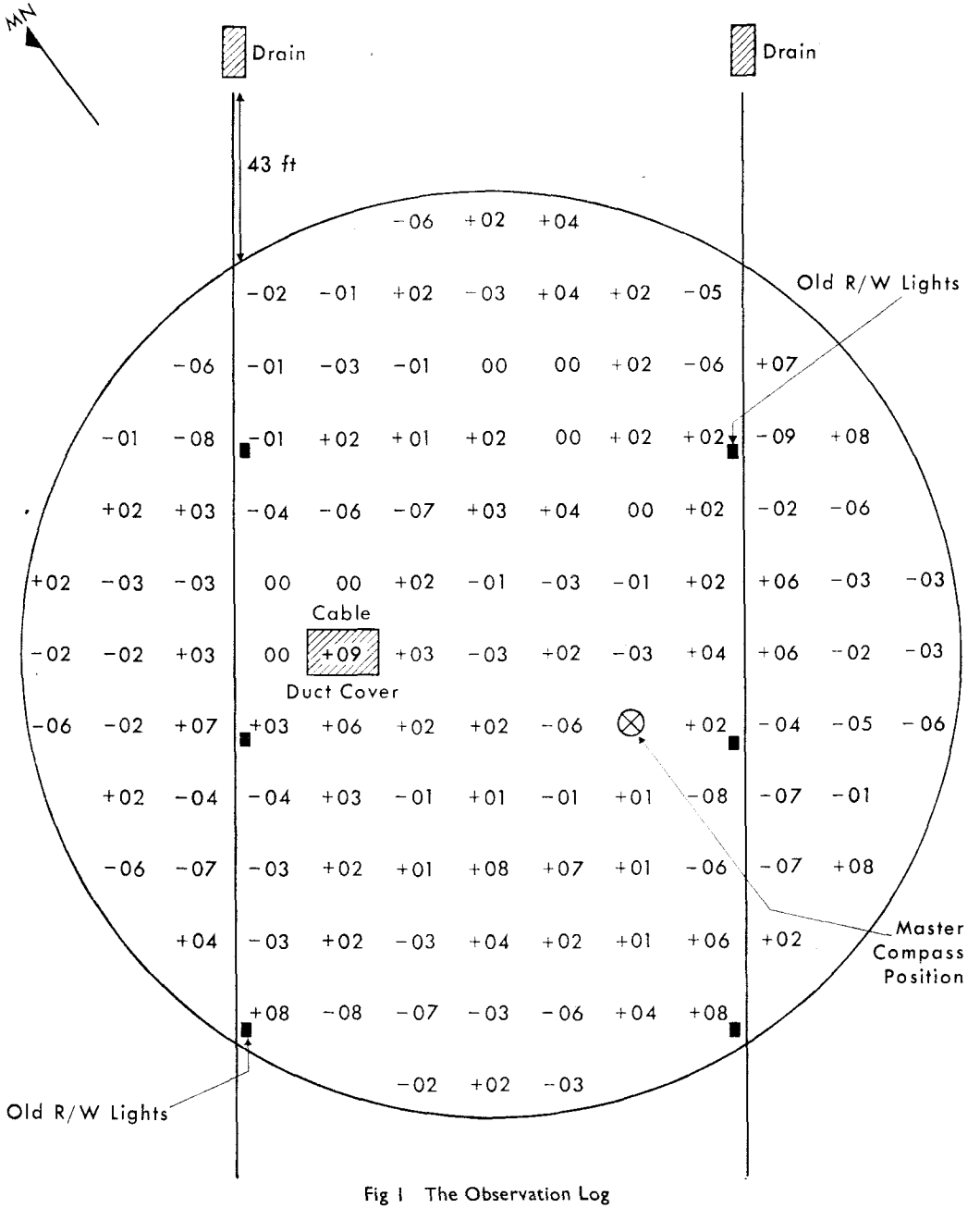


Fig 1 The Observation Log

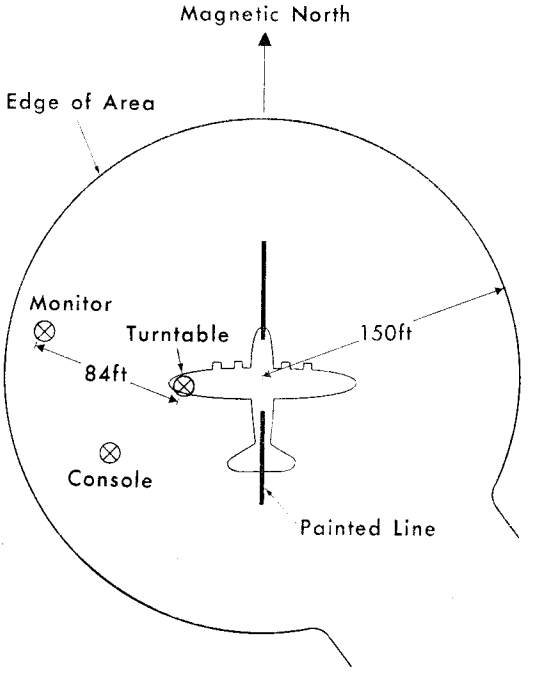


Fig 2 Equipment Layout

results. The procedure is:

a. Place the tripod in the likely centre of the compass base.

b. Carry out the instrument correction (para 15).

c. Position the mobile compass at various points around the site to give good coverage, take bearings between the compasses, and log the deviations.

d. If no deviations exceed $\pm 0{\cdot}1°$ for Class 1 or $\pm 0{\cdot}25°$ for Class 2 bases a full survey is justified. If the first few readings are outside the limits it may help to reposition the master compass.

23. **The Detailed Survey.** Both the Establishment Survey and the Periodic Re-Survey are to be done using the detailed survey method shown below:

a. Carry out the initial survey procedure of para 22.

b. Mark the site temporarily with stakes or paint at intervals of 20 feet to ensure accurate compass positioning.

c. Carry out the instrument correction (para 15).

d. Covering the site quadrant by quadrant find the deviations at intervals of 20 feet. Investigate anomalies as they are found (para 9).

e. Record the deviations on the observation log as shown in Fig 1.

f. Re-align the compasses in the master compass position at least once per hour, or whenever a change of variation is suspected.

g. Take bearing of permanent landmarks to pinpoint the site centre.

h. Mark out the site permanently on completion of the survey (para 5).

j. Send the particulars and copies of the observation log to the addresses shown in para 2.

24. **Area Survey.** Detailed procedures for doing an area survey using the MC-1M compass calibrator are contained in TO 5N3-3-7-1.

Printed for Her Majesty's Stationery Office by Wilfred Edmunds Ltd., Station Road, Chesterfield
781/808923/12,500/2-73/468/2.

PART 3

DR NAVIGATION SYSTEMS

Section

1 **Control Systems**

2 **Computing Principles and Circuits**

3 **Position Computing and Display Systems**

4 **Inertial Navigation Systems**

PART 3

SECTION 1

CONTROL SYSTEMS

Chapter

1 **Remote Indication and Control**

2 **Servomechanisms**

CHAPTER 1

REMOTE INDICATION AND CONTROL

CONTENTS

	Paras
Introduction	1–5
DC Systems	
Desynn Transmission Systems	6–9
M-Type Transmission System	10–17
AC Systems	
Introduction	18
Torque Synchros	19–21
Control Synchros	22–24
Differential Synchros	25–27
Resolver Synchros	28–33
Summary	
Summary Tables	34

Introduction

1. Instances frequently occur in aircraft instrument systems when the angular motion of a shaft has to be accurately reproduced at some other location. Direct mechanical linkage is often not suitable because of the distance involved or the resulting poor accuracy. In these cases a remote electrical indication system is often employed.

2. These remote indication systems translate movement of a shaft into electrical signals by means of a transmitter unit or transducer, which is electrically connected to a receiver unit located in the desired position. The movement of the first shaft is duplicated by the receiver which positions a second shaft, thus giving the remote indication of the first shaft's movement. By convention the first shaft is known as the input shaft and the second, the output shaft.

3. Simple systems may be employed consisting of a transmitter and receiver, electrically connected. Only small torques are developed such as is required to move a light pointer over a graduated scale. This is adequate for the remote indication of, for example, DF bearings or the position of a radar scanner (*see* Fig 1).

4. There are, however, many occasions when the accurate remote control of the position of a heavy load is required (*eg* remote rotation of a radar scanner). To provide the necessary torque servomechanisms (*ie* amplifiers and servo motors) are normally employed.

5. A number of different devices are used to give remote indication of angular position or to control the movement of heavy loads from a distance. Both DC and AC systems are used and these are discussed below.

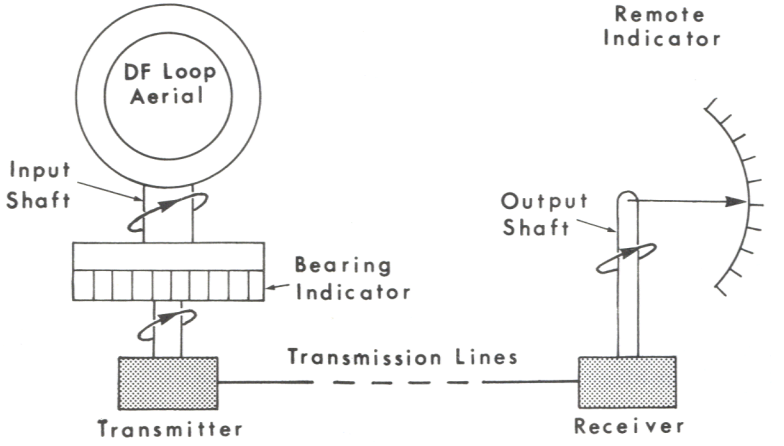


Fig 1 Simple Electrical Remote Indication

DC SYSTEMS

Desynn Transmission System

6. The Desynn Transmission System is a simple transmission system with low torque characteristics which is used for the remote indication of angular position. It is often used where a simple pointer and scale is adequate, *eg* remote indication of flap, rudder and elevator positions, or to repeat the reading of an instrument at a remote point. The accuracy of the system is ± 2° (1σ).

7. **The Transmitter.** The transmitter (*see* Fig 2) consists of a continuous resistance ring (toroidal potentiometer) having three fixed tappings A, B and C spaced 120° apart which are connected to the receiver. The input shaft carries two spring loaded sliding contacts or wipers diametrically opposed in

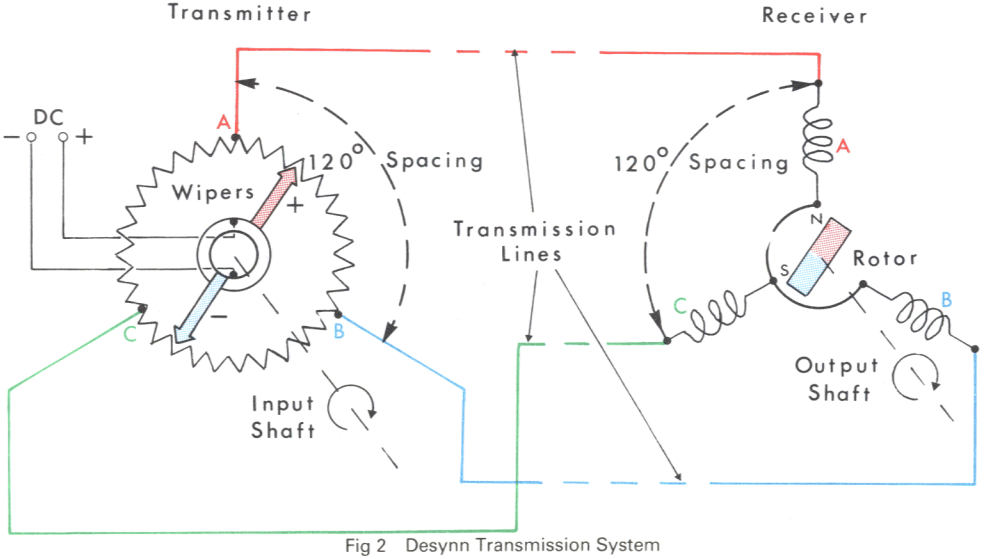


Fig 2 Desynn Transmission System

contact with the potentiometer. The wipers are fed via slip rings and brushes with DC.

8. **The Receiver.** The receiver (*see* Fig 2) consists of three high resistance coils whose axes are spaced 120° apart, with a permanent magnet rotor pivoted at their centre carrying a pointer. The three coils are connected to the tapping points A, B and C in the transmitter.

9. **Desynn Operation.** When DC is applied to the transmitter wipers, the voltages at the tapping points A, B and C produce a current flow in the three stators of the receiver and a resultant magnetic field is produced. The rotor magnet aligns itself with this magnetic field. The magnitude and polarity of each tapping point voltage varies according to the position of the wipers and thus, if the input shaft is rotated, the change of voltages at A, B and C produces a variation in the current flowing in the stator coils and rotation of the resultant magnetic field in sympathy with the rotation of the input shaft. The rotor magnet remains aligned with this field at all times and so rotates in synchronism with the input shaft. This operation is shown in Fig 3 a and b:

a. In Fig 3a the voltage distribution around the potentiometer is such that point A is at +24 V while B and C are both at +8 V. Thus, as the voltage at A differs from that at B and C by the same amount, current flows from A through coil A in the receiver then divides equally at the star point with half the total current flowing through coil B and half through coil C back to the transmitter. The resultant magnetic field in the receiver, with which the rotor magnet aligns itself, is compounded from the vectors representing the individual fields.

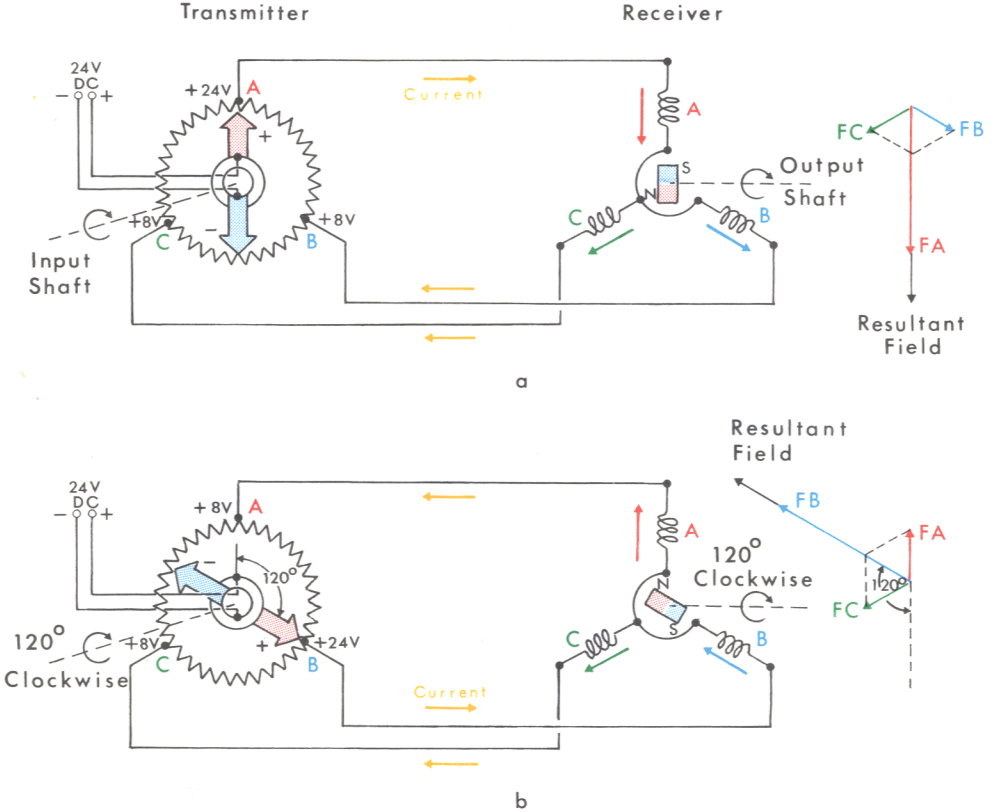


Fig 3 Operation of Desynn Transmission System

b. If the input shaft is rotated through 120° in a clockwise direction as shown in Fig 3b, the voltage distribution around the potentiometer is such that current flows from B through coil B in the receiver then divides equally to flow through coils A and C back to the transmitter. The vectors show that the resultant magnetic field also rotates through 120° clockwise and the rotor shaft aligns itself along this new axis.

Thus if the wipers in the transmitter are placed in any position by the input shaft, the resultant field at the receiver and hence the rotor magnet take up corresponding positions. If a pointer, moving over a calibrated scale, is attached to the rotor, remote indication of the position of the input shaft is immediately available.

M-Type Transmission System

10. The amount of torque produced by the Desynn system is limited by the amount of current which can be taken by the low resistance toroidal potentiometer before overheating occurs. Where moderate torque is required to rotate fairly substantial indicators or comparable devices, an M-type or step by-step-transmission system can be used. In the M-type system the transmitter is modified considerably from that used in the Desynn system but the receiver operates on the same principle.

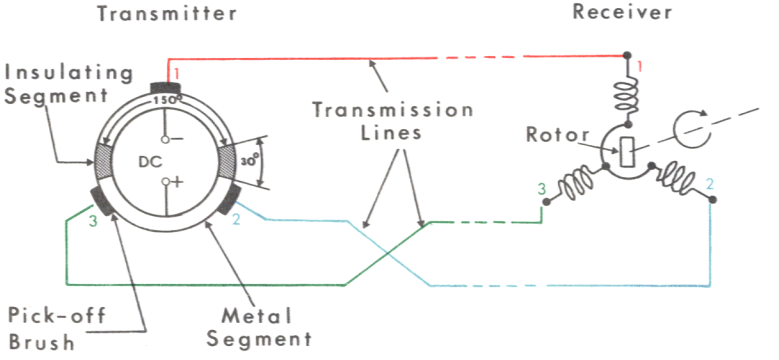


Fig 4 M-Type Transmission System

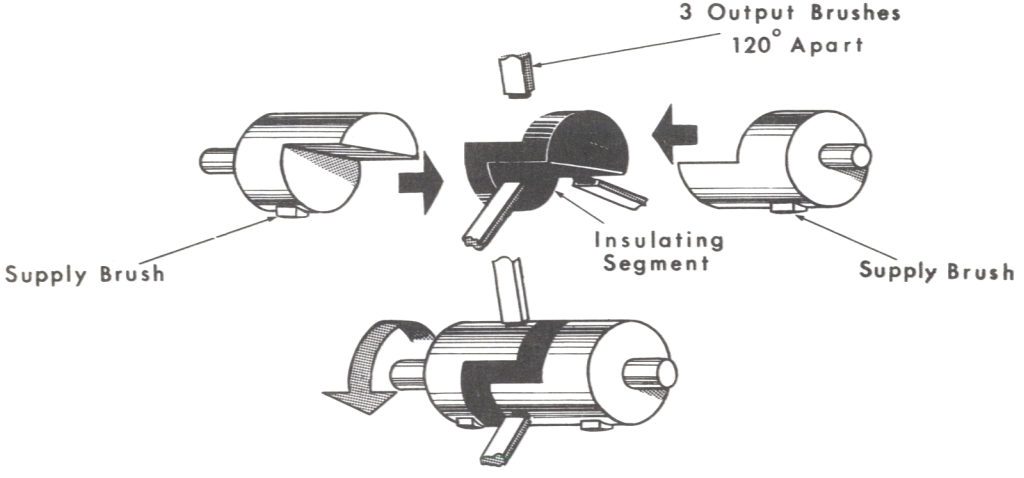


Fig 5 M-Type Drum Transmitter

11. The essential features of a simple M-type transmission system are shown in Fig 4. The transmitter is basically a drum type switch, the drum consisting of two segments each spanning an arc of 150° separated by two sections of insulating material each extending over 30°. The two metal segments are connected to opposite poles of a suitable DC supply and three pick-off brushes are disposed around the drum at intervals of 120° (*see* Fig 5).

12. The receiver unit is similar to that in the Desynn system, although the rotor may be either a permanent magnet or a laminated soft-iron core. The outer end of each coil in the receiver is connected to one of the three pick-off brushes in the transmitter. More than one receiver may be operated from a single transmitter.

13. **System Operation.** Operation of the M-type transmission system is shown in Fig 6.

a. In position 1 of the input shaft, brush 1 is connected to the negative supply and brushes 2 and 3 to the positive. These polarities are applied to the three coils in the receiver so that the current divides through coils 2 and 3 with all the current flowing through coil 1. Magnetic fields F1, F2 and F3 are produced and vector resolution produces the resultant field as shown.

b. Rotation of the input shaft through 30° clockwise (position 2 in Fig 6) produces a condition where brush 1 is negative, brush 2 is disconnected by the insulated segment and brush 3 is positive. At the receiver,

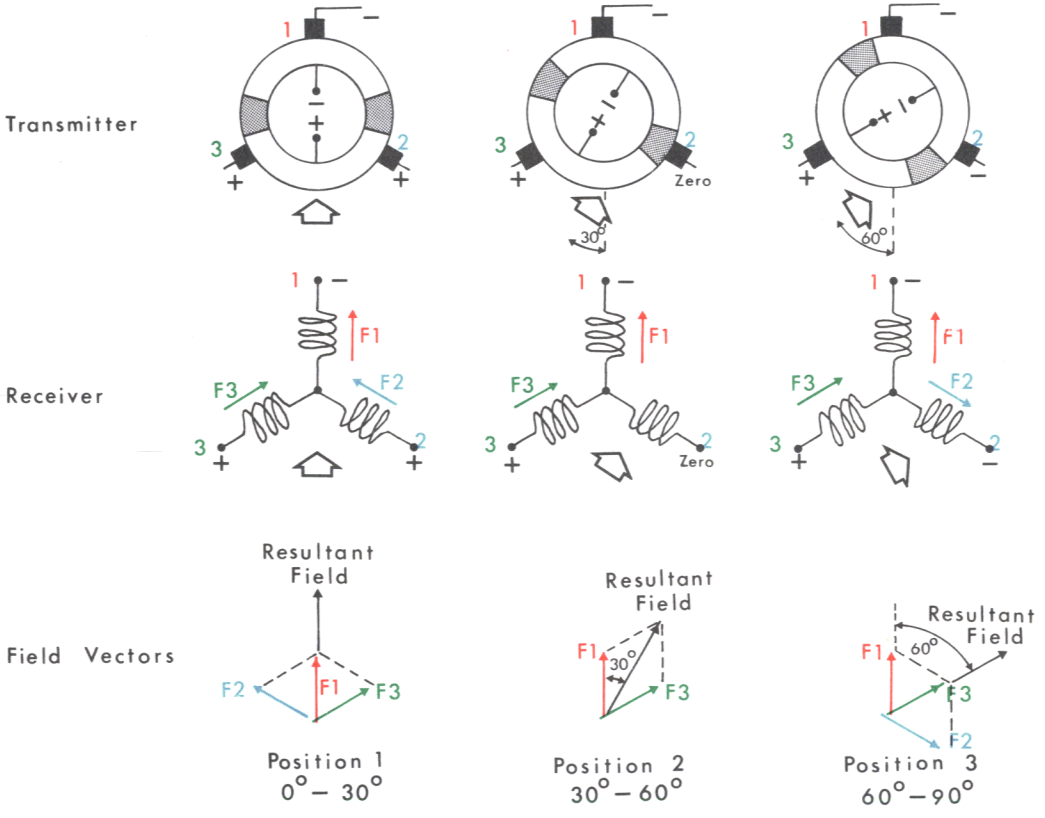


Fig 6 Operation of M-Type Transmission

equal currents flow through coils 1 and 3, while coil 2 carries no current. Resolution of fields F1, F2 and F3 now produces a resultant field which is seen to have rotated through 30° clockwise in sympathy with the input shaft.

c. The condition after a further 30° rotation of the input shaft is shown at position 3. The resultant field, again following the input shaft, is now rotated 60° from the initial position.

The receiver rotor aligns itself with the axis of the resultant field and hence the angular movement of the input shaft, but only in discreet steps of 30°. There is a change of pick-off brush polarity at one or other of the brushes each time the input shaft is turned through 30°. The complete pattern of brush polarities through 360° of rotation is given in Table 1 which shows that there are 12 steps in a complete rotation. For certain purposes, the 30° step is too large and a modified system, giving 24 steps of 15° each, may be used to improve the sensitivity of the system.

15. **Types of Receiver.** The rotor of the the receiver may be either of the soft iron (inductor) type or a permanent magnet. The inductor rotor is built up of iron and aluminium laminations and continuously aligns itself with the axis of the resultant field in the stator to offer the path of lowest reluctance, *ie* when the laminations are in line with the resultant flux. Since this type of rotor is non-polarized it is possible for it to align itself in either of two positions 180° apart. The permant magnet rotor, which is more commonly used, does not suffer from this ambiguity. Due to the relatively strong magnetic field produced by the magnet, the rotor torque is considerably higher than that of the induced type and, being polarized, the rotor lines up in one position only.

16. **Synchronization of Transmitter and Receiver.** The fact that the receiver rotor in

Step	Transmitter Position(°)	Pick-off Brush Polarity		
		1	2	3
1	0 – 30	–	+	+
2	30 – 60	–	0	+
3	60 – 90	–	–	+
4	90 – 120	0	–	+
5	120 – 150	+	–	+
6	150 – 180	+	–	0
7	180 – 210	+	–	–
8	210 – 240	+	0	–
9	240 – 270	+	+	–
10	270 – 300	0	+	–
11	300 – 330	–	+	–
12	330 – 360	–	+	0

Table 1 Polarity Changes During 360° Rotation of Transmitter

14. **Types of Transmitter.** Two other types of transmitter are in common use in M-type transmission systems. These are commutator and eccentric cam type transmitters. The former is a development of the drum transmitter and gives 24 × 15° steps. The latter does not suffer from brush wear and is preferred when the rotation rate is high.

an M-type transmission system only moves in 30° (or 15°) steps is a disadvantage. Greater sensitivity can be achieved by gearing up the input shaft to the transmitter shaft. A 60:1 gearing system is commonly

used, the transmitter shaft completing 60 revolutions for each revolution of the input shaft. The receiver is geared down by an equal ratio if a 1:1 output to input ratio is required. Although a 60 times increase in sensitivity is obtained in this case, the possibility of ambiguity is introduced. One revolution of the transmitter shaft now represents a rotation of 6° of the input shaft producing 12 steps of ½° each (with a 30° step transmitter) and there are 60 different positions in the full 360° movement of the input shaft, each separated by 6°, into which the receiver can "lock" and still follow the M-type sequence. In all but one of these positions, the output shaft will be out of synchronization with the input shaft. Initial course synchronization is therefore necessary and this is normally achieved by manual adjustment before the transmission system is used.

17. **Accuracies.** Because of frictional and resistive losses, the accuracy of M-type transmission systems is seldom better than ± 1°. This accuracy is adequate for the remote transmission of shaft rotation rates such as analogues of ground speed, but presents problems in the transmission of actual shaft position, *eg* heading.

AC SYSTEMS

Introduction

18. The application of the DC systems described above is limited to the remote indication of shaft position and the transmission of moderate torques to remote indicators or other devices. AC systems are generally preferred for high accuracy applications and also where servomechanisms are involved. The AC systems are self-synchronous (hence the name – synchro) and are divided into four groups:

a. Torque synchros.
b. Control synchros.
c. Differential synchros.
d. Resolver synchros.

Torque Synchros

19. The basic torque synchro consists of a transmitter (TX) and a receiver (TR), both of which are very similar. Each has a stator made up of three windings, star connected at 120° to each other, and a rotor which is a single winding energized by an AC supply. Fig 7 shows a diagrammatic representation of a torque synchro system, and the actual construction is shown in Fig 8. The TX and TR rotors differ in that the TR rotor is

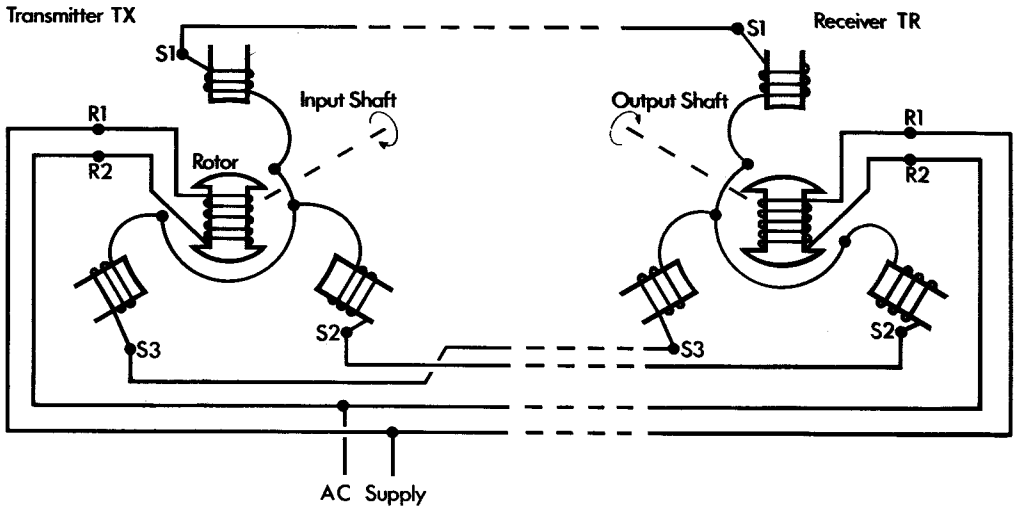


Fig 7 Basic Torque Synchro System

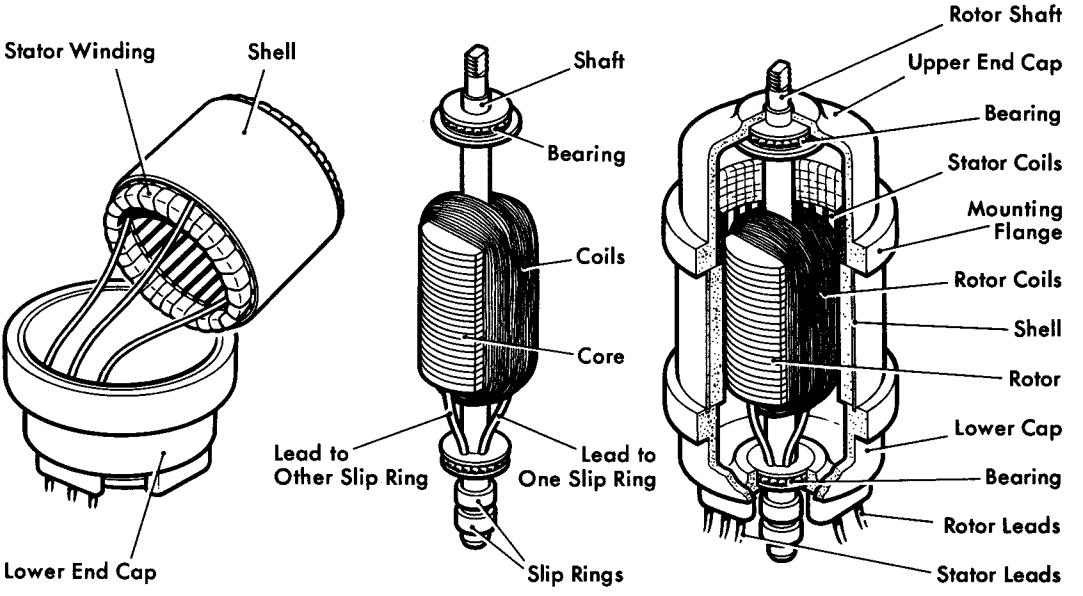


Fig 8 Construction of Torque Synchro

normally fitted with a mechanical damper to prevent oscillation.

20. **Torque Synchro Operation.** The operation of torque synchro is shown in Fig 9. The TX rotor, energized by the AC supply, has an associated alternating field which cuts the windings of the TX stator coils producing an induced emf. Because the TX stator windings are in closed circuit with the TR stator windings a current flow occurs which, by Henry's Law, must be of such a direction and magnitude as to produce a field associated with the TX stator which is equal in

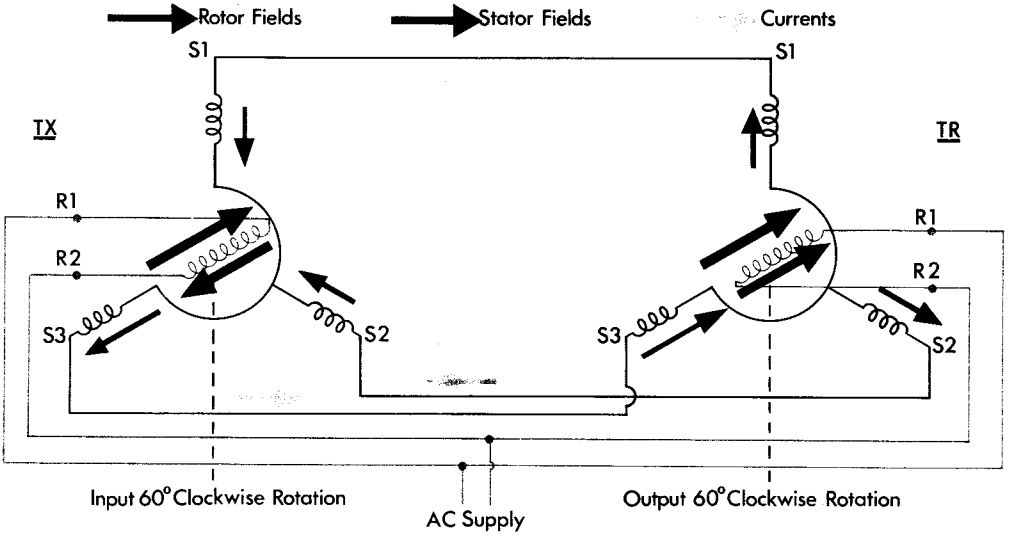


Fig 9 Operation of the Torque Synchro

strength, but in the opposite direction to the TX rotor inducing field. A similar field, parallel to the TX rotor field, is produced in association with the TR stator. The TX and TR rotors are energized by the same AC supply and thus have associated magnetic fields. The presence of both rotor and stator fields within the TR causes the rotor to turn to align its field with that of the stator and thus with the fields of the TX stator and rotor. As the phase of the AC supply changes, all the field directions simply reverse and the system remains in alignment. As the two rotors reach alignment, they induce equal but opposite emfs in the two stators; current ceases to flow and the fields collapse. The stator coils are of low impedance and any rotor misalignment produces sufficient current flow to produce reasonable torque.

21. **Torque Synchro Accuracy.** As the torque synchro approaches synchronization, the field structure collapses and the available torque falls off. If a high degree of accuracy is required the load must be limited; lightly loaded torque synchro accuracy is ± 1° (1σ).

Control Synchros

22. If it is required to move heavier loads a control synchro, employing a separate servomotor to provide the necessary torque amplification, may be used. Control and torque synchros are similar; both have three-winding stators and single-winding rotors. The control transmitter (CX) rotor is AC energized, but not the control transformer (CT) rotor. A control synchro system is shown at Fig 10.

23. **Control Synchro Operation.** The CX rotor, fed from an AC supply, produces an alternating field which, by Henry's Law induces an opposing field in the CX stator. The circuit current causes a magnetic field associated with the CT stator and parallel to the CX rotor field. When the CT rotor is at 90° to the CT stator field there is no induced emf (or error signal) in the rotor: the rotor is said to be in the "null" position. If the CT rotor is displaced from the null position, an emf, proportional to the angular displacement from the null position, is induced in the rotor, the phase of this induced emf depending upon the direction

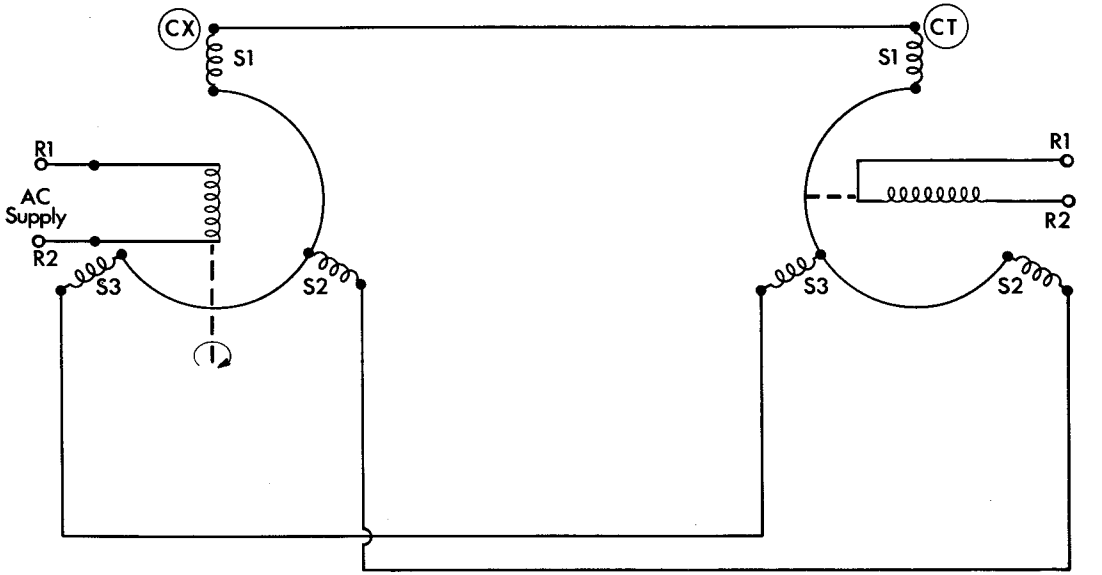


Fig 10 Control Synchro System

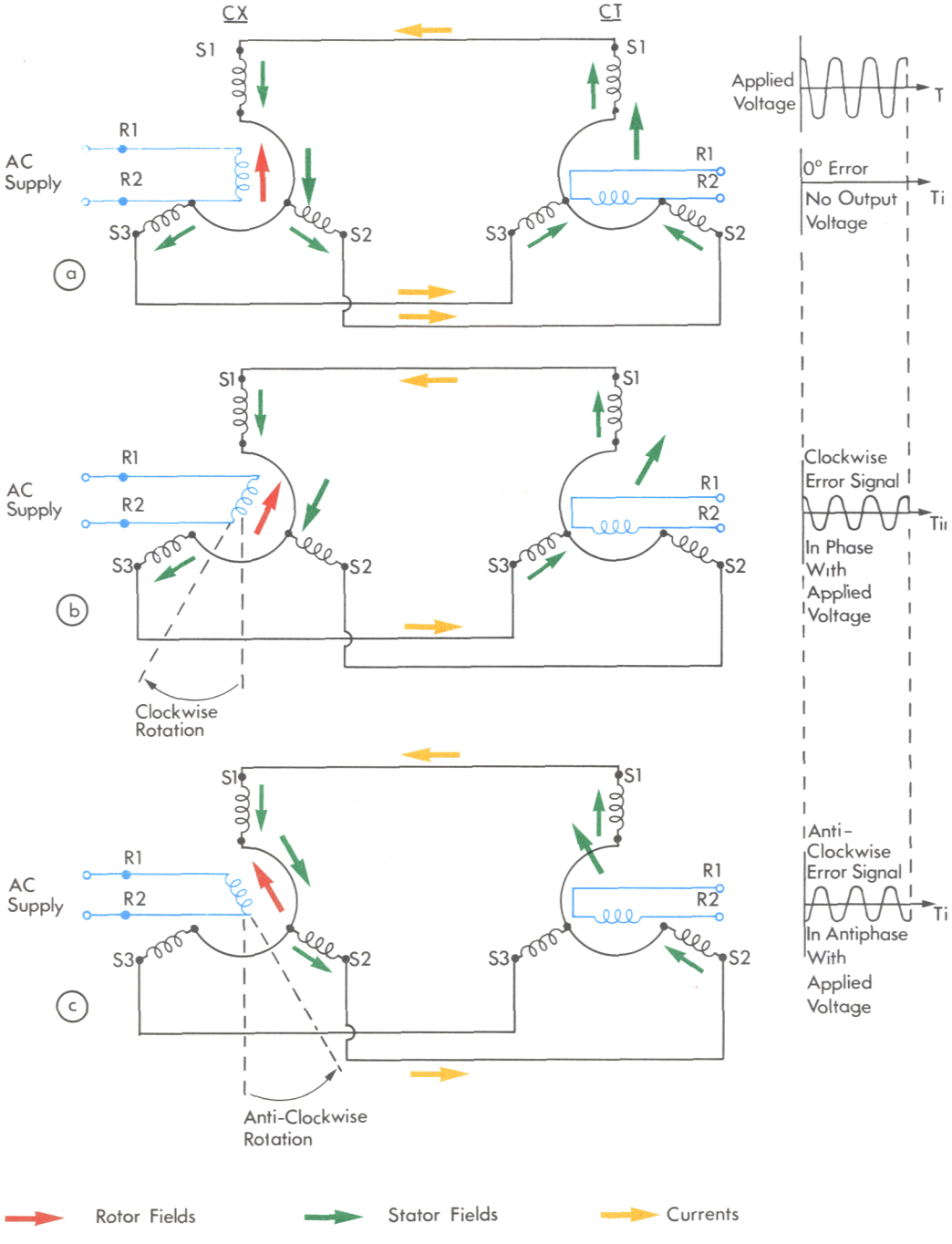


Fig 11 Operation of a Control Synchro System

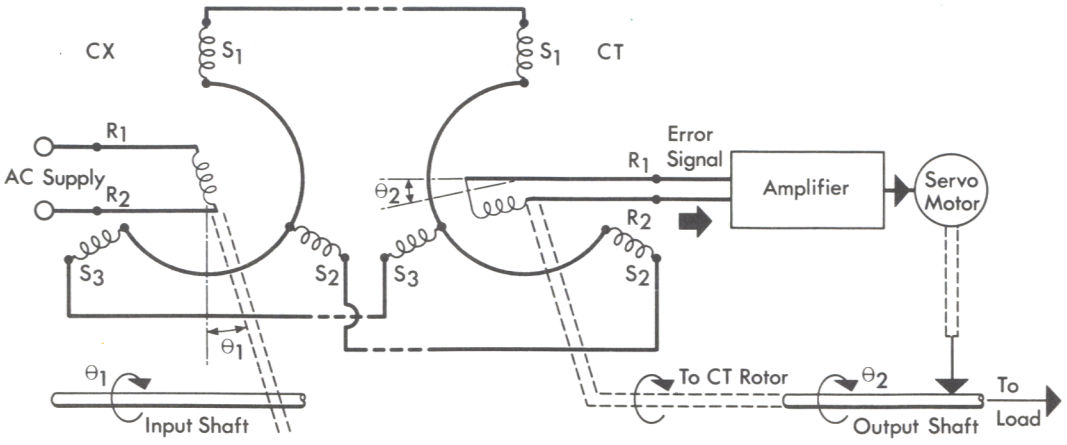


Fig 12 Complete Control Synchro System

of displacement. The operation of the control synchro system is shown in Fig 11. The induced error signal is amplified and fed to one phase of a two phase servo-motor which drives the output shaft of the CT rotor. The second phase is supplied by the same AC source supplying the original CX rotor input. The motor drives the output shaft and the CT rotor until the induced error signal is zero; the direction of movement being determined by the phase of the error signal. A complete control synchro system is shown at Fig 12.

Note: Current flow is continuous in the control synchro, the current magnitude being limited by employing high impedance stators.

24. **Control Synchro Accuracy.** Using a suitably powered motor, the control synchro accuracy is independent of load. Typical accuracy figures are ± 6 minutes of arc (1σ).

Differential Synchros

25. Differential synchros may be used to add or subtract two shaft rotations. The

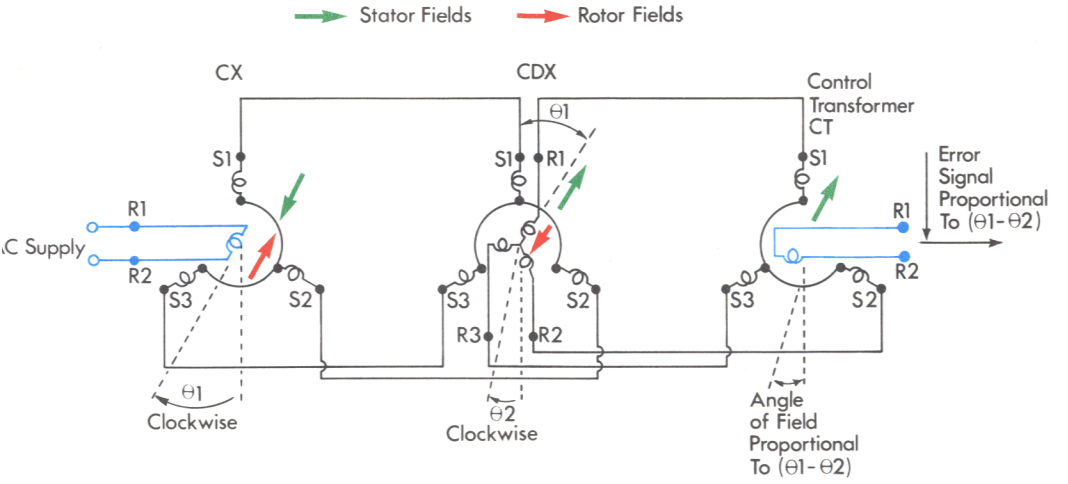
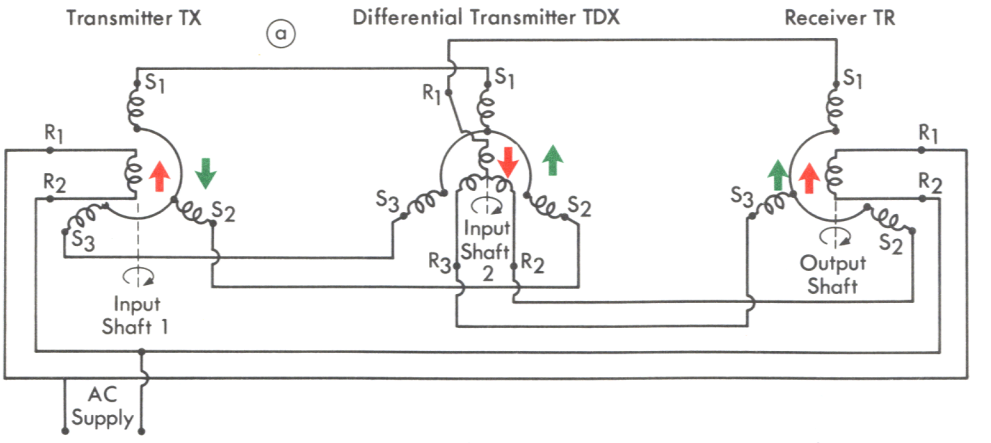
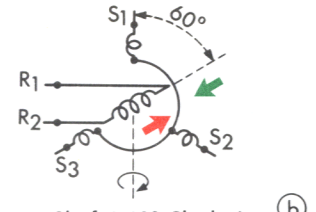


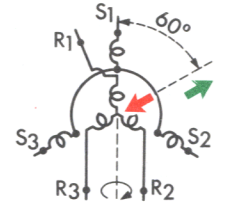
Fig 13 Differential Synchro System



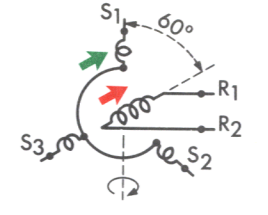


Input Shaft 1 60° Clockwise

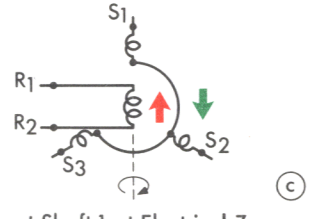
b



Input Shaft 2 at Electrical Zero

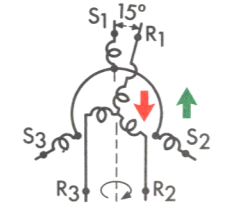


Output Shaft 60° Clockwise

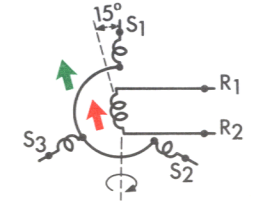


Input Shaft 1 at Electrical Zero

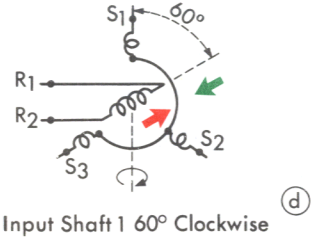
c



Input Shaft 2 15° Clockwise

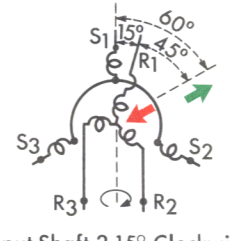


Output Shaft 15° Anti-Clockwise

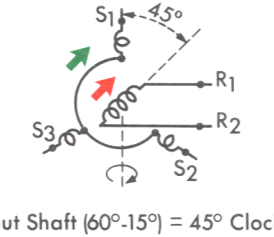


Input Shaft 1 60° Clockwise

d



Input Shaft 2 15° Clockwise



Output Shaft (60°-15°) = 45° Clockwi

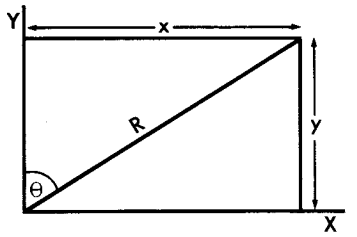
Stator Fields Rotor Fields

Fig 14 Action of the Differential Synchro

differential synchro (CDX) consists of a three-winding stator and a three-winding rotor. The control system in Fig 13 includes a CDX. The CDX becomes a TDX when used in a torque synchro system.

26. **Differential Synchro Operation.** The operation of a differential synchro within a torque synchro is illustrated in Fig 14 (the operation within a control synchro system is similar). Shaft rotation 1 is fed to the TX rotor in the normal manner causing an induced field associated with the TDX stator parallel to the TX rotor field. Shaft input 2 is fed to the TDX rotor. An emf is induced in the TDX rotor coils by the TDX stator alternating field. The TDX rotor coils are connected to the TR stator coils and, consequently, the current flow produces a magnetic field associated with the TDX rotor which opposes the field in the TDX stator. A magnetic field is also induced in association with the TR stator coils and alignment of the TR rotor takes place as explained above (para 20).

27. **Application of Differential Synchros.** Although the operation of both TDX and CDX are identical in theory, their windings are different because of the different system current flows: torque systems have zero current flow when aligned, whereas control systems have continuous current flow. Several differential synchros can be included in a system, *eg* two could be used, in tandem, to add variation and drift to magnetic heading to give an output of true track.



$$x = R\sin\theta$$
$$y = R\cos\theta$$
$$R = \sqrt{x^2 + y^2}$$
$$\theta = \tan^{-1}\frac{x}{y}$$

Fig 15 Relationship Between Polar and Cartesian Co-ordinates

Resolver Synchros

28. **Co-ordinate Systems.** The relationship of one point to another may be defined in either of two ways:

a. Polar co-ordinates (range and bearing)

b. Cartesian co-ordinates (distances X and Y along orthogonal axes).

The two co-ordinate systems are shown in Fig 15, together with the equations relating one system to the other.

29. **Use of Resolvers.** The resolver synchro is used to convert one co-ordinate system to the other. Ground speed and track define a vector in polar co-ordinates. The same vector may be expressed in northings and eastings in cartesian form. Similar resolvers are used to convert from polar to cartesian and vice versa, but the modes of operation are slightly different. The resolver synchro consists of a stator and a rotor, both having two orthogonal windings. The resolver synchro is illustrated in Fig 16.

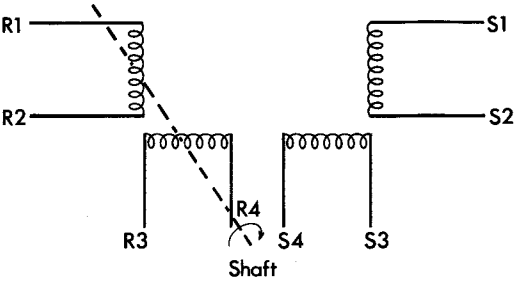


Fig 16 Resolver Synchro

30. **Resolver Synchro (Resolving).** In the resolving mode, the resolver synchro converts polar to cartesian co-ordinates *eg* ground speed and track to northings and

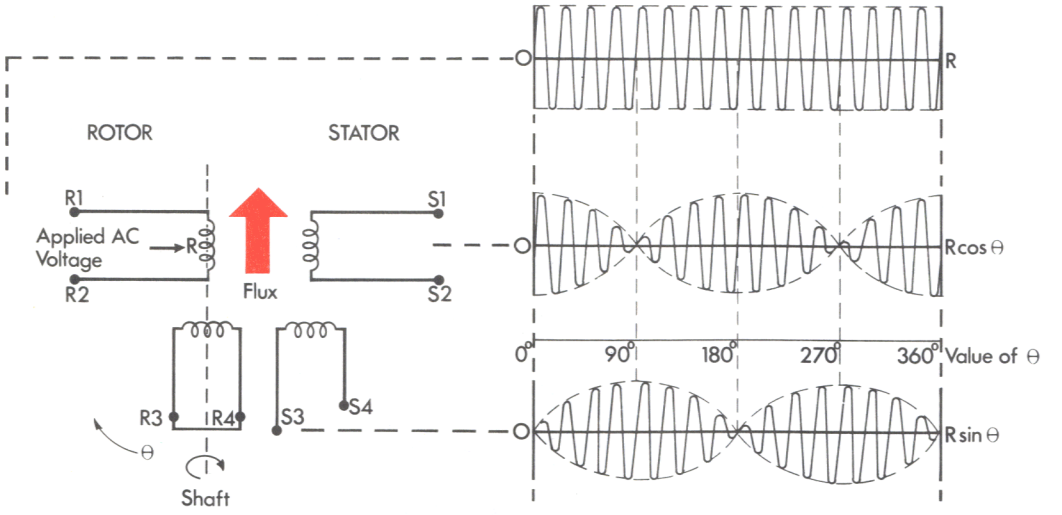


Fig 17 Conversion of Polar to Cartesian Co-ordinates

eastings. In this example R (an electrical analogue of ground speed) is applied, as an AC voltage, to one rotor winding. The rotor is then turned through angle θ (track). The rotor field has components of Rcos θ (northings) and Rsin θ (eastings) along the stator winding axes; the voltages induced in the stator windings are proportional to Rcos θ and Rsin θ. The operation of the resolver synchro (resolving) is shown in Fig 17.

31. **Resolver Synchro (Compounding).** A similar resolver is used to convert cartesian to polar co-ordinates, but in this case additional components are needed as shown in Fig 18. The Y and X co-ordinates

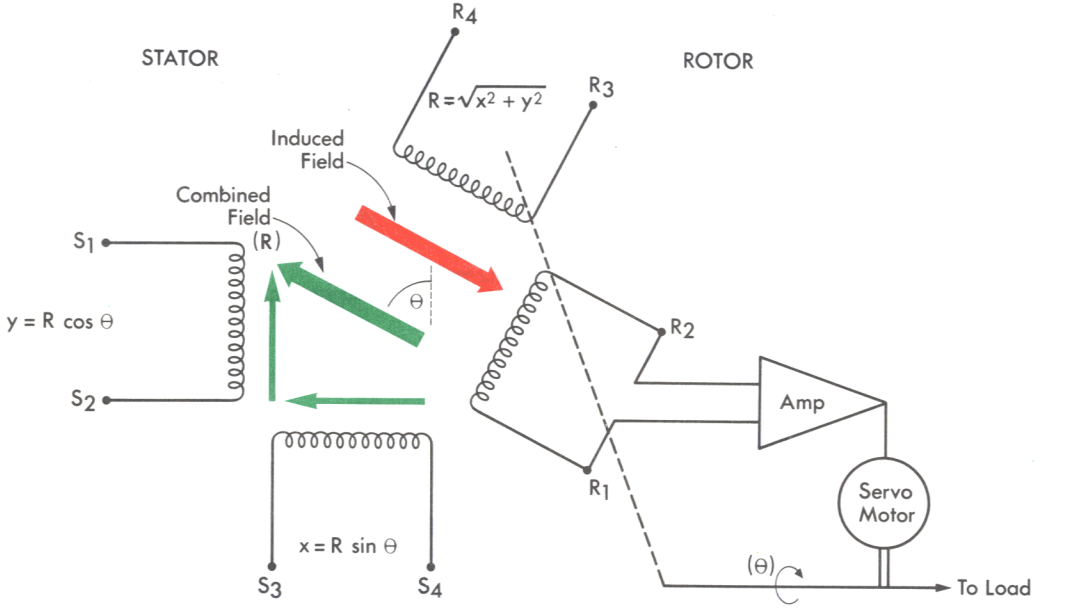


Fig 18 Conversion of Cartesian to Polar Co-ordinates

(northings and eastings) are fed to the stator windings as AC voltages, the associated fields combining to produce a stator field of magnitude R at an angle θ (R and θ are analogues of ground speed and track). One of the rotor windings is connected to an amplifier and servo-motor in the same manner as a control receiver (CT). It is therefore driven to a position at 90° to the stator field and the output shaft is turned through the angle θ, thereby deriving track. When the CT connected rotor winding is at 90° to the total stator field, the other rotor winding lies parallel to that field. In this position a field proportional to R is produced in association with this rotor winding and hence a voltage analogue $\sqrt{x^2 + y^2}$ of ground speed may be obtained.

32. **Resolver Synchro (Differential).** It is often necessary to produce the sine and cosine of the sum of two angles multiplied by a given value, *eg* northings and eastings relative to true North represented by R (ground speed), multiplied by cosine and sine θ (true track) may be required as grid northings and grid eastings. If the angle between true North and grid North is represented by ϕ, then required outputs are Rcos (θ + ϕ) and Rsin (θ + ϕ) as illustrated in Fig 19.

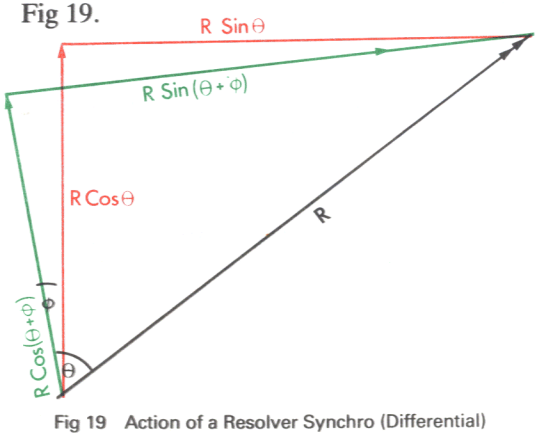


Fig 19 Action of a Resolver Synchro (Differential)

33. **Operation of Differential Resolver Synchro.** The operation of the differential resolver synchro is shown at Fig 20. There are three inputs; Rcos θ (true northings) and Rsin θ (true eastings) both fed as voltage analogues to the stator coils, ϕ (convergence) is fed as a shaft rotation to position the rotor coils relative to the stator. The fields associated with the stator coils may be resolved into 4 sub-fields, two parallel to each rotor coil. From Fig 20, two of the sub-fields are shown to be additive and two subtractive. The sub-field values are:

a. Rcos θ cos ϕ − Rsin θ sin ϕ = Rcos (θ + ϕ).

b. Rsin θ cos ϕ + Rcos θ sin ϕ = Rsin (θ + ϕ).

Output voltages taken from the rotor coils are analogues of R cos (θ + ϕ) (grid northings) and R sin (θ + ϕ) (grid eastings). Thus the differential resolver synchro redefines cartesian co-ordinates about a new datum direction. The versatility of the device may be illustrated by imagining inputs of northings, eastings and desired track. The outputs would then be distance gone along and across desired track.

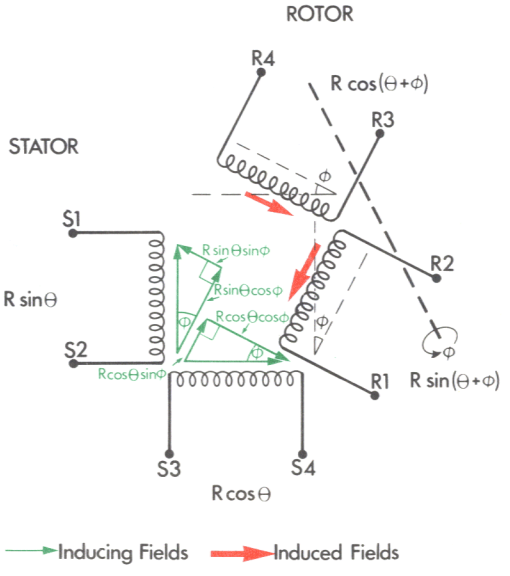


Fig 20 Operation of Differential Resolver Synchro

SUMMARY

Summary Tables

34. Table 2 summarizes the remote indication systems discussed in the preceding paragraphs. Table 3 summarizes the pertinent detail of the various types of synchro mechanisms.

System	Remarks
Desynn	DC. Provides only sufficient torque to operate small instruments: gives remote indication of dial readings to an accuracy of about ± 2°.
M-Type	DC. Provides moderate torque, sufficient to drive small mechanisms: accurate to about ± 1°. Typical use is to rotate the scanning coils in a CRT in synchronism with a radar aerial.
Torque Synchro	AC. Provides only sufficient torque to operate small instruments: efficient and accurate to within ± 1°: often used to transmit data such as radar bearings to the place where the information is required.
Torque Differential Synchro	AC. As for the torque synchro, but provides summation of two input shaft angles, *eg* to combine a DF loop reading and a compass reading to give true bearing.
Control Synchro	AC. Gives an electrical output that is dependent on the error in alignment between driving shaft and load shaft. The error signal is normally used as the input to a control system driving a heavy load. Accuracy about ± 6′ arc.
Control Differential Synchro	AC. As for control synchro, but provides summation of two input shaft angles.
Resolver Synchro	AC. Used in computers to give either cartesian or polar co-ordinates of an input, and for conversion of one to the other: can also be used in a manner similar to that of a control synchro.
Resolver Differential Synchro	AC. Gives an electrical output in the form of sine and cosine values of the sum or difference of two input angles.

Table 2 Summary of Remote Indication Systems

Component	Code	Inputs	Outputs	Uses
Torque Transmitter	TX	Mechanical rotation of rotor	Electrical from stator	Transmits angular information
Torque Receiver	TR	Electrical to stator	Mechanical rotation from rotor	Operates low torque equipment
Torque Differential Transmitter	TDX	Electrical to stator and mechanical rotation of rotor	Electrical from rotor	Transmits the sum of angular inputs
Torque Differential Receiver	TDR	Electrical to stator and rotor	Mechanical rotation from rotor	Provides low torque equipment with the sum of two angular inputs
Control Transmitter	CX	Mechanical rotation of rotor	Electrical from stator	Transmits angular information
Control Transformer	CT	Electrical to stator	Error signal to servo loop	Controls position of servo mechanism
Control Differential Transmitter	CDX	Electrical to stator and mechanical rotation of rotor	Electrical from rotor	Transmits the sum of two angular inputs
Control Transmitter with rotatable stator	CXB	Mechanical rotation of stator and rotor	Electrical from stator	Transmits the sum of two angular inputs
Control Receiver with rotatable stator	CTB	Electrical to stator and mechanical rotation of stator	Error signal to servo loop	Provides a position servomechanism with a control signal which is the sum of two angular inputs
Resolver Synchro (Resolving)	RS	Electrical to rotor and mechanical rotation of rotor	Electrical from stator	Resolves polar co-ordinate inputs to cartesian co-ordinate outputs
Resolver Synchro (Compounding) OR Arc/Tan Resolver	RS	Electrical to stator	Electrical from rotor and mechanical rotation of rotor	Compounds cartesian inputs to polar outputs
Resolver Synchro (Differential)	RS	Electrical to stator and mechanical rotation of rotor	Electrical from rotor	Redefines cartesian co-ordinates about a new datum direction

Table 3 Synchro Details

CHAPTER 2

SERVOMECHANISMS

CONTENTS

	Paras.
Introduction	1–2
Simple Control System	3–8
Automatic Control System	9–11
Servomechanisms	12–13
Types of Servos	14–16
EXAMPLES OF SERVOMECHANISMS	
Introduction	17–19
Typical Position Control Servo	20–26
Typical Speed Control Servo	27–30
APPLICATIONS OF THE SERVO PRINCIPLE	
Introduction	31
Gyro-Magnetic Compass	32–33
Accelerators	34–36
Computer Applications	37–38
PERFORMANCE OF SERVOMECHANISMS	
Introduction	39–40
Response	41–42
Step Input—No Friction	43-46
Ramp Input—No Friction	47
Effect of Restraints	48–49
Steady State Errors	50–52
Summary	53
IMPROVEMENT OF TRANSIENT RESPONSE	
Introduction	54
Viscous Damping	55–59
Velocity Feedback Damping	60–63
Summary	64
REFINED SERVOMECHANISMS	
Introduction	65
Error-Rate Damping	66–68
Transient Velocity Damping	69–70
Integral Control	71–72
CONCLUSION	
Summary	73–76
Reference	77

Introduction

1. The transmission systems described in Chapter 1 of this Section included many devices capable only of remote indication on light pointers. At least one, however, the control transmission system, could do more than this. It could not only transmit the information over considerable distances, but its receiving element included parts which released much greater power at the output than was available at the input. A lightly applied movement at the input could control the position of a heavy load.

2. The receiving elements of the control transmission system are members of a large family of control devices known as servomechanisms, all of which have this ability to amplify the input force. To define a servomechanism (or servo) properly, however, its pattern of operation must follow a particular sequence. This sequence, which need not involve remoteness of control, will now be examined.

Simple Control System

3. Suppose that we wish to control the position of a radar scanner or other heavy load. In the first instance we could couple up a motor capable of driving the scanner and fit some means of controlling it. Eventually an arrangement such as that in Fig 1 could be put together.

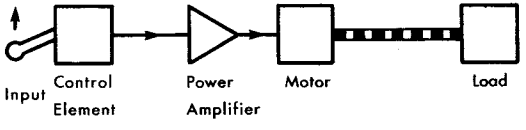


Fig 1 Elements of Control System

4. The control element, perhaps a variable resistor, applies the input to a power amplifier, which drives the motor in the required direction. The motor, in turn, moves the load.

5. The control element could be calibrated with a scale indicating the angle through which the input is turned. When the input is moved a voltage proportional to this angle is applied through the amplifier to the motor. The motor accelerates at a rate compatible with the load inertia and with restraints, such as friction, until it reaches a steady speed with the driving torque equal to the restraining torques.

6. Since restraining torques increase with speed, the load speed, and not its position, is controlled by a device of this type. Clearly the load will not stop at the required position unless some further action is taken.

7. Several courses of action are possible but perhaps the simplest and most obvious is to brief an operator to watch the load movement. He could slow the motor down as the load closed on the required position by drawing back on the control element, finally bringing it to rest. His actions would probably be such as to allow high speeds for large load movements and low speeds for small movements. In other words he would, at any rate during his first few attempts, move the control element by an amount proportional to the required angle. The voltage would then be regulated by the difference between the load angle and the input angle.

8. This control system, however, is not automatic; it can only be used when the operator can see the load and when fatigue on his part is unlikely. Once the possibility of prolonged operation is envisaged, or when the operator cannot read the load position or if the changes of input are too rapid for him to follow, then an automatic system must be used.

Automatic Control System

9. A simple automatic system can be designed to work in precisely the same way as the operator. The load position is fed to some device which compares it with the input and the difference between them regulates the voltage to the amplifier. The link between the load and the comparison device is known as *feedback;* the difference be-

tween the load angle and the input angle is called the *error* and the comparison device is termed the *error detector*. The voltage to the amplifier is called the *error signal* and it is usually produced within the error detector.

10. A block schematic diagram of the automatic system is illustrated in Fig 2. The essential features are as follows:–

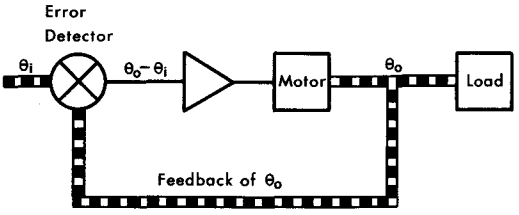


Fig 2 Automatic Control System

a. Application of the input angle, θ_i, to the error detector.

b. Feedback of the load position, θ_o, to the error detector.

c. Subtraction of θ_i, from θ_o to produce the error.

d. Production of an error signal proportional to $\theta_o - \theta_i$.

e. Control of the amplifier output by the error signal.

f. Control of the motor movement by the amplifier output.

g. Movement of the load by the motor in a direction which reduces the error.

The new load position is fed back to the error detector and the sequence b to g continues until the error is zero, when the error signal disappears and movement stops.

11. The automatic control system described operates by continuous cycling of the load position through the loop formed by the feedback, error detector, amplifier, and motor. Control mechanisms in which this loop can be identified are known as *closed loop* systems, while those which do not have feedback are known as *open loop* systems.

Servomechanisms

12. We are now in a position to define the servomechanism more clearly. To be classed as a servomechanism, an automatic control system must be capable of continuous operation and have:–

a. Error actuation.
b. Power amplification.
c. Closed loop control.

13. Thus the system in Fig 2 is a servomechanism. It is actuated by the error since the net input to the amplifier is the error signal and not a voltage representing the input angle. It has power amplification and closed loop control; it is fully automatic and capable of continuous operation.

Types of Servo

14. There are two main classes of servomechanism—position control servos and speed control servos:–

a. *Position Control Servos.* Position control servos are used to control the angular or linear position of a load. The input also will normally be an angle or position, but may be found in other forms.

b. *Speed Control Servos.* Speed control servos are used to control the speed of a load. In this case the input will not normally itself be a speed; inputs are usually in the form of voltages or shaft angles.

15. It should be noted that the control system shown in Fig 1 is not a speed control servomechanism. It has no feedback and is therefore an open loop system for speed control.

16. The classification into position and speed control servos is a convenient one in view of the applications of the servo princi-

ple met in normal service equipments. In general, however, the servo can control many things not embraced by these terms. Thermostatic control of a gas oven uses the servo principle, being actuated by the error in oven temperature; the control of the concentration of a solution in a chemical process is another example. Indeed the input and output may take so many forms that it is common practice to use non-committal descriptions such as input demand for θ_i and load behaviour for θ_o.

EXAMPLES OF SERVOMECHANISMS

Introduction

17. Servomechanisms of either classification, can be operated by a.c. or d.c. power supplies. In general the a.c. system is capable of greater accuracy and stability, but is limited to low power applications. The d.c. servo is used in high torque situations, but more often a hybrid a.c./d.c. servo, combining the merits of both, will be found when heavy loads are involved.

18. The synchro control transmission system has already been mentioned as an example of a servomechanism. It is illustrated in Fig 3 with the servo terms in red to assist the reader in identifying the features enumerated in para 10.

19. It is now proposed to examine two typical systems, one a position control servo, the other a speed control servo.

Typical Position Control Servo

20. Imagine a situation in which a delicate movement represents the measure of some quantity, and is required to actuate several stages of an analogue computer. The movement may be the deflection of a capsule, an ammeter or a voltmeter, or some other device. Whatever it is we suppose it is quite inadequate to the task. Power magnification is evidently needed, but in producing this the accuracy must not be degraded. These requirements are exactly those which can be filled by a servomechanism.

21. The outline of the requirement is shown in Fig 4, while Fig 5 shows elements which would typically achieve it.

22. The input shaft carries the demanded angle, θ_i, while the output shaft indicates, at that moment, θ_o. The error detector is known as a C and Y inductive pick-off, and consists of two pieces of soft iron material, one shaped C, the other Y. They are pivoted concentrically, the Y turning within the C. Wound on one limb of the Y part is an a.c. energized coil which sets up alternating fluxes through the other limbs. When the C part surrounds the Y symmetrically the flux flow through the limbs is distributed equally, as in Fig 6a, but if the C is turned the flux flow becomes unbalanced, Less flux flows at A because of the greater air gap, while more flows at B; the greater the movement of C, the greater the flux unbalance.

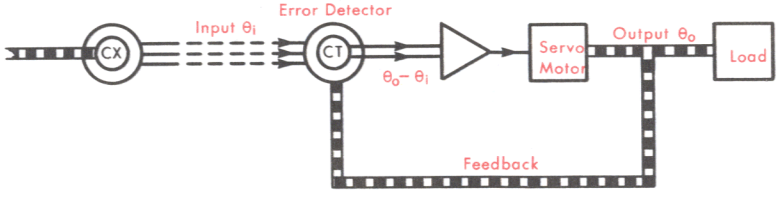


Fig 3 Servo Elements of Control Synchro

Fig 4 The Requirement

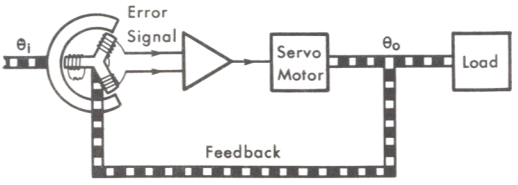


Fig 5 Servo Components

23. Identically wound secondary coils are placed on limbs 2 and 3 and the alternating flux induces voltages in them. When the C and Y are symmetrical the voltages will be equal, but when C is moved they will be unequal. These coils are connected in series so that the induced voltages are in anti-phase, and therefore cancel one another in the symmetrical position. In the unsymmetrical position, however, the voltage unbalance allows the output of one coil to dominate the other and a current flows. The phase of the output will indicate the direction of the movement of C while the amplitude represents the magnitude of the displacement.

24. Returning to Fig 5, the input and output shafts are coupled to the error detector, θ_i to C and θ_o to Y, in such a way that when $\theta_o = \theta_i$ the C and Y are symmetrical and no current flows. This then is the null position of the error detector. Any difference between θ_o and θ_i will result in a C and Y displacement and produce an error signal indicating the direction and size of the difference. The error signal is applied to the amplifier which activates the servomotor to move the load. The load turns towards θ_i, at the same time moving the Y around. When symmetry is reached the signal becomes zero and the load is at θ_i.

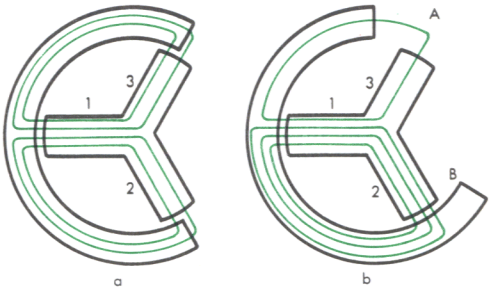


Fig 6 Flux Flow in Pick-off

25. The servomotor is usually a two-phase a.c. induction motor or hysteresis motor. In each case one winding is permanently fed at a reference phase, and the motor turns when the other winding is energised by the amplified error signal. The sense of rotation depends on the phase of the error signal relative to the reference phase. This in turn depends on the sense of the error. One point of operation, important in a later discussion, is that the torque delivered by these motors depends on the magnitude of the error signal.

26. The amplifier may be conventional valve or transistor, but magnetic amplifiers are becoming increasingly popular because of their robustness and freedom from drift.

Typical Speed Control Servo

27. The situation calling for a speed control servo in navigation equipments is most commonly that of converting a voltage representing airspeed or groundspeed into an angular velocity. A shaft turning at this angular velocity can then be used to display distance gone. The device used is called a velodyne and its components are illustrated in Fig 7.

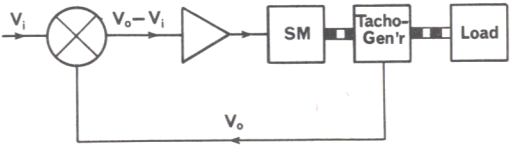


Fig 7 Speed Control Servo, the Velodyne

28. The input voltage V_i is applied through a power amplifier to turn a servometer which accelerates the load towards the required speed. Comparison between the load speed and the input voltage is made possible by converting the speed into a voltage. The conversion is effected by a tachogenerator coupled to the output shaft. This is a special type of generator which gives a voltage proportional to its speed of rotation. It can be very small and absorbs little power since only a voltage, with negligible current, is required.

29. The tachogenerator output, V_o, is fed back to be subtracted from V_i at the amplifier input. The motor is therefore controlled by the difference in voltages and will speed up or slow down until the difference is zero.

30. In practice the equality of voltages is never quite reached and a small residual difference is necessary to counter friction. Nevertheless a suitable choice of components can ensure an input-output relationship which is very closely linear over the operating range.

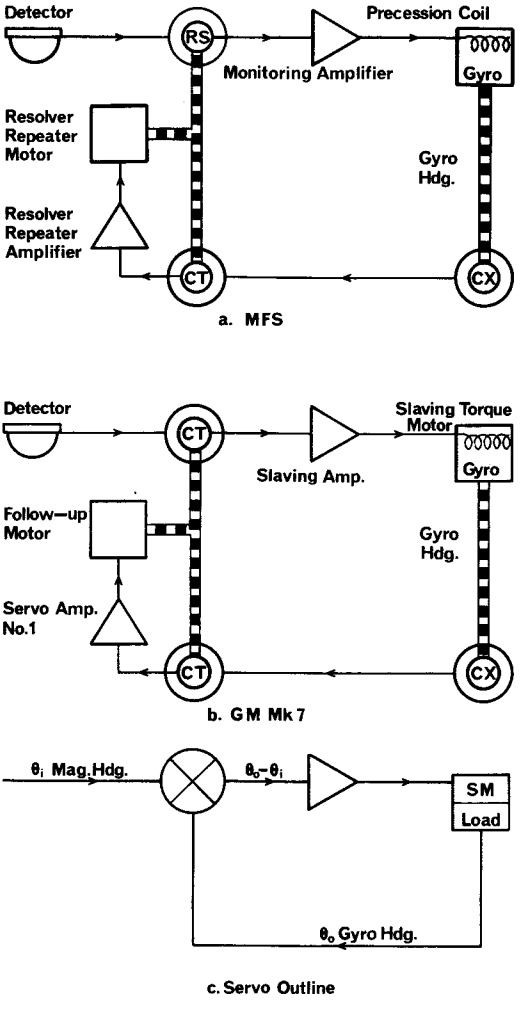


Fig 8 Gyro-Magnetic Compass Servo Loop

APPLICATIONS OF THE SERVO PRINCIPLE

Introduction

31. The servo principle can be identified in many navigation equipments. Velodynes are used extensively to drive latitude and longitude counters, and control transmission systems are used to provide remote indication of heading. There are many more examples; three are discussed below.

Gyro-Magnetic Compass

32. Fig 8a is a diagram of a Military Flight System Master Compass Loop, and Fig 8b is the same part of the GM Mk 7 compass. Fig 8c shows the servo action of both in block form. The precession coils replace the servomotor, and the gyro spin axis is the load. Magnetic heading, θ_i, is applied to the resolver synchro together with gyro heading. Gyro heading is the load position θ_o, and the resolver synchro the error detector. The error in gyro heading actuates the system.

33. The compass is therefore essentially a servomechanism, and the fact that a second servo is used to provide the load position feedback is a matter of design convenience. Indeed the G4F compass uses a direct mechanical link for the feedback. Whatever the feedback method, the servo principle

can be identified in all gyro-magnetic compasses.

Accelerometers

34. The accelerometers used in inertial navigation systems are often small servomechanisms. The arrangement of Fig 9 is typical. A ring of mass M is magnetized across its thickness, and mounted on a rod fixed to the aircraft. By careful choice of materials and clearances the ring can slide along the rod freely, so that any acceleration of the rod along the sensitive direction will displace it. Such a displacement is sensed by an E and I inductive pick-off, the I part of which is fixed to the mass. An a.c. excited coil on the centre limb of the E part sets up an alternating flux which induces unbalanced outputs from windings on the outer limbs when the I part is so displaced. An output is provided indicating the magnitude and direction of the displacement. For detail of the action the reader should compare the E and I with the C and Y which is similar in principle.

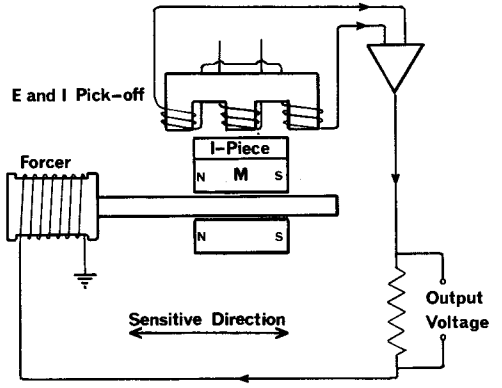


Fig 9 Typical Accelerometer

35. The error, or displacement, signal is amplified and applied to energize a solenoid, the forcer. The forcer acts on the mass to prevent further displacement, *ie* to accelerate it at the same rate as the rod. If the forcer is applying a force F to do this then, since $F=Ma$, F is a measure of the acceleration, a. The force applied is proportional to the current flow from the amplifier, so the current also is a measure of the acceleration, and a resistor in the circuit will provide a voltage output representing acceleration.

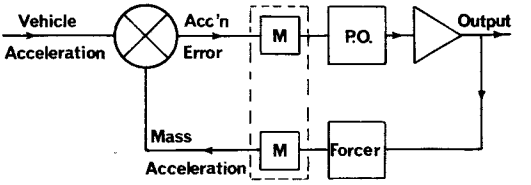


Fig 10 Accelerometer Servo Action

36. The block diagram in Fig 10 is typical of many which could be drawn to illustrate the servo action. Notice that the fundamental error is in acceleration. The mass acts as a transducer transforming the acceleration difference into the more tangible displacement form. The forcer coil also acts on this transducer to transform the voltage back into an acceleration. The error detector symbol in this diagram does not indicate a component; the difference in acceleration exists as a physical fact, but is not readily measured without the transformation.

Computer Applications

37. Wide use is made in analogue computers of the servo principle. One example is illustrated in Fig 11 where a voltage V_1 is to be divided by another V_2. V_1 is fed to the

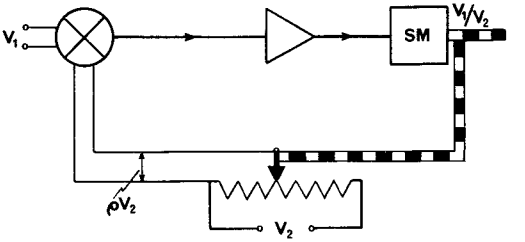


Fig 11 Division by Servomechanism

error detector directly, but V_2 is applied across a potentiometer so that only a proportion of it, say ρV_2 dictated by the wiper

position, is used. The error detector applies the difference $V_2-\rho V_2$ to the amplifier and the servomotor drives the wiper until $V_1=\rho V_2$ when the error signal is zero. Since ρ is proportional to the angle turned by the motor, and also ρ now equals $\frac{V_1}{V_2}$, the motor movement represents the quotient required.

38. In this case one of the inputs, V_2, is used as an element of the feedback, but the loop is easily identified.

PERFORMANCE OF SERVOMECHANISMS

Introduction

39. The descriptions given in the preceding paragraphs of servo action are rather superficial, and are on occasion ambiguously termed in order to avoid difficulty. It is intended, however, to discuss some of the more sophisticated members of the family and before this can be done the behaviour of the simple servo must be studied in greater detail.

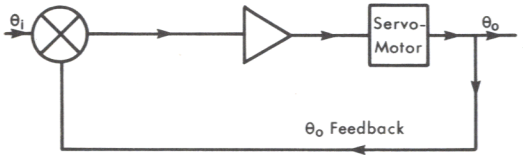


Fig 12 Simple Servomechanism

40. The servo illustrated in Fig 12 will be chosen as the model. The discussion which follows applies equally to the position servo and the speed servo, so that θ_i and θ_o may represent positions or speeds. For simplicity Fig 12 will be taken to be a position servo.

Response

41. The response of a servo is the pattern of behaviour of the load when a change is made to the input condition. It has so far been assumed that if the input moves to θ_i the load will simply follow, its response being a reproduction of the input movement. The paragraphs which follow will show that matters are not as simple as this.

42. Two important factors affecting response are the form which the input change takes, and the various restraints, friction etc., which act on the output. These are now considered in turn. Two types of input change will be covered, one when the input suddenly changes to a new position, the other when it suddenly moves at a constant speed. The first is known as a step input, the second a ramp input, the names deriving from the curves of input against time shown in Fig 13. Both are discussed without considering restraints in the first instance.

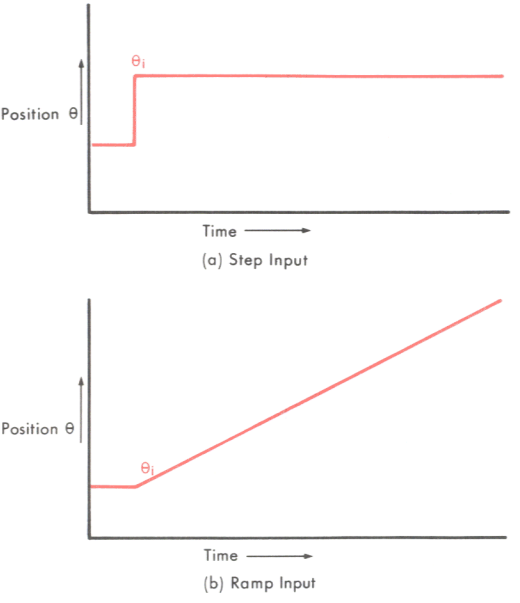


Fig 13 Types of Input

Step Input—No Friction

43. For this discussion we will assume that the input and output were aligned at θ_o, until the input suddenly changed to θ_i. An error signal proportional to $\theta_o-\theta_i$ appears at the amplifier input and the motor is energized to null the error.

44. One important point must now be emphasized. The torque delivered by the motor to the load is directly proportional to the error; it acts only on the inertia of the load which therefore accelerates at a rate proportional to the error. As the error reduces so the acceleration reduces, until it reaches zero with zero error.

45. But this is not a satisfactory state of affairs, for the load acceleration is in one sense only and that to increase its velocity. Saying that the acceleration is zero at zero error simply means that the load has reached a steady speed when we require it to be stationary. Further, since there is nothing to stop it, it keeps moving past the required position.

46. The error signal produced, and, therefore, the torque applied to the load, now reverse in sense to slow down the load. Since, however, the components operate symmetrically about the null, the pattern of deceleration is a mirror image of the original acceleration. The load stops when it has overshot by the initial error, and from there the performance is repeated. The resulting load oscillation about the demanded position is illustrated graphically in Fig 14.

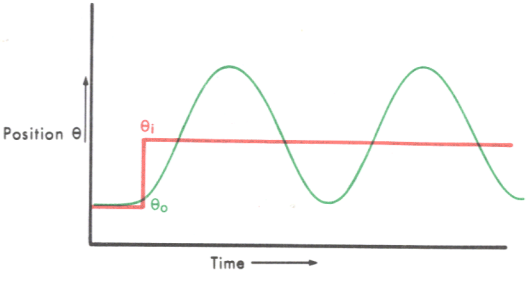


Fig 14 Oscillating Response to a Step Input

Ramp Input—No Friction

47. The description of the response can be followed in Fig 15. In the early stages of the ramp, while the error signal is small, the load accelerates slowly and lags behind the input. The error signal grows as the lag increases, building up the acceleration. Eventually the load speed equals the input speed but since a substantial position error exists it continues to accelerate. When its speed exceeds that of the input the position error starts to decrease; the acceleration reduces and the load reaches a constant speed at zero position error with no error signal. The load speed, however, exceeds the input speed and an overshoot results. That the outcome is a continuous oscillation can be easily imagined from this point.

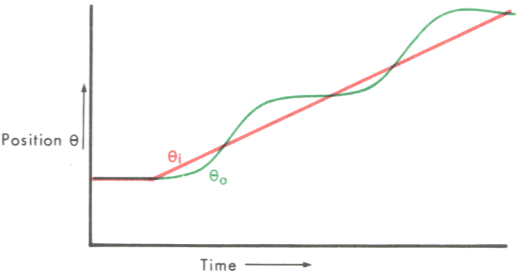


Fig 15 Oscillating Response to a Ramp Input

Effect of Restraints

48. The oscillatory responses are obviously not desirable, and luckily, restraints on the load have a stabilizing effect. Various inherent factors act to oppose the load movement; they include static friction, kinetic friction, eddy currents, air resistance, viscous lubricants and many others. Lumping them all together for the moment the general effect is to reduce the amplitude of each successive swing until gradually the output becomes steady. The oscillations are known as *transients* and they are effective during the *transient response period,* or *settling time.* Once the output has settled it has reached the *steady state.*

49. While restraints are beneficial in stabilizing, or damping, the response, they do have certain detrimental effects. One of these is that power is wasted, another is the introduction of error in the steady state.

Steady State Errors

50. Examination of the various restraints present would show that their effect is in part due to a small constant magnitude force known as *coulomb friction* and in part to *viscous friction* which increases with speed.

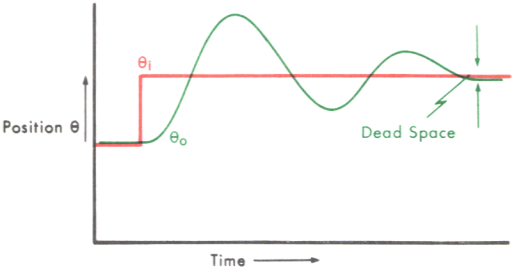


Fig 16 Response with Coulomb Friction to Step Input

51. The resistance due to coulomb friction tends to degrade the sensitivity of a servo, for a torque which overcomes it must be generated before any movement of the load takes place. To provide this torque the load error must reach some finite size, and any errors less than this will not be corrected. Fig 16 shows the effect of coulomb friction on the response to a step input. The load comes to rest somewhere within a band of error, known as the *dead space,* the width of which depends on the amount of coulomb friction. For most modern servos the coulomb friction is very small, and its effect is often neglected.

52. Viscous friction does not produce a dead space in the step input case since it has no value when the speed is zero. It does however produce a similar effect when the ramp input is considered. In the steady state the load is moving with constant speed; it is therefore being resisted by viscous friction. An error signal must be produced to overcome this, therefore an error must exist. The response is illustrated in Fig 17, and the error necessary to overcome the friction is known as *velocity lag*. Coulomb friction may be considered small compared with viscous friction during a ramp input, but, of course, it also contributes to this error. However, the greater part is due to viscous friction, and since this increases with speed the error is generally reckoned to vary directly with speed.

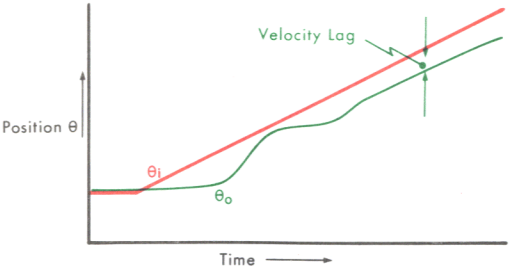


Fig 17 Response with Viscous Friction to Ramp Input

Summary

53. The simple servo oscillates in response to either a step or ramp input. Friction damps the oscillation, but leads to dead space and velocity lag.

IMPROVEMENT OF TRANSIENT RESPONSE

Introduction

54. For many applications the simple servo using its inherent friction for damping is perfectly adequate. This is usually the case for small position servos, but when large loads are involved the transient response is unsatisfactory. Time and energy are wasted during this period, and bearing wear is increased. It is evidently desirable to reduce the number of oscillations, and also the response time. Two methods commonly employed are described.

Viscous Damping

55. This method is simply a controlled increase of the inherent viscous damping to achieve the required response. One device in use is the eddy current damper shown in Fig 18.

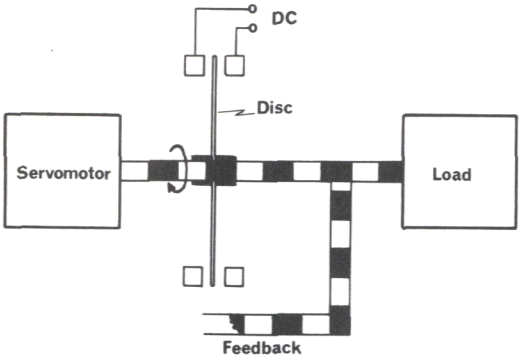


Fig 18 Eddy Current Damper

56. This simple device consists of a thin disc of metal with high electrical conductivity (usually aluminium) which is attached to the output shaft. It spins between the poles of electromagnets mounted round its periphery. Eddy currents are induced of magnitude proportional to the field strength and to the disc velocity. These eddy currents set up magnetic fields which act against the inducing fields and forces opposing the disc rotation are created. These forces are closely proportional to the disc velocity, and therefore provide parallels to the inherent viscous forces. They can be controlled by adjusting the current flow to the electromagnets.

57. Varying degrees of damping can be applied. Fig 19 shows some of the stages, coulomb friction being ignored for simplicity. Using only inherent friction *light damping* is achieved. Too much extra viscous friction will produce a very sluggish response and the system is *heavily damped.* The degree of damping which just prevents any overshoot is known as *critical damping.* Slightly less damping than this, to allow one small overshoot, is *optimum damping* which gives the smallest settling time. Most designs are aimed at this condition.

58. The effect on the transients for a ramp input can be similarly adjusted to produce optimum damping. A snag arises, however, for any increase in viscous friction also

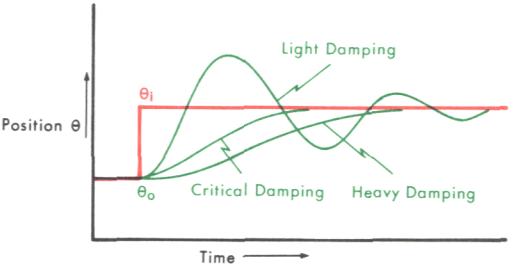


Fig 19 Degrees of Damping – Step Input

increases the velocity lag. Thus to remove the transient oscillations completely a considerable velocity lag must be expected. Fig 20 illustrates the response for two degrees of damping for a ramp input.

59. The response achieved by additional viscous damping can be made adequate, but it has the great disadvantage of wasting energy. The second method attacks this problem.

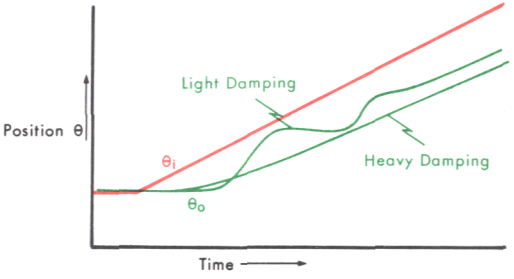


Fig 20 Degrees of Damping – Ramp Input

Velocity Feedback Damping

60. Viscous damping acts by absorbing motor torque. It does so by applying a force at the motor output proportional to the output speed. Examining these statements we see that the damping effect is produced by reducing the motor torque in the desired proportion, while the friction force applied to do so is the cause of energy waste. If, therefore, the motor torque can be reduced in the same proportion by some means other than an opposing force the damping action will be retained, but power no longer

wasted. Velocity feedback damping acts in this way.

61. Motor torque can be lowered by cutting off part of the amplifier output, and a simple way of doing this is to cut down the error signal. For effective damping the reduction must be on the lines indicated by viscous friction, that is it must be proportional to the output speed. We therefore feed back a voltage proportional to the load velocity and apply it in opposition to the error signal at the amplifier input. The feedback voltage is provided by a tachogenerator on the output shaft. The arrangement is shown in Fig 21. Since a voltage with negligible current is required the additional output load can be neglected.

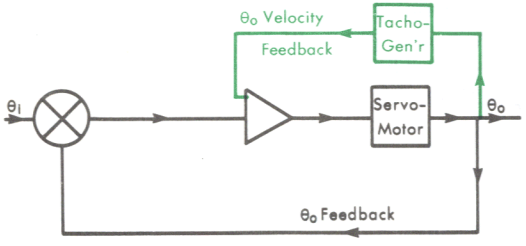


Fig 21 Velocity Feedback

62. Varying degrees of damping can be achieved by adjustment of the feedback and much greater precision is possible than with viscous friction. Once again optimum damping is sought.

63. Velocity feedback increases velocity lag just as did the viscous friction method, but for a different physical reason. In this case the steady state velocity of the load imposes a signal on the amplifier input which must be cancelled in some way if the steady velocity is to be maintained. The cancellation can only be made by an equal error signal, which means that an error must exist.

Summary

64. Transient response can be improved in two ways, by applying extra viscous friction or by velocity feedback. Both increase velocity lag in the response to ramp inputs, but of the two velocity feedback is to be preferred since power is not wasted.

REFINED SERVOMECHANISMS

Introduction

65. While most of the servos used in aircraft systems are damped by inherent friction, extra viscous friction, or velocity feedback some applications require greater accuracy in the steady state. Methods have therefore been developed which improve sensitivity by reducing dead space and velocity lag. Three are discussed, the first two aimed at velocity lag, and the third at steady state errors in general. In all cases a small sacrifice in transient response is made and the design operating environment must be stringently observed if the system is not to revert to an oscillatory state.

Error-Rate Damping

66. Velocity lag can be lessened by introducing a new signal at the amplifier input which cancels the velocity feedback when the input and output velocities are equal. This leaves the position error signal free to reduce the lag.

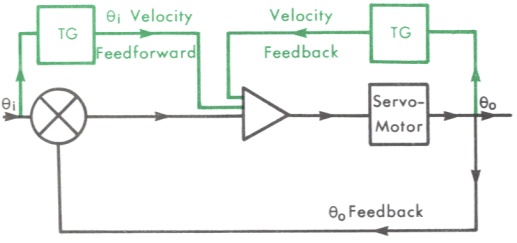


Fig 22 Feedforward of Input Velocity

67. One method is to fit a tachogenerator on the input shaft, as shown in Fig 22. During a ramp input a steady state is eventually reached in which the tachogenerators are applying equal and opposite voltages to the amplifier. If any velocity lag exists then the position error signal torques the motor to reduce it. The system has no effect on velocity lag due to inherent friction.

68. An improvement of this method arises when it is realized that the tachogenerators are together, at any time, feeding into the amplifier the difference between the input and output shaft velocities. But this difference is simply the error rate, for the error is itself the difference between the shaft positions. The same effect is therefore achieved by dispensing with both tachogenerators and simply applying the rate of change of the error signal to the amplifier. Fig 23 shows a differentiator, D, feeding this rate in. The circuit and action are entirely equivalent to that of Fig 22.

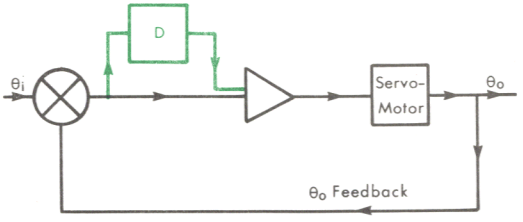


Fig 23 Feedforward of Error Rate

Transient Velocity Damping

69. This type of damping is developed from velocity feedback. The guiding principle is that velocity feedback makes no useful contribution to response during the steady state and is better cut out.

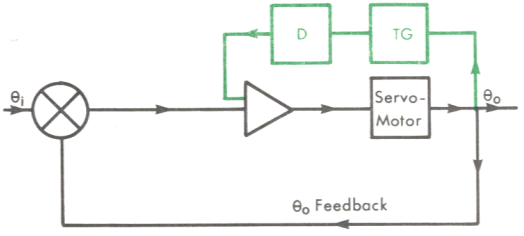


Fig 24 Transient Velocity Damping

70. A circuit which will do this is illustrated in Fig 24. It consists of the normal loops, but with a differentiator in the velocity feedback line. Thus only the derivative of the load velocity reaches the amplifier. The result is that damping is effective only during the transient response period, that is when a rate of change of load velocity exists. Once the steady state is reached there is no further rate of change, the derivative is zero, and feedback ceases. Velocity lag is therefore once again reduced. This type of damping is also known as acceleration feedback.

Integral Control

71. The methods so far described reduce velocity lag, but have no effect on lag and dead space caused by inherent friction. A common method of dealing with these residual steady state errors is known as Integral Control. The arrangement is illustrated in Fig 25, used there in conjunction with feedforward of error rate.

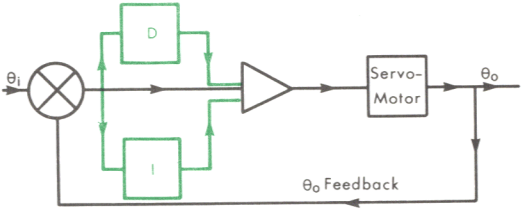


Fig 25 Integral Control

72. Differentiator D operates in the normal way as described in para 68, but the situation is modified by the integrator, I, which feeds the time integral of the error signal into the amplifier. The effect on the transient response is negligible, but as the error settles to its steady state so its integral increases, superimposing on the amplifier a signal which provides additional torque at the load. The load is moved by this torque towards the correct position. Adjustment of the proportion of the integrator output can be made to ensure, when the error signal is zero, that the subsequent constant integrator output is just sufficient to counter the inherent friction. Thus velocity lag is zero. For a step input the dead space error signal is integrated until large enough to zero the error, and adjustment of the damping differentiator output ensures stability.

CONCLUSION

Summary

73. The simple servomechanism uses an error detector to actuate an amplifier and motor until the load conditions conform to the input demand.

74. This system is oscillatory and although inherent friction damps the vibration certain additional elements are necessary to ensure minimum transients. The elements used may be additional viscous friction or velocity feedback. Viscous damping is usually sufficient if the servo is small, but where power waste is intolerable velocity feedback is preferred.

75. When transient response is improved by these methods an increase of velocity lag is experienced. To counter this either error-rate or transient velocity damping is used.

76. Finally, the system can be rid of most residual errors by integral control.

PART 3

SECTION 2

COMPUTING PRINCIPLES AND CIRCUITS

Chapter

1 **Analogue Computers**

2 **Digital Computers**

CHAPTER 1

ANALOGUE COMPUTERS

CONTENTS

	Paras.
Introduction	1–3
Types of Computers	4–5
Methods of Computation	6–9
Comparison of Methods	10–16
Aircraft Analogue Computers	17–19
EXAMPLES OF COMPUTATION	
Introduction	20
Case 1, $y=f_1(x)$	21–23
Case 2, $y=f_2(a, x)$	24
Case 3, $y=f_3(a, b, x)$	25–29
Alternative Methods	30
TYPICAL AIRBORNE ANALOGUE COMPUTER	
Introduction	31
The Problem	32–34
Ball and Disc Resolver	35–36
Sphere Resolver	37–38
Differential Gear	39
Stage 1	40
Stage 2	41–42
Stage 3	43–44
Stage 4	45
Complete Computer	46–47
Other Examples of Computers	48
References	49

Introduction

1. The effectiveness of a navigator is limited by the accuracy and speed with which he can solve various numerical problems. Sometimes these problems are simple, but more often they are not and require a number of arithmetical processes, each giving an opportunity for error. Often, too, the human computer is quite inadequate; the calculation of bombing angle, for example, requires continuous evaluations in a complicated equation concurrently with the changing variables.

2. Accuracy and speed, however, are not dependent only on the difficulty of the calculation. Errors occur increasingly even in the simplest of computations as repetition tires the human computer.

3. To assist the navigator in all these cases he is supplied with a number of calculating instruments, varying in complexity from the simple slide rule to large electronic machines. In this section automatic computers will be considered, and the reader is referred to Part 1, Sect. 1 of this volume for descriptions of manual computers.

Types of Computers

4. An examination of automatic computers would show that they calculate in one or other of two quite distinct ways, allowing a useful classification. In one type, the digital computer, calculations are performed by essentially arithmetical methods. Numbers are operated on directly, much as they are in cash registers or desk calculating machines. The second type, known as the analogue computer, solves by creating analogies of the problem and finding solutions through them. The numerical values in the calculations are represented by physical quantities such as voltages, lengths, angles, *etc*. These are then combined by the physical laws relating them to simulate the computation.

5. This chapter is concerned with the elements of those analogue computers commonly used in navigation systems, while Chapter 2 deals with digital computers. It is convenient here, however, to discuss further the general features of both, and since these are more often directly the result of method of calculation rather than hardware this can be done without presuming particular equipments.

Methods of Computation

6. To amplify the remarks on method let us take two simple examples:—

(*a*) Evaluate y in

$$y=ax^2+bx+c \qquad (1)$$

for various values of x.

(*b*) Calculate the distance travelled by a car during given periods of time when the speed v varies continuously. In other words, integrate v with respect to time.

7. **Arithmetical Methods.** Both problems can be solved using purely arithmetical processes as follows:—

(*a*) By long multiplication work out x^2, then ax^2 and bx and add ax^2, bx and c together. Repeat for each value of x.

(*b*) Calculate consecutive small elements of the distance travelled by taking spot values of v at small intervals δt of time apart. The elements of distance δs are given by $v.\delta t$ and the integral over any given period is given by summing the appropriate elements. For close approximations it is necessary to choose a small δt.

8. **Alternative Methods.** There are a number of alternative methods available which avoid the rather tedious arithmetic of paragraph 7. Choosing just one for each problem we can have:—

(*a*) Using simple values of x construct a graph of (1), with y along one axis and x along the other. Enter with the required values of x and extract y.

(*b*) Clearly the distance travelled can be found by counting the number of revolutions of the wheel, *i.e.* by noting the total angle through which it turns. We may remove the hazards involved by simulating the rotation of the wheel on a shaft and conveniently presenting its total angle on veeder counters. The counters may be calibrated to read distance rather than angle.

9. **General Remarks.** The arithmetical solutions are those used by the digital computer, while the alternatives are akin to those often used by analogue machines. (There are so many analogue methods that it is impossible to be more specific.) It may be interesting to note that until the advent of the electronic digital computer the arithmetical methods were excessively tedious, and therefore avoided as often as possible. The analogue computer is the descendant of methods of calculation such as geometrical constructions, graphs, nomograms, and simple simulating devices (such as the mileage indicator at paragraph 8(*b*)), all devised to avoid arithmetic. The general features of the computers can now be found by comparing the methods of paragraphs 7 and 8.

Comparison of Methods

10. **Versatility.** It is evident that the arithmetical approach is much more versatile. Provided that the human computer can add, subtract, multiply and divide, he can solve both problems. Further he can extend his methods to calculate a bank balance or the area of a room, the only difference being a rearrangement of the individual processes of addition, *etc.*, to suit the occasion. The digital computer is capable of all arithmetical processes and it requires only to be told the sequence of calculation, *i.e.* to be programmed, to complete the solution. On the other hand each analogue method is applicable only to a particular type of problem. The graph produced to solve (1) has no bearing whatsoever on the bank balance or on the area of the room. In the same way each analogue computer is best regarded as a single purpose machine, solving a single equation and useless for any other. There are exceptions to this rule, but they are never found in airborne analogue computers.

11. **Accuracy.** In the arithmetical method of solution of (1) the error in y is determined exactly by any errors in the values of a, b, c, and x used, since the method of calculation itself admits of no error. Indeed we can say that the arithmetical method is absolutely precise. This precision can, however, be blunted somewhat by rounding. For example if a result, exact at ten decimal places, is rounded to five for convenience, it clearly becomes less precise. For the digital computer the same remarks apply. In theory the method is absolutely precise, but in practice there is a limit to its precision, dictated by the number of digits with which it can work. By increasing the number of digits it is possible to produce almost any degree of precision. The analogue computer, in contrast, has many sources of error. Looking again at the graphical solution of (1) it can be seen that however well a, b, c and x are specified there is a limit to the accuracy of entering and interpreting the graph. Evidently the scales along the axes, the thickness of the lines, and the care taken in the original drawing all affect the errors in y. Similar problems afflict the analogue computer, where the scale of representation, friction and inertia in moving parts and manufacturing tolerances all contribute to the final error. In principle there are no limits to the precision of the digital computer beyond those imposed by time, bulk and cost, while it is unusual to find an analogue computer accurate to better than 0.1%.

12. **Speed of Computation.** Just as the graphical solution of equation (1) is faster than the arithmetical one, so the analogue is quicker than the digital computer. Indeed the analogue computer often gives its results concurrently with the input, in the sense that entry with x on the curve immediately gives y. The relative slowness of the digital computer is not, however, always significant; a modern electronic digital computer would evaluate expression (1) in about one thousandth of a second.

13. **Calculus Operations.** The analogue computer deals very easily and directly with integration with respect to time. The example given of a car-mileage indicator is a simple case, but in fact most airborne computer applications are just as easy to understand. In other calculus problems, however, the analogue computer is not so useful; integration with respect to a variable other than time, and differentiation tend to be clumsy operations. On the other hand, the digital computer can cope with almost any type of calculus process, using one of the many numerical methods of integration or differentiation available. This means, as implied in paragraph 7(*b*), that it calculates the result for one set of conditions, then for the next, and so on. It has some disadvantage when dealing with the variable time, for if the calculations take longer than the time interval chosen, then the results quickly lag the events which they represent. In other words the digital computer may not operate in real time unless special precautions are taken; the analogue computer readily operates in real time and it is largely because of this that it tends to dominate the airborne computer field.

14. **Equipment Bulk.** It has already been indicated that a single digital computer can do many different types of calculation, whereas the analogue computer is single purpose. If many different calculations have to be performed then it is possible, by suitable programming, to use a single digital computer for them all, and this may effect a considerable weight and volume saving over the various analogue computers which would be needed. On the other hand each analogue computer would probably be smaller than the single digital computer, and if there are only a few stereotyped calculations then the analogue computer's total bulk may be less. Further, there is a limit to the number of calculations, each taken in turn, that a single digital computer can cope with in the time scale allowed and for very complex navigation and weapons systems two may be required. There can be no general answer, and the comparison can be carried out fairly only in the light of the aircraft's operational role.

15. **Form of Input and Output.** The digital computer deals in discrete values; its inputs and outputs are individual numbers, which lend themselves to simple presentation on counters or by printing. The analogue computer works with continuous quantities and its results are, therefore, more suitable for presentation on a scale of some sort. For example, the slide rule is a simple manual analogue computer in which the values are represented on a continuous scale. Many analogue computers however, have a counter output, using the mechanism which would normally give a scale pointer deflection to turn the counters. Such a counter output is sometimes called a digital readout, but the phrase in no way implies that the computation method is digital.

16. **Summary.** The digital computer is more accurate, and versatile, and may be less bulky when many calculations are to be made. The analogue computer is, however, fast, works in real time, and is more compact (and also cheaper) when the calculations are simple and repetitive. Most aircraft navigation systems require fast, continuous results to a few fairly simple and repetitive problems. Thus the versatility of the digital computer is not often needed. Further, the accuracy of the inputs to many navigation computers is rarely very high so that the precision of the digital computer is seldom a significant factor. In general, digital computers are considered as competitors in the airborne field only when fully automatic and integrated navigation and weapons systems are contemplated.

Aircraft Analogue Computers

17. Aircraft analogue computers can be mechanical, electrical or, more commonly, electro-mechanical. The numerical values may be represented by angles, lengths, voltages, currents, or by any other quantity which is convenient. The analogue may change several times in the same computer if a change allows a simpler or more accurate form of calculation.

18. All analogue computers can be described as devices which simulate a mathematical process. Many of the devices are not peculiar to computers; they are often familiar mechanical and electrical networks which in other contexts would be used differently. An ordinary resistor, for example, in theory provides a method of simulating division or multiplication. The current, I, voltage, V, and resistance, R, are connected by the law $V=IR$. If two quantities a and b are to be multiplied we may represent a by I and b by R. The product ab $(=c)$ is given by V. To divide c by b we simply measure the current.

19. A great many devices exist and new ones appear with each new instrument. It is not possible, therefore, to provide an exhaustive catalogue, and in this publication explanations will be given with each instrument in its own chapter. In the paragraphs that follow, a few typical examples are examined. Descriptions of other common mechanisms and circuits are to be found in A.P. 3302, Part 1a, Book 3, and in A.P. 3274, Volume 5.

EXAMPLES OF COMPUTATION

Introduction

20. The evaluation of $y=ax^2+bx+c$ will be described in the first place when a, b, and c are constants, then when a as well as x is variable, and finally with variables a, b and x. More concise statements of the problem may be made by writing $y=f_1(x)$, *i.e.* y is a function of one variable, x; then $y=f_2(a, x)$ when y is a function of two variables; and finally $y=f_3(a, b, x)$.

Case 1, $y=f_1(x)$

21. We will consider two simple methods of solution. One is purely mechanical, the other electro-mechanical. Both are commonly used, and the choice depends on the form of analogue in which the result is required. Both methods have similar limitations. They can be used only for single valued and continuous functions. Such functions as $y=\sqrt{x}+3x^2$ and $y=\frac{1}{1-x^2}$ would cause trouble. For the first function only one root would be evaluated, and for the second the discontinuity at $x=1$ or -1 would render the methods impracticable around those values. The accuracy to be expected is about 1%.

22. **Mechanical Solution.** The independent variable (or input) x, is represented by the angle of rotation of a shaft which turns a plane cam, A in Fig. 1. Spring-loaded against the cam is an arm B, known as the follower, pivoted at C. As A turns through x, so B is turned about C by an angle which evidently depends on the cam shape and the follower size. The cam can be cut so that the angle moved by B (the output) represents y. The device can be regarded as a mechanized

graph, and has similar limitations and accuracy. It has, however, further limitations which may be of passing interest. The steepness of the cam slope must not be so great as to lock the follower; the spring must be powerful enough to drive the succeeding computing elements; y must be fairly small compared to x to avoid large leverages. When these limitations are added to those of paragraph 21 it becomes clear that the plane cam is not necessarily the best method of solution for all problems of the type $y=f(x)$. Despite this it has many applications and is found in most navigation computers.

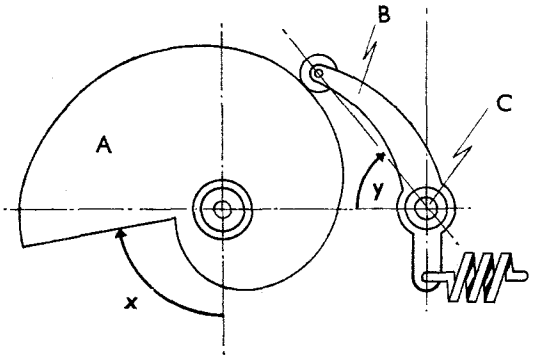


Fig. 1. Plane Cam

23. **Electro-Mechanical Solution.** The simple potentiometer can often be arranged to provide a voltage proportional to y. Resistance wire is wound on to a shaped card as shown in Fig. 2.

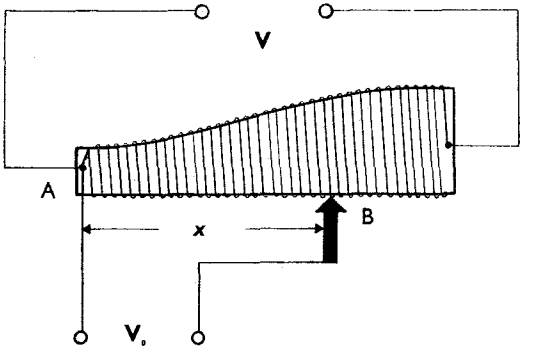


Fig. 2. Shaped Card Potentiometer—Principle

A constant voltage V is applied across the resistor so formed, and a wiper B is moved along the wire at the straight edge. The distance moved by the wiper from the point A is made proportional to the input x. The voltage, Vo, between the wiper and A depends on x, on the voltage V, and on the card shape. The shape is made to conform to the functional relationship so that Vo is proportional to y. The card is often bent into a circle as in Fig. 3 and x applied as an angular analogue. A common application of the shaped card potentiometer is in the evaluation of the circular functions cos x and sin x.

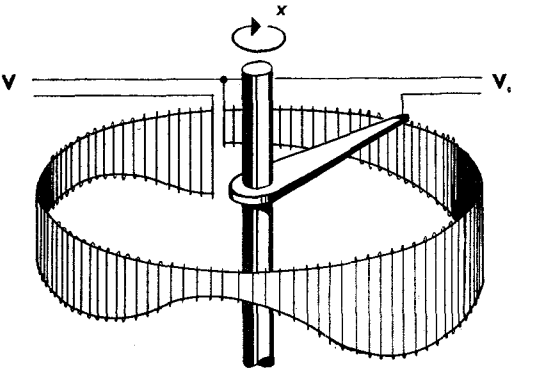


Fig. 3.
Shaped Card Potentiometer—Practical Configuration

Case 2, $y = f_2(a, x)$

24. Direct evaluation may be obtained in many cases by using a 3-dimensional cam (Fig. 4). One input, a, moves the cam laterally and the other, x, rotates it about the axis zz. The output y is picked off by the follower moving vertically. The action may be seen more clearly from Fig. 5. For each value of a the solution is simply that of an equation of the form $y=f(x)$, thus each segment is a plane cam evaluating the simpler expression. The complete 3-D cam is formed by joining all the plane cams within the range of a required. The limitations are the same as for the plane cam.

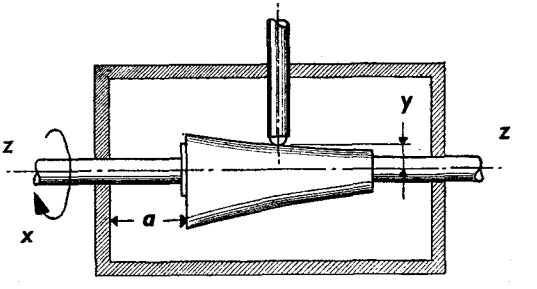


Fig. 4. 3-Dimensional Cam

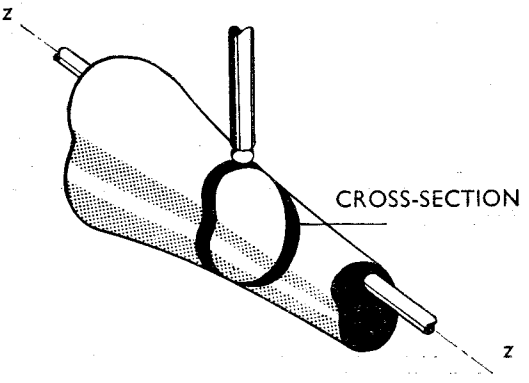


Fig. 5. Segment of a 3-D Cam

Case 3, $y = f_3(a, b, x)$

25. The solutions given in paragraphs 22 and 23 amount to the modelling of a function of x in two dimensions. When dealing with $y=f_2(a, x)$ the model can still be constructed, this time in three dimensions. When $y=f_3(a, b, x)$, however, direct modelling would require the physically impossible fourth dimension, and we therefore reduce the problem to a number of simpler two- or three-dimensional ones each of which gives part of the result and can be simply combined to give y. We will discuss a typical solution in which the parts ax^2 and bx are calculated and then added to c to give y. In this example x will again be represented by an angle of rotation of a shaft, but the a, b, and c analogues will be the amplitudes of alternating voltages. The method of solution could apply equally well to the evaluation of $y=f_1(x)$ or $y=f_2(a, x)$.

26. The quantity ax^2 will be found by multiplying a by x to give ax, then multiplying again by x. The multiplication ax can be achieved in principle by ordinary transformer action. Fig. 6 shows a transformer in which the turns ratio, $\frac{N_1}{N_2}$, can be varied by moving a brush A across the secondary winding. The output voltage E_2 is given by $\frac{E_1 N_2}{N_1}$. Thus if E_1 is the analogue of some quantity p, and N_2 (*i.e.* the distance moved along by the wiper) represents another, q, then E_2 is proportional to the product pq. A more compact design is achieved by using an auto-transformer, in which there is only one winding. Fig. 7 shows an auto-transformer producing the same output E_2 representing pq. In this context the auto-transformer is often called an inductance potentiometer or I-pot.

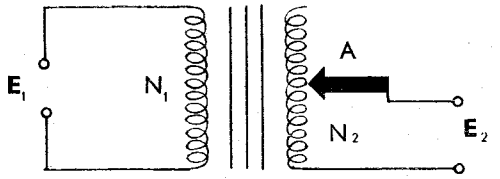


Fig. 6. Multiplying Transformer

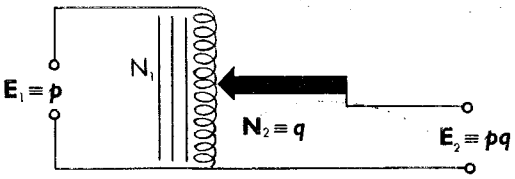


Fig. 7. Multiplying Auto-transformer

27. In practice the winding is formed as a toroid wound on a ring of high permeability material such as mumetal. This shape, Fig. 8a, minimizes leakage flux. It also allows the application of an angular analogue of q as an input. The analogue of p is still applied across the whole winding and the output pq obtained as before. Fig. 8b shows the diagram symbol usually used.

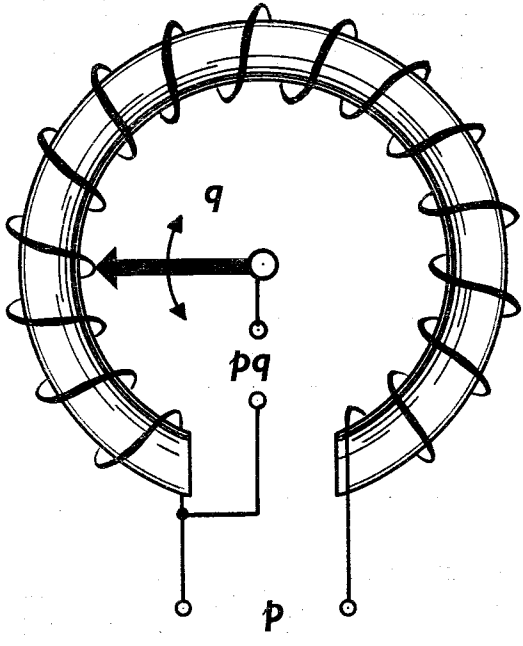


Fig. 8a. I-pot Configuration

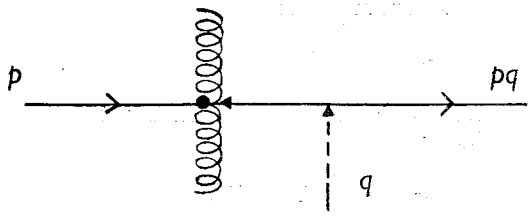


Fig. 8b. I-pot Diagram Symbol

28. The computation of $y=ax^2+bx+c$ can now be carried out using three I-pots to calculate the three products ax, ax^2 and bx. The products ax^2 and bx can then be added to the analogue of c by applying each to the primary winding of a 1:1 transformer. The secondary windings are connected in series as in Fig. 9, giving a voltage amplitude representing the sum.

Alternative Methods

30. The methods of solution described are typical, but many others are in common use. In particular, had the analogues been direct voltages completely different circuits would have been employed. However, few modern airborne analogue computers use d.c. analogues since the components tend to be heavier and larger than the equivalent a.c. parts. One notable d.c. analogue computer is the Navigation and Bombing Computer, N.B.C., and for information on its techniques the reader is referred to A.P. 2894.

TYPICAL AIRBORNE ANALOGUE COMPUTER

Introduction

31. The computations considered in the previous paragraphs were of a general nature. It is now proposed to examine a typical navigation computer in a little more detail than is usual for

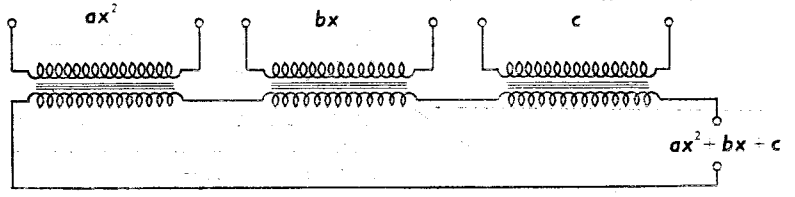


Fig. 9. Addition of Alternating Voltages

29. The complete computer is shown in Fig. 10 using the accepted symbols.

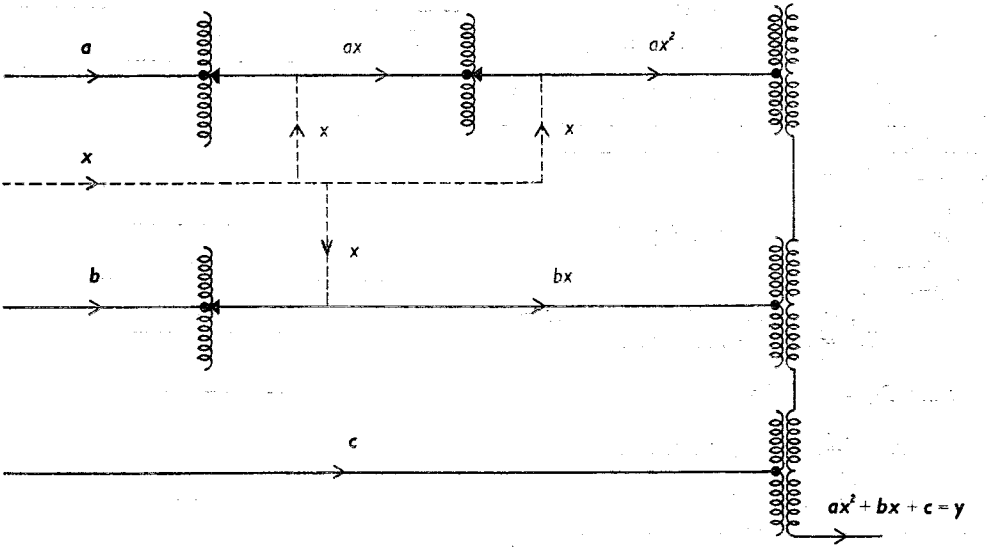


Fig. 10. Computation of $y=ax^2+bx+c$

this publication, to include mention of factors such as scale of representation.

The Problem

32. Consider the evaluation of:—

$$A=\frac{U}{R}\tan \lambda_1 \text{ radians/hour} \quad \ldots \quad (2)$$

where A=the rate of true meridian convergence
U=easting groundspeed in kts
λ_1=present latitude, in minutes
R=Earth radius in n.m.

from the inputs G (groundspeed) and T (true track). The analogue method of calculation will depend largely on the form of the inputs. Let us specify them as:—

(*a*) An angular velocity proportional to G where 60 radians/hr corresponds to 1 kt
$\therefore$ G kts is represented by 60 G radians/hr or G radians/min.

(*b*) A shaft angle of rotation equal to T.

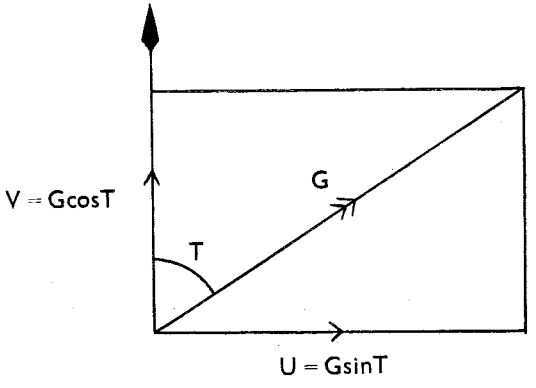


Fig. 11. Trigonometrical Relationships

33. A simple mathematical analysis is a necessary preliminary to show the connections between G and T and the equation (2) for A. This may be done as follows:—

Note that,
$U=G \sin T$ ... from Fig. 11
And also that,
$\lambda_1=\lambda_o+D_N$
where λ_o=start point latitude, in minutes
D_N=distance gone north in n.m.

Further, $D_N=\int_{t_o}^{t_1} V\,dt$

where V=northing groundspeed
t_o=time at λ_o
t_1=time at λ_1

Or, in words, that the distance gone north is the integral with respect to time of the velocity north in the time interval (t_1—t_o) hours.

By substitution therefore,

$$\lambda_1=\lambda_o+\int_{t_o}^{t_1} V\,dt$$

But $V=G \cos T$ ... from Fig. 11
So that

$$\lambda_1=\lambda_o+\int_{t_o}^{t_1} G \cos T\,dt$$

Since the Earth's radius is a constant we have now reduced all the variables to terms containing known quantities λ_o, R, G and T.

Substituting for U and λ_1 in (2)

$$A=\frac{G \sin T}{R}\tan\left\{\lambda_o+\int_{t_o}^{t_1} G \cos T\,dt\right\}$$

This is the expression which the computer must solve. It is not as fearsome as it seems, and in fact is solved very simply by stages.

34. **Computation Stages.** The computer will solve all stages simultaneously, but it is easier in the first instance to draw a diagram of the apparent flow of information and to list the operations at each stage separately. Thus following through on Fig. 12 we have:—

(*a*) *Stage* 1. Apply G and T to some mechanism or network which will provide G sin T and G cos T.

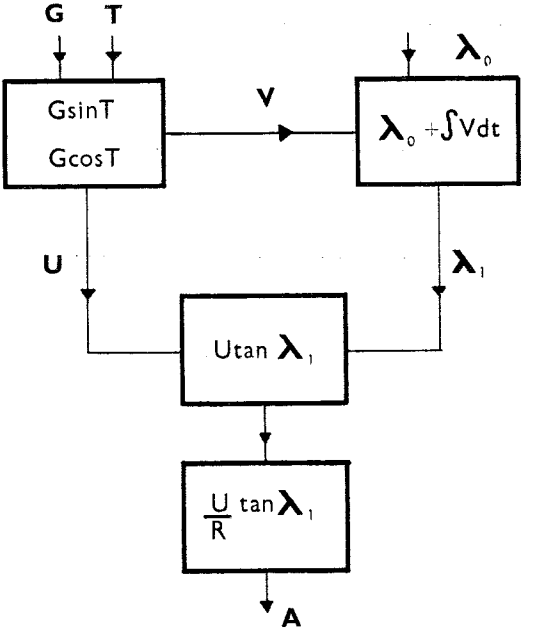


Fig. 12. Computation Stages

(*b*) *Stage* 2. Take the output G cos T (=V) and pass it through a further mechanism which will integrate it with respect to time. Add in λ_o at this point to give an output of λ_1.

(*c*) *Stage* 3. Feed in G sin T (=U) and λ_1 to a device which will give U tan λ_1.

(*d*) *Stage* 4. Divide by R to give A.

The computation therefore involves operations with trigonometrical ratios, integration, addition, and division. It is convenient here to discuss some of the elements which will be used, before considering the stage solutions.

Ball and Disc Resolver

35. A pair of ball and disc resolvers, one giving sine, the other cosine functions is shown in Fig. 13. It consists of a gear P which is turned through some input angle θ. The movement drives two crank pins O and Q through θ from their datums. Over each pin is fitted a slotted rod constrained to move in the directions AA, so that a ball at the foot of each rod moves twice across its disc for each revolution of the crank-pin. The distance of the ball from the centre of the disc can be seen to be L sin θ for that on the diagram left and L cos θ for that on the right, where L is the crank radius.

36. A clearer view of the ball, disc and roller is shown in Fig. 14. If the disc is turned through one revolution, then a point on the ball moves through a distance 2 π L cos (or sin) θ, and in so doing imparts the same movement to the roller. Thus for one revolution of the disc the roller turns through $\frac{2\pi L \cos\theta}{2\pi r} = \frac{L}{r}\cos\theta$ (or $\frac{L}{r}\sin\theta$) where r is the roller radius. The outputs from the rollers are therefore $\frac{L}{r}\cos\theta$ and $\frac{L}{r}\sin\theta$ times the disc speeds.

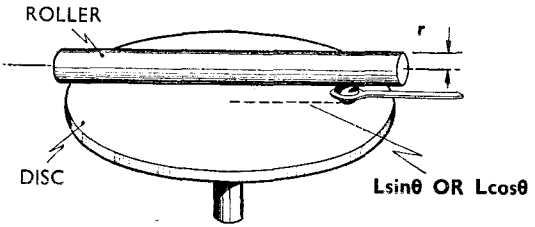


Fig. 14. Ball, Disc and Roller

Sphere Resolver

37. An alternative solution to the same problem is provided by the sphere resolver illustrated in Fig. 15. It consists of a hardened steel sphere supported by an input roller (1), two output rollers (2) and (3), and two spring-tensioned idling rollers which are not shown in the illustration. One input X, is transmitted through bevel gearing to rotate the sphere by means of roller (1), while the other input, θ, rotates a collar containing the shaft of roller (1). The two inputs are arranged concentrically, the X drive shaft passing through the θ drive gear, to the bevel gears. The output shafts are positioned mutually at right-angles, and the two idling rollers press the sphere against the input and output rollers to provide operating friction.

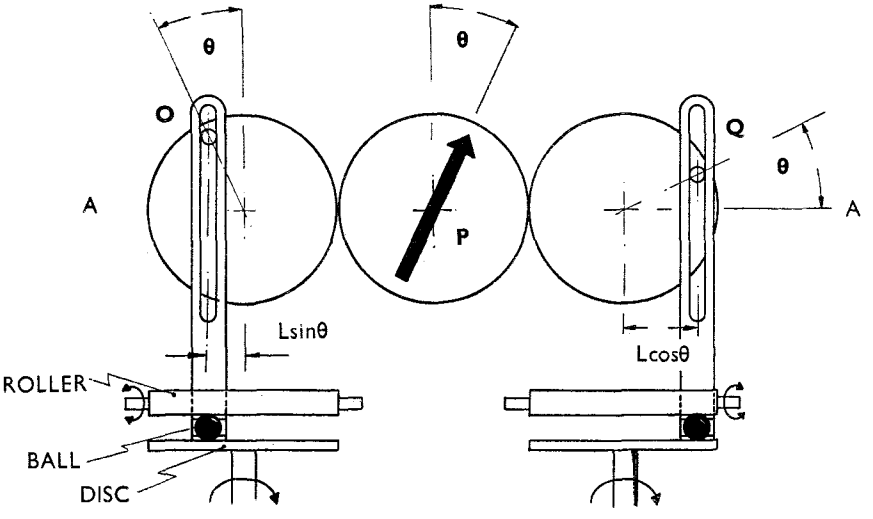


Fig. 13. Ball and Disc Resolvers

38. When the input angle θ is zero, the carriage is in the position shown, and the X drive is conveyed entirely to roller (2), the other roller remaining stationary. When the angle is not zero, the axis around which the sphere rotates is displaced, and the motion is imparted to both rollers (2) and (3) in the relationships X cos θ and X sin θ.

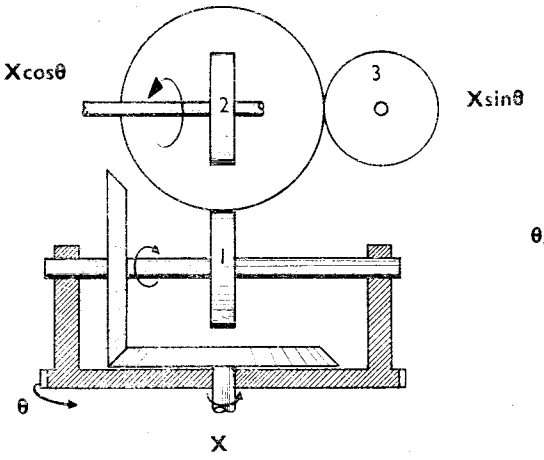


Fig. 15. Sphere Resolver

Differential Gear

39. A differential gear is a mechanical cogwheel device constructed so as to combine algebraically two angular shaft inputs. A typical gear is shown in Fig. 16. Shafts 1 and 2, whose rotations are to be added, carry bevel gears 3 and 4 which both mesh with idler gears 5. The idler gears are symmetrically disposed in a cage which is free to rotate round an axis collinear with the input shafts. Both idler gears thus remain continuously in mesh with the gears 3 and 4. The cage carries a ring gear 6 meshing with a bevel 7 on the output shaft with a step-up ratio of 2:1. If gears 1 and 2 are turned simultaneously in the same direction through equal angles, say θ, they will carry the gears 5 around with them, and the cage and 6 will turn through θ. Now hold 2 stationary and turn 1 through θ. Gears 5 will walk around 4 taking 6 with them through some angle, say ø. Now hold 1 stationary and turn 2 through θ. By symmetry the gear 6 will again turn through ø. Thus we have, when θ is applied independently to each input a total rotation of 6 equal to 2 ø, and when θ is applied simultaneously to each a total rotation of θ. Hence $2\,ø=\theta$ and $ø=\frac{\theta}{2}$, Now if different rotations are applied to 1 and 2, say θ_1 and θ_2, then 6 will turn through $\frac{\theta_1}{2}+\frac{\theta_2}{2}$ and the output, 7, rotates by $\theta_1+\theta_2$ owing to the 2:1 step-up. Notice that the output is the algebraic sum of the inputs. If they are in opposite senses then the output is their difference.

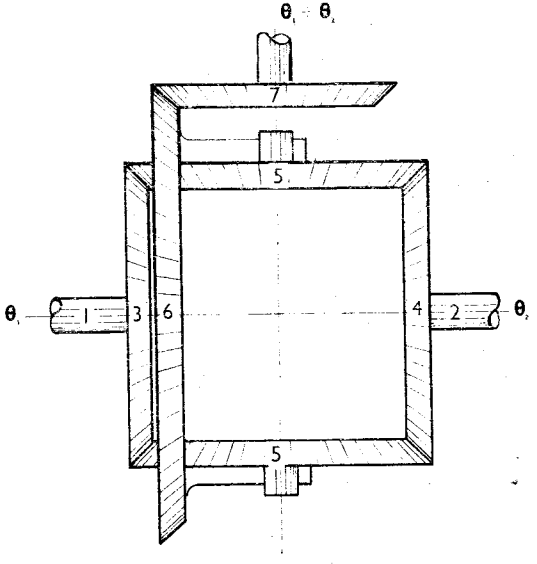


Fig. 16. Differential Gear

Stage 1

40. For the solution of stage 1 we will choose a ball and disc resolver pair. The cranks are turned through T and the discs driven at G radians/min (*see* paragraph 32). The roller outputs are therefore $G\frac{L}{r}\sin T$ and $G\frac{L}{r}\cos T$ radians/min. If $\frac{L}{r}=2$, say, then the outputs are analogues of U and V at a scale of 1 kt corresponds to 2 radians/min. U kts are therefore represented by 2U radians/min, V kts by 2V radians/min.

Stage 2

41. The output 2 V radians/min, or 120 V radians/hr, from stage 1 is an angular velocity which represents V kts. Thus the total angle turned in the required period $t_o—t_1$ is proportional to the distance flown north, D_N, in the same period. The distance D_N in n.m. is represented by 120 D_N radians on the shaft.
Using suitable gearing 1 n.m. can be clocked on veeder counters for every 120 radians turned by the shaft carrying V. Again, since 1 n.m.=1 min of latitude the change of latitude over the period can be displayed directly.

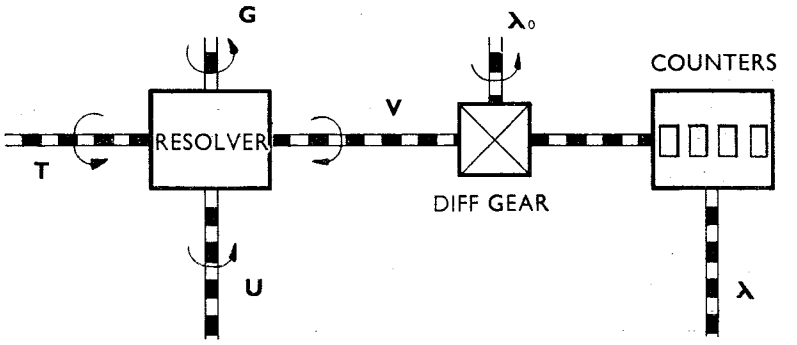


Fig. 17. Stages 1 and 2

42. The addition of λ_o at this point is done by the insertion of a differential gear in the V drive to the counters. Thus D_N turns one input to the gear while λ_o is applied manually at the same scale, 120 radians for each minute of λ_o required. The total reading on the counter is then λ_1. Fig. 17 shows a schematic diagram of the computer to date. An output of λ_1 from the counters can be obtained using suitable gearing at a 1:1 correspondence.

Stage 3

43. It would appear that the first part of this stage might be to find tan λ. Direct evaluation of the tangent, however, is difficult. It is easier to use the identity $\tan \lambda = \frac{\text{Sin } \lambda}{\text{Cos } \lambda}$ and a neat way of doing this is illustrated in Fig. 18. The angular velocity 2U radians/min is applied to the rotor of a CX and transmitted to the stators of a CT. Any error signal from the CT rotor energizes a servomotor which drives the input roller of a sphere resolver. The cosine output roller is coupled to turn the rotor of the CT. The angle λ_1 is applied to offset the input roller of the sphere resolver.

44. Imagine the shaft S (which carries the analogue of U) to be stationary and the CT rotor in its null. If S is turned the motor drives the rotor to follow the movement. If S is turned continuously then the rotor will be driven at the same speed. Let the motor speed be p radians/min when S is turning at 2U radians/min; then the rotor turns at $p \cos \lambda_1$ radians/min and $p = \frac{2U}{\cos \lambda_1}$ radians/min. Consider now the sine output disc of the resolver. It turns at

$$p \sin \lambda_1 = \frac{2U}{\cos \lambda_1} \sin \lambda_1 = 2U \tan \lambda_1 \text{ radians/min.}$$

The sine disc therefore provides the required output.

Stage 4

45. It now remains to divide by R, an action which requires no physical process. The division by R means simply that the scale of representation has changed and that the actual value of A in radians/hour is given by $\frac{1}{2R}$ x output. This is taken into account when matching for any further computation using A, or when calibrating any display.

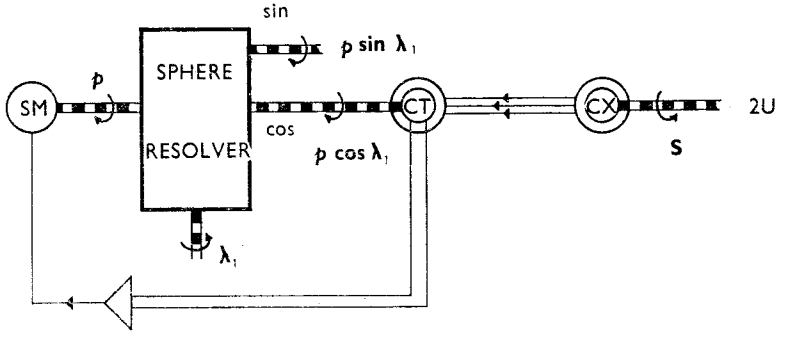


Fig. 18. U tan λ_1 Servomechanism

Complete Computer

46. The stages of computation may now be combined to illustrate the complete unit. Fig. 19 is a schematic data flow in diagram, *i.e.* it is one which shows the sequence of the computation with just sufficient detail of the devices to give a clear picture of the function of the unit.

47. A second diagram used to represent computers is the block diagram. Fig. 20 shows how this computer would appear. The block diagram has the virtue of diagrammatic simplicity, but can sometimes be more difficult to follow than the schematic flow diagram.

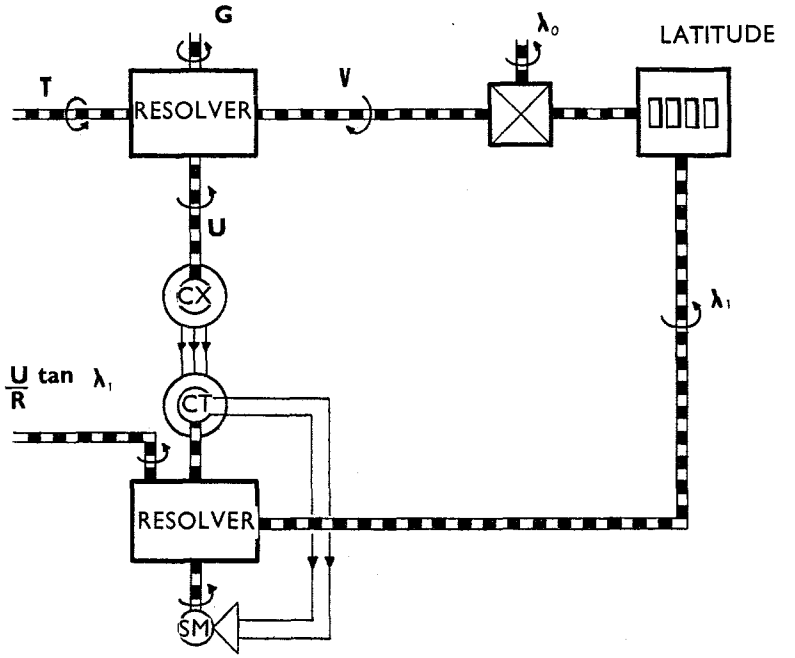


Fig. 19. Schematic Data Flow Diagram

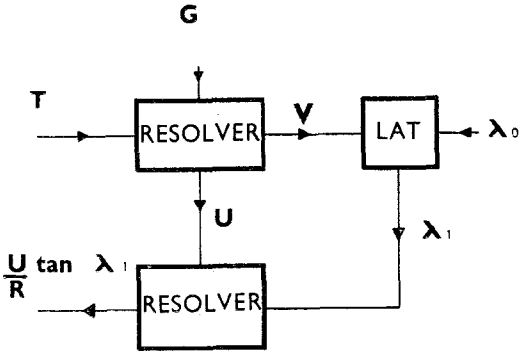


Fig. 20. Block Diagram

Other Examples of Computers

48. Most navigation computers are analogue computers. Examples such as the T.A.U. Mk. 1, A.P.I. Mk. 1, N.B.S., and the G.P.I.s will be found throughout this publication.

References

49. Further reading on the subject is contained in the following A.P.s: 3302, 2890, 1469, 1275B, 4685 and 2894.

CHAPTER 2

DIGITAL COMPUTERS

CONTENTS

	Paras.
Introduction	1-3
THE PRINCIPLES OF GENERAL PURPOSE COMPUTERS	
Organization of the Computer	4-5
Process of Computation	6-10
Representation of Instructions and Data	11-13
The Binary Number System	14-17
Arithmetic Unit	18-25
Types of Store	26-31
Parallel and Serial Machines	32-33
Magnitude, Precision and Word Length	34-36
Programming Languages	37
Flow Diagrams	38-43
A SPECIAL PURPOSE COMPUTER	
The Problem	44
The Inputs	45
Principle	46-47
Equipment	48-50
Computation	51-52
Selection of Gates	53-54
Timing	55-57
Negative Error Angles	58
Summary	59
GLOSSARY OF COMPUTING TERMS	

Introduction

1. Chapter 1 of this Section considered in general terms the differences between analogue and digital computers. This chapter is concerned with the elements of digital computers, and its purpose is to dispel the mystique and help aircrew to gain confidence in their equipment. No attempt is made to cover all devices or all aspects of computer technology; for details of a particular computer reference should be made to the appropriate technical A.P. Other information will be found in chapters (to be issued later) in A.P. 1234B; the topics included there will be electronic devices, the mathematics peculiar to computers and discussion of the various uses to which the computer can be put.

2. There are many different kinds of computers; some can be made to perform several different types of calculation, and these we refer to as General Purpose (GP) computers; others are built to carry out one type of calculation, and we call these Special Purpose computers. In airborne installations one GP computer might be fitted to cope with all the problems of navigation, flight control, weapon delivery, and so on, but it is likely that many airborne digital computers will be small SP units. In navigation the essential computing problems are the resolution of vectors into components and integration, and many small, lightweight and robust special purpose computers exist which will do one or both of these.

3. In this chapter two hypothetical computers are described; one is GP, the other special purpose. These computers should not be regarded as typical, for the typical computer is as elusive as the average man; it does not exist, and all generalities lead eventually to contradiction.

THE PRINCIPLES OF GENERAL PURPOSE COMPUTERS

Organization of the Computer

4. The digital computer comprises a number of units each of which apes one of the functions performed by human beings when they calculate. When homo sapiens goes to work, he first reads up, is told, or perhaps remembers, the method to be used. He then writes down the numbers involved and taking them two at a time, he adds, subtracts, multiplies or divides, as indicated by the method until the answer is reached. At all stages the intermediate results are written down or otherwise retained until next needed. When the answer is obtained he writes it down and passes it on to whoever wants it. The functions are:—

(*a*) *Communication.* Man obtains the method and the numbers and supplies the result by communicating with others through reading, writing, and talking.

(*b*) *Retention.* Man either remembers or writes down the method, the numbers and the intermediate and final results.

(*c*) *Arithmetic.* The ability to add, subtract, multiply and divide is clearly an essential.

(*d*) *Regulation.* The process of calculation must be controlled in such a way that the numbers are operated on in the way, and in the order, given by the method.

5. These functions are achieved in the computer by units as follows:—

(*a*) *Communication.* The computer has an input unit and an output unit. It calculates by manipulating pulse groups which represent numbers and these groups are created at the input unit when punched paper or card is fed in; the punched paper is obtained from a special typewriter with a plain language keyboard. The output unit reverses the process by translating the pulses back into plain language and printing the result, or by preselection by the operator, can supply punched paper with the answer for subsequent decoding.

(*b*) *Retention.* The computer has a storage unit into which the method, the numbers and all results, intermediate and final, are written.

(*c*) *Arithmetic.* The arithmetic unit comprises many circuits which, although not independent of one another, are best thought of as being arranged separately into sub-units, one for addition, one for subtraction, one for multiplication and one for division. When it is required to add two numbers, for example, the two pulse groups are drawn from store, applied to the adding network and the resulting pulse group is returned to store.

(*d*) *Regulation.* The computer operates on two numbers at a time and to avoid confusion it does so in a strictly timed rhythm. To supply the timing it uses a crystal oscillator (the clock) and it works on a two-beat cycle; during the first beat it seeks an instruction, add 4 and 3, say, and during the second beat it carries out the operation and puts 7 into store. The timing sequence and the connection of the correct number to the appropriate part of the arithmetic unit are the responsibility of the control unit.

The arrangement of the various units is shown in Fig. 1.

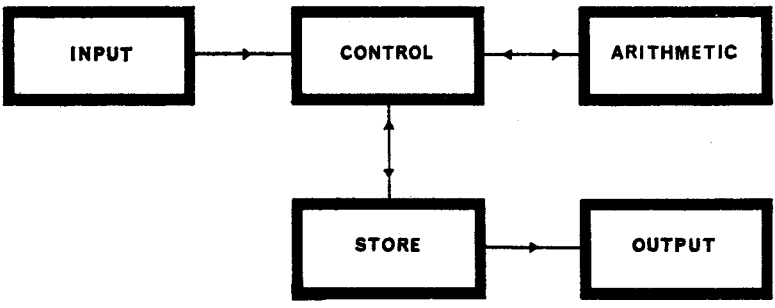


Fig. 1 Units of the Computer

Process of Computation

6. The process of computation starts with the human being, and his first task is to find a method of solution which the computer can cope with, *i.e.* one in which intuition and artistry play no part. Having found such a method he reduces it to a series of stolid, logical steps each one representing a single arithmetical operation. This process is known as programming and in a large, complicated problem the production of a program can take weeks or even months to complete.

7. Programs are written out as a series of instructions using a special code, in which each operation is described in two parts: the function, add, subtract, *etc*, and the location of the number to be used in the operation. (In the computer storage unit numbers are held in individual store locations each one identified by a location number, and in the program reference is made, not to the value of a number, but to the store location.) To simplify matters one number in each operation is first put into a special single number store, known as the accumulator, and the immediate result of the subsequent operation automatically replaces it. The following example may clarify the position:—

Compute:

$$(3 + 6) \times (4 - 2)$$

Action Required	*Function Code*
Read 3, 6, 4, 2 into store locations 100, 101, 102, 103.	L 100
Put 3 into the accumulator.	H 100
Add 6 to the accumulator.	A 101
Put the number in the accumulator into store location 104.	T 104
Put 4 in the accumulator.	H 102
Subtract 2 from the accumulator.	S 103
Multiply the accumulator by 9.	M 104

The number in the accumulator at the end is 18.

8. The program is punched out using the special typewriter onto paper tape and presented to the input unit. The complete program is to be directed by the control unit into store, and to achieve this the programmer attaches an administrative instruction, say Y, to the head of his program. Other administrative instructions are added as necessary; if he wishes to have the result printed out he could add a further letter, P, to the end; this he would follow with a further symbol, say Z, telling the computer to stop; to complete the program he would append yet another symbol, say X which tells the control unit that the input of program is complete and that the computation is to start. Thus the complete program would appear as follows:—

Function Code	
Y1	Store the program which follows in store locations 1, 2, 3 *et seq.*
L 100	
H 100	
A 101	
T 104	
H 102	
S 103	

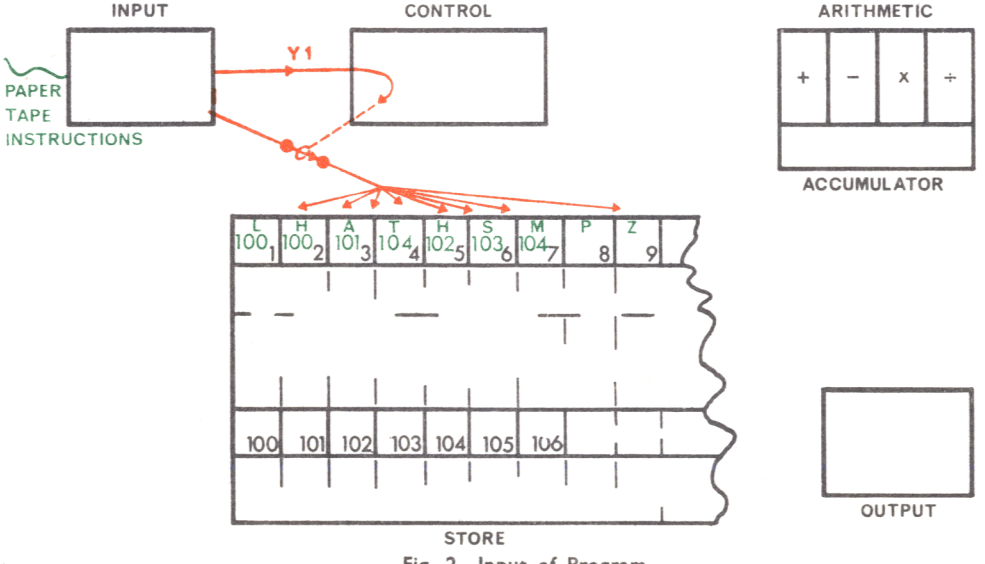


Fig. 2 Input of Program

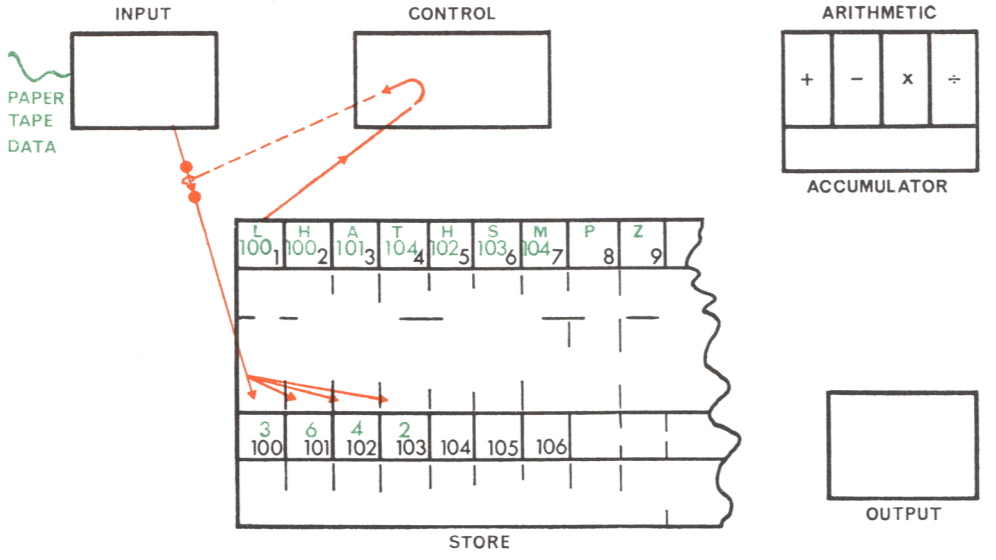


Fig. 3 Input of Data

M 104
P Print
Z Stop computing
X Start computing.

The four numbers 3, 6, 4 and 2 would follow on a separate tape, known as the data tape.

9. Fig. 2 shows the first action of the computer. The signal Y 1 from the input unit causes the control unit to read the instructions one by one into store locations 1, 2, 3, 4 *etc.* When X is received the control unit immediately refers to the first stored instruction L 100. This causes it to look to the input unit for the data, which it puts in the store locations required, *i.e.* 100 *et seq*, as in Fig. 3.

10. The control unit now seeks the next instruction, H 100, and moves 3 to the accumulator. Numbers are not erased when taken from store, so that 3 now appears in two places (Fig. 4). The next instruction brings 6 to the adding circuit, along with 3 from the accumulator, and the result, 9 replaces 3 in the accumulator. The remainder of the program is completed in the same way, P moving the result, 18, to the output unit for printing, and Z stopping the machine.

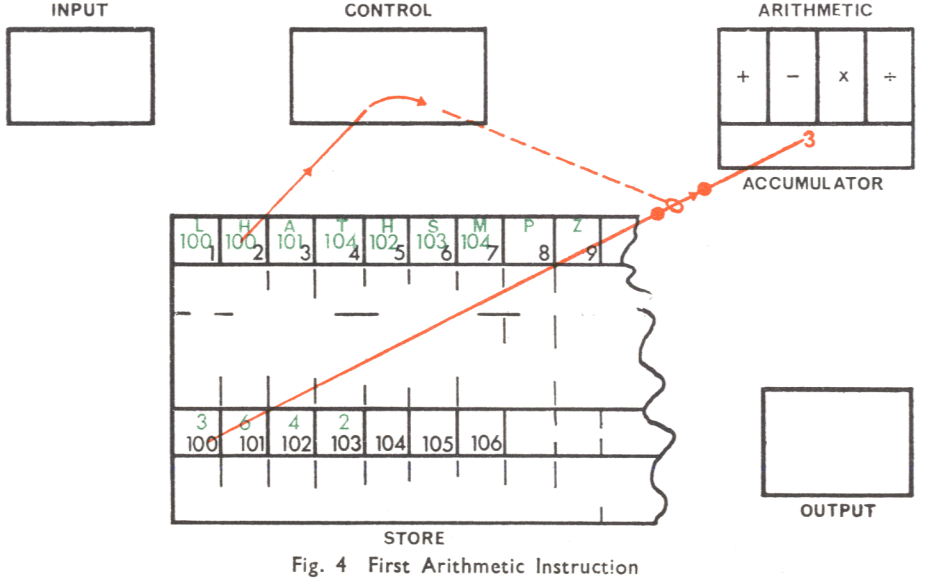


Fig. 4 First Arithmetic Instruction

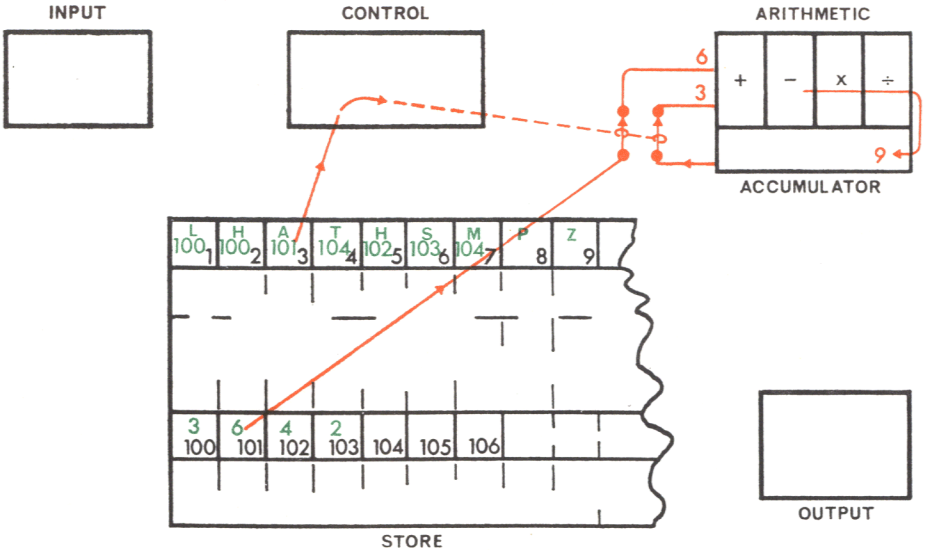


Fig. 5 Second Arithmetic Instruction

Representation of Instructions and Data

11. Numbers are represented within the computer by voltage pulse groups using a code which depends on the binary number system. In this system there are only two digits, 0 and 1, so that a number such as

0 1 0 1 1

can be indicated by a pulse train in which a pulse represents 1 and no pulse an 0. Binary digits are often referred to as bits.

12. Instructions are represented in exactly the same way and a pulse group

1 0 1 0 1 1 1 0 1

can only be distinguished as one or the other by its appearance in the computer at a specified time and place. Thus groups occurring during beat 1 are instructions and are interpreted as such, while those during beat 2 are numbers. Of course an instruction comprises both a function and a store location number and these are distinguished by their relative positions within the group, the first few digits, perhaps, being the function part and the remaining digits the store number. A complete pulse group is known as a word, and the number of digits the word length.

13. In store, numbers and instructions are represented by the conditions of bistable devices. These may be regarded as switches which are set to on or off, each location having one switch for each digit. In Fig 6 we have a single location showing the storage of a word of 6 digits, 0 0 1 1 0 1. To store a number the pulses are applied to electronic devices, C, which move the switches accordingly; to unstore we apply a complete word of pulses at A and the pulse group appears at B. The impulsing of stores is carried out by a pulse generator under the control of the clock in the control unit (hence the term clock

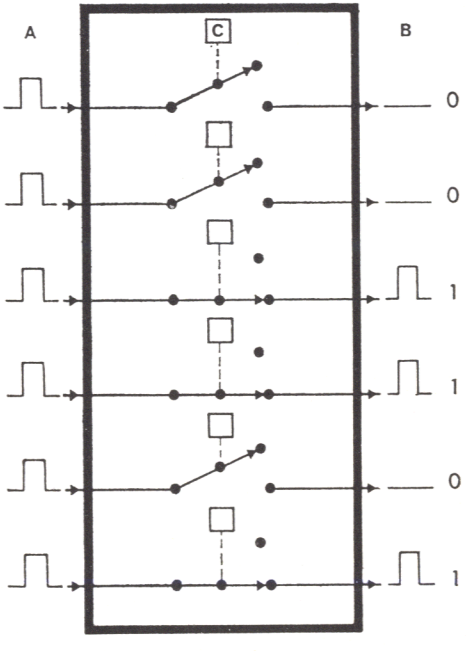


Fig. 6 Storing by Switches

pulses). When we say that the control unit refers to store for an instruction or number we mean that it selects the location and releases pulses by impulsing it.

The Binary Number System

14. The number system which we use every day is so written that each digit has a particular significance indicated by its position relative to the others. When we see the number 534 we know that this means 500 + 30 + 4 because convention dictates that the extreme right digit indicates the number of units and the next, moving left, indicates the number of bundles of ten while the 5 indicates the number of bundles of one hundred. Such a number system is known as a positional number system, and decimal notation 534 is really a shorthand method of writing:—

$$(5 \times 100) + (3 \times 10) + (4 \times 1)$$

or $(5 \times 10^2) + (3 \times 10^1) + (4 \times 10^0)$.

15. We use the decimal system because it is convenient: we have 10 fingers and 10 toes against which items can be counted using a one to one correspondence. But there is no real restriction and we can count in bundles of any number and represent them using positional notation. In particular we can count in bundles of two, and this turns out to be convenient for electonic computation since only two digits 0 and 1 are required. The extreme right digit as before represents the number of units, but when we have two units we have one bundle of two and would appear as a 1 to the left of the unit digit. Two bundles of two are one bundle of four and would appear as a 1 to the left of the bundles of two digits. For example decimal 3 is one unit and one bundle of 2 and would appear as:—

$$1\ 1 \quad i.e.\ (1 \times 2^1) + (1 \times 2^0)$$

and the binary number

$$1\ 0\ 1\ 1\ 0\ 1\ 1$$

$$i.e.\ (1 \times 2^6) + (0 \times 2^5) + (1 \times 2^4) + (1 \times 2^3) + (0 \times 2^2) + (1 \times 2^1) + (1 \times 2^0)$$

is in decimal $64 + 16 + 8 + 2 + 1 = 91$.

16. **Arithmetic.** Addition, subtraction, multiplication and division are extremely simple in binary. For example:—

(*a*) Add 1 0 1 0 to 1 0 1 1 1.

1 0 1 0	
1 0 1 1 1	
1 1 1 1	CARRY
1 0 0 0 0 1	SUM

The carry digit results from $1 + 1 = 1\ 0$ *i.e.* $(1 \times 2^1) + (0 \times 2^0)$.

(*b*) Multiply 1 0 1 by 1 1 0

1 0 1	
1 1 0	
0 0 0	mult by 0
1 0 1	mult by 1 0
1 0 1	mult by 1 0 0
1 1 1 1 0	

Multiplication is often performed as a shift-add operation, that is by shifting the number to the left for every power of two by which it is multiplied and adding the successive results.

17. **Binary Fractions.** Fractions in binary comprise the negative powers of two, and the integral part of a number is separated from the fractional part by the binary point. We have, then,

$\frac{1}{2} = 2^{-1} \equiv 0.1$,
$\frac{1}{4} = 2^{-2} \equiv 0.01$,
$\frac{1}{8} = 2^{-3} \equiv 0.001$,
$\frac{7}{8} = \frac{1}{2} + \frac{1}{4} + \frac{1}{8} \equiv 0.111$

and binary 10101.1011

$\equiv 2^4 + 2^2 + 2^0 + 2^{-1} + 2^{-3} + 2^{-4}$
$= 16 + 4 + 1 + \frac{1}{2} + \frac{1}{8} + \frac{1}{16}$
$21\frac{11}{16} = 21.6875$ in decimal.

Arithmetic with binary fractions follows the same simple pattern as with binary integers. For example:—

(*a*) Add 10.101 to 110.11

10.101
110.11
1001.011

i.e. 2.625 + 6.75 = 9.375 in decimal

(*b*) Multiply 0.101 by 0.01
shift binary point 2 places to the left,
answer = 0.00101

Arithmetic Unit

18. The function of the arithmetic unit is to generate, as required, a pulse group representing the sum, difference, product or quotient of two pulse groups applied to it. To achieve this the computer has many different circuits into which numbers are directed, but a proper study of these circuits, known as logic circuits, is beyond the scope of this chapter. Here we will limit the discussion to what one or two of them do, rather than how they do it, and will further restrict ourselves to addition.

19. Consider first the form of output required from an adding network when two digits *A* and *B* are applied to it. They may take either of the values 0 or 1 and the following table summarizes the results:—

A	*B*	*Sum*
0	0	0
1	0	1
0	1	1
1	1	10 *i.e.* 0 carry 1.

The adding network is to be of such a design that when two digits are applied to it an output first digit is required only when *A* and *B* are different. We therefore specify that a device is needed which will give an output digit only when $A \not\equiv B$ (*A* is not equivalent to *B*); such a device is readly engineered, and we will refer to it as a not-equivalent circuit. Its symbol and operation are summarized in Fig. 7.

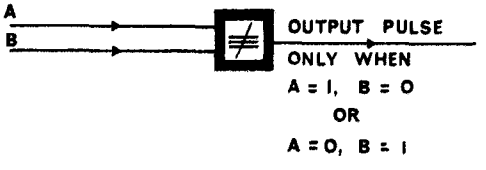


Fig. 7 Not-Equivalent Circuit

20. The carry digit is required only when both *A* and *B* are pulses, and is obtained from a special circuit element known as an AND gate; this provides an output pulse only when *A* AND *B* are applied together. Our adding mechanism now consists of the elements arranged as in Fig. 8, but it is hardly very practical since most numbers have more than one digit and a means of moving the carry digit on to the next column is required; for this reason the device at Fig. 8 is known as a half-adder.

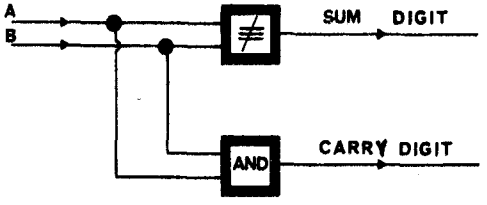


Fig. 8 Half-Adder

21. The calculation so far can be represented symbolically as follows:—

digit	*A*
digit	*B*
sum digit	$A \not\equiv B$
carry digit	*A* AND *B*

using the interpretations outlined already: $A \not\equiv B$ means 1 if $A \not\equiv B$, while *A* AND *B* means 1 if *A* and *B* are both 1. When we have numbers of two digits we may use the same symbolism. Thus if the digits are *P*, *A* and *Q*, *B*:—

digits	*P*	*A*
digits	*Q*	*B*
sum digit	$P \not\equiv Q$	$A \not\equiv B$
carry digit	*P* AND *Q*	*A* AND *B*

22. But the *A* AND *B* digit must be carried into the *P*, *Q* column and be added to the result of $P \not\equiv Q$ to give the second column sum digit; similarly the *P* AND *Q* digit belongs in the third column, thus we find:—

Serial	*3rd digit*	*2nd digit* *P* *Q*	*1st digit* *A* *B*
1. Sum digit 2. Carry digit	*P* AND *Q*	$(P \not\equiv Q)$ *A* AND *B*	$(A \not\equiv B)$
3. Sum digit 4. Carry digit	$(P \not\equiv Q)$ AND (*A* AND *B*)	$(P \not\equiv Q) \not\equiv (A$ AND $B)$	

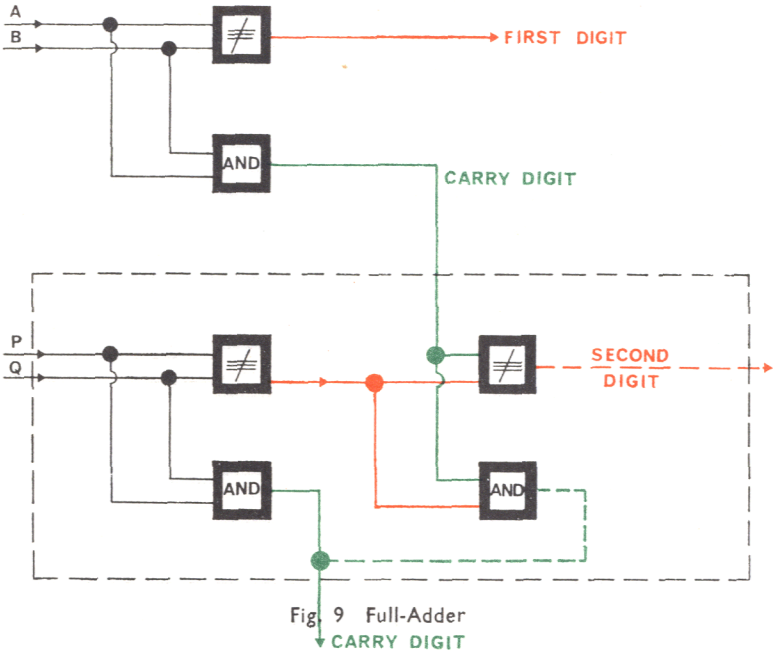


Fig. 9 Full-Adder

23. In this computer each digit is carried on a separate line so that all the digits in each number arrive at the arithmetic unit simultaneously. The complete adder is built up of units which behave as indicated in the symbolic addition above. Following through on Fig. 9 the operation is as follows:—

(*a*) *A*, *B* arrive at one not-equivalent gate and *P*, *Q* arrive at another. The first digits of *A* plus *B* and *P* plus *Q* are given on the full red lines. These correspond to $A \not\equiv B$ and $P \not\equiv Q$ at 1 above.

(*b*) The carry digits *A* AND *B* and *P* AND *Q* are produced by their AND gates on the full green lines. These correspond to *P* AND *Q* and *A* AND *B* at serial 2 above.

(*c*) The carry digit *A* AND *B* is added to the result of $P \not\equiv Q$ to produce finally the second digit $(P \not\equiv Q) \not\equiv (A \text{ AND } B)$ on the broken red line. This corresponds to serial 3.

(*d*) The carry digit from operation (*c*) is $(P \not\equiv Q)$ AND $(A$ AND $B)$; this is produced by an AND gate and its output is connected to the *P* AND *Q* carry digit line to move up to the next column of digits. We notice that digit outputs (*P* AND *Q*) and $(P \not\equiv Q)$ cannot exist together, hence this final carry digit is (*P* AND *Q*) or $(P \not\equiv Q)$ AND $(A$ AND $B)$.

24. A complete adder which will accept numbers of 5 digits is shown in block form in Fig. 10, each adding unit consisting of the gates enclosed within the black hatched line in Fig. 9.

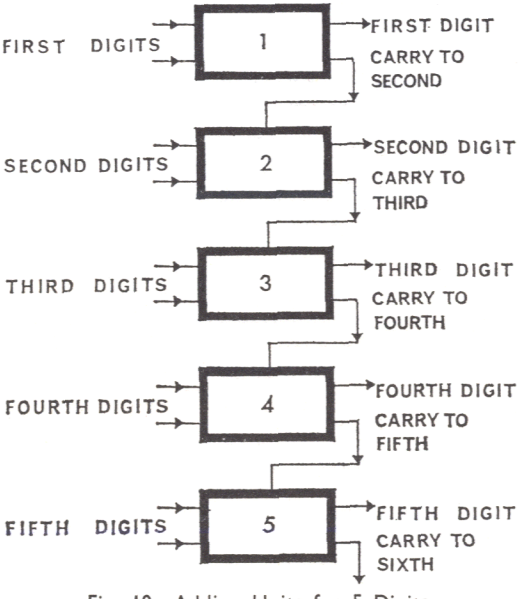


Fig. 10 . Adding Units for 5 Digits

(A.L. 7, Dec. 65)

25. Further information on the processes involved, including methods of subtraction, multiplication and division, is to be found in A.P. 4776, Circuit Techniques in Digital Computers and Data Handling Equipment.

Types of Store

26. **General.** Many different methods of storing exist and the store used in a particular computer is determined after consideration of the following:—

(*a*) *Capacity.* How many computer words are stored?

(*b*) *Non-volatility.* Does the store retain information until erasure is desired?

(*c*) *Access Time.* How long does it take to find, and then to read a word from or write a word into, any given location?

(*d*) *Size and Cost.* Is it a physically manageable system? What is the cost per bit stored?

(*e*) *Associated Circuitry.* What ancillary equipment is required to use the storage medium? What is the cost of this equipment? What are the servicing problems?

Ideally a storage system would combine the largest capacity, fastest access time, smallest size and cost, and simplest associated circuitry, but in general these are incompatible requirements. A compromise is effected in any given machine, although many computers tend to have storage systems comprising elements of more than one type. The usual system is to have a fast access store of reasonable size and a slower access store of larger capacity to which less frequent reference is required. Some of the more common storage systems are described below.

27. **Magnetic Tape.** Magnetic tape storage has a large capacity, but its random access time is high. It is therefore used to store information such as past records and accounts and other data seldom required. The tape consists of a non-magnetic base material coated with a magnetic material. It may have several tracks, and to speed access each track has its own reading and writing head. To write onto the tape a transport mechanism (Fig. 11) moves the tape at constant speed under the writing head. Pulses of current flowing in the head cause flux linkage with the tape surface and consequent magnetization. The head is usually dual purpose and to read the stored instruction or data the tape is passed beneath the read leg of the head. An emf is induced in the head as each bit passes. This type of store is non-volatile, and occupies little space, but it may require complicated associated circuitry.

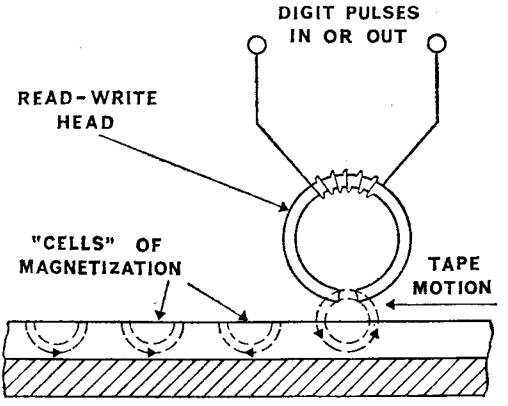


Fig. 11 Magnetic Tape Read/Write

28. **Magnetic Drum.** The magnetic drum (Fig. 12) uses the same principle of operation as magnetic tape, but has a faster random access. Capacity is, however, reduced. It consists of a number of closed tracks on magnetic material mounted on a drum, each track having its own read/write head.

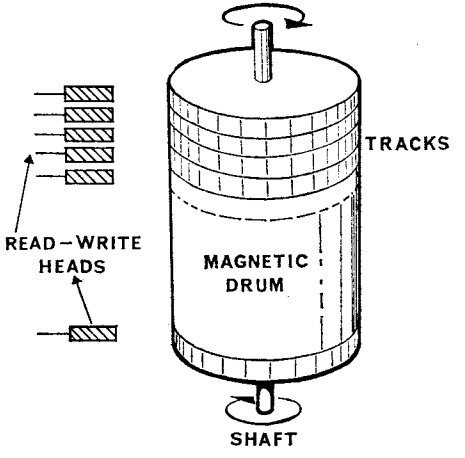
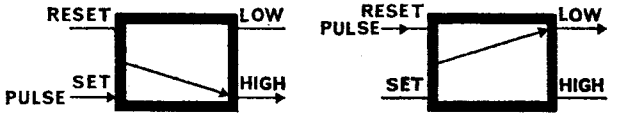


Fig. 12 Magnetic Drum Store

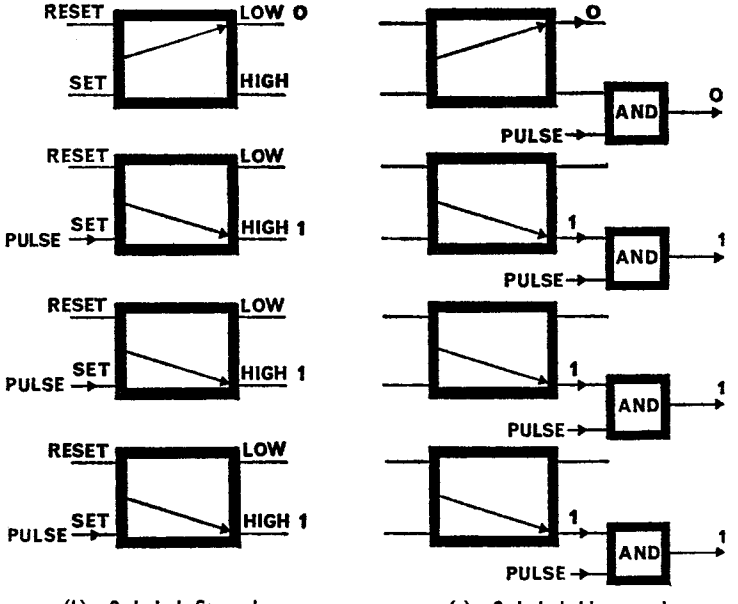
29. **Magnetic Discs.** Magnetic discs are another variant of the same principle. A number of discs having circular closed tracks on each side are mounted concentrically. Read/write heads are provided for each face of each disc and they are moved radially to the required track. Magnetic discs provide a larger store capacity than the magnetic drum without any loss of access time.

30. **Ferrite Core.** A ferrite core store consists of many small rings of material (they can be about the size of a pin head) which exhibit almost square BH hysteresis curves. Each core is magnetized by passing a current in the positive sense through a wire threading it to store a 1 or in the negative sense to represent 0. When an output is required negative current passed and the change of magnetic state induces a pulse in a secondary wire, if the core were storing a 1. This type of store is extremely fast, there being no moving parts, but it is volatile since each reading destroys the stored bits. Core stores are used as the immediate access store of most computers.

31. **The Bistable Flip-Flop.** Many computers use transistorized bistable flip-flops both for storage and for the control of entry to the arithmetic unit. For our purposes the flip-flop (Fig. 13 (*a*)) may be regarded as a device with two inputs and two outputs. The input points are usually referred to as set and reset, while the output is either a high or low voltage. A pulse applied to the set input will cause continuous high voltage output, and this condition will remain whatever is done subsequently to the set input until a pulse is applied to the reset input. A pulse at the reset input will switch the output to the low voltage and this condition will persist until a set pulse is received. For storage purposes we use the flip-flop with an AND gate, the high voltage output point being permanently connected to one AND input; one such combination is used for each digit. To empty the store a pulse is applied at all reset inputs; thus all outputs are low voltage. A number is then stored by applying the pulse pattern to the set inputs, causing continuous high voltage outputs at each location which received a 1 (Fig. 13 (*b*)). Thus for each 1 stored there is an AND gate with a permanent input. To unstore all the AND gate second inputs are pulsed; those with a permanent input give a pulse output and those without give 0 (Fig. 13 (*c*)).



(*a*) Bistable Flip-Flop Symbol and Action



(*b*) 0 1 1 1 Stored (*c*) 0 1 1 1 Unstored

Fig. 13 Bistable Flip Flop Store

Parallel and Serial Machines

32. The computer described in paras. 4-25 used one wire, one switch, one adder, *etc.* for each digit. Thus all the digits in a word were manipulated independently and at the same time. Such a machine is said to use parallel operation.

33. Many computers, however, transmit whole words on one wire, the digits being separated in time. The complexity of such a machine is reduced since one adding circuit can, for example, operate on digits one pair at a time to add the complete numbers. This is the serial method of computation, and although serial machines are smaller, less complex, and cheaper than parallel machines they are very much slower. Indeed since a parallel machine operates on a complete word in one move, and the serial machine takes n moves to work through the n digits in the word it may be n times slower.

Magnitude, Precision and Word Length

34. It has been shown how each digit in store requires its own switch, or other bistable element, and when being operated on each requires its own wire and set of gates. Clearly there is a limit to the number of digits if complexity and bulk are to be minimized. For most computers a working compromise is found at between 30 and 50 digits. However word length alone tells us neither the precision nor the range of numerical magnitude of the computer, for the bits (binary digits) can be used in different ways. Usually the first bit is reserved for the sign of the number, the common convention being 0 for positive and 1 for negative numbers, but this apart the significance of the remaining bits depends on the method of number representation; the programmer can, with most computers choose between two methods, known as fixed point and floating point representation.

35. **Fixed Point Representation.** In the fixed point method the position of the binary point is determined at the beginning of the calculation and is held there throughout. Thus 1010.111 would be stored as 0001010.11 in a machine with 9 available bits, if the programmer decided to use two places of binary fractions. All numbers appearing throughout a calculation are rounded to the chosen number of places, but the great disadvantage of the method is that an estimate must be made in advance of the order of magnitude of every number appearing in the course of a calculation to avoid occasions when the most significant digits overflow and are lost. Most often nowadays the fixed point method is used only when all the numbers and calculations results are integers, and is particularly useful when counting particular items. It has therefore a rather limited application in mathematical calculations, but is much used in commercial computations where classification and sorting are the main tasks.

36. **Floating Point Representation.** In decimal notation, standard form provides an economical method of writing very large or very small numbers. For example the velocity of light is approximately 29,800,000,000 cms/sec, but this is more conveniently written 2.98×10^{10} cms/sec, while the mass of the α-particle (0.000 000 000 000 000 000 000 006 644 gms) is usually written 6.644×10^{-24}. Any decimal number can be written in the standard form $A10^n$, and clearly the number is defined by the mantissa, A, and the index (or exponent) n; the exponent indicates the number of places which the decimal point in the mantissa should be moved to reproduce the conventional form. In the binary system halving and doubling have the same effect on the binary point as dividing and multiplying by 10 have in the decimal system, hence binary numbers can be standardized in the form $B(2)^m$. The radix 2 is not usually written and the other common convention is that the mantissa B is a binary fraction of modulus not less than $\frac{1}{2}$, thus binary + 111 000 000 000 can be specified by +111 +1100, since the original number is reproduced by moving the binary point in .111 twelve (1100) places to the right. Fractions can be represented by negative exponents. In computers this method of representation is known as floating point, and its great advantage is that it allows the expression of numbers very much larger or smaller than those in the range of magnitude implied by simple word length (the example shows a 13 digit number being indicated by 9 bits, including the sign). Of course the compensating disadvantage is also evident, for if in the example the number had been + 111010001000 it too would be represented by 9 bits in the form +111 +1100, that is the method implies some rounding. The actual allocation of bits to mantissa and exponent results from a compromise between precision, indicated by mantissa length, and range, indicated by the exponent. A not unusual arrangement is that of the Mercury computer where 37 available bits (excluding sign bits) are distributed 29 to the mantissa and 8 to the exponent, increasing the range from a nominal quantity of order decimal

10^{11}, to decimal 10^{75} while allowing an adequate precision of 9 significant decimal figures. The likelihood of overflow is small with floating point and it is usually preferred in mathematical calculation.

Programming Languages

37. The program described in paras. 7 and 8 is written in a function code for a particular computer; such writing is a slow process and the production of good, economical programs is difficult without considerable experience. To speed the writing of programs programming languages have been devised which can be used in a wide range of computers and can be written without reference to the function code. These languages, for example Algol, select certain words in plain language and assign particular meanings to them. When encoded by special typewriter onto paper tape or cards and applied to the computer they cause it to react in a way very much as the English word would indicate. Thus all programs in Algol, for example are initiated by using the word "begin" and are closed by the word "end". Encoded versions of "start" and "stop" would have quite different effects. Further these languages permit complete algebraic expressions to be written directly using special symbolism. For example to solve

$$x = \frac{-b + \sqrt{(b^2 - 4ac)}}{2a}$$

we are permitted to encode directly an instruction to put the result of the computation on the right hand side into a store designated x; for interest sake this would be written out as

$$x: = (-b + \text{sqrt}\ (b*b - 4*a*c))/(2*a);$$

in Algol. When using such languages a special program has to be written for each computer and stored in the machine before the problem is read in. This special (compiler) program translates the language into function code, and the calculation then proceeds automatically on that.

Flow Diagrams

38. Regardless of the language used to express a problem solving procedure, the steps of the procedure must be clearly thought out before the program is written. One aid to visualising a problem of any complexity is the flow, or block, diagram; this is a diagram made up of a set of boxes of various shapes, each containing a description of some operation, and linked by arrows showing the flow of control between operations. For computer users the diagram provides a useful indication of the routine followed in a particular calculation, and it is probable that it, will become the standard method in this publication of describing and explaining airborne installations. Some simple examples follow.

39. A rectangular box is used in this publication to indicate a processing operation, and an oval box an input or output operation; a series of such boxes linked one following another indicates a straightforward sequence of operations. For example a section of program is to read in six numbers, a, b, c, d, p, and q and evaluate the quantities

$$r = \frac{pd - qb}{ad - bc} \text{ and } s = \frac{qa - pc}{ad - bc}.$$

A flow diagram outlining the necessary steps is shown in Fig. 14.

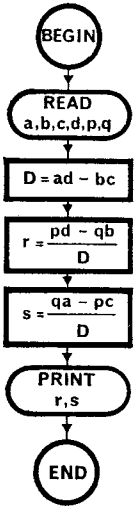


Fig. 14 Simple Flow Diagram

This example is rather trivial, but it at least shows that the programmer will arrange to calculate $ad - bc$ once only.

40. **Decisions.** At certain points in most programs it is necessary to follow different paths according to the results of comparisons or decisions. A diamond shaped box is used to indicate a decision. In the example at Fig. 14 the calculation will breakdown if $ad - bc = 0$ (the results r and s will be infinite or indeterminate and computers cannot work with such quantities; the usual reaction is to stop). This eventuality can be catered for by introducing an alternative sequence; in Fig. 15 it is indicated that the computer will be programmed to print out "$D = 0$" rather than attempt to continue with the calculation.

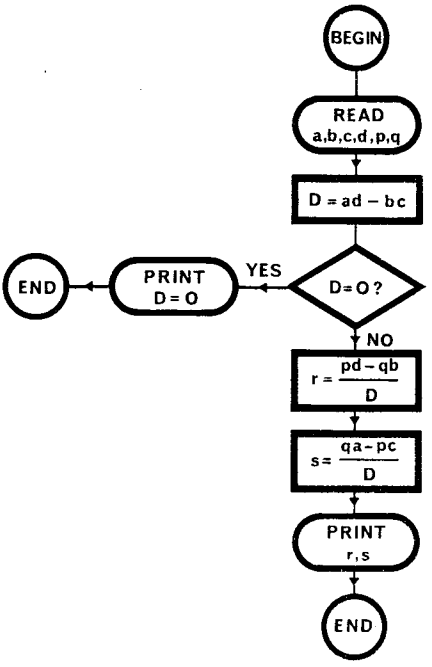


Fig. 15 Decision in Flow Diagram

41. **Loops.** A sequence of instructions obeyed repeatedly is termed a loop. If for example several sets of data a, b, c, d, p, q were to be manipulated the program would be designed to read in the next set and repeat the calculation, rather than stop after printing. This would be indicated by the lines on the diagram in Fig. 16; the computer would wait for further instructions should no new data be available.

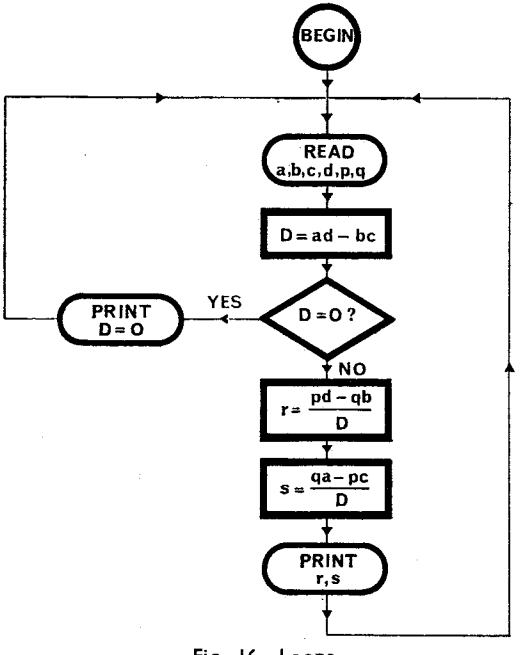


Fig. 16 Loops

42. **Counting.** All computers can be programmed to count the number of times a loop is repeated, and can be made to stop when a given count is reached. In Fig. 17 the example being followed is refined by a count, n, of loop repetitions and the computer is to stop after calculating r and s from 5 sets of data.

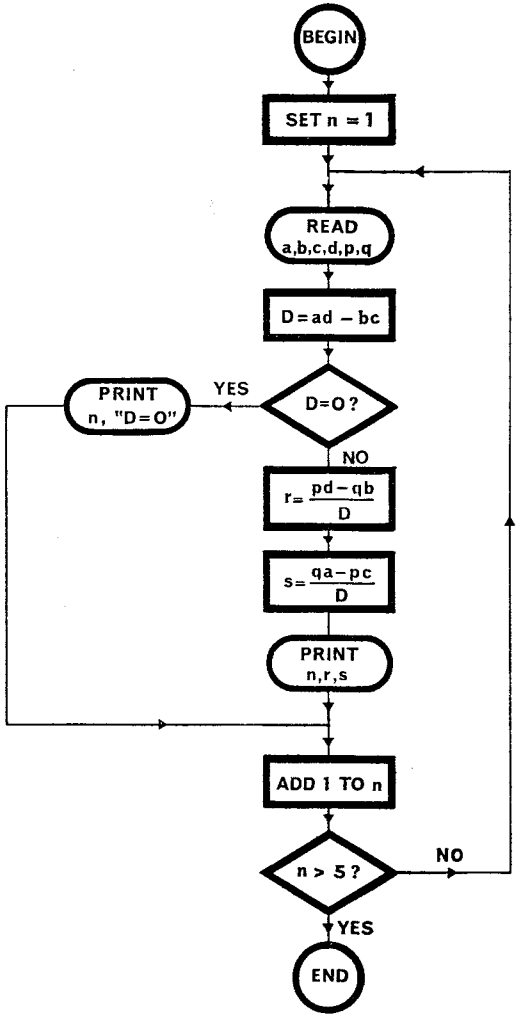


Fig. 17 Counting

43. **Iteration.** By its great speed of calculation the digital computer lends itself to iterative methods, that is, to methods in which a calculation is repeated with slight variations in the data until sufficient accuracy has been reached. For example the series

$$\frac{1}{1^2} + \frac{1}{2^2} + \frac{1}{3^2} + \frac{1}{4^2} + \ldots\ldots\ldots\ldots$$

can be summed directly to any given accuracy by an iterative loop which will stop when a term in the series has been found which is less than the prescribed quantity. In Fig. 18 a flow diagram outlines a procedure which might be adopted if terms less than .00005 are to be ignored.

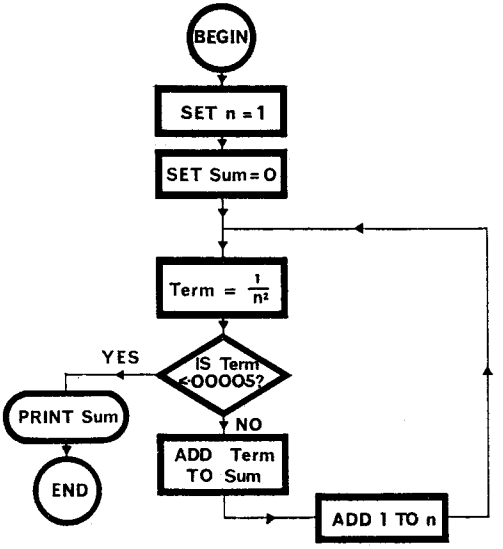


Fig. 18 An Iterative Loop

A SPECIAL PURPOSE COMPUTER

The Problem

44. The computer chosen for this discussion will calculate components of groundspeed along and across a required track, but the principle can be adapted to the resolution of any vector quantity into any components. The problem is illustrated in Fig. 19; groundspeed, *G* kts, along the TMG produces components *G* sin ε across track and *G* cos ε along track, where ε is the track error angle.

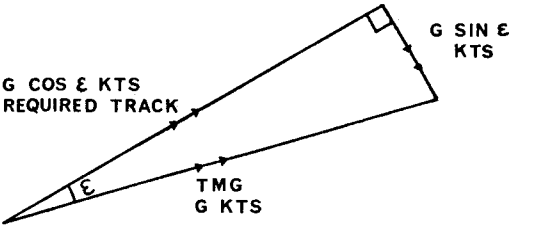


Fig. 19 Resolution into Components

The Inputs

45. The inputs to the computer are *G* as a pulse train from Doppler, each pulse representing, say, 1/40 n.m. flown, and ε, derived from a suitable differential synchro arrangement and presented to an "M" type motor in the computer.

Principle

46. The computation consists essentially of two multiplications of *G*, one by sin ε, the other by cos ε. In a GP computer the values of sin ε and cos ε are usually calculated from first principles using the value of ε, but this requires a large range of GP logic circuits and in the special purpose computer we prefer to provide what amounts to a table of sine and cosine values from which numbers can be called into the calculation as required. The table is kept to a manageable size by limiting its range (ε is restricted to8½°) and by tabulating at ½° intervals. Thus if ε is between ½° port and ½° starboard, cos α will be 1 and sin ε = 0. The complete table is (all numbers rounded to 3 decimals):—

ε	*Angle Used*	*Sine*	*Cosine*
0	0	0	1.000
½°	1°	0.017	1.000
1½°	2°	0.035	0.999
2½°	3°	0.052	0.999
3½°	4°	0.070	0.998
4½°	5°	0.087	0.996
5½°	6°	0.105	0.995
6½°	7°	0.122	0.993
7½°	8°	0.139	0.990
8½°			

47. Since the required multiplication is always by a fraction, or at most by one, the number of pulses output never exceeds the number input, and a process of methodical rejection of pulses is adopted to achieve the required outputs. For example, to multiply by cos 3° (0.999) we simply reject 1 pulse in every 1000 and so on.

Equipment

48. Two electronic devices form the backbone of the computing system. One of these is the AND gate, already described, while the other is a special bistable circuit.

49. The action of the bistable is to provide, on receipt of a pulse train, two outputs each one equal to half of the input; thus if six pulses are received, as in Fig. 20, the first is output at A, the second at B, the third at A, and so on. The chronological order is not disturbed, 2 lying between 1 and 3, and 4 between 3 and 5; we say that the B output is 180° out of phase with the A output.

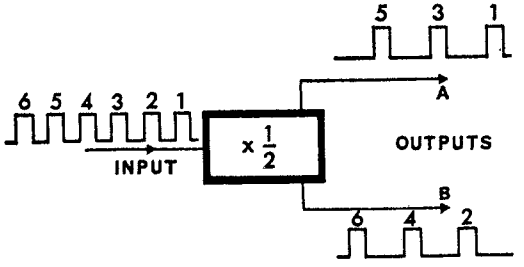


Fig. 20 Bistable Symbol and Action

50. If now the output at A is applied to another bistable the final outputs from the second will each be one quarter of the original input. A series of such devices can be strung together to provide, in turn, ½, ¼, ⅛....., of the input. In Fig. 21 ten bistables are connected in series; if 1024 pulses are applied to unit 1, 512 are passed to unit 2, 256 to unit 3, and finally 1 pulse is output from unit 10. During the passage of the pulses along the A channel, 512, 256, 128 *etc.*, pulses are made available at the B output points.

Computation

51. The output points of a series of 11 AND gates are connected to a common output line, and the B output from each bistable of a system of 10 is applied to an associated AND gate input as shown in Fig. 22; the final bistable's output is applied to a pair of AND gates. To obtain a given fraction of the input, say 0.017 (Sin 1°), AND gates 6 and 9 are opened by feeding a +ve voltage (which may be regarded as a permanent pulse) to the second inputs to these gates; over the period of receipt of the 1024 pulses only those at AND gates 6 and 9 are released, *i.e.* the output

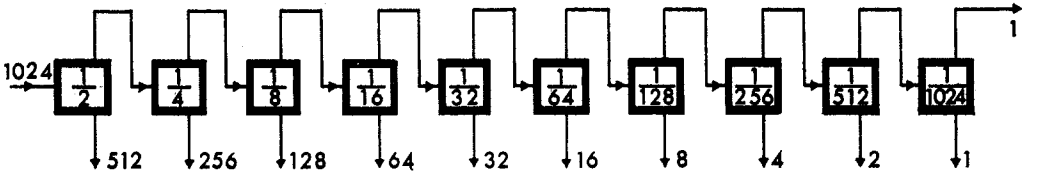


Fig. 21 Bistables in Series

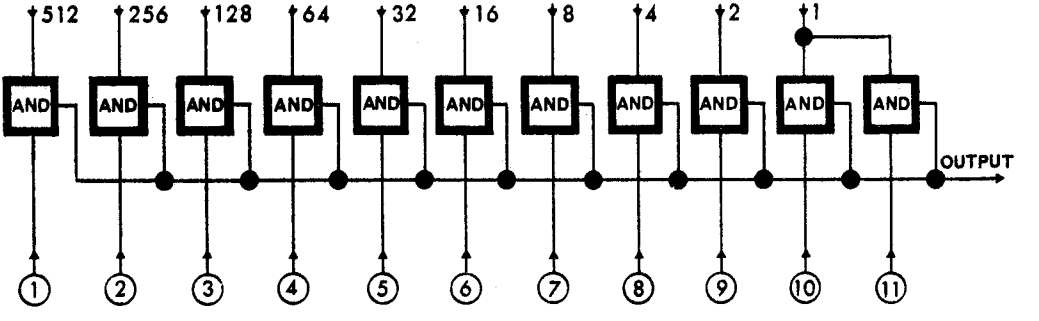


Fig. 22 Pulse Rejection Process

is 16 + 2 = 18 = 0.017 of 1024. To provide an output representing $G \cos 5°$ we require $1024 \times \cos 5° = 1024 \times 0.996 = 1020$ pulses to pass over the period; this is done by opening gates 1, 2, 3, 4, 5, 6, 7 and 8, *i.e.*
512 + 256 + 128 +64 + 32 + 16 + 8 + 4 = 1020 pulses, and keeping 9 and 10 closed.

52. Finally, to release all pulses, *i.e.* $G \cos 0°$, all the gates are opened, including the double gate at unit 10. To provide a facility for both sines and cosines a duplicate set of AND gates is used.

Selection of Gates

53. The selection of the gates is provided by a 9 position switch driven by ε (Fig. 23) through suitable gearing. Each switch position is connected to two rectilinear wire networks, one known as the cosine matrix, the other the sine matrix, through which the + voltage is applied to the appropriate gates.

54. To summarize the action so far we take $\varepsilon = 3°$. The input 3° moves the switch to position 3 and lines 3 on both matrices are energized. Gates 5, 6, 8 and 9 are opened to give 54 pulses (= 0.053 × 1024) which corresponds to $G \sin 3°$, while all but the last cosine gates are opened to give 1023 pulses corresponding to $G \cos 3°$.

Timing

55. It is important that an alteration of the track error angle switch should not take place

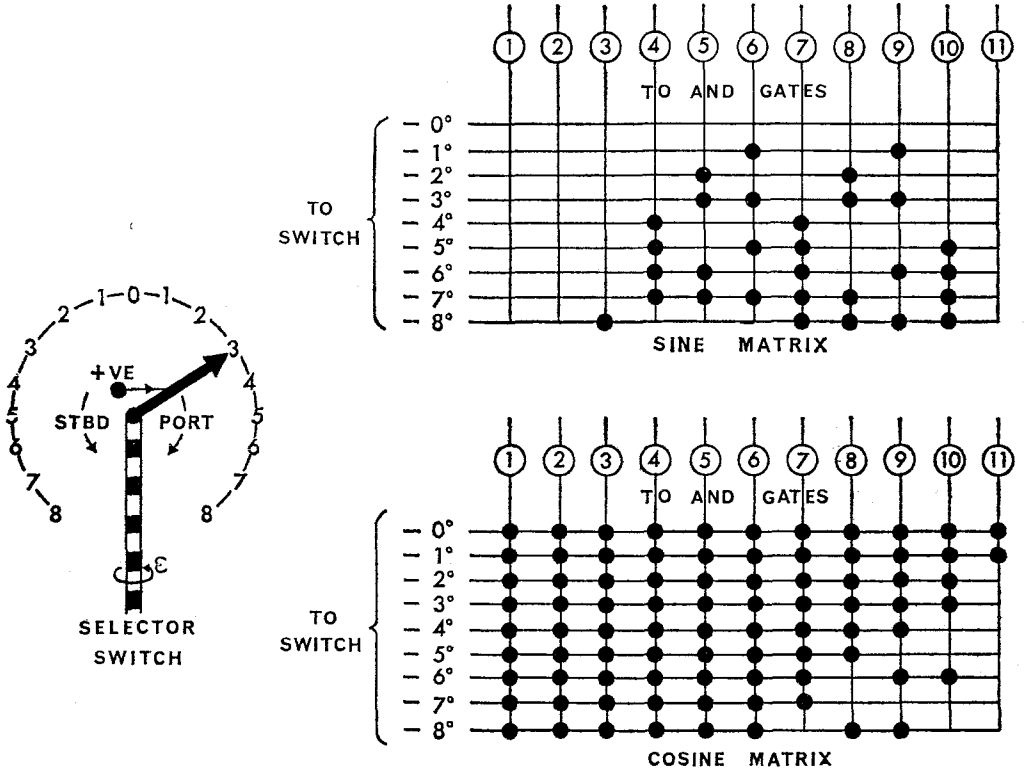


Fig. 23 Selection of Gates

while a pulse is being processed through the multiplier. A timing device prevents this from happening.

56. An oscillator governs the production of regularly spaced pulses at a rate somewhat in excess of the maximum rate of groundspeed input. Incoming groundspeed pulses are synchronized with these "clock" pulses before being passed to the multiplier unit, as in Fig. 24.

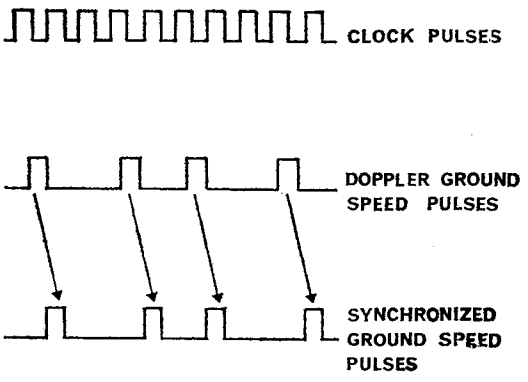


Fig. 24 Pulse Synchronization

57. Track error angle is received by an "M" type motor and the input to the motor is passed through an inhibitory circuit into which the clock pulses are also fed. This circuit prevents the motor from stepping during a clock pulse.

Negative Error Angles

58. The description so far has dealt with positive error angles. If ε becomes negative (*i.e.* TMG to port of the required track) the sine output drive has to change direction. A switch is actuated by ε as it passes through 0° to change the direction of the across track display drive.

Summary

59. The complete computer is assembled in block form in Fig. 25.

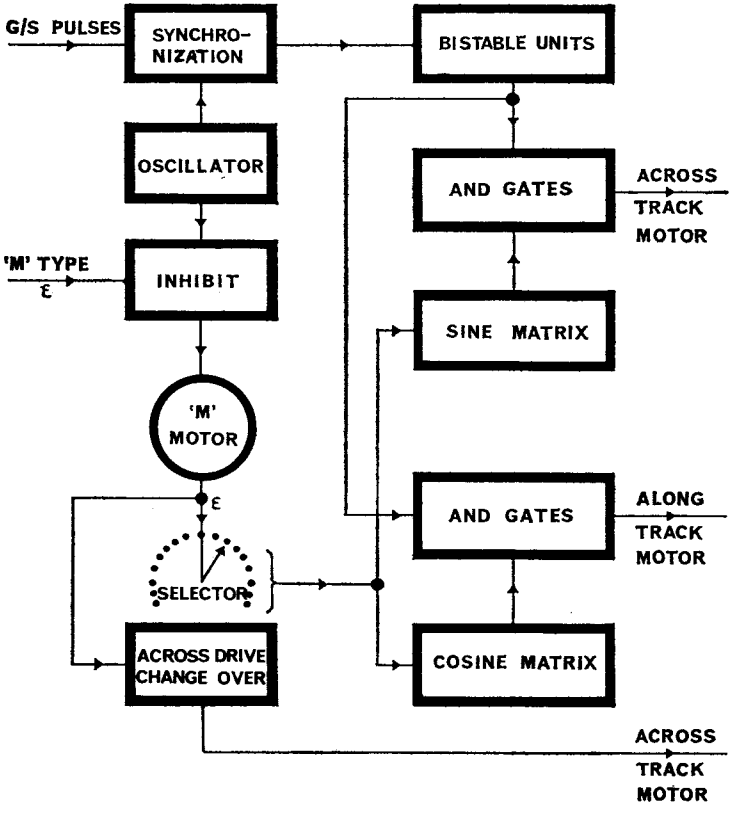


Fig. 25 Track Error Angle Computer

GLOSSARY OF COMPUTING TERMS

Note.—Those terms used in the text are followed by a reference paragraph number in brackets.

ACCESS TIME (26)	The time required to read or write a word from or to a store.
ADDRESS	The label or co-ordinates of store location.
ALGOL (37)	A generalized autocode intended for use on many types and makes of computers. Derived from the words algorithmic language.
ALPHA-NUMERIC	Alphabetic and/or numeric.
ANALOGUE	A continuously variable representation of one quantity by means of another physical quantity (*e.g.* the representation of temperature by the voltage from a thermocouple).
ARITHMETIC UNIT (18)	That which performs arithmetic operations in a computer.
AUTOCODE	A system of automatic coding using problem oriented language, in which no knowledge of the computer machine code (*see* below) is required.
BACKING STORE	A large (slow access) store which is used to load the smaller (fast access) working store.
BINARY SYSTEM (14)	The system of arithmetic, using digits 0 and 1, in which numbers are represented as the sum of powers of two (*e.g.* $23 = 16 + 4 + 2 + 1 = 10111$).
BIT (11)	A binary digit; the quantum of information.
CENTRAL PROCESSING UNIT	The name given to the combination of the arithmetic and control units.
CHARACTER	A single decimal cypher, letter or symbol coded in binary form.
CLOCK (5)	The device which generates the fundamental timing pulses for a computer.
COMPILER (37)	The programme which assembles the autocode program into machine code.
CONDITIONAL JUMP	The instruction which diverts the course of a program into one of two channels depending on whether a tested number is positive or negative.
CONTROL UNIT (5)	The logical circuitry which ensures that the program is obeyed.
CONTROL REGISTER	The register which contains the instruction currently being obeyed.
DECISION (40)	A choice between two or more sequences of instructions.
DECODE	Transform a signal from digital to analogue form (Opposite to ENCODE).
DIAGNOSTIC PROGRAM	A program designed to show up and locate faults in the operation of a computer.
ENCODE	*See* DECODE.
FERRITE CORE (30)	A small ferrite ring capable of storing one binary digit.
FIXED POINT (35)	A number representation in which the decimal point is in a fixed position on the register.
FLOATING POINT (36)	A number representation in which part of the word expresses the mantissa (form) of the number, and the rest expresses the exponent (the power of ten next above the number). $2397.4 = .23974 \times 10^4$ Written as $+ .23974 + 4$.
FLOW DIAGRAM (38)	A diagrammatic representation of the course of the computation.
FORTRAN	An autocode devised by IBM.
FUNCTION (7)	The computer operation initiated by each instruction.

FUNCTION CODE (7)	A list of arithmetic and organizational operations of which the machine is capable, together with their numeric symbols.
GATE (20)	A device which allows a signal to pass if, and only if, a control signal is present.
HARDWARE	Strictly speaking the electronic logic curcuits of a computer, the term is often used to describe the equipment in toto.
INPUT ROUTINE	The program which accepts data punched in an alpha-numeric code and assembles it in the appropriate storage location. *See* OUTPUT ROUTINE.
INSTRUCTION (8)	A statement specifying a single step in the computer program.
ITERATION (43)	The repetition of a process in a cyclic manner.
JUMP	An alteration in the course of a program. *See* CONDITIONAL JUMP and UNCONDITIONAL JUMP.
MACHINE CODE	The numerical language, based directly on the functional logic of the computer, used to express instructions.
MAGNETIC FILES	Devices such as magnetic discs, drums and tapes which are capable of storing a large amount of information in a binary form. Used as backing stores.
MATRIX STORE (30)	A matrix of ferrite cores used as the immediate access or working store.
ON-LINE	Connected to, and under the control of, a computer (Opposite to OFF-LINE).
OUTPUT ROUTINE	A program which assembles data for punching or printing in a given format. *See* INPUT ROUTINE.
PARALLEL MODE (32)	The method of performing arithmetic operations in which the whole of a word is examined at once and not bit by bit.
PARITY DIGIT	A bit attached to a character or word to assist in error detection.
PERIPHERAL EQUIPMENT	Equipment associated with the computer.
PROGRAM (7)	The sequence of instructions followed by a machine to carry out a computation.
READ	To transfer information from store.
REGISTER	A device for temporarily storing a word or group of characters.
SERIAL MODE (32)	The method of peforming arithmetic operations in which each word is examined bit by bit.
SHIFT (16)	To divide or multiply a word by a given power of two.
SOFTWARE	The library of programs maintained by computer operators.
STORE (5, 26)	A device for retaining information for subsequent use.
SYMBOLIC CODE	The alpha-numeric language, containing mnemonic functions and addresses, which may be translated by a computer program to produce instructions in machine code.
UNCONDITIONAL JUMP	A change in the course of a program which does not depend on a decision.
WORD (12)	A sequence of binary digits, fixed in number, which represents alpha-numeric data or an instruction.
WORKING STORE	The fast access store which contains a limited number of program steps and the relevant data.
WRITE	To transfer information into store.

PART 3

SECTION 3

POSITION COMPUTING AND DISPLAY SYSTEMS

Chapter	
1	**Air Position Indicator, Mark 1B**
2	**Wind Finding Attachment, Mark 1B**
3	**Ground Position Indicator, Mark 1B**
4	**Ground Position Indicator, Mark 4 Series**
5	**Type 9476 Computers**
6	**Type 947 and 9478 Display Units**
7	**The Ferranti Navigation Display and Computer**
8	**Ground Position Indicator, Mark 7**
9	**The Tactical Air Navigation System (TANS)**

Part 3 Sect 3 Chap 1

CHAPTER 1

AIR POSITION INDICATOR, MARK 1B

CONTENTS

	Paras.
Introduction	1– 2
Theory of Operation	3– 6
DESCRIPTION	
Components	7
Compass Repeater System	8
Resolving Gears	9–10
Secant Gear	11–12
Counter Mechanism	13–16
ERRORS AND ACCURACY	
Errors	17–21
Accuracy	22
OPERATIONAL USE	
General	23
Pre-Flight Check	24
In-Flight Use	25–28
After-Flight Drill	29

Introduction

1. The air position of an aircraft can be determined by plotting the heading of the aircraft and the air distance flown on a map or chart. This method can provide reasonably accurate results when the heading and airspeed are constant over long periods of time, but where these factors continuously vary the accurate calculation of air position becomes extremely difficult, if not impossible. The requirement to maintain an airplot under all flight conditions, therefore, led to the development of the Air Position Indicator (A.P.I.).

2. The Air Position Indicator, Mk. 1B, is the current model of a series of air position indicators. It is an analogue computer with inputs of true heading and air distance flown, and outputs of air position in latitude and longitude. The input of true heading is obtained from any remote indicating compass using M-type transmission for its output, and the air distance flown is obtained from an A.M.U. The latitude and longitude output of air position is presented on veeder counters.

Theory of Operation

3. An aircraft flying from a point A for an air distance of n nautical miles on a true heading of $\theta°$ will reach an air position B. Examination of Fig. 1 will show that it is possible to calculate the change in the aircraft's air position in terms of north/south and east/west coordinates. Using the conventional terms of northing for movement north or south and easting for movement east or west, we have—

$$\text{Northing} = n \cos \theta \text{ nautical miles}$$

$$\text{Easting} = n \sin \theta \text{ nautical miles}$$

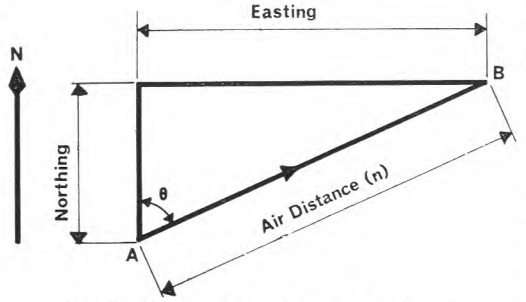


Fig. 1. Coordinates of an Air Position

4. For presentation in terms of latitude and longitude, the northing output can be used to drive the latitude counter mechanism, as the change of latitude in minutes is equal to the northing in nautical miles. The easting output, however, requires modification before it can be presented as change of longitude.

5. From a basic knowledge of the form of the earth it will be known that at any given latitude the length of one minute of longitude is equal to one nautical mile multiplied by the cosine of the latitude. Thus, for latitude λ.

$$1 \text{ n.m.} \times \cos\lambda = 1 \text{ min. of longitude}$$

$$\therefore 1 \text{ n.m.} = \frac{1}{\cos\lambda} \text{ or } \sec\lambda \text{ min. of longitude}$$

$$\text{and } n \sin\theta \text{ n.m.} = n \sin\theta \sec\lambda \text{ min. of long.}$$

6. The air position of B in latitude and longitude results therefore from a solution of the following equations

$$\text{Lat. B} = \text{Lat. A} + n\cos\theta$$
$$\text{Long B} = \text{Long A} + n\sin\theta\sec\lambda$$

and the A.P.I. provides a mechanical means of continuously solving them. The coordinates of point A are manually set on the counters and thereafter the air position of the aircraft is continuously resolved, and displayed on the counters, even though frequent changes of heading or airspeed may be made.

DESCRIPTION

Components

7. The basic components of the Air Position Indicator, Mk. 1B, are as follows:—

(*a*) The compass repeater system

(*b*) Two resolving gears

(*c*) The secant gear

(*d*) The counter mechanism.

These basic components, together with their accessories and interconnections, are mounted on a baseplate and are enclosed by two covers, one above and the other below. All necessary controls are brought out to the face of the instrument

Fig. 2. Air Position Indicator, Mk. 1B

(Fig. 2). The controls for an A.M.U. Mk. 1 or 2 are also fitted to the A.P.I., but where an A.M.U. Mk. 4 is used, a separate control panel is fitted and the A.M.U. controls on the A.P.I. are inoperative (*see* Part 1, Sect. 4, Chap. 5 of this Volume).

Compass Repeater System

8. The M-type transmission of true heading from the remote indicating compass system is received in the compass repeater motor of the A.P.I. This motor drives a gear train which positions the compass repeater pointer and the two resolving gears according to the heading of the aircraft (Fig. 3). A synchronizing shaft, positioned above the compass repeater and operated by the standard compass corrector key, enables the compass repeater system to be synchronized with the remote indicating compass. Two parallel lines and an arrow head on the face of the compass repeater may be set against any desired heading by turning the knob below the dial.

Resolving Gears

9. The two resolving gears are the mechanisms used to produce the sine and cosine functions of heading and air distance. Each consists of a crank which moves, through a linkage, a ball carriage across the face of a disc (Fig. 3). As the crank is rotated by the compass repeater system to accord with the aircraft heading, the ball carriage is moved across the disc a distance proportional to the sine or cosine of the heading. The disc is rotated by the air distance input from the A.M.U., so that the balls in the carriage will rotate at a rate proportional to the air distance multiplied by the sine or cosine of the heading. This rotation is transferred to a roller, in contact with the ball carriage, which will also rotate at a rate proportional to $n \cos \theta$ or $n \sin \theta$ depending upon the function determined by the crank. These resolving gears, or ball and disc resolvers as they are called, are discussed more fully in the chapter on analogue computers.

10. Of the two resolvers, the one providing the output proportional to $n \cos \theta$ is connected, through gears, to the latitude counters, whilst the one providing the output proportional to $n \sin \theta$ is connected to the secant gear. The output rollers of both resolvers, however, are connected to M-type transmitters which transmit the northing and easting output of the rollers to equipments such as the Ground Position Indicator, Mk. 1B, and the Wind Finding Attachment, Mk. 1B, which require inputs in this form.

Secant Gear

11. The secant gear is also a ball and disc resolver, of similar construction to that producing the cosine function. In this case, however, the crank is positioned by a worm drive from the latitude counters. The ball carriage is moved across the disc a distance proportional to the cosine of the latitude set on the counters but, as the drive in this case is from the roller to the disc, the output is proportional to the inverse function, *i.e.* the secant of the latitude. The drive to the secant gear is the output of the sine resolver so the secant gear output is proportional to $n \sin \theta \sec \lambda$, and this output is used to operate the longitude counters.

12. In this type of secant gear, the gear ratio increases rapidly as the ball carriage approaches the centre of the disc until it is theoretically infinite at the centre. There is, however, a limit on any friction drive gear, above which the loading will cause the gear to slip. The ratio at which slipping will start corresponds to a latitude in the region of 70° to 75° N or S, the exact value depending upon the particular instrument. There is no immediate indication that the gear is slipping so that, except when using special techniques, the use of the A.P.I., Mk. 1B, is limited to latitudes less than 70°.

Counter Mechanism

13. The counter mechanism has four veeder counters, one for each cardinal direction, which are geared together in pairs, north/south and east/west. In front of each pair of counters is a shutter system which allows only one counter of each pair to be seen at any one time. When both counters of each pair reach zero the mechanism moves the shutter which then covers the decreasing counters and uncovers the increasing ones.

14. Each counter consists of four drums, two digits for degrees and two for minutes, so that the counter normally reads up to 99° 59′. This range is sufficient for the latitude counters but not for those giving longitude. To overcome this difficulty, the shutter of the longitude counters moves over a further stage, to bring into view a figure 1 engraved on the shutter itself, as the counters turn over from 99° 59′ to 00° 00′. The counters can, therefore, indicate up to 199° 59′, thus allowing

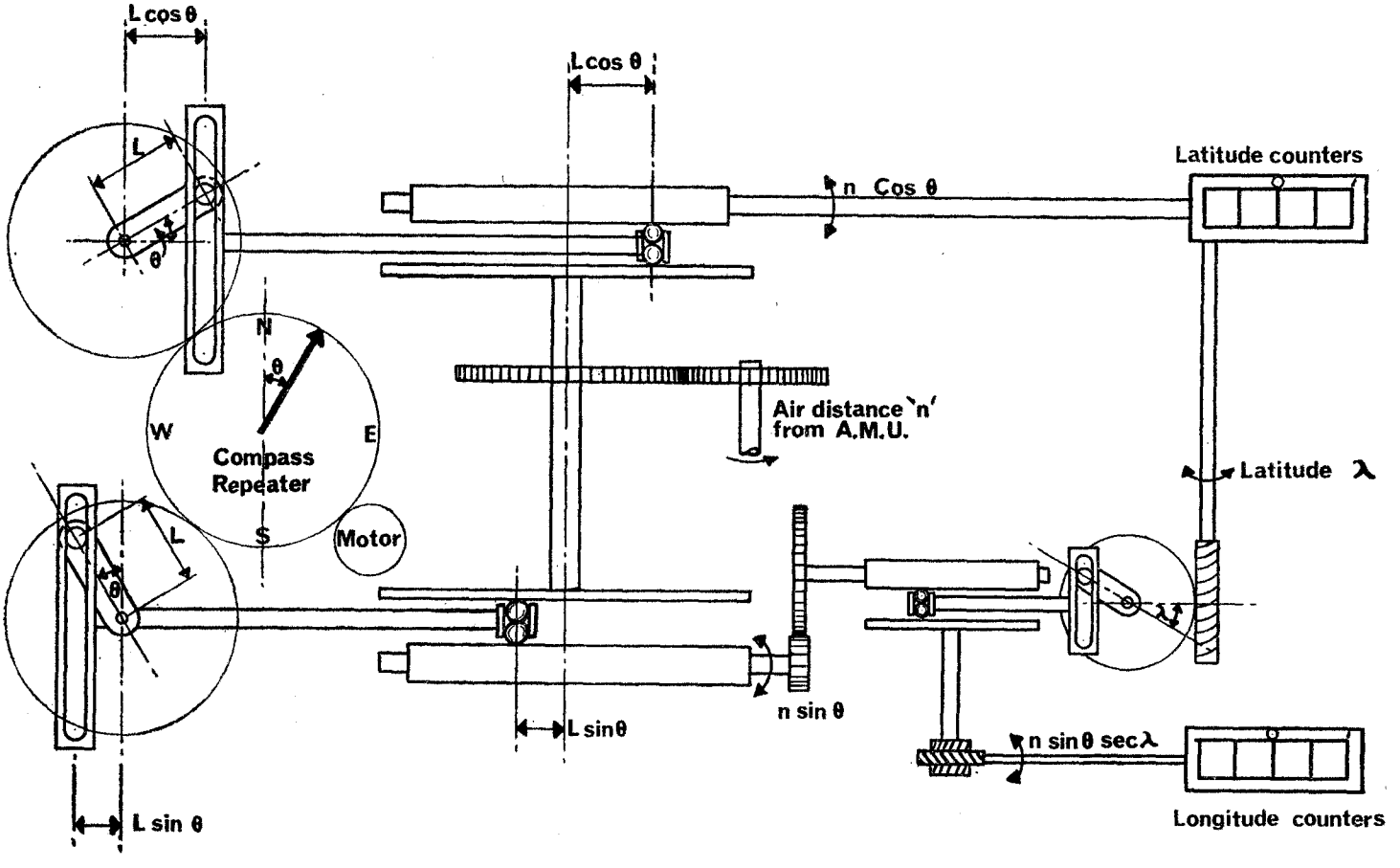


Fig. 3. Air Position Indicator, Mk. 1B—Schematic

a small margin when flying near the Greenwich anti-meridian. There is, however, no changeover from east to west, or *vice versa*, when crossing the meridian.

15. Associated with each pair of counters is a resetting knob which, when depressed, disengages the normal drive to the counters and engages a direct drive from the knob. The counters can therefore be adjusted by depressing and turning the appropriate resetting knob. Each knob is provided with a pointer which registers against a scale to show the amount and sense of the correction applied, one complete rotation of the knob altering the counter reading by 20 minutes of latitude or longitude. Since the A.P.I. does not record on the channel when the knob is depressed, it is essential that resetting is performed as rapidly as is compatible with accuracy. The north/south resetting knob does not disengage the drive from the counters to the secant gear, which is always set according to the latitude counter reading.

16. For clarity of reading, the windows displaying the counter figures are of such a size as to display only one row of figures, and adjacent figures cannot be seen. When the counters are turning over from one figure to the next, therefore, only a part of each can be seen and occasionally they cannot be recognized. An upward movement of the small knob situated between the resetting knobs moves up a slide which increases the width of the windows and the upper figure is uncovered. On release, a spring returns the knob and slide to its normal position. A rotary switch, situated between the synchronizing shaft and the A.M.U. ON/OFF switch on the face of the instrument, operates a rheostat switch which switches on and controls the intensity of the lighting for the counters and compass repeater.

ERRORS AND ACCURACY

Errors

17. The accuracy of the A.P.I. is limited by a number of errors, which can be summarized as follows:—

(*a*) Instrument errors.
(*b*) Slipping of the secant gear.
(*c*) Altitude and latitude errors.
(*d*) Longitude error.

These errors are those of the instrument itself, but it must also be remembered that the inputs to the A.P.I. from the compass system and A.M.U. may have errors. In this case the A.P.I. error will include those input errors.

18. **Instrument Errors.** These errors arise from manufacturing tolerances in the various components, and some may remain, even after calibration. In fact, the acceptable error after manufacture or overhaul is $\pm 1\%$ of the distance flown.

19. **Slipping of the Secant Gear.** As stated in paragraph 12, slipping of the secant gear can cause an error in the conversion of eastings to minutes of longitude if the A.P.I. is used at latitudes greater than 70° N or S. If the instrument is used normally at moderate or low latitudes, and special navigational techniques are used above 70° N or S, then this error will not occur.

20. **Altitude and Latitude Errors.** The design of constant-scale measuring instruments such as the A.P.I. is based upon the assumption that:—

(*a*) At all altitudes and latitudes the length of a nautical mile is 6,080 feet and is equal to one minute of latitude.
(*b*) A distance of 6,080 feet multiplied by the cosine of the latitude is equal to one minute of longitude.

These assumptions are true only at sea level at latitudes 47° 42′ N and S, and at any other altitude or latitude must introduce errors. These errors are maximum at high altitudes and in low and high latitudes, but even then are small enough to be ignored particularly when the A.P.I. is regularly reset to a fix. A full explanation of these errors is given in A.P. 1234B, Section 4, Appendix to Chapter 1.

21. **Longitude Error.** The conversion of eastings to minutes of longitude should be achieved by use of the secant of the ground latitude at which the aircraft is flying. The instrument, however, uses the secant of the air latitude, which introduces an error whenever there is a difference between air and ground latitudes. The size of the error depends on the difference between air and ground latitude and also upon the actual figure of latitude. For this reason, the A.P.I. should normally be reset whenever the difference between air and ground latitudes approaches one degree and the ground latitude is greater than 15° N or S.

22. The errors described in paragraphs 17-21 will all affect the accuracy of the instrument output when it is used in circumstances where these arrors arise. Any errors of heading and air distance inputs will also affect the accuracy of the

A.P.I. output. With reasonably accurate inputs, no secant gear slip and no great difference between air and ground latitudes, the air position on the A.P.I. is accurate to within ±2% of the air distance flown since the last reset.

OPERATIONAL USE

General

23. The actual operational use made of the A.P.I. will vary with the role of the aircraft and the type of navigation employed. Similarly any drills or checks will have minor modifications depending upon aircraft type. Those given in this chapter are intended to serve only as a guide.

Pre-Flight Check

24. The navigator's pre-flight check of his equipment should include the following sequence to test the functioning of the A.P.I.:—

(*a*) Switch on compass.
(*b*) Check that the A.P.I. repeater is synchronized with the compass master indicator.
(*c*) Put on any other switches required by the particular installation.
(*d*) Switch on A.M.U. and allow about 30 seconds for it to warm up.
(*e*) Press A.M.U. ground test push-button and check that the A.M.U. indicator lamp winks regularly, thus indicating that the A.M.U. is functioning.
(*f*) Keep test button depressed and check that the A.P.I. counters rotate in a direction appropriate to the heading.
(*g*) Check lighting of A.P.I. face.
(*h*) Set compass error on V.S.C.
(*j*) Set counters to latitude and longitude of take-off point.

In-Flight Use

25. In flight it is necessary to check that the A.M.U. indicator lamp continues to wink, that the compass operates correctly and that the correct compass error is always set on the V.S.C. Accuracy checks and resetting should be carried out as required.

26. **Accuracy Checks.** Errors in the heading and distance information fed into the A.P.I. may not be readily apparent, but will cause the instrument to register incorrect air positions. The accuracy of the A.P.I. must therefore be checked as soon as possible after take-off and, if the flight is a lengthy one, again at the halfway stage. The check consists of plotting two A.P.I. readings, approximately 60 n.m. apart, while the aircraft is flying a constant heading and true airspeed. The heading is then drawn in from the first air position and along this line is measured the air distance flown between the times of the first and second A.P.I. readings. This will give the manual air position for the time of the second A.P.I. reading, and these two positions should be within 3 n.m. of each other. This procedure should also be carried out if, for any reason, the accuracy of the A.P.I. output is in doubt.

27. **Resetting.** If the A.P.I. is allowed to run for a considerable time the wind vector may become inconveniently long, and any large difference between air and ground latitude may introduce errors (*see* paragraph 21). To maintain accuracy and keep the wind vector within reasonable limits, therefore, it is necessary to reset the A.P.I. Whenever possible the A.P.I. should be reset to a reliable fix, but where this is impossible, it may be reset to an M.P.P. or D.R. position. The most usual method of resetting is by calculation and application of reset factors, *i.e.* the change of latitude and longitude to bring the air position into coincidence with the ground position.

Example:—	1040 Fix	5134N	0328W
	1040 Air position	5111N	0408W
	Reset Factors	23N	40E

When the factors have been calculated and checked, they are then applied by depressing and turning the appropriate resetting knob the required number of divisions, in the correct sense, shown on the scale against the resetting knob. After resetting by this method, the counter readings should be plotted to check that the reset has been accurately done. It should be noted that, although the resetting may be done at any time subsequent to the time of the fix, the airplot has been restarted from the time of the fix.

28. **Faults.** The A.P.I. is a reliable instrument and incorrect operation can usually be traced to the compass or A.M.U. which supply the inputs. However, if the compass and A.M.U. appear to be operating correctly but the A.P.I. counters are not moving, with the transmission switches, if fitted, in the on position, then the flexible drive connections to the A.P.I. should be checked. If the winking lamp goes out but the A.P.I. continues to show changes of latitude and longitude, it is most probable that the lamp has burnt out. The A.P.I. can still be used, but its accuracy should be checked at regular intervals, and the defective lamp replaced before the aircraft is

flown again. There are no other faults which can be rectified in the air.

After-Flight Drill

29. After flight the following actions should be carried out:—

(*a*) Set V.S.C. to zero, unless this has been done on joining the circuit.

(*b*) Switch off A.M.U.

(*c*) Switch off any other switches used in the particular installation.

(*d*) Reset counters to datum required for the next flight.

(*e*) Switch off compass.

CHAPTER 2

WIND FINDING ATTACHMENT, MARK 1B

CONTENTS

	Paras.
Introduction	1– 2
Principle	3– 4
Description...	5–10
Operational Use	11–19
Accuracy	20

Introduction

1. An air plot wind velocity for a given period of time may be obtained by determining the displacement of the aircraft's air position from its ground position during that time. This is normally achieved by plotting air and ground positions on a chart and measuring the vector difference. Such a method, however, cannot produce an acceptable result in a very short period of time using a normal plotting chart, since errors must arise in measuring the small vectors involved, while air and ground positions can themselves be plotted only to the nearest half mile. These errors are greatly magnified when such a small vector is used to calculate the wind effect for an hour. To find an air plot wind velocity over a short period of time therefore requires:—

(*a*) Large-scale plotting.

(*b*) Observation of air and ground positions to a high degree of accuracy.

(*c*) Precise timing of observations.

2. The A.P.I. is sufficiently accurate to determine the displacement of the air position from the ground position after a run of only three or four minutes. Unfortunately, the A.P.I. counters can be read only to the nearest half mile, which is too inaccurate. An attachment, with a sufficiently open scale, is therefore provided, to enable the displacement to be indicated with the required degree of accuracy. This attachment is known as the Wind Finding Attachment, Mk. 1B (W.F.A.).

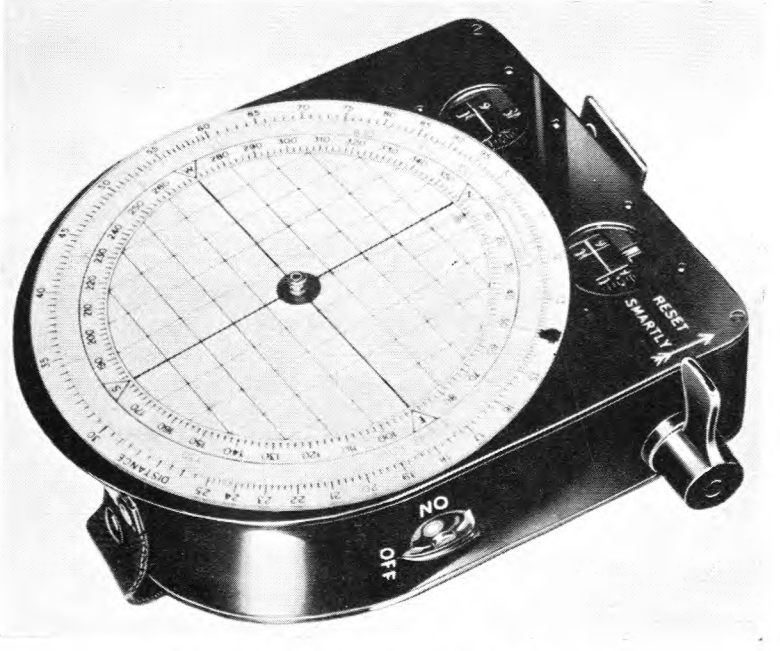


Fig. 1. Wind Finding Attachment, Mk. 1B

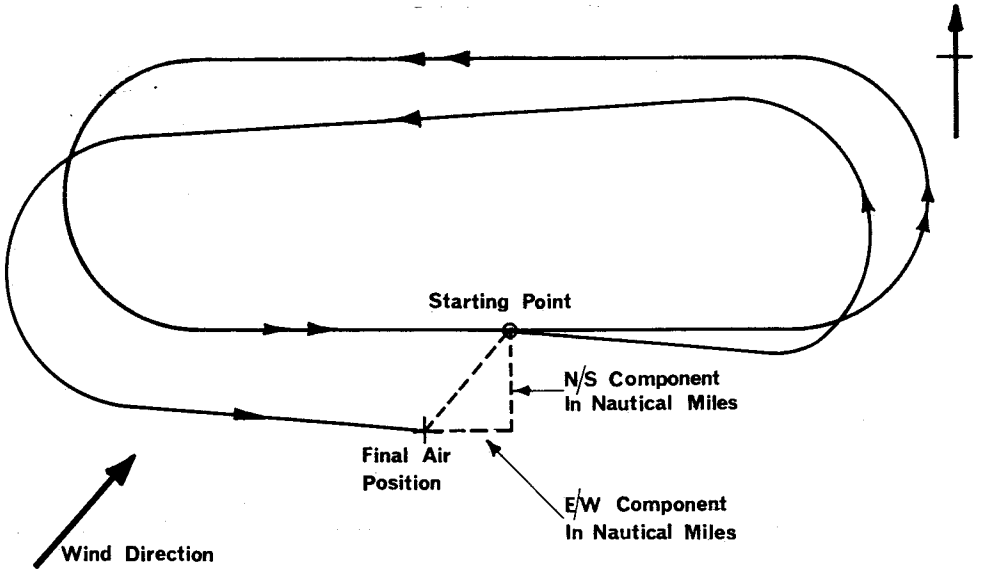


Fig. 2. Principle of Wind Finding Attachment

Principle

3. If an aircraft completes a closed ground circuit, its change in air position during the time of the circuit represents the wind effect for that time (Fig. 2). This vector, representing the wind effect, can be resolved into N/S and E/W components, representing the coordinates of the final air position, based upon the starting (and finishing) point. Similarly, if the changes in northings and eastings from an A.P.I. are plotted, then their resultant will be the wind effect for the time of the circuit.

4. The W.F.A. is designed to operate on the outputs of the M-type transmitters driven by the resolving gears of the A.P.I. These outputs of northings and eastings are indicated directly on the counters of the W.F.A. for the period of time for which the W.F.A. was switched on, *i.e.* the time of the circuit. The face of the W.F.A. carries a combined plotting disc and computer enabling the coordinates to be plotted and the wind direction and speed determined.

Description

5. The Wind Finding Attachment, Mk. 1B, takes the form of a box, weighing approximately 3 lb., connected to the A.P.I. by a length of electrical cable. When in use the instrument may be held in the hand, and at other times placed in the stowage bag without removing the connection to the A.P.I. To facilitate holding the W.F.A. a webbing strap is provided which is secured to two metal studs on the case, a third stud giving one alternative position for the strap.

6. On the face of the instrument are two sets of counters which are driven by the W.F.A. repeater motors. These motors, which are of the Desynn type, are electrically connected to the M-type transmitters in the A.P.I. A two-position rotary switch on the side of the W.F.A. controls the supplies to the repeater motors from the A.P.I. transmitters.

7. Each set of counters consists of two calibrated drums and a shutter mechanism. One set is marked SN and the other WE, the shutter operating to cover one letter on each set thus indicating, by the exposed letter, the direction of the coordinates. Engraved on the shutter is a datum line, against which the counters may be read. One counter drum is calibrated in nautical miles and the other has ten divisions representing tenths of a nautical mile. Each tenth division is divided into five, representing fiftieths of a nautical mile, but interpolation to one hundredth of a nautical mile may easily be made. Thus the counter reading shown in Fig. 3 is 2.75 n.m. north of the datum point.

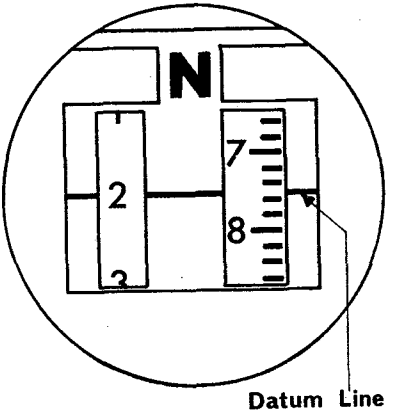


Fig. 3. Counter Reading

8. The counters can be reset, independently of the A.P.I., by moving the resetting lever, on the side of the W.F.A., to the limit of its travel and then releasing it. An arrow engraved on the case by the lever shows the correct operating direction. The resetting movement should be carried out briskly, otherwise the counters may fail to reach the null position. A check should always be made, therefore, to ensure that the dots engraved on the decimal counter are adjacent to the datum line on the shutter. It is important to reset the counters every time the W.F.A. is used, not only to avoid subtraction of the previous reading but to avoid introducing an error should the nautical mile counter drum turn unnoticed from 9 through 0 to start again at 1.

9. When the counters are reset they, in fact, register a negative value of 0.05 n.m. (Fig. 4). For mechanical reasons the two reciprocal decimal scales are staggered to the extent of 0.1 n.m., so that whichever way the counter drum rotates a N or S (or E or W) component of 0.05 n.m. is flown before the counters read zero. For a three-minute circuit this factor could cause an error in wind speed of approximately 1.4 knots. This error may be remedied by adding 0.05 n.m. to both counter readings before computing the wind.

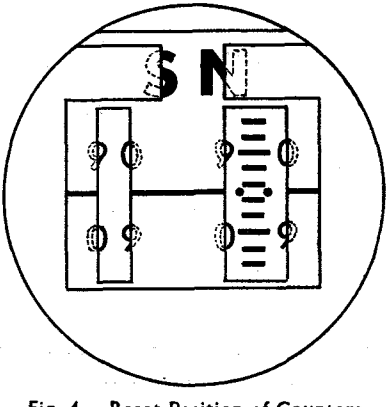


Fig. 4. Reset Position of Counters

10. Below the counters on the face of the instrument is mounted the computer for determining the resultant wind speed and direction. It consists of:—

(*a*) A fixed base plate, bearing around its circumference a time scale and two index marks against which the wind speed may be read in knots and m.p.h. A graduated lubber line with an arrow head runs from the centre to the top of the plate.

(*b*) A rotatable, matt-surfaced, perspex disc concentric with the base plate. A distance scale is marked around the circumference of the disc and is read against one or other of the index marks on the base plate. The central area is gridded and surrounded by a compass rose.

Operational Use

11. Exact timing is essential during a wind finding operation with the W.F.A. and it is advisable to use a stop watch, although a navigator's watch should be satisfactory if used carefully. The A.P.I., and in consequence the W.F.A., will not record accurately during sideslip and it is therefore essential that correctly banked turns are made during the wind finding circuit. It is recommended that rate 1 turns are not exceeded and that a circuit time of 3 to 4 minutes is employed.

12. Since the aircraft must be flown accurately over the datum point at the start and finish of the circuit, an accurate assessment of the vertical must be made. It has been found that an experienced observer can judge the vertical to within 5 degrees, the error in wind speed caused by this inaccuracy being small enough to be ignored below 5,000 feet. For example, a 5-degree error at 5,000 feet can produce an error of 1.7 knots in a wind found after a three minute circuit. Since the error in wind speed is directly proportional to the height, it is recommended that above 5,000 feet a vertical sighting device is used.

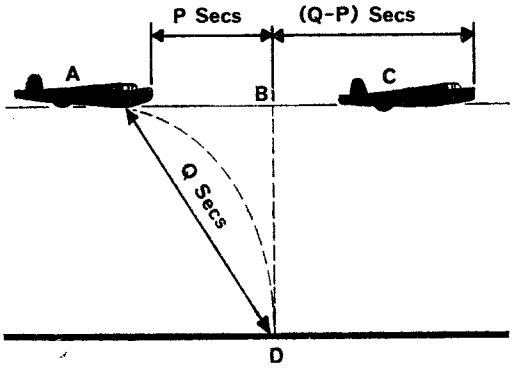


Fig. 5. Fall of Marker in Still Air

13. Over land it is normally easy to select some prominent object on or near track as a datum for the start and finish of the circuit, and the procedure is straightforward. Over the sea, however, the use of some form of sea marker is necessary and the procedure is more complicated. To turn the aircraft back after dropping the marker, in order to start the circuit over it, would involve a certain loss of time. To obviate this, time delay tables have been compiled for various markers, by means of which the circuit time may be adjusted to allow for the ballistic properties of the marker. The procedure is to start a stop watch when the marker is dropped, and start the W.F.A. *P* seconds later. The circuit is then continued until the aircraft is vertically above the marker, when the watch and W.F.A. are stopped. The circuit time is then found by deducting *Q* seconds from the time recorded on the stop watch. An example of a time delay table giving the *P* and *Q* factors is given in Table 1 at the end of this chapter.

14. The necessity for this procedure may be explained by considering three cases; still air, headwind and crosswind. Fig. 5 illustrates the case in still air. The marker is released and the stop watch started when the aircraft is at A. After *P* seconds the aircraft is at B, vertically above the point of impact of the float, and the W.F.A. is started. The marker, however, takes *Q* seconds to arrive at D, its point of impact, by which time the aircraft is at C.

15. Now consider the effect of a headwind (Fig. 6). When the W.F.A. is started after *P* seconds the aircraft will have been displaced from B to B_1 by the wind effect for *P* seconds, while after *Q* seconds the point of impact of the marker will have been displaced from D to D_1 by the wind

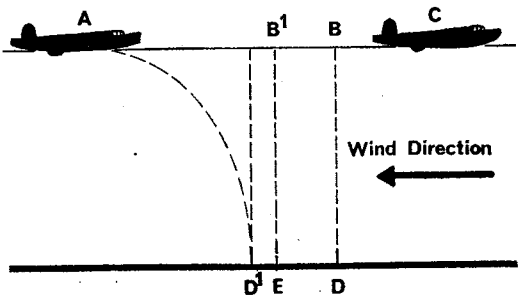


Fig. 6. Effect of Headwind

effect for *Q* seconds. The circuit is therefore begun over E and completed over D_1, resulting in an undermeasurement by the W.F.A. equal to the wind effect for (*Q*—*P*) seconds. To compensate for this, it is necessary to deduct (*Q*—*P*) seconds from the time of the circuit. Since the stop watch was started *P* seconds before the W.F.A., it is therefore necessary to deduct a total of (*Q*—*P*)+*P* *i.e.* *Q*, seconds from the recorded stop watch time.

16. This correction applies equally to the case where a crosswind is experienced (Fig. 7). If the marker is released at point A, in still air its point of impact would be X. The aircraft would reach X after *P* seconds, while the marker would take *Q* seconds. The wind effect for *P* seconds will displace the aircraft to B where the W.F.A. is switched on, while the wind effect will displace the marker to D where the circuit is completed. The effect of this is to displace the final air position downwind of B by a distance BD which is equal to (*Q*—*P*) seconds wind effect. The wind found, whilst correct for direction, would be too small unless (*Q*—*P*) seconds were deducted from the circuit time. Since the stop watch was started *P* seconds early, a total of (*Q*—*P*)+*P*, *i.e.* *Q*, seconds must be subtracted from the recorded stop watch time to obtain the correct circuit time.

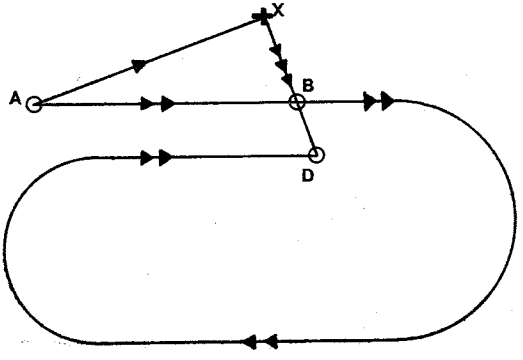


Fig. 7. Effect of Crosswind

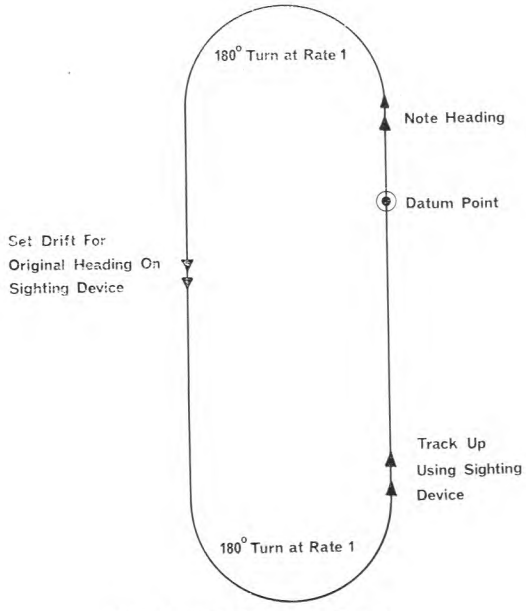


Fig. 8. Wind Finding Circuit

17. **Use of the W.F.A. over Land.** Over land the circuit procedure is as follows:—

(*a*) Check that the A.P.I. is synchronized with the compass repeaters.

(*b*) Check that the W.F.A. switch is in the OFF position and operate the re-setting lever to zero the counters.

(*c*) Select a prominent ground feature for use as a datum point and notify the pilot of the intention to find a W.F.A. wind with this object as a datum point.

(*d*) Track the aircraft over the datum point and when the aircraft is vertically above the point switch on the W.F.A. and start the stop watch.

(*e*) Note the aircraft heading and instruct the pilot to turn through 180° at rate 1.

(*f*) Maintain the reciprocal of the original heading for 45 seconds and while doing so set the drift for the original heading on the sighting device.

(*g*) Turn back onto original heading at rate 1 and track up to the datum point.

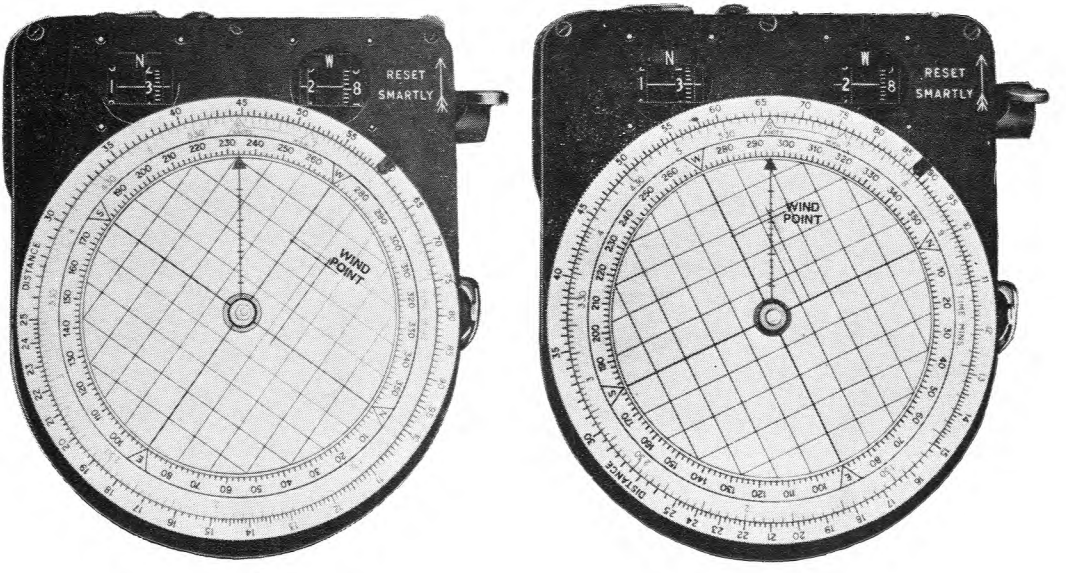
(*h*) When the datum point is vertically beneath the aircraft switch off the W.F.A. and stop the stop watch.

(*j*) Use the counter readings and circuit time to find the wind, as detailed in paragraph 19.

18. **Use of the W.F.A. over the Sea.** Over the sea the procedure is as follows:—

(*a*) Check that the A.P.I. is synchronized with the compass repeaters.

(*b*) Check that the W.F.A. switch is in the OFF position and operate the re-setting lever to zero the counters.



(*a*) (*b*)

Fig. 9. Plotting Coordinates and Reading Wind Velocity

(*c*) Select a marker and extract the *P* and *Q* factors from the relevant tables. Inform the pilot of the intention to fly a wind-finding circuit.

(*d*) Release the marker and start the stop watch. After *P* seconds start the W.F.A. Note the heading and instruct the pilot to turn through 180° at rate 1.

(*e*) Maintain the reciprocal of the original heading for 45 seconds and while doing so set the drift for the original heading on the sighting device.

(*f*) Turn back on to original heading at rate 1 and track up to the datum point.

(*g*) When the datum point is vertically beneath the aircraft switch off the W.F.A. and stop the stop watch.

(*h*) Subtract *Q* seconds from the recorded stop watch time to obtain correct time for circuit.

(*j*) Use the counter readings and circuit time to find the wind, as detailed in paragraph 19.

19. **To Determine Wind Velocity.** The counter readings and circuit time are used to determine wind velocity as illustrated in the following example:—

Example: To find wind velocity given counter readings

1.3N and 2.81W

circuit time 4 minutes 14 seconds.

(*a*) Add 0.05 to both counter readings (*see* paragraph 9). Corrected readings are now 1.35N and 2.86W.

(*b*) Rotate computer until N is against the arrowed lubber line. Plot the N reading up the lubber line by drawing a horizontal line 1.35 divisions from the centre of the computer.

(*c*) Rotate computer until W is against the arrowed lubber line. Plot the W reading up the lubber line by drawing in a horizontal line 2.86 divisions from the centre of the computer.

(*d*) Rotate the computer until the intersection of the two drawn lines, the wind point, is over the lubber line. Read off the wind direction, 295°, against the arrow head (Fig. 9(*b*)).

(*e*) Read the wind effect where the wind point coincides with the lubber line, *i.e.* 3.15.

(*f*) Set the circuit time on the time scale against the wind effect on the distance scale, *i.e.* 4 mins 14 seconds against 3.15 (Fig. 9(*a*)).

(*g*) Read off wind speed against knots pointer, *i.e.* 44.7 knots.

The wind velocity is therefore 295°/44.7 knots.

Accuracy

20. The instrumental accuracy of the W.F.A. is approximately $\pm\frac{1}{2}$ knot, after allowing for the counter construction error mentioned in paragraph 9. Errors in timing the circuit, starting and stopping the W.F.A., and inexact tracking over the datum point will all introduce errors in the wind velocity found. These errors can be kept to a minimum by careful operation, and the accuracy of the wind so found can be expected to be of the order of $\pm 1°$ and $\pm 2\frac{1}{2}$ knots. The wind found, unlike normal air plot winds, is a local wind.

Time Delay Table for Use with Flame Floats of 300ft/sec TV								
Height (feet)	Time Delay (factor *P*) for IAS (knots)							Time of Fall (factor *Q*)
	100	120	140	160	180	200	220	
1,000	7	7	6	6	6	6	6	9
1,500	7	7	7	7	7	6	6	11
2,000	9	9	8	8	8	8	7	13
2,500	9	9	9	9	8	8	8	15
3,000	10	10	9	9	9	9	8	17
3,500	10	10	10	10	9	9	9	19
4,000	11	11	10	10	10	9	9	20
4,500	11	11	10	10	10	10	9	22
5,000	12	11	11	11	10	10	10	24
6,000	12	12	11	11	11	10	10	27
7,000	13	12	12	11	11	11	10	30
8,000	13	13	12	12	11	11	11	33
9,000	13	13	12	12	12	11	11	36
10,000	14	13	13	12	12	11	11	39

TABLE 1. EXAMPLE OF TIME DELAY TABLE

CHAPTER 3

GROUND POSITION INDICATOR, MARK 1B

CONTENTS

	Paras.
Introduction	1– 2
Principle	3– 4
Wind Vector Mechanism...	5– 8
Graticule Mechanism	9
Traverse Mechanism	10–12
Installation...	13
Controls	14
Chart Alignment	15
Pre-Flight Checks	16
Operation in the Air	17
Accuracy	18–19

Introduction

1. The Ground Position Indicator, Mk. 1B, is designed to maintain automatically and continuously a dead reckoning track plot by compounding a manually set wind velocity with an air position electrically transmitted from the Air Position Indicator, Mk. 1B. It projects on to a chart fixed to the navigator's table, an illuminated graticule, the position of which indicates ground position, and the orientation of which shows aircraft heading.

Principle

2. The instrument is designed as a small-area navigation aid and is used principally for tactical work such as patrols and searches. It can be used with either a 1:72,915 or a 1:500,000 scale chart (conical orthomorphic projection). The equipment is shown in Fig. 1(*a*) and the principle of operation is depicted schematically in Fig. 1(*b*).

3. The instrument is installed at a fixed height above the navigator's table and is connected electrically to the A.P.I., Mk. 1B. Two electrical transmitters in the A.P.I. drive corresponding repeater motors in the G.P.I. according to the air mileages in a N-S and E-W direction. At the same time, a vector resolving mechanism, manually set to the wind speed and direction, drives two shafts at rates proportional to the wind mileage N-S and E-W. Differential gears combine the air and wind mileages to give N-S and E-W D.R. ground mileages.

4. A mirror in the optical projection system is rotated about one axis in proportion to the N-S ground mileage and about another, at right-angles to the first, in accordance with the E-W ground mileage. The graticule image is then reflected from this mirror and traverses the map according to the D.R. position.

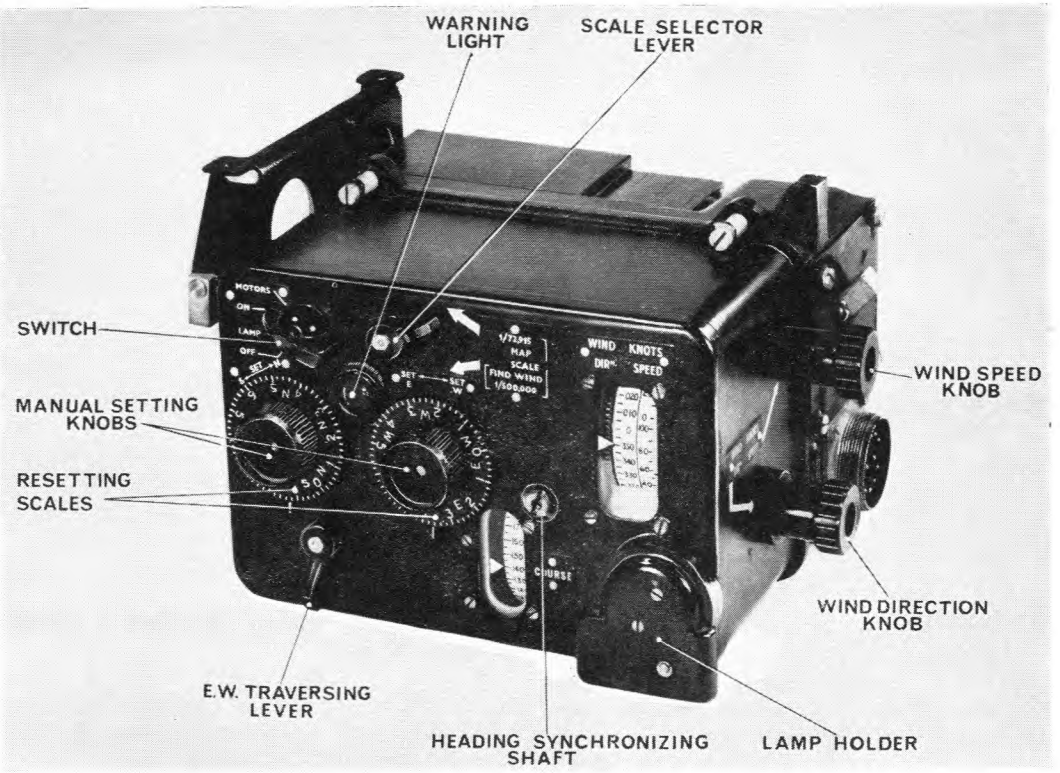
Wind Vector Mechanism

5. This mechanism comprises the wind vector resolving mechanism and the variable speed gears. Rotary knobs are provided for setting wind speed and direction. The mechanism is shown in Fig. 2.

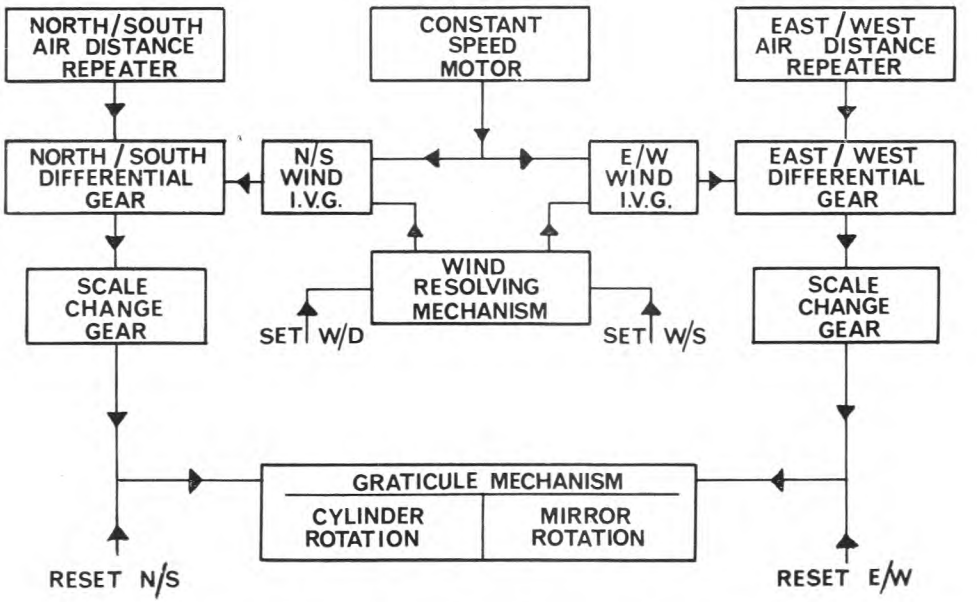
6. The wind velocity is resolved into its two components by a crank mounted on a drum. In Fig. 3(*a*), the crank pin, A, is positioned a distance from the centre of the drum proportional to the wind speed OA. The drum is rotated from its north datum direction, ON, through the angle of wind direction NOA. Thus the crank pin is displaced from the centre datum as follows:—

(*a*) In a direction parallel to the north datum equal to wind speed X cosine wind direction, *i.e.* N-S wind component, OC.

(*b*) In a direction perpendicular to the north datum equal to wind speed X sine wind direction, *i.e.* E-W wind component, AC.



(*a*) General View



(*b*) Operation—Schematic

Fig. 1. G.P.I., Mk. 1B

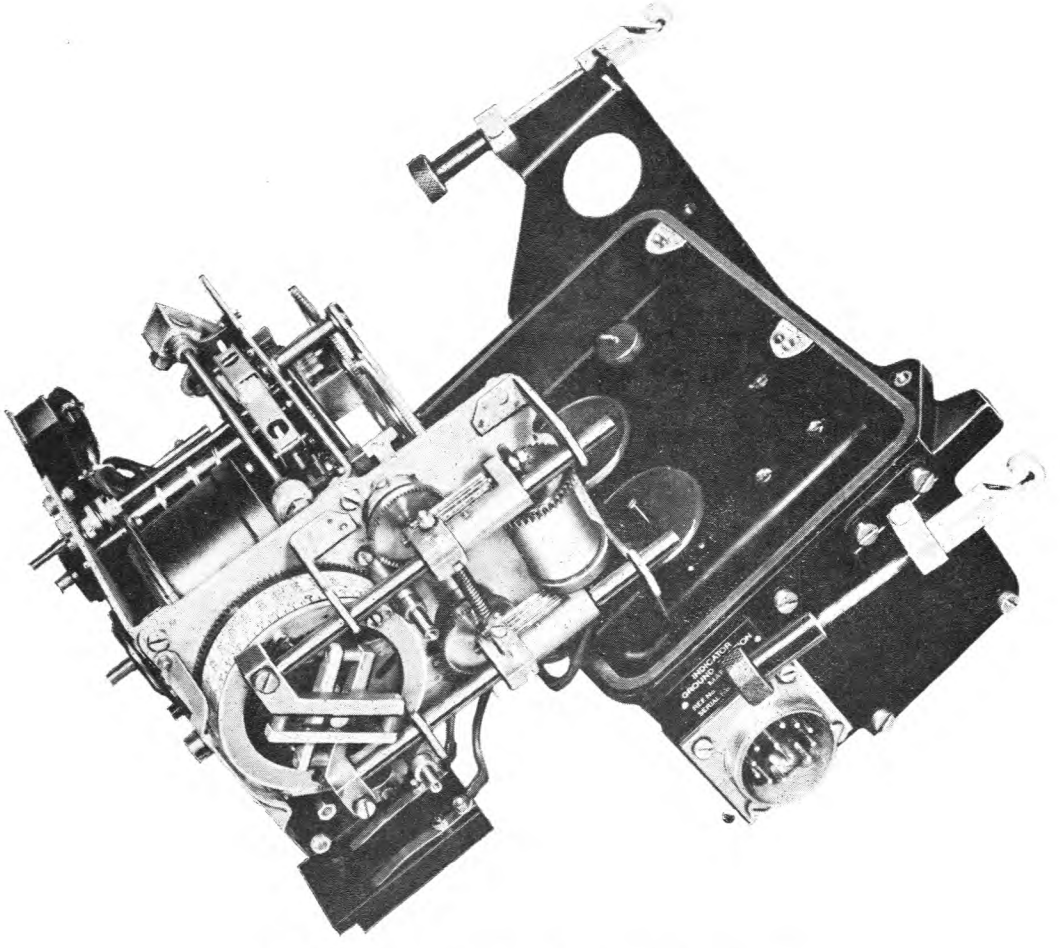


Fig. 2. Wind Vector Mechanism and Variable Speed Gears

7. Two take-off slides fit over the crank pin as shown in Fig. 3(*b*) and are mounted so that they may move only to left or right. When the N-S slide is displaced from the zero position to the left or right a distance X proportional to the N-S wind component, the E-W slide is displaced to the left or right of the zero position a distance Y proportional to the E-W wind component.

8. An electric motor drives two discs at a constant speed. On each disc a smaller disc rests edge-on at a position determined by the take-off slides of the wind resolving mechanism. The revolutions of the smaller discs are thus proportional to the N-S and E-W wind mileages respectively and constitute the wind components. The dimensions of the mechanism and the speed of the constant-speed discs are calculated to provide wind mileages up to a maximum speed of 100 knots. The arrangement is shown in Fig. 4.

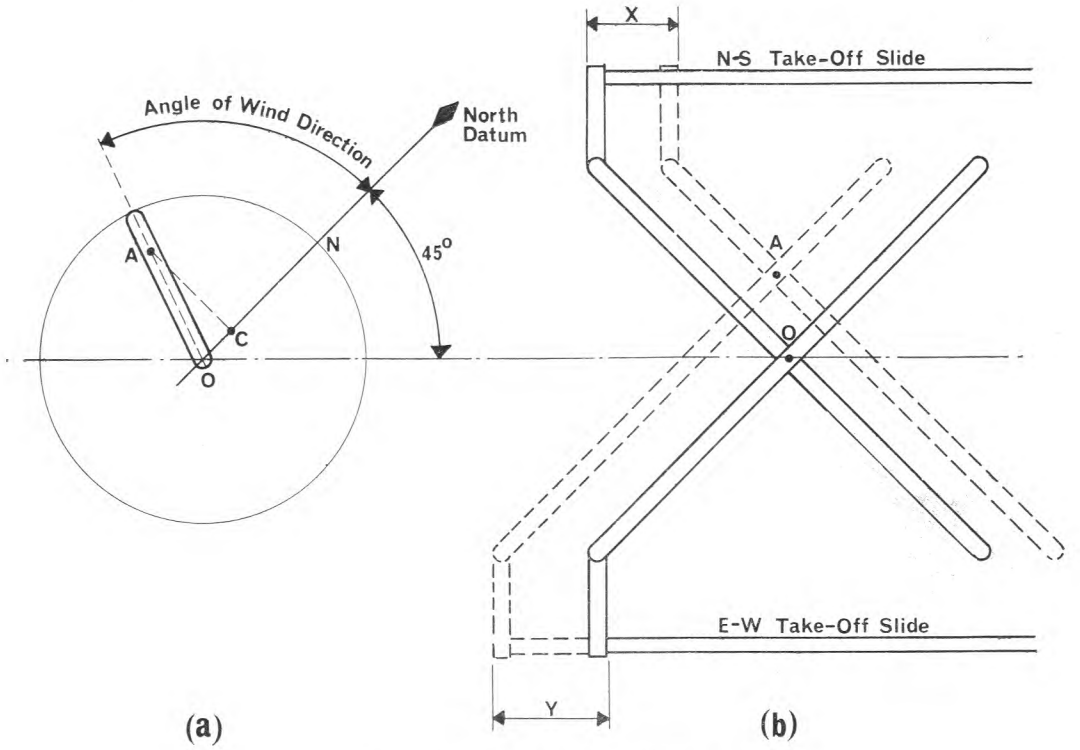


Fig. 3. Wind Resolving Mechanism

Graticule Mechanism

9. The graticule is electrically illuminated and focussed by a lens for projection *via* a mirror on to the chart. It is projected centrally along a cylinder which can rotate about its axis, this axis being parallel to the east-west direction of the chart. In the cylinder a mirror is mounted so that it can rotate about the horizontal diameter of the cylinder. As the cylinder rotates, the mirror axis will remain in a vertical plane which will be N-S across the chart. As the mirror rotates about its N-S axis the graticule will move E-W across the chart. The graticule is orientated by a compass repeater so that when projected on to the chart it indicates aircraft heading. The arrangement is shown in Fig. 5.

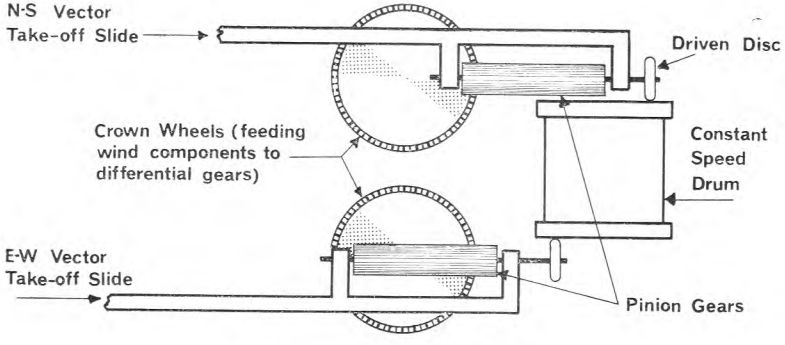


Fig. 4. Wind Mileage Mechanism

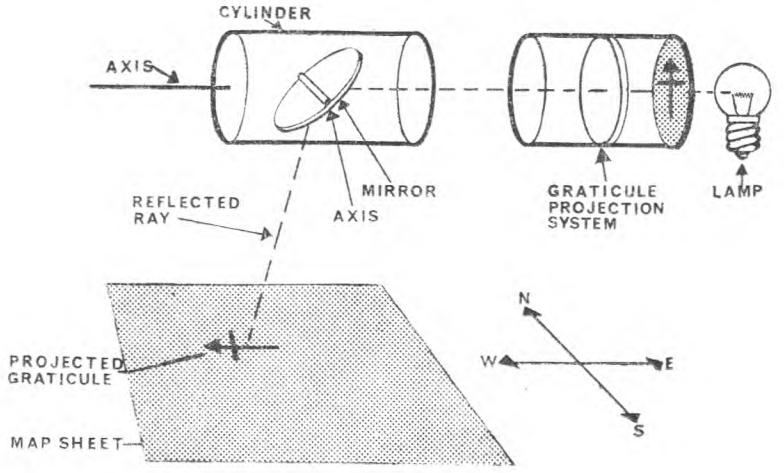


Fig. 5. Projection System

Traverse Mechanism

10. **N-S Traverse Mechanism.** The N-S ground mileage rotation, through a worm drive and block, operates a tangent arm and rotates the cylinder causing the graticule to traverse the map in the N-S direction. The operative edge of the tangent arm is at right-angles to the mirror axis and this edge, the cylinder axis and the normal to the mirror are all in one plane (Fig. 6). The light reflected from the mirror therefore swings through the same angle as the cylinder is rotated and the graticule traverses a distance on the map exactly proportional to the movement of the worm-driven block.

11. **E-W Traverse Mechanism.** The E-W ground mileage similarly operates a tangent arm through a worm drive and block. Since it is required that the light reflected from the mirror will turn through the same angle as the tangent arm the rotation of the mirror must be half that of the tangent arm (a reflected ray moves through twice the angle the mirror is turned). This halving is achieved by a lever system (Fig. 7) as follows:—

(*a*) In its central position the plane of the mirror is at 45 degrees to the axis of the projector, and an arm terminating in a flat plate is rigidly attached to the back of the mirror. This plate is spring-loaded onto a small ball mounted in a pivoted bracket, which in turn is spring-loaded on to an adjusting screw in a second bracket pivoted about the same axis as the first. This second bracket is rotated by the E-W tangent arm.

(*b*) The height of the centre of the ball is adjusted to be on the axis of the projector when the mirror is at 45 degrees to this axis. The distance between the centre of the ball and the mirror pivot axis is twice that between the centre of the ball and the ball pivot axis. The mirror thus turns through half the angle turned through by the ball bracket, *i.e.* half the angle turned through by the E-W tangent arm.

12. The E-W traverse of the graticule is affected by the N-S traverse movement. A given angular rotation of the mirror causes the graticule to traverse a smaller map distance when projection is oblique N-S than when projection is vertical. The tangent arm and ball are pivoted in the frame of the instrument but the mirror and plate are pivoted in the cylinder. When projection is vertically downwards rotations of the mirror and the tangent arm are in the same plane, but if projection is to the north or south of the central position the two rotations are in different planes. The mirror will then be rotated through less than half the angular movement of the tangent arm. The effect can be seen in Fig. 8 which is an end-on view of Fig. 7 but with the cylinder now rotated through an angle $\emptyset$ degrees about the E-W axis. While the ball has moved a distance AB from its rest position the plate has moved a distance AD, where AD is less than AB. The error introduced amounts to approximately:—

(*a*) 1.2 n.m. in travelling 108 n.m. from the centre to the corner of a 1:500,000 chart.

(*b*) 0.2 n.m. in travelling 16 n.m. from the centre to the corner of a 1:72,915 chart.

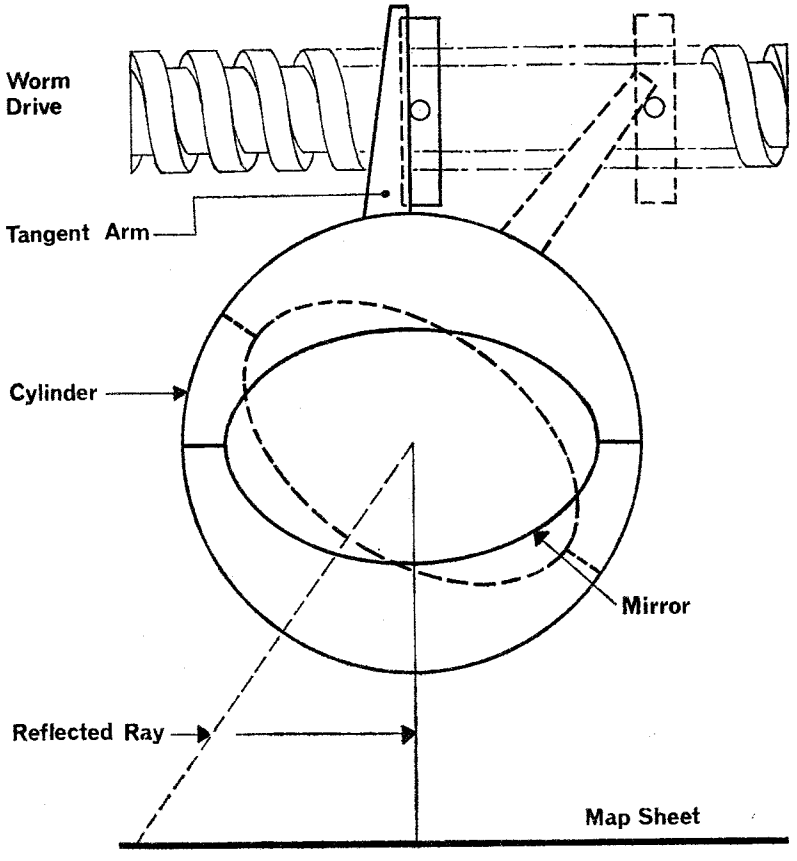


Fig. 6. N-S Traverse Mechanism

Installation

13. In an aircraft the G.P.I. is mounted on a mounting ring and its position is set by three adjustable studs in this ring. The axis of the mirror and cylinder must be 21 inches above the chart. This is achieved by setting the studs on the mounting ring 25.84 inches above the chart table.

Controls

14. The following controls are on the face or right-hand side of the instrument (*see* Fig. 1):—

(*a*) *Function Switch.* This is the main ON/OFF switch and can be set to lamp only, or to lamp and motors, or to motors only.

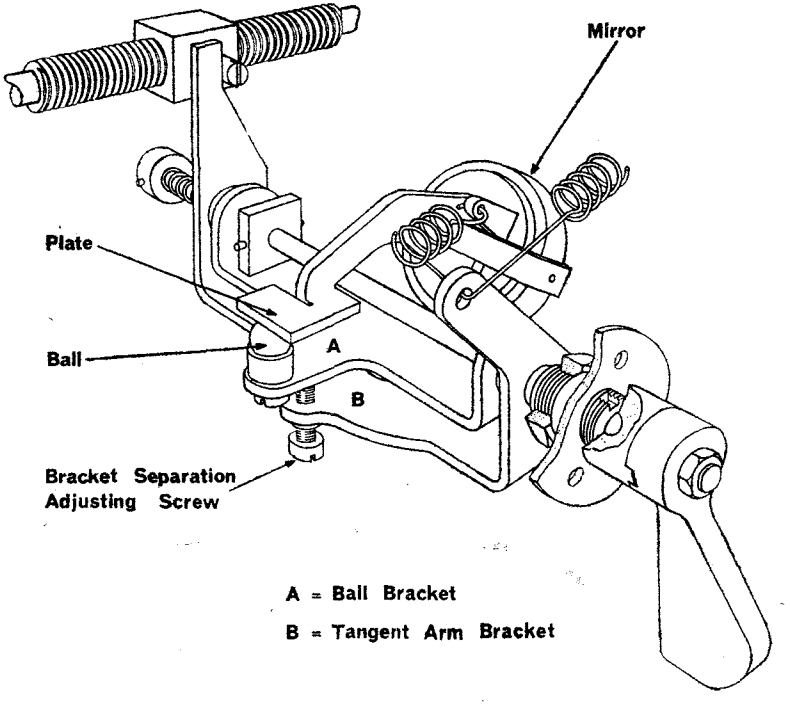
(*b*) *Scale Selector Lever.* This lever changes gears in the gear boxes making it possible to use either a 1:72,915 or 1:500,000 chart.

(*c*) *Heading Synchronizing Shaft.* Synchronization of the heading repeater is achieved by turning this shaft. The repeater controls the orientation of the graticule.

(*d*) *Manual Setting Knobs.* The position of the graticule on the chart may be manually adjusted by means of the N-S and E-W setting controls.

(*e*) *Resetting Scales.* The scales around the manual setting knobs can be used to move the graticule the required number of nautical miles N-S and E-W to reset the graticule to a fix.

(*f*) *E-W Traverse Lever.* This lever is used for rapid traversing of the chart by the graticule during the alignment process.



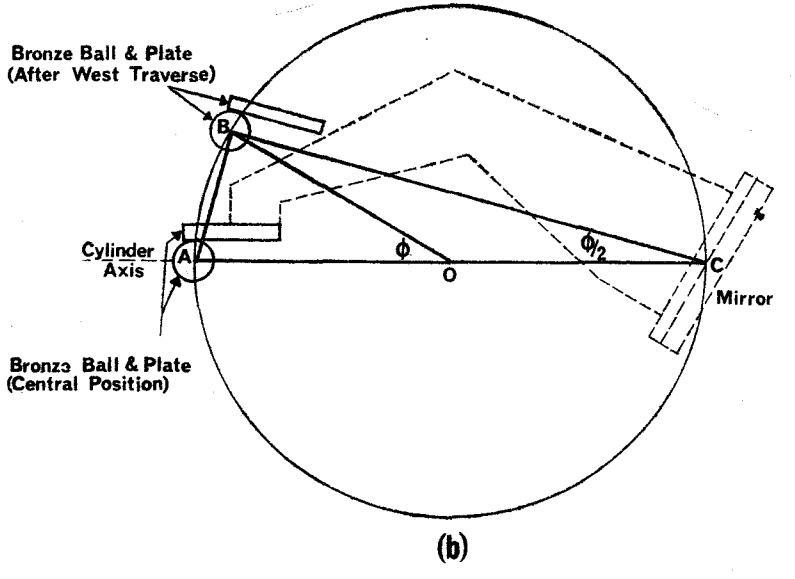


Fig. 7. E-W Traverse Mechanism

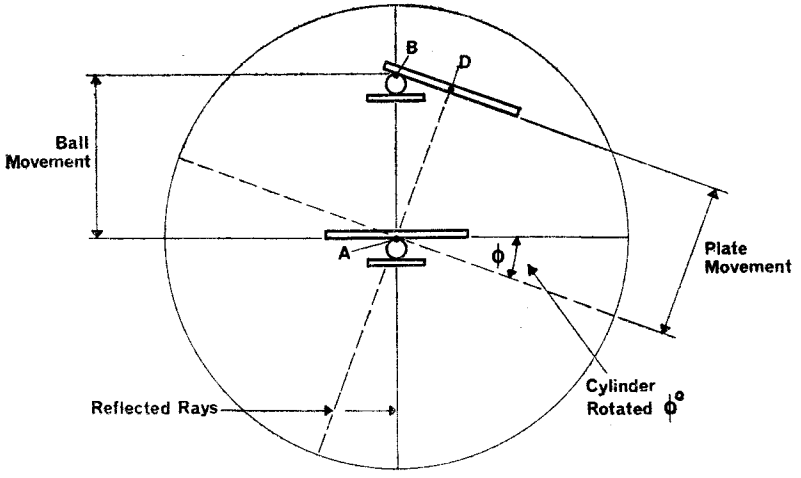


Fig. 8. Effect of N-S Traverse on E-W Traverse

(*g*) *Warning Lamp.* The red warning light is operated when the graticule approaches within approximately 1¼ inches of the coverage limits. There is no storage mechanism so that if the graticule reaches the limiting stops and there is an interval before coverage is again entered, subsequent graticule indications will be in error until resetting to a fix has been carried out.

(*h*) *Lamp Holder.* The holder is easily withdrawn for lamp changing.

(*j*) *Wind Speed Control.* Speeds up to 100 knots may be set. The speed scale is repeated 5 times around the circumference of the dial so that one scale is always visible. The control sets a movable datum (also repeated 5 times) engraved on a ring against the wind speed scale.

(*k*) *Wind Direction Control.* The wind direction scale is engraved on the same drum as the wind speed scale. The setting control moves the scale against a fixed datum.

Chart Alignment

15. The following procedure is recommended:—

(*a*) Switch on the lamp and set the graticule immediately under the G.P.I.

(*b*) Place the chart on the table with the graticule at the centre.

(*c*) Using the traverse lever swing the graticule across the chart.

(*d*) Adjust the position of the chart until the graticule intersects the centre of each side of the chart when traversed as in (*c*) above. Owing to curvature of the parallels the graticule will not exactly follow a parallel across the chart.

(*e*) Pin the chart down.

If only a portion of the chart is to be used set the centre of that portion under the graticule and adjust the chart so that the graticule follows as nearly as possible the central parallel of the portion of chart required.

Pre-Flight Checks

16. (*a*) Switch on the lamp. If the lamp does not light, withdraw the holder (lower right-hand corner of instrument) and fit a new lamp of the same type.

(*b*) Switch on the motor. Check that it is running by listening or by setting a high wind speed and checking that the wind scales turn.

(*c*) Align the chart to be used. If two charts are to be used align the second and secure it and align the first on top.

(*d*) Synchronize the heading scale with the compass.

(*e*) Select the scale required, ensuring that the selector lever is fully home and the gears engaged.

Operation in the air

17. (*a*) Set wind speed and direction.

(*b*) Switch lamp on; leave motor switched off.

(*c*) By means of the graticule setting knobs set the centre of the graticule to the map position of an impending fix.

(*d*) At the time of the fix switch on the motor.

It will be seen that operation from a fix requires prior knowledge of its coordinates. In cases where

such information is not available the G.P.I. may be set to a D.R. position.

Accuracy

18. Traverse error, discussed in paragraph 12, is the only significant instrumental error. The curvature of the parallels and the divergence of the meridians introduce another, non-instrumental error. This error has a value of approximately 2 n.m. on a run from the centre to the corner of a 1:500,000 chart. The result will be an incorrect indication of latitude and longitude.

19. The system error will be a combination of compass, A.M.U. and A.P.I. error, wind error, traverse error and map error. The accuracy to be expected, assuming that an accurate wind velocity is used, is 5% of the distance run.

CHAPTER 4

GROUND POSITION INDICATOR, MARK 4 SERIES

CONTENTS

	Paras.
Introduction	1–2
GPI MARK 4A	
Components	3
Indicator	4–9
Azimuth Drive	10–14
Resolving Mechanism	15–18
Storage Mechanism	19–28
Errors	29–33
Operation	34–37
Flight Planning—Presentation	38
GPI MARK 4	
Description	39–40
GPI MARK 4C	
Description	41–43

Introduction

1. The Ground Position Indicators Mks. 4, 4A, and 4C are electromechanical computers designed for use with Doppler radars and remote indicating compass systems. The inputs are heading by M-type transmission from the heading reference, drift transmitted by synchros from the Doppler aerial, and ground distance gone by M-type transmission from the Doppler tracker unit. The output is the computed position in Northing/Easting, Latitude/Longitude, or Along/Across Track coordinates. A storage system is built into the equipment, permitting the freezing and correction of the position counters.

2. Since the three computers are very similar, this chapter deals primarily with the GPI Mk. 4A, the Mk. 4 is dealt with briefly in paras 39–40 and the Mk. 4C in paras 41–43.

GPI MARK 4A

Components

3. There are two components; an indicator unit and an amplifier unit. The latter is used in conjunction with a servo system to provide a power drive of track angle: it has no controls, is mounted remotely, and is therefore not described in this chapter. All controls are on the face of the indicator (Fig 1).

Indicator

4. **Main Switch.** The ON/OFF switch, which is at the bottom of the unit, controls the 28V dc supply to the motors and transmitters within the GPI. It also controls the transmission of groundspeed to the instrument. Heading transmission becomes operative as soon as the compass is switched on, and transmission of drift and ground

Fig 1 Indicator

mileage occurs a few seconds after the Doppler is switched on.

5. **Heading and Track Indicator.** A compass rose on the face of the instrument carries two pointers. One of these has a double arrow head to indicate track, while the other has a single arrow indicating heading. The pointer indications are accurate to about half a degree. The heading pointer is synchronized with the compass master indicator by the use of the knob at the left-hand side of the compass rose. Synchronization should be carried out with the drift servo powered (*ie*, Doppler switched on). The track pointer then follows the heading, lagging or leading by an amount equal to the drift reading on the Drift and G/S indicator. If the drift servo is not powered the track pointer will remain stationary until it lags by 36° from the heading. A limiting stop in the azimuth synchro unit then engages, dragging the track pointer along 36° behind the heading pointer. When this

happens the drift conditions in the GPI may differ by more than 36° from the drift input from the Doppler *eg*, Doppler drift of 20° port and GPI drift condition of 20° starboard. In these circumstances, as soon as the Doppler is switched on, both pointers will be driven continuously round the dial in one direction. This is because the azimuth resolving mechanism has a null position every 72°, allowing it to deal with normal drifts of up to 36° relayed from the Doppler; but when more than 36° of drift is presented to the mechanism it tries to reach a null position in the wrong direction. The servomotor used in the azimuth resolving mechanism has sufficient power to overcome the power of the heading repeater motor, and therefore both pointers are driven continuously in one direction by the servomotor. To stop this continuous rotation, the drift on the Doppler indicator should be inched until its drift indication is within 36° of that on the GPI; the continuous rotation will stop and the track pointer will be pulled into synchronization. The heading must then by resynchronized.

6. **Presentation Selection Switch.** A three-positioned switch in the centre of the indicator allows a selection of any one of three types of position presentation. Movement of the switch applies the appropriate form of drive to the counters, and adjusts the counter shutters to give the correct presentation. It must be noted that a change of presentation in flight requires a complete counter resetting. The three presentations are:—

a. *GRID.* Position is indicated as nautical miles travelled in north or south and east or west directions from any point at which the counters have been set to zero. The range obtainable on this setting is 4,000 n.m. on each counter, with an overshoot of approximately 200 n.m.

b. *LAT. LONG.* The two counters give degrees and minutes of latitude and longitude. The minutes are read on the right-hand drum against a mark on the shutter, and the other drums give the number of whole degrees. At the zero changeover, the minute mark moves with the shutter, and any reading must be delayed until the changeover is complete. The range on each counter is 400°. Slipping of a secant gear in the resolving mechanism usually causes errors in latitudes above 70° north or south.

c. *A/A SET TRACK.* When A/A is set, the counters appear the same as for GRID, but they give nautical miles gone along and across any track which has been set on the set track drum. The normal north/south counters record distance along the selected track, the counters increasing north; the east/west counters record distance across track, increasing east for distance to the right of the set track and west for distance to the left. The east/west counters are marked R (right) and L (left) respectively.

7. **Set Track Drum and Knob.** A drum marked off at two-degree intervals, and numbered every 10°, appears through a window in the lower right-hand corner of the GPI face. The drum is rotated by a setting knob near the window, one revolution of the knob representing 4°. The knob is divided into four by projecting points at 90° intervals, and setting this knob scale against a fixed datum mark above the knob allows very accurate track setting. The track set on this control does not affect the track pointer on the compass rose, which continues to give the true track of the aircraft. A locking device prevents movement of the drum from its zero position unless the A/A presentation is chosen and also prevents return of the presentation switch to GRID or LAT. LONG. until the drum has been returned to zero. Care should be taken when change of presentation is made, since the locking device can be forced and this will result in unserviceability of the GPI.

8. **Counter Resets.** Resetting of the counters can be carried out either manually or with the aid of an electric motor. A knob alongside each set of counters controls both

manual and electrical resetting, the knob being pulled out for manual control. On manual resetting, rotation of the knob gives a corresponding rotation of the indicator dials. With electrical resetting, rotation of the knob gives a rate of reset of about 700 n.m. (on A/A and GRID presentations) or about 70° (on LAT. LONG. presentation) per minute. This high rate of reset permits a rapid change of the counters when selecting a different position presentation.

9. **Storage System.** The storage system is controlled by the NORMAL/FIX switch, two hand reset knobs and two drum indicators which are illuminated by concealed lamps at all times when any information is in store. To store information, the NORMAL/FIX switch is turned to FIX, whereupon the indicator lamps light, and the position shown on the position counters remains constant. Counter resetting can then be carried out. When the switch is put back to NORMAL, electric motors automatically feed all the stored mileage into the position counters until the storage is cleared, when the indicator lamps will go out. Should the automatic system fail, manual unstoring can be carried out. A handle, shown in its stowed position in Fig 1, is inserted in the upper keyway (N/S). The cross and dot indicator nearest to this keyway shows the direction in which the handle must be turned, and a line dividing dots from crosses indicates the neutral position. The handle is then inserted in the lower keyway (E/W) and a similar procedure carried out with reference to the other cross and dot indicator.

Note.—The storage indicator markings cover a range of 160 n.m. for each component, although in effect 200 n.m. can be stored, corresponding to ½ revolution of the indicator drum. Should 200 n.m. be exceeded, the counter affected would be 400 n.m. in error after unstoring (*see* para. 27). The rate of automatic unstore is equivalent to approximately 3,000 knots.

Azimuth Drive

10. The azimuth unit combines heading from the compass with drift from the Doppler to give a power drive representing track. It comprises a differential synchro control transmission system (*see* Sect. 1. Chap. 1) with the transmitter in the Doppler aerial array and the receiver in the GPI Mk. 4A conditions in the receiver are modified by signals fed into it from the compass.

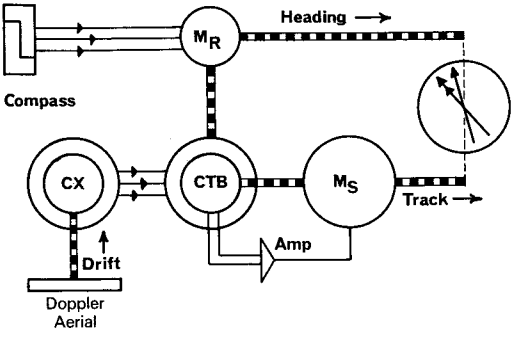


Fig 2 Azimuth Drive—Schematic

11. **Drift.** Drift is determined by the direction of the aerial array relative to the aircraft fore-and-aft axis, and is transmitted by a synchro transmitter (CX in Fig 2) to the GPI Mk. 4A. The signal is received by the stators of a differential control transformer (CTB in Fig 2), the rotor of which is maintained in the null position by its associated servomotor (MS). The CTB rotor is thus always positioned relative to the stators by the drift angle. To improve sensitivity and accuracy the transmission of drift is geared up so that one revolution of the CTB rotor represents 72° of drift.

12. **Heading.** An M-type transmission of heading from the compass is received by a repeater motor (MR) in the GPI Mk. 4A and, through a 60:1 gearing is presented on the compass rose on the face of the instrument. The motor output is also transferred through suitable gearing to rotate the

CTB stator by the angle of heading in such a way that one revolution of the stator represents 72° of heading.

13. **Track.** The direction which the CTB rotor takes up relative to the stator is the angle of drift. Since however, the stator is itself turned relative to a datum on the instrument case by the heading angle, so the CTB rotor position relative to this datum must be heading ± drift, *ie* track. The rotor position is transferred through suitable gearing for indication on the compass rose by the track pointer.

14. **Azimuth Synchro Stops.** In certain circumstances (*eg* Doppler switched off) in which the CTB servomotor is not powered, the CTB rotor would remain stationary while the stator responding to heading changes, moved. The rotor and stator could become desynchronized from one another by complete revolutions, and on power eventually being switched on the heading and track pointers would be separated by 72° or multiples of 72°. To overcome this danger a mechanical stop is fitted: the stop limits movement of the stator relative to the rotor to a total of 356°, 178° either side of the zero drift position. Since 360° rotation of the rotor is equivalent to 72° of drift, 178° rotation is equivalent to 35° of drift, and this is the maximum drift which the system can accommodate.

Resolving Mechanism

15. The resolving mechanism (Fig 3) consists of ball-disc-roller assemblies, similar to those used in the API Mk. 1B, which resolve the ground mileage into components mutually at right angles to each other.

16. Ground mileage is fed from the Doppler to the disc via a repeater motor in which the shaft completes 48 revolutions per nautical mile. The output rotations of the two rollers represent the aircraft displacement in two directions at right angles to each other, each shaft rotating $\frac{1}{10}$ revolution per nautical mile.

17. The track shaft from the azimuth drive mechanism drives into one side of a differential gear, the centre of which is geared to a set track knob and drum. When the set track drum is at zero. the resolving cranks in the GPI are positioned by the track drive from the azimuth unit, and ground mileage is resolved in accordance with either the GRID or LAT. LONG. functions of the equipment. When the A/A function is selected and a track is set on the drum, the output from the differential is the difference between TMG and the set track; the departure from set track is then resolved into components along and across the set track.

18. The method of obtaining the presentation of position varies with the presentation selected:—

a. *Grid.* When GRID is selected, the resolving gear is set according to TMG, and the component N/S and E/W mileage outputs are fed directly to the counters.

b. *Latitude and Longitude.* When LAT. LONG. is selected, the resolving gear is set according to TMG. The N/S drive passes to the counters through an additional 6:1 reduction gear; this is necessary because the one set of veeder counters must register a maximum of 4,000 n.m. on A/A, or 400° of latitude, equivalent to 24,000 n.m. on LAT. LONG. The E/W counter is switched to a ball-disc secant gear where the mileage input is from a repeater motor driven by the E/W mileage transmitter and the ball position is set by the latitude counter. The resultant E/W output is therefore proportional to input miles multiplied by the secant of the latitude. A 6:1 reduction gear is incorporated in the secant gear drive to the counters which thus read degrees and minutes of longitude.

c. *Along and Across Track.* When A/A is selected, the ball carriage arm in the initial resolving gear is positioned relative to the track on the set track drum instead of relative to true north. The output drives therefore represent actual miles

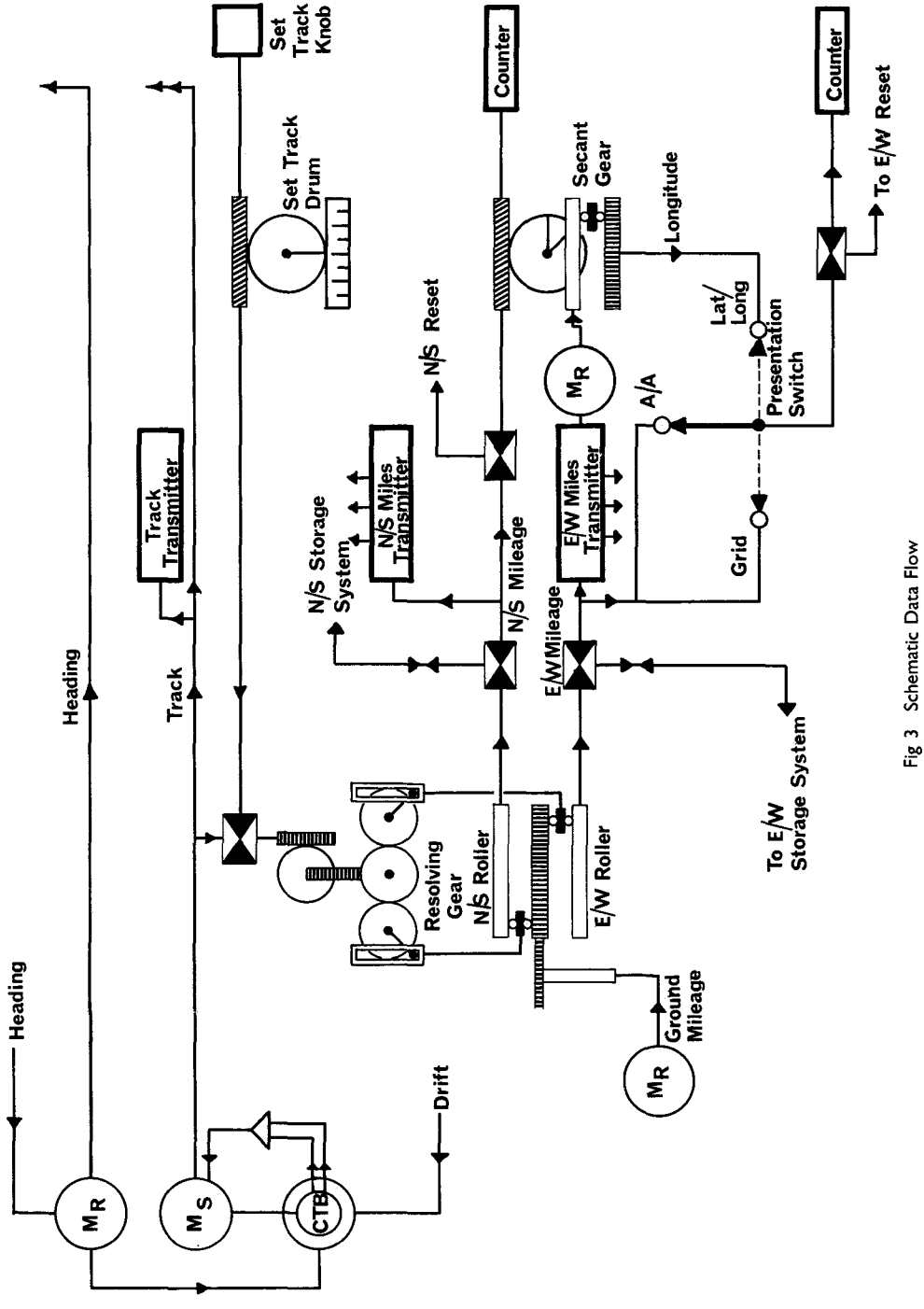


Fig 3 Schematic Data Flow

along and across the selected track. Track can be set only when the presentation switch is in the A/A position, and other presentations cannot be obtained until the set track drum is returned to 000 degrees.

Storage Mechanism

19. The storage system consists of two identical mechanisms, one for N/S mileage and one for E/W mileage. A schematic diagram of one storage mechanism, greatly simplified is shown in Fig 4.

20. The drive from the resolving gear feeds into a differential, the output from which passes either to the counters or to the storage mechanism, depending on the position of the NORMAL/FIX switch. When passing to the storage mechanism the drive rotates a storage disc, indicator drum, and a cam, W.

21. The storage disc has a slot cut in its periphery into which a locking plate can drop. The locking action can happen only when the disc is in a position corresponding to zero storage. The plate can be lifted clear of the disc by an arm attached to a shaft coupled to the NORMAL/FIX switch. The latter is in fact the storage control knob.

22. Cams on the plate assembly and the control knob shaft operate two electrical contacts A and B. A push-rod operating a reversing relay C rests on the cam W but can be lifted clear by an arm on the control knob shaft.

23. Geared to the drive operating the disc is an electric motor, the drive from which passes through a mechanically operated clutch. The motor is powered through relay C, the power being governed by contacts A and B, both of which must be closed before the motor will start.

24. **NORMAL—No Mileage in Store.** When the storage switch is at NORMAL and there is no mileage in store the locking plate is in the storage disc slot, and cam W is positioned so that the rod operating relay C at one of the contour change points on the cam. The rod itself will be in contact

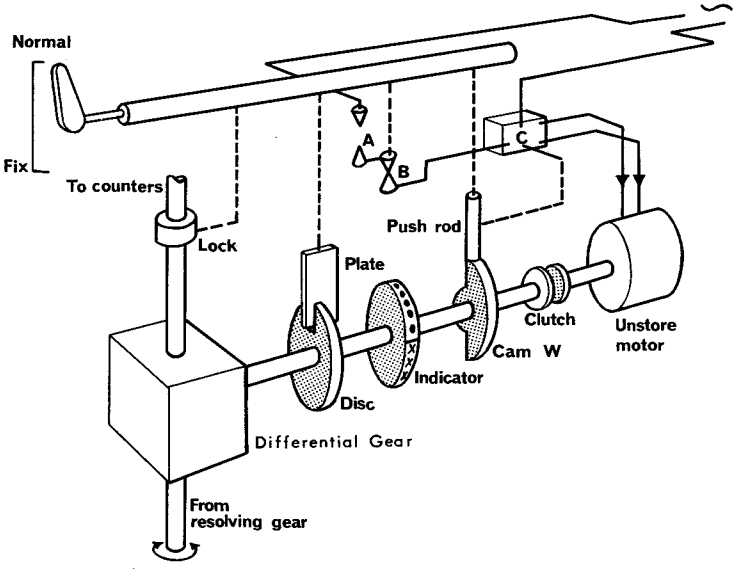


Fig 4 Simplified Storage Mechanism—No Mileage Stored

with the cam face either at the larger radius or the smaller radius according to the direction in which the cam has returned to the zero position. Contacts B are closed, but contacts A are open and, therefore, no power can reach the motor. Since the storage mechanism is locked the mileage passes straight through the differential to the counters.

25. **FIX—Mileage being Stored.** when the switch is put to FIX an arm coupled to its shaft engages with the plate, lifts it clear of the disc and raises the push rod clear of the cam. At the same time a lock is applied to the drive to the counters, and the clutch in the unstore motor drive is opened. Since the output of the resolving mechanism is prevented from reaching the counters it passes through the differential into the storage system, and the disc, cam W, and the indicator drum begin to rotate. The indicator drum, viewed through a window, shows by means of dots and crosses the sense of the stored mileage. Contacts A close but contacts B open and the unstore motor is, therefore, still unpowered. A lamp lights, illuminating the indicator drum.

26. **NORMAL—Mileage being Unstored.** When the switch is returned to NORMAL the locking plate is lowered but cannot drop into the disc slot since there is mileage in store. Contacts A remain closed. Contacts B and the clutch to the unstoring motor close. The push rod drops on to cam W and in so doing energizes relay C which in turn powers the motor. The position of the rod (on the large or small cam radius) determines the direction of rotation of the motor, the drive being the shortest way to the neutral position. The lock is removed from the counter drive and the unstore motor commences to drive the disc towards its zero position. The output from the storage mechanism is added in the differential to the incoming mileage from the resolving mechanism, and their algebraic sum is passed to the counters. The unstore motor continues to drive the disc and drum until they reach their zero position. At this point the plate drops into the slot under spring action and contacts A open, thus breaking the power feed to the motor and extinguishing the indicator lamp. The storage system is now locked and normal drive from the resolving mechanism to the counters is resumed.

27. **Excessive Storing.** One-half rotation of cam W is arranged to represent 200 nautical miles, and the counters will be at least 400 n.m. in error after unstoring if this figure of 200 is exceeded. If between 200 and 400 n.m. were stored the push rod would be in contact with cam W at the wrong radius, and the unstore motor would be powered in the wrong direction. In returning the disc to its zero position by the shortest route the unstore motor would drive the counters in the opposite direction to that required; *eg*, if 230 n.m. North were stored, the motor would drive the disc back to zero and unstore 170 n.m. South. Between 400 and 600 n.m. stored would mean more than one complete rotation of cam W and although the cam would be in the correct relative position the slot would reach zero once the unstore motor had unstored the mileage in excess of 400 and this amount of mileage (400) would be lost.

28. **Manual Unstoring.** A manual unstore knob, coupled to the unstore motor shaft, is provided as a safeguard in case of motor failure. The direction in which the knob should be turned to unstore is indicated by the appearance of dots or crosses in the window. Although 200 n.m. storage can be accommodated, the indicator drum is graduated only to 160 n.m. in either direction. Thus, if the latter figure has been exceeded, and only manual unstorage is available, there is no indication of the correct unstore direction and large errors may be introduced.

Errors

29. The GPI Mk. 4A is subject to instrument error, latitude and altitude error, and input error.

30. **Instrument Error.** Distance resolution is accurate to within 0.5% and track resolution is accurate to within 0.5 degrees. Slipping of the secant gear, which is likely to occur at latitudes higher than 70°N or S, will increase the instrument error. A further inherent source of error is back lash in the transmission and resolution gear. Backlash error is kept to a minimum by ensuring that the GPI reset knobs are at their central position when not in use.

31. **Latitude and Altitude Error.** The design of constant-scale measuring instruments such as the GPI Mk. is based on the assumptions that:—

a. At all altitudes and latitudes the length of a nautical mile is 6,080 feet and is equal to one minute of latitude.

b. A distance of 6,080 feet multiplied by the cosine of the latitude is equal to one minute of longitude.

These assumptions are true only at sea level at latitudes 47°42′ N and S, and at any other latitude or altitude must introduce errors. These errors are maximum at high altitudes and in high and low latitudes; altitude error always causes an overreading in distance while latitude error causes overreading in high latitudes and underreading in low latitudes. The overall error at 50,000 feet at 45°N is an overreading of approximately 0.25%. A full explanation of this error and methods of minimizing it are given in AP 3456G, Pt 2, Sec 1, Ch 1 Annex.

32. **Input Error.** Despite the inherent accuracy of the GPI Mk. 4A the ultimate accuracy depends on the quality of the inputs. The drifts and groundspeed values supplied by Doppler are accurate to about 0.3°, and 0.7% respectively. The accuracy of the true heading input is much less than that of the drift and is, in fact, the greatest source of error. Very accurate compass calibration and precise alignment of the M-type transmission system, together with careful application of corrections for variation, deviation and coriolis effect are necessary to minimize heading error.

33. **System Error.** The error which concerns the operator is that which arises when the GPI Mk. 4A, Doppler and the heading reference are used together in a particular type of aircraft, *ie* the system error: discussion of this error is not within the scope of this chapter.

Operation

34. The following drills apply to the GPI Mk. 4A in any installation.

35. **Pre-Flight.**

a. Set NORMAL/FIX switch to NORMAL and GPI main switch to OFF.

b. Switch on and synchronize the compass; switch on Doppler.

c. Synchronize heading and track pointers with compass.

d. Set zero drift and 600kt groundspeed on the Doppler.

e. Set presentation selection switch to A/A and the track drum to the value indicated by the heading and track pointers plus 45 degrees.

f. Switch GPI main switch to ON and check N/S and E/W electrical resetting in both directions. Return counters to zero and switch off GPI.

g. Note the time, and simultaneously switch on the GPI and the Doppler distance gone counter.

h. Switch to FIX and note that both storage lights come on, the counters freeze, and a dot or cross appears on each storage drum.

j. Switch to NORMAL and note that both storage lights go out together as the drums reach their neutral positions.

k. Exactly two minutes after switching on in g. switch off the GPI and the Doppler distance gone counter.

l. Check that the distance gone counter reads 20 n.m. and that the GPI counters read 14.1 n.m. North and 14.1 n.m. Left.

m. Set the presentation switch and track drum as required for flight.

n. Set the GPI counters to take-off co-ordinates and the Doppler distance gone counter to zero.

p. Switch off GPI.

q. Immediately before take-off check that the GPI heading repeater is still synchronized with the heading reference.

36. **Take-off.**

a. As aircraft becomes airborne switch on GPI and distance gone counter.

b. Carry out Doppler switching-on procedure as appropriate.

37. **Operation in Flight.** Particular in-flight procedures are laid down by the user Commands.

Flight-Planning Presentation

38. Before flight, the navigator must decide which of the three position presentations he will use. In addition to tactical considerations, the following factors are relevant:—

a. When a single track is to be flown, A/A track presentation permits immediate checking of track-keeping accuracy and of distance to go. The GPI counters can be set so that the actual distance to go is shown, by setting the South counter to read that distance before take-off. The counters should then read zero at destination. With doppler groundspeed readily available frequent and accurate revision of ETA can be calculated. For a flight involving a return to base, the outbound track can be left set on the GPI and the flight started with both counters reading zero. The distance along track accrued on the outward run is progressively reduced on the homeward run, and thus reads distance to go to base. It should be remembered that when a reciprocal track to that being flown remains set on the GPI the left and right across-track indications are reversed, *eg* an indicated 4 miles to left of track is actually 4 miles to right of track.

b. When large track changes are to be made, accuracy is lost if the A/A track presentation is used. Therefore the latitude and longitude or grid presentation is normally used whenever the proposed route is other than a single track.

c. For flights in polar regions when a grid technique must be used, the GPI presentation must be set to GRID.

GPI MARK 4.

Description

39. The GPI Mk. 4 was designed to operate with the Green Satin Doppler which can only measure drift up to ±20 degrees. Although externally the Mk. 4 is identical to the Mk. 4A the gear ratios in the drift transmissions are different, limiting the Mk. 4 to 20 degrees of drift.

40. As in the Mk. 4A heading synchronization should be carried out only when the drift servo is powered, but in this case, owing to the overall gear ratio of 8:1, there is a null position every 45°. Thus if the drift conditions in the GPI differ by more than 22½° from the drift input from Green Satin, continuous rotation of the track and heading pointers will occur.

GPI MARK 4C

Description

41. The Mk. 4C equipment is basically a GPI Mk. 4A fitted to accept outputs from a surface movement corrector. The latter is set manually by the navigator to offset Doppler velocity errors introduced by sea surface movement; it resolves surface motion into north/south and east/west components, each of which drives an M-type transmitter, from which the information is fed to repeater motors in the GPI Mk. 4C.

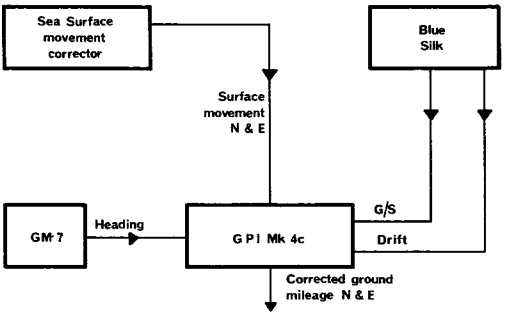


Fig 5 GPI Mk. 4c—Schematic

42. The repeater motors feed the north/south and east/west ground mileages to one side of a differential gear box. The second input is the two ground mileage components obtained from the conventional disc-roller-ball resolving gear of the GPI itself. The output from the differential is the summation of the ground mileage components, which is then used to drive the position counters.

43. A typical block schematic data flow is shown in Fig 5. Operating drills for the GPI Mk. 4C are promulgated by the user Command.

Printed in the United Kingdom for Her Majesty's Stationery Office
(2462/87) Dd8061677 12/87 G3371 10170

CHAPTER 5

TYPE 9476 COMPUTERS

CONTENTS

	Paras
Introduction	1–4
PRINCIPLES OF OPERATION	
General	5–6
Brief Description of a Digital Resolver	7
Operation of the Digital Resolver in the 9476 Computers	8–18
DESCRIPTION OF CONTROLLING SEQUENCES	
Timing	19–21
Digital Switch Unit	22–24
9476B AND F COMPUTERS	
Introduction	25
Description	26–29
9476C COMPUTER	
Introduction	30
Description	31–34
Principle of Store Operation	35–43
Features of the 9476C Computer	44–45
ACCURACY AND LIMITATIONS	
Accuracy	46
Limitations	47–48

Introduction

1. An aircraft's position may be displayed in a variety of forms *eg* latitude and longitude, distance along and across track or as a point on a moving map display.

2. Some airborne computers can provide outputs in any of these forms, however if a single fixed form of display can be tolerated, then the associated computer will be simpler and therefore cheaper, lighter and smaller. In the Decca 9476 series of computers, which are single purpose devices, these features are further enhanced by using digital computing techniques and employing solid state circuitry in the construction.

3. There are three variants, B, C and F, of the 9476 computer and all resolve doppler distance gone about a resolution angle (E). The definition of E, and hence the computer's output, depends on the type and role of the computer thus:—

a. The 9476B and F computers provide distance along and across required track, and E is the difference between track made good and required track.

b. In the latitude/longitude role the 9476C computer requires E equal to true track; but when providing the drive to the Roller Map Mk. 4 an input of E equal to the difference between track made good and chart angle is necessary.

4. Although the resolution angle takes various forms the computers are very similar and the main differences are:—

a. The C model has a store.

b. Different scaling circuits are necessary to reduce the doppler input to the rate required for the appropriate display system.

PRINCIPLES OF OPERATION

General

5. The resolution of doppler distance-gone (G) into distance along and across required track (or chart angle) is shown in Fig 1b and that of G about true track is shown in Fig 1a. In the 9476 computers this process is performed by the digital resolver which is the heart of this computer series.

6. The operation of the digital resolver uses the method explained in Part 3, Sect 2, Chap 2 of this volume and a knowledge of that chapter is assumed in this explanation. However, a brief resume is given below for completeness.

Brief Description of a Digital Resolver

7. The reader will recall that the doppler distance gone input, G, is a digital pulse train. Since required components G sin E and G cos E are always less than G, they are obtained by methodically rejecting some of the pulses as they pass through a scaler board. This rejection is programmed in the sine and cosine matrices. The angle E is sensed by a rotary switch on the M-type motor's shaft and the lines in the matrices corresponding to sine E and cosine E are energized. These are connected to the AND gates on the scaler board so as to reject the required number of pulses. This sequence is summarized in Fig 2.

Operation of the Digital Resolver in the 9476 Computers

8. In practice the digital resolver is not quite so simple. Since E can lie anywhere between 0° and 360° it would not be practicable to produce matrices to cover the complete resolution angle range. However, just as trigonometrical tables for angles between 0° and 90° enable, with an elementary knowledge of trigonometry, the sine and cosine of any angle to be calculated, so the resolver can be arranged to perform in a similar manner.

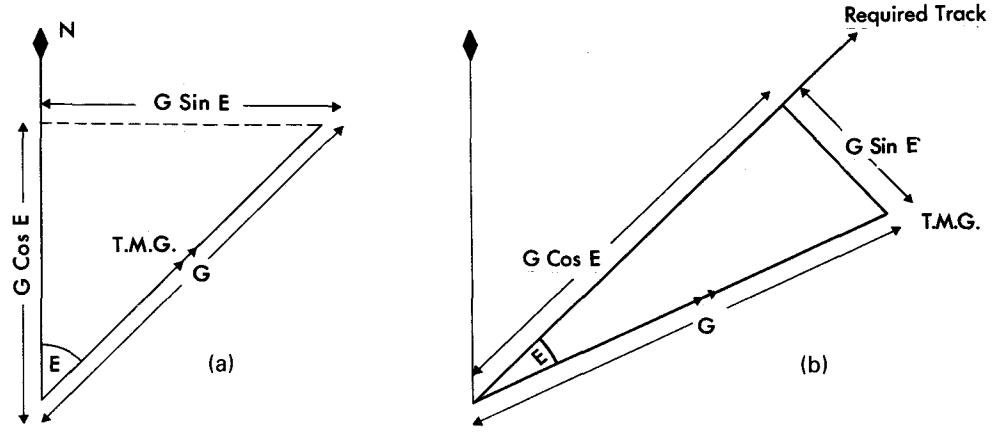


Fig 1 Resolution of Doppler Distance Gone

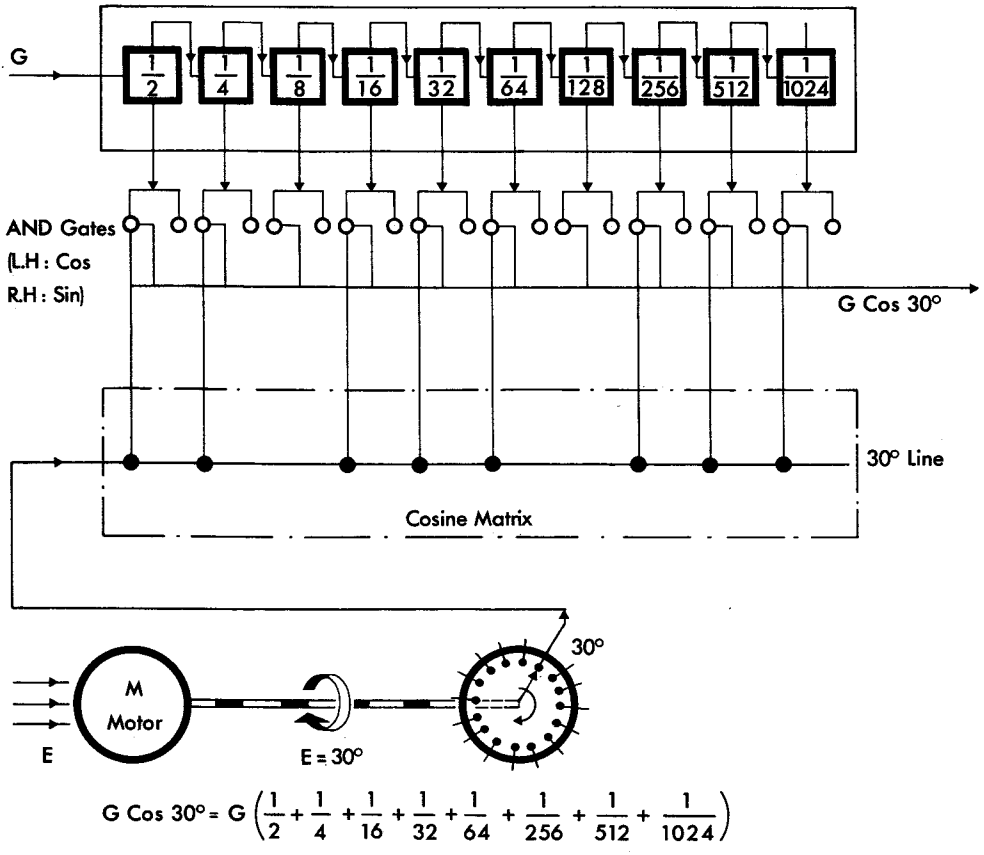


Fig 2 Principle of the Digital Resolver

9. From Fig 1b it can be seen that when E is less than 90° the cosine component provides the along track drive and the sine component the across track drive. When E is greater than 90° the correct movement can be achieved by interchanging the inputs to these drives and/or reversing the direction of movement, *see* Table 1. A similar relationship can be deduced for the sine and cosine components and the northings and eastings in Fig 1a.

10. In the following table E is the difference between track made good and required track (or chart angle) measured clockwise from the required track, alternatively it can be true track measured in the conventional manner.

11. To derive the two components of G it is not even necessary, and certainly not practicable, to have matrices with lines corresponding to each individual value of α from 0° to 90°. Again let us consider the analogy of the trigonometrical tables. Their size could be reduced by increasing the interval between each tabulated angle and, provided the interval was reasonably small, the trigonometrical ratios for intermediate values could be found quite accurately by linear interpolation. This principle is adopted in the 9476 computer where the interval selected between "tabulated values" is $\frac{90°}{16} = 5.625°$. Two matrices, each with 15 input lines produce the sine and cosine for all multiples of 5.625° between 0° and 90°; no lines are provided for cos 90° or sin 0°.

Value of E	Along Track/Northings Drive	Across Track/Eastings Drive
1st Quadrant $E = \alpha$	G cos α (forwards or northwards)	G sin α (right or eastwards)
2nd Quadrant $E = \alpha + 90$	G cos $(\alpha + 90)$ $\equiv -$G sin α (backwards or southwards)	G sin $(\alpha + 90)$ $\equiv$ G cos α (right or eastwards)
3rd Quadrant $E = \alpha + 180$	G cos $(\alpha + 180)$ $\equiv -$G cos α (backwards or southwards)	G sin $(\alpha + 180)$ $\equiv -$G sin α (left or westwards)
4th Quadrant $E = \alpha + 270$	G cos $(\alpha + 270)$ $\equiv$ G sin α (forwards or northwards)	G sin $(\alpha + 270)$ $\equiv -$G cos α (left or westwards)

Table 1 Allocation of Components to Axes

12. The resolution angle is sensed by three rotary switches attached to the shaft of the M-type motor. The gearing between these is such that the quadrant is sensed by one, and the most significant (MS) and least significant (LS) portions of the equivalent first quadrant angle (α in table 1) by the other two. The MS portion is an exact multiple of 5.625° and the LS part is the remainder.

13. To illustrate the ideas outlined in paras 12 and 13, consider a resolution angle of 45.7032°.

The MS portion is 45° (*ie* $8 \times 5.625°$) and the LS part is 0.7032°. Thus to find G sin 45.7032° by the interpolative method outlined in para 11 we need:—

$$\left[G \sin 45° + \frac{0.7032}{5.625} (\text{Sin}50.625° - \text{Sin}45°)\right]$$

or in general terms:—

$$G\left[\sin \text{MS part} + \frac{\text{LS part}}{5.625} (\text{Differential sine factor})\right] \quad \ldots\ldots\ldots\ldots\ldots\ldots(1)$$

where $\left[\sin (x + 5.625)° - \sin x°\right]$ is termed a differential sine factor, with x an exact multiple of 5.625°. The sines of all (15) MS values are programmed into the MS sine matrix, which controls the ground-speed pulse output from its associated scaler board, thus G sin MS part can be produced. Moreover the fraction $\frac{\text{LS part}}{5.625°}$ can be programmed into a matrix for specific values of the LS part, the values chosen will govern the size of the matrix and the final accuracy of the sine of the resolution angle. As a compromise 16 values are selected, at intervals of 0.3516° (*ie* $\frac{5.625°}{16}$). Thus one can "enter" the matrices with a resolution angle accurate to ± 0.1758°.

14. For each MS position there is a corresponding differential sine and cosine factor and the programme to produce the sixteen values of each is built into a Differential Matrix. The input lines to the Differential Matrix are connected to the appropriate contact on the MS rotary switch. Therefore if this switch is in, say, the 45° position the Differential Matrix input line wired to produce (sin 50.625° − sin 45°) is energized, as well as the 45° input line in the MS matrix.

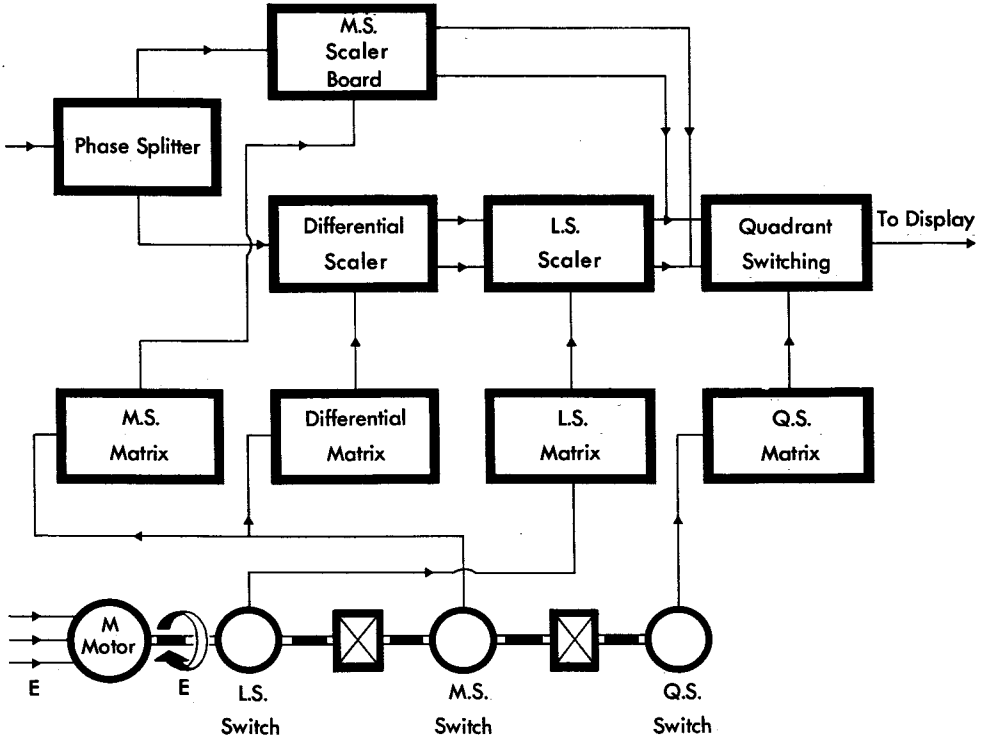


Fig 3 Digital Resolver

15. We are now in a position to produce $G \times \frac{\text{LS part}}{5.625°} \times$ Differential factor to add to G sin MS part as in (1). The process is shown in Fig 3. The ground speed pulse train is fed through a phase splitter and divided into two equal, but 180° out of phase, outputs at half the input rate, *see* Fig 4. The phase splitter is the last stage in scaling the groundspeed input rate to the requirements of the M-type motors in the display, and the pulse trains are the true analogues of groundspeed. One path from the phase splitter leads through the MS scaler board which, being controlled by the MS switch and MS sine and cosine matrices, produces outputs corresponding to G sin MS and G cos MS. The other path passes through the differential scaler board and, by a similar process as on the MS board, outputs of G × differential sine factor and

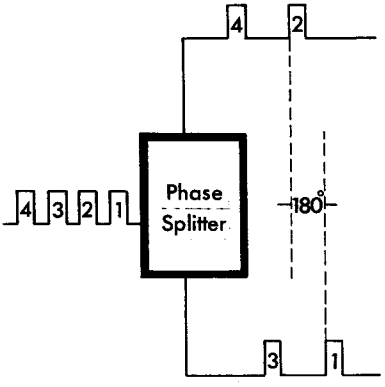


Fig 4 Action of Phase Splitter

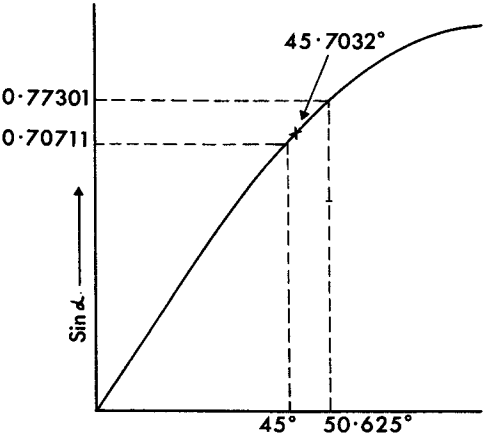
G × differential cosine factor result. These are fed to the LS sine and cosine scaler board which, programmed by the LS switch *via* the LS matrices, produces the required fractions of the differential factors.

16. Since the outputs from the LS and MS boards are 180° out of phase, as a result of the phase splitter, the corresponding outputs can be joined directly, without any interference, to produce G sin α and G cos α.

17. The process of combining the outputs is simply one of addition, and though this readily fits the sine channel (as the sine increases with the angle) the cosine channel would appear to pose problems. This is overcome at the MS switch which energizes the cosine matrix line carrying the MS value above (*eg* 50.625°) that sensed (45°), therefore the output from this scale board is less than that required. The differential cosine matrix is programmed to produce the positive differential (*ie* cos 45° − cos 50.625°), and the LS switch energizes the line in the matrix which will release $\frac{\text{5.625-LS Part}}{\text{5.625°}}$ of the differential input. Thus the combined output is correct. Fig 5 illustrates the calculations for the sine and cosine of 45.7032°.

18. **Quadrant Selection**. There are four input pins, accepting forward, reverse, left and right instructions, for the two M-type motors in the display. The sine and cosine components are fed to the respective pins through a matrix which is controlled by the quadrant selection (QS) switch. The QS matrix is shown diagramatically at Fig 6.

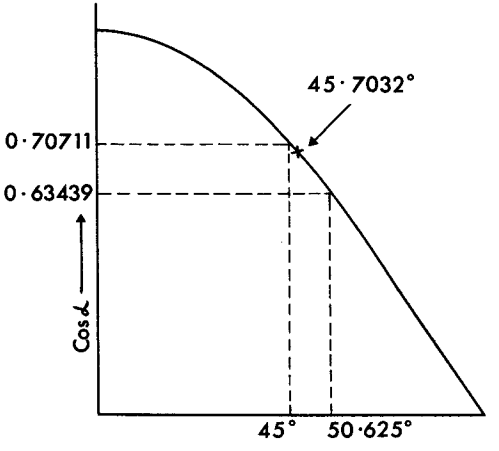
It can be seen that the diodes block the sine and cosine inputs unless a voltage, from the QS switch, is applied to the other side.



Sin 45.7032 = sin 45° + $\frac{0.7032}{5.625}$ (sin 50.625° − sin 45°)

= 0.70711 + $\frac{1}{8}$ (0.0659)

= 0.71535 (c.f. tabulated value of 0.71569)



Cos 45.7032°

= cos 50.625° + $\frac{5.625-0.7032}{5.625}$ (cos 45 − cos 50.625°)

= 0.63439 = $\frac{7}{8}$ (0.7272)

= 0.69802 (c.f. tabulated value of 0.69842)

Fig 5 Incremental Calculations of Sine and Cosine

DESCRIPTION OF CONTROLLING SEQUENCES

Timing

19. To ensure that G is correctly resolved about a discrete resolution angle it is important that the shaft of the M-type motor carrying the three sending switches should not move while a pulse is being processed through a scaler board. Any change in the state of the AND gates should be made in between pulses. A timing sequence ensures that this happens.

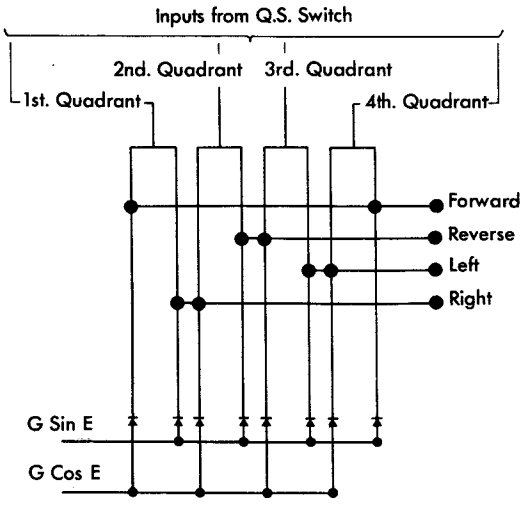


Fig 6 QS Matrix

20. An oscillator within the computer produces regularly spaced pulses at a rate in excess of the maximum rate of groundspeed input. Incoming pulses are synchronized with these "clock" pulses, as in Fig 7, before being passed to the resolver.

21. The resolution angle is received by the M-type motor *via* an inhibiting circuit into which the clock pulses are also fed. This circuit prevents the motor from moving during a clock pulse, and hence during a ground speed pulse since the two are synchronized.

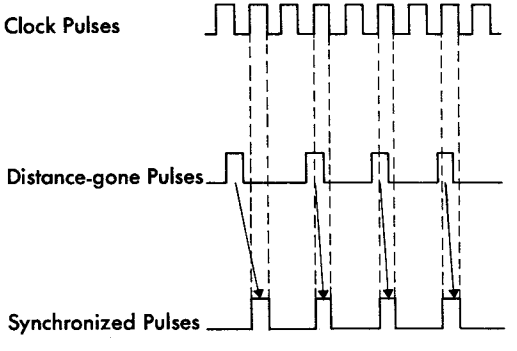


Fig 7 Synchronization of Input Pulses

Digital Switch Unit

22. The digital switch unit comprises the synchro control transformer, an M-type motor and the three sensing switches all geared together, as in Fig 8.

23. An input representing resolution angle is fed to the stators of the control transformer (CT) and any misalignment between the rotor and stator drives the M-type motor, *via* the servo board containing the inhibiting circuit of para 21. The motor drives the CT rotor to the null position, thus completing the servo loop.

24. The motor also drives through the gear trains the LS, MS and QS switches. The gear ratios are as shown in Fig 8, so that one revolution of the CT rotor is produced by $\frac{1024}{3}$ revolutions of the motor which produces 64 revolutions of the LS switch, 4 revolutions of the MS switch and $\frac{1}{4}$ revolution of the QS switch.

9476B AND F COMPUTERS

Introduction

25. So far we have examined the operation of the digital resolver, the timing sequence and the digital switch unit. In so doing we have almost described the operation of the 9476B and F computers. These two models drive displays giving along and across required track information; a block schematic of the 9476B computer is at Fig 9.

Description

26. In Fig 9 the reader should be able to identify the digital resolver, the digital switch unit and the timing circuit. The only portion not covered so far is the input stage which is devoted to scaling the doppler pulse train. It is in this area that the differences between the B and F models occur.

27. **Scalers.** The reduction of the doppler input, to a rate compatible with the output

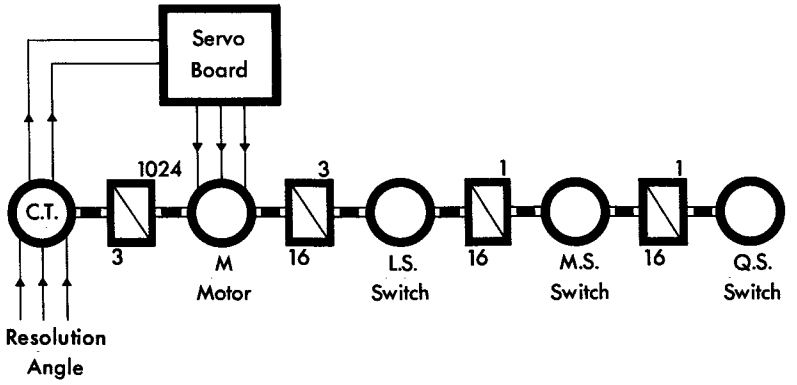


Fig 8 Digital Switch Unit

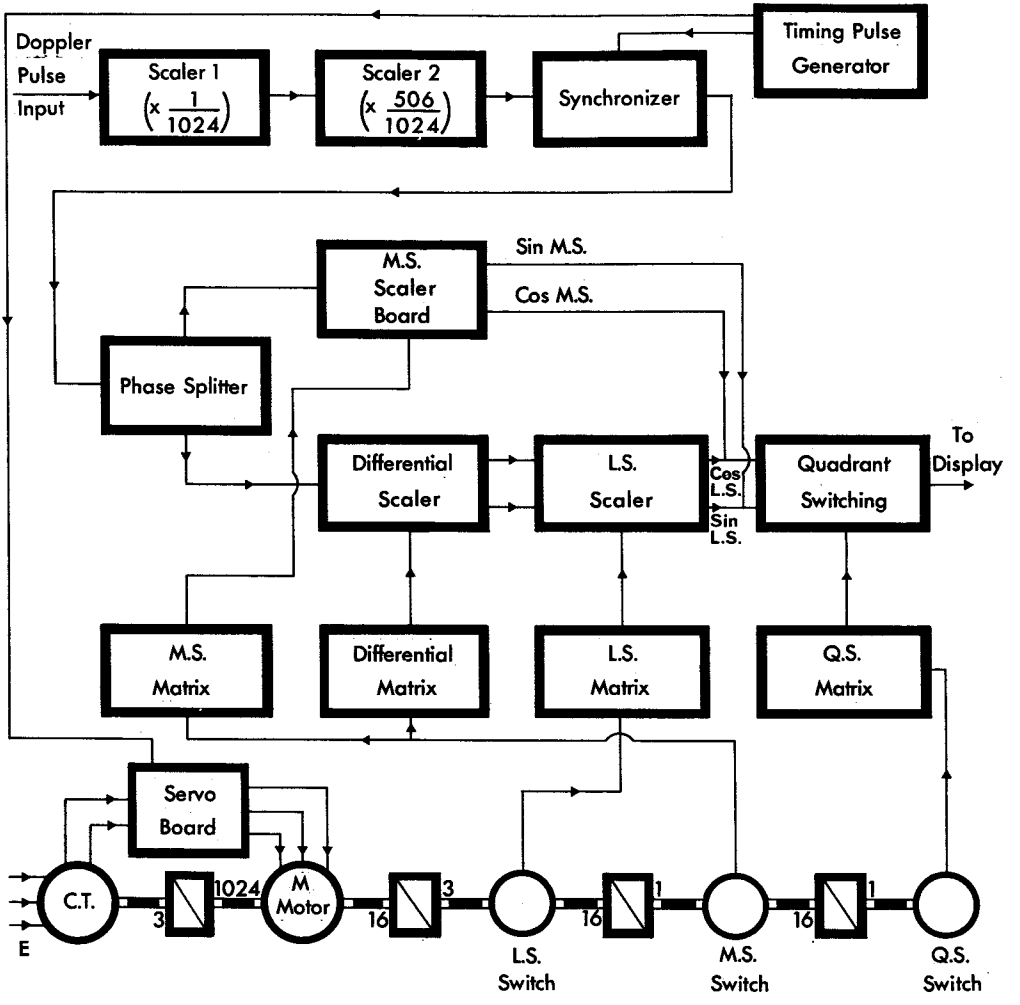


Fig 9 Block Schematic of the Type 9476B Computer

gearing on the M-type motors in the display units, is made in scalers 1 and 2 and the phase splitter. The latter makes the final reduction, dividing by two, but the reduction factors imposed by the scalers can be varied. These scalers are similar in construction to the scaler boards in the digital resolver and they can be programmed with one selected reduction factor, as in the B model, or alternative factors may be introduced *via* a suitable matrix as in the F model.

28. **9476B**. The factors required to reduce the Decca 62M doppler inputs (80,649 pulses/nm, to the rate required by the associated 9478B display are shown in Fig 9. The scaled input, once synchronized, is fed *via* the phase splitter into the digital resolver. Between pulses the shaft of the digital switch unit, and hence the switches on it, are positioned according to the resolution angle fed from the associated 9478B display unit described in Part 3, Sect 3, Chap 6. Consequently the outputs from the LS, MS and QS switches energize the appropriate lines in their matrices and the resolver outputs are fed to the display. The computer weighs 9 lbs.

29. **9476F**. Although the B and F computers present positional information in a similar form, the F model provides an additional facility which enables the scale of the along and across displays to be increased by ten. In this mode the limit of the along track display become 99.9 nms and that of the across track display 9.99 nms (as against the normal limitations of 99 nms and 99.9 nms). This scale change is made in scaler 2 once DROP is selected on the NORMAL-DROP switch at the auxiliary display. The two displays associated with this computer are:—

a. Along/Across Track Display Unit Type 9478F.

b. Auxiliary Across-Track Display Type 9478G.

These are described in Part 3, Sect 3, Chap 6. All the controls for the computer are on the display units. This computer also weighs 9 lbs.

9476C COMPUTER

Introduction

30. The 9476C Computer drives the Type 947 Lat/Long Display Unit and provides outputs of True Track and Groundspeed to other equipments. It can also drive a Roller Map if required. The 9476C differs from the B and F versions as follows:—

a. The 9476C has a store facility.

b. The type 9476C, when operating in the true track mode, regenerates true track information through its track repeater unit (*see* para 45 and Fig 10) to feed other equipments. This method reduces the load which would otherwise be imposed on the track source by multiple sub-systems, and so maintains the accuracy of the track information throughout. The 9476C computer also regenerates the groundspeed pulse input, thus providing a source of groundspeed information which can be used without increasing the load on the doppler. In the Roller Map role the repeater units of the 9476C are redundant.

Description

31. A block schematic diagram of the type 9476C computer in the true track mode (*ie* driving the latitude/longitude display) is at Fig 10. The doppler distance gone pulse input is fed to scalers 1 and 2 which, in the true track mode, are biased to provide the $\frac{1}{2}$M scale required by the display. However in the Roller Map role these scalers are controlled *via* the scaler matrix and the scale change switch on the display head. This enables the display heads to use $\frac{1}{2}$M, $\frac{1}{4}$M and $\frac{1}{10}$M maps.

32. The output from the scalers, suitably reduced according to the scale selected, is synchronized with the internally generated timing pulses before passing to the digital resolver. The resolution angle is fed to the CT on the digital switch from:—

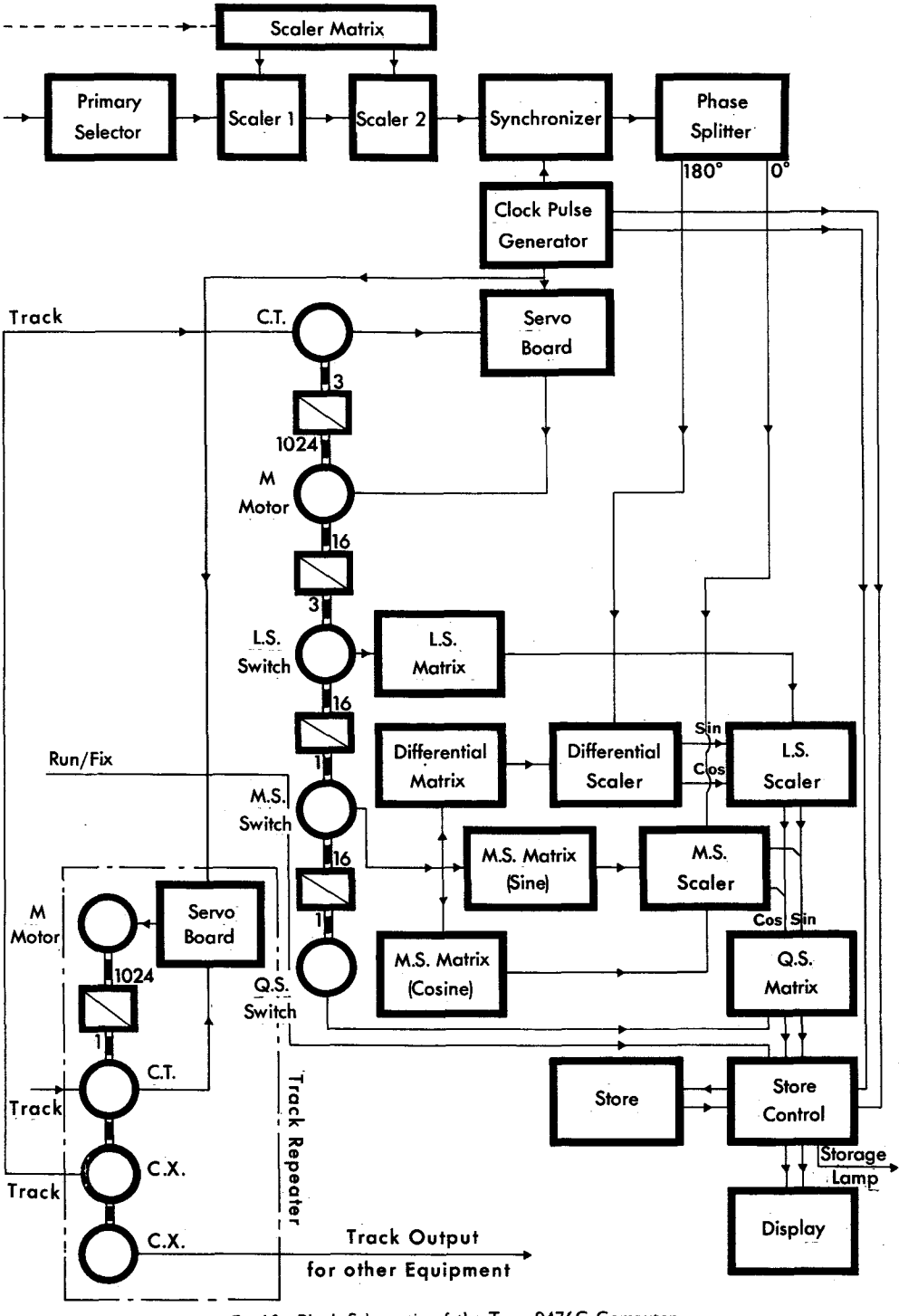


Fig 10 Block Schematic of the Type 9476C Computer

a. The track repeater when the 9476C is operating in true track mode or

b. The control differential transmitter (CDX) in the display head when the computer driving the Roller Map.

33. The outputs from the digital resolver are fed to the store control the functions of which are:—

a. To direct the components of distance gone to the display when operating in the normal RUN condition, or to the associated store boards when operating in the FIX condition.

b. To direct the stored data to the display during the unstoring sequence *ie* after switching from FIX to RUN.

c. To control the storage capacity warning lamp in the display head. This warning circuit is discussed in para 44.

34. The along and across track stores and the display follow the store control.

Principle of Store Operation

35. **The Bi-stable.** The basic unit of the along and across stores is the bi-stable flip-flop. For our purposes this bi-stable can be considered simply as a unit with one input to two transistors. A and B, each having its own output. The device is so arranged that one transistor will conduct while the other is off. This state can be reversed by the application of a pulse to the input, *ie* if A was previously conducting it will now be switched off and B, which was off, will now conduct. A further pulse would reverse the state yet again, as will each pulse subsequently applied to the input. These two states, shown diagrammatically in Fig 11, are quite stable and are maintained until switched by an input pulse.

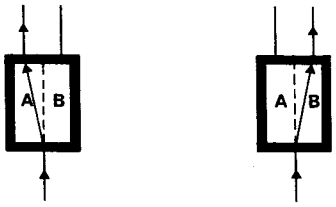


Fig 11 The Two States of a Bi-stable

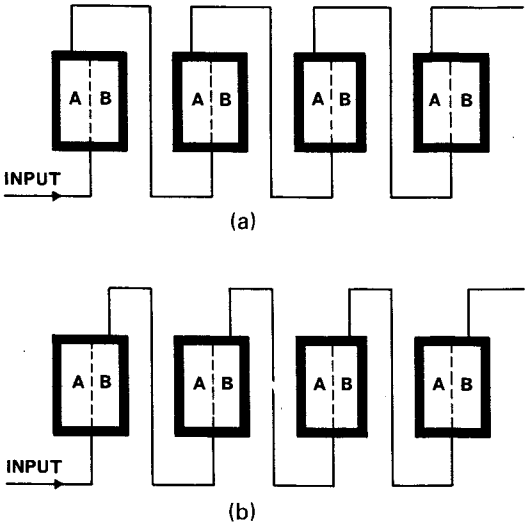


Fig 12 Store Construction

36. **Store Construction.** The along and across stores are identical, each consists of a 15 stage bi-stable chain so arranged that the output of one bi-stable provides the input to the next. The store can be set so that either all the A outputs or all the B outputs are used as the successive inputs, as shown in Fig 12.

For simplicity only the along store will be explained and furthermore only four bi-stables will be used to illustrate the operation of this unit.

37. **Store Operation.** When the FIX/RUN switch is to FIX the pulse train, representing G cos α, is directed to the store by the store control. Consider the store in its initial state with all the A transistors conducting, this is shown in Table 2, where a transistor conducting is indicated with a 1 and the store is set so that the B outputs are providing the successive inputs, as in Fig 12b. The first pulse into bi-stable 1 will cause transistor 1B to conduct and will thus pass onto bi-stable 2 causing transistor 2B to conduct which in

Bi-stables	1		2		3		4	
	A	B	A	B	A	B	A	B
Pulses Initial State	1		1		1		1	
1st pulse state		1		1		1		1
2nd	1			1		1		1
3rd		1	1			1		1
4th	1		1			1		1
5th		1		1	1			1
6th	1			1	1			1
7th		1	1		1			1
8th	1		1		1			1
9th		1		1		1	1	
10th	1			1		1	1	
11th		1	1			1	1	
12th	1		1			1	1	
13th		1		1	1		1	
14th	1			1	1		1	
15th		1	1		1		1	
16th	1		1		1		1	
17th		1		1		1		1

Table 2 Forward Storage Progression

turn prompts transistor 3B and 4B to conduct. The second pulse into bi-stable 1 will reverse the state and 1A will conduct, however since 1A is not connected to bi-stable 2 the remainder of the units will maintain their previous state. Pulse 3 into bi-stable 1 causes 1B to conduct which in turn causes bi-stable 2 to switch over so that 2A is now conducting. However, bi-stables 3 and 4 remain unaltered since there is no input from 2B. The rest of the progression can be followed through on Table 2.

38. **Direction of Store.** The QS matrix *via* the store control determines whether the A or B outputs provide the inputs to successive bi-stables. If reverse store is necessary, that is if the resolution angle lies in the second or third quadrant, then the A outputs provide the successive inputs throughout the store.

39. **Ambiguity.** Now re-examining Table 2 in the light of this, it can be seen that considerable ambiguity could occur. If we start at state 16 (which is identical to the initial state) and read back through states 15, 14, 13 *etc*, these are the states we would expect after 1, 2, 3 *etc* pulses had been stored in reverse: thus state 10 could represent 10 pulses stored in forward mode or 6 pulses stored in reverse. Fortunately this confusion can be overcome by using the last bi-stable as a "steering device", if the B transistor is conducting it is a forward store state and *vice versa* if the A transistor is conducting. From Table 2 it can be seen that with this convention the direction of store reverses after the eighth pulse, or in general terms after 2^{n-1} pulses where n = number of bi-stables. Thus the absolute capacity of the store in either direction is 2^{n-1} pulses.

40. **Readout from Store.** The removal of the pulses from store could be achieved if we could simultaneously feed a pulse to the display with one to the store which would reverse the direction of the progression by one step. By the time the store had returned to its initial state as many pulses would have been fed to the display as were originally put into store. This is the principle governing the unstoring sequence: the reversal of the progression is achieved through the store control, by switching to the alternative output/input sequence. The pulses used in the unstoring sequence are clocking pulses.

41. **Unstoring Sequence.** Consider the store state when four pulses have been stored in a forward direction, this is shown in Table 3. The RUN/FIX switch is now put to RUN which changes the output/input sequence, thus the outputs from the A transistors pass to the bistables, Fig 12a. Bearing this in mind Table 3 can be followed through, it can be seen that the initial state (all As' conducting para 38) has been reached after four pulses have been fed out to the display *ie* the number that was stored.

42. **Control of Clocking Pulses.** With the RUN/FIX to FIX the clock pulses are blocked from the store control and thus no clock pulses can enter the store and storage will go ahead as in para 38. When RUN is selected the clock pulses are allowed into the store control and, if there are any pulses stored, will pass into the store and reduce the contents as in paras 41 and 42 at a speed of

	1		2		3		4	
	A	B	A	B	A	B	A	B
4 pulses stored	1		1			1		1
1st clock pulse		1	1			1		1
2nd	1			1		1		1
3rd		1		1		1		1
4th	1		1		1		1	

Table 3 Unstoring Sequence

4,000 kts. An equal number of pulses would have been fed to the display during the same period. A flow diagram for the store control and store is given in Fig 13.

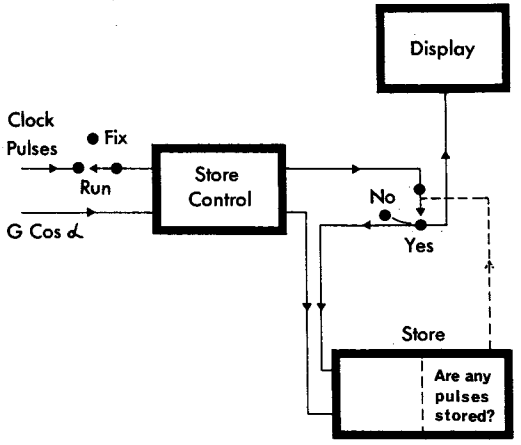


Fig 13 Flow Diagram for Store and Store Control

43. **Store Capacity**. Pulses can only be removed from store in the correct direction before 2^{n-1} pulses are stored, thereafter if RUN is selected the store would empty in the opposite direction. Referring to Table 2 if eleven pulses had been stored in the forward direction (*ie* 2^{n-1} pulses had been exceeded) when RUN was selected, then five pulses would be fed to the display in reverse direction, whereupon the initial state would be reached. This is obviously undesirable and some indication of the state of the store is required by the navigator. This is provided by the last two bi-stables, for as soon as their states differ a warning lamp flashes. This occurs after 2^{n-2} pulses have been stored, *see* Table 2. A hundred per cent safety margin is thus incorporated, for it will be recalled that the store reverses its direction after 2^{n-1} pulses have been stored. As soon as the warning lamp begins to flash RUN should be selected, for there is no indication of the reversal of the storage direction. This is not so troublesome as it may first appear, for with 15 bi-stables as in the 9476 stores the lamp will begin to flash after 2^{13} (8192) pulses have been stored which, on $\frac{1}{4}$M scale, is equivalent to 105.26 nm. Thus a further 105.26 nm would have to be put into store before the reversal of storage direction occurred.

Features of the 9476C Computer

44. Earlier in this chapter reference was made to the track repeater unit of the 9476C computer, this is a straight forward servo loop system, *see* Fig 10. Track made good information from the CDX on the doppler 62M aerial is accepted by the CT and transmitted to other equipments, and to the digital switch, *via* the CX's mounted on the shaft. This unit is not used when the 9476C computer is driving the Roller Map; in this role the resolution angle is fed directly from the display head to the digital switch.

45. In the Roller Map installation the track repeater unit is made redundant only by the external wiring. Thus in a system employing a latitude/longitude display and a Roller Map, the computers can be interchanged should the "master" repeater unit become unserviceable.

ACCURACY AND LIMITATIONS

Accuracy

46. Broadly, the accuracy of the resolver is ± 0.18° this being half the LS interval, but, since the resolver computes its components on the basis of linear interpolation between MS points, very small additional errors are introduced. Added to these considerations we have transmission errors and the overall accuracy claimed is:—

a. In azimuth—within 0.5°.

b. Along track—0.1% of distance gone or 0.1125″ on the Roller Map whichever is the greater.

Limitations

47. **Speed Range**. The type 9476 computers can handle speeds up to a maximum of 1,000 kts.

48. **Maximum Rate of Turn**. The digital resolver in the type 9476 computers can cope with rates of turn up to a maximum of 360° in 25 secs.

49. **Compass Safe Distances**. The compass safe distance is eleven inches for all type 9476 computers. This is the minimum distance from the compass at which the equipment, after being magnetized in a DC field of $1\frac{1}{2}$ oersted, shall be placed in order that the maximum change of compass deviation shall not exceed 1° for the direct indicating compass and $\frac{1}{8}$° for the remote indicating compass.

CHAPTER 6

TYPE 947 AND 9478 DISPLAY UNITS

CONTENTS

TYPE 947 DISPLAY UNIT

	Paras
Introduction	1
Description	2–5
Principle of Operation	6–7
Displays and Controls	8

TYPE 9478 DISPLAY UNITS

General	9–11

TYPE 9478B AND F DISPLAY UNITS

Principle	12
Description	13–17
Operation	18–23

TYPE 9478G DISPLAY

Introduction	24
Description	25–27
Operation	28–31

Introduction

1. The type 947 display unit is the latitude/longitude display for the 9476 computer; essentially it is the latitude/longitude portion of the GPI Mk. 7 (*see* Part 3, Sect 3, Chap 8 of this Volume) with a variation control system.

Description

2. The display unit, *see* Fig 1, provides latitude and longitude read-outs on two five-digit in-line counters reading to one minute with a sixth letter indicator providing N-S and E-W indications. The counters are arranged to change direction of rotation at the equator and at the Greenwich meridian respectively, with the appropriate N or S and E or W indication.

3. Controls are provided for both fast and slow reset of each counter; this operation is facilitated by the store of the associated type 9476C computer. As indicated in Fig 1 the variation setting control is also on the display unit.

4. Where the Roller Map and/or the along/across track systems are fitted in addition to the latitude/longitude display, their track inputs are provided from the type 9476C computer linked to the type 947 display. This computer is designated the master unit. No on/off switch is incorporated in the type 947 display, thus the inadvertant shut-down of the subsidiary systems is prevented.

5. The unit measures $5\frac{3}{4}'' \times 4\frac{1}{2}'' \times 11\frac{1}{2}''$ and weighs 12 lbs.

Principle of Operation

6. **General.** The northings and eastings components of the doppler distance gone are fed to the type 947 display as pulse trains

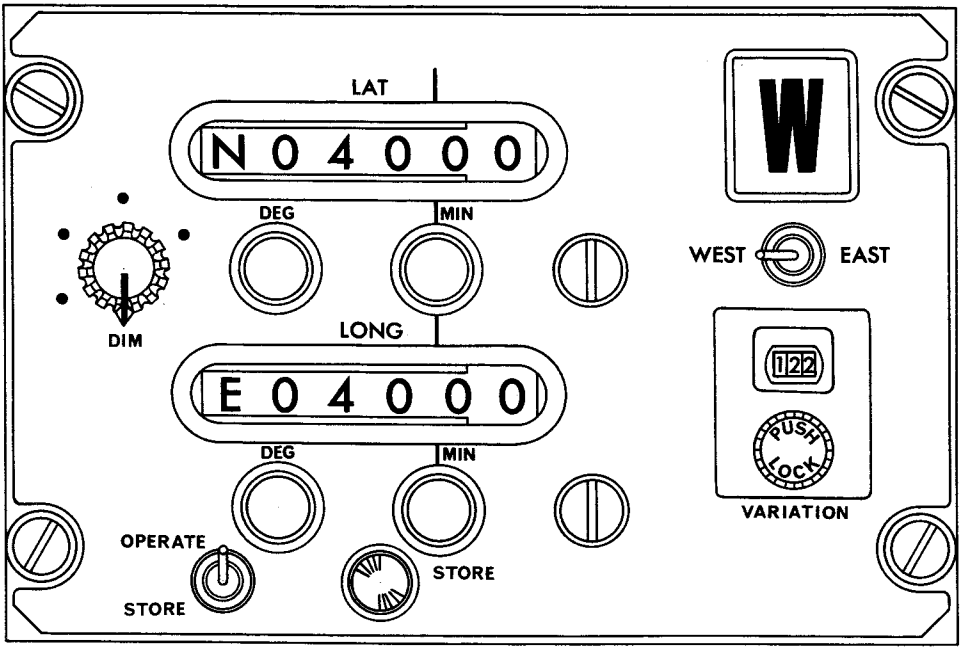


Fig 1 Type 947 Display Unit

from the computer. The pulse trains can be transformed into shaft rotations *via* M-type motors and so drive the veeder counters of the display. However, the eastings have to be multiplied by the secant of the latitude to produce change of longitude; this is achieved by using a sphere resolver as in the GPI Mk. 7.

7. **Variation Control System.** In the C130 (Hercules) aircraft installation, magnetic heading is fed to the differential synchro in the doppler aerial (CDX1) which thus provides an output of magnetic track. Variation is applied to this quantity at the type 947 display before it is fed to the type 9476C computer. The variation control system is a differential synchro (CDX2) in which the rotor is positioned according to the variation value and the magnetic track is fed to the stators. The order in which the lines are connected between CDX1, CDX2 and the CT in the type 9476C computer determines whether variation is added to or subtracted from the magnetic track. Thus on the type 947 display the value of variation is set manually, and the sense is determined by a switch which controls the order of the connections between CDX1, CDX2 and CT, *see* Fig 2.

Displays and Controls

8. Fig 1 shows the front panel of the type 947 display, on it are located the following displays and controls:—

a. *Latitude/Longitude Veeder Counters.* Two five-digit in-line counters display latitude and longitude values to one minute, with a sixth letter indicator providing N-S and E-W indications.

b. *Reset Controls.* Below each veeder counter are two reset controls marked DEG and MIN; each control provides a two-speed reset facility. When rotated clockwise they increase the degrees (DEG) or minutes (MIN) display and decrease the display when turned anti-clockwise. When in the fully out (normal) position the controls give the fast resetting rate, the

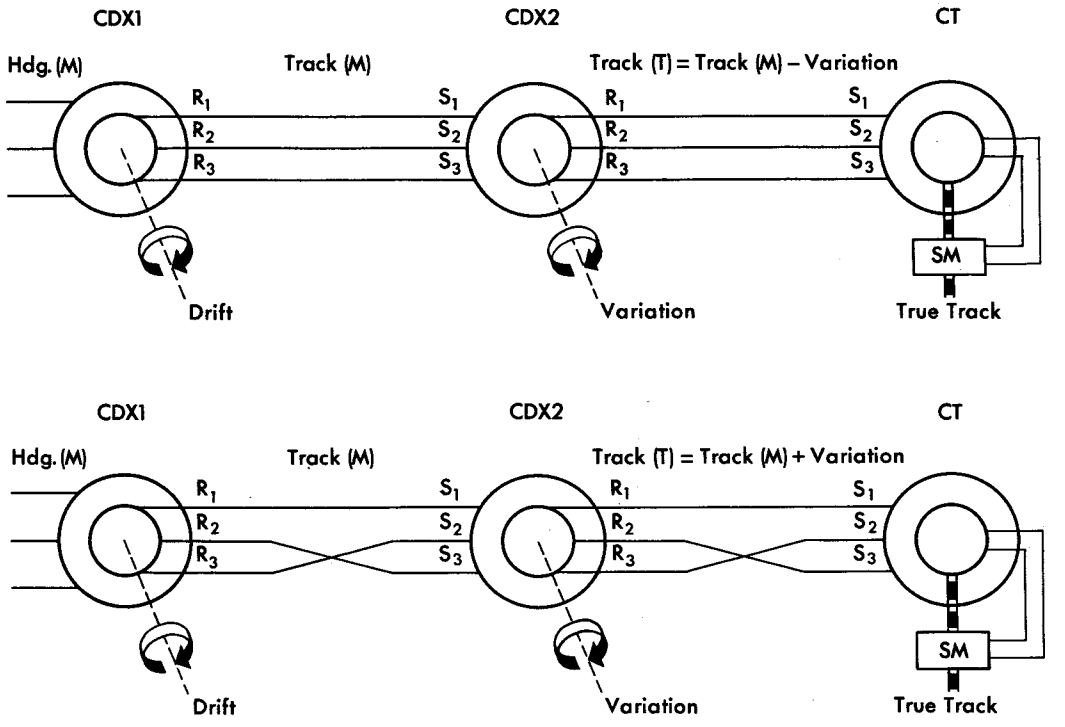


Fig 2 Variation Control System

counters being driven electrically. A slow rate can be achieved by depressing the reset control and turning; in this way the displays can be altered manually *via* a gear train, and precise values can be set. The operator should ensure that the reset controls return to the fully out position after a manual reset has been made, otherwise the displays may jam.

c. *Dimming Control.* The degree of illumination on the face of the display is determined by the dimming control on the left of the panel.

d. *OPERATE/STORE Switch.* A store is provided in the 9476C computer to prevent the loss of incoming information while resetting. This facility is brought into operation by selecting STORE on the OPERATE/STORE switch; OPERATE is the normal operating position. This switch corresponds to the RUN/FIX switch on the Roller Map Mk. 4.

e. *Store Warning Lamp.* The state of the store is indicated by the store warning lamp which is to the right of the OPERATE/STORE switch. The lamp glows steadily while storage is progressing but blinks once either of the channels reaches its nominal capacity. Once this occurs, OPERATE should be selected as soon as it is conveniently possible. As explained in the main chapter the lamp begins to flash once the store is half full, but there is no change of indication when the store is completely full.

f. *Variation/Warning Display.* In the top right hand corner of the unit is an illuminated window in which three displays can appear: OFF, W and E. The power to the unit is controlled *via* a circuit breaker, when this is open the OFF flag appears. The W or E indicates the position of the Variation Sense Switch.

g. *Variation Sense Switch.* The switch marked WEST/EAST determines the sense of the variation applied to the incoming magnetic track.

h. *Variation Setting Control.* The magnitude of variation is manually set by rotating the knob marked PUSH LOCK. Before a setting can be made the knob must be pulled out; once the value has been set the knob is depressed to lock it in position thus preventing its inadvertant adjustment. The value of variation set is displayed to one degree on a three-digit in-line counter.

TYPE 9478 DISPLAY UNITS

General

9. The Display Units types 9478B and F both provide a continuous veeder counter display of the aircraft's position along and across a required track. The F model also has the capacity to feed the autopilot with a signal proportional to the across track displacement.

10. The Display Unit type 9478G, an auxiliary across track display, supplements the facilities of the 9478F unit by providing an expended across track read-out.

11. Associated with these displays is the type 9476 computer, *see* Part 3, Sect 3, Chap 5, and these items together with the compass and doppler 62M radar constitute the complete system.

TYPE 9478B AND F DISPLAY UNITS

Principle

12. The 9476 computer receives a resolution angle, equal to TMG—required track, from the 9478 display. The doppler distance gone is resolved about this angle into along and across required track components which are then fed, as pulse trains, to the 9478 display. M-type motors in the display unit convert the pulse trains into shaft rotations to drive the veeder counters.

Description

13. The display unit shows the aircraft's displacement from the required track, either to the left or right, up to a maximum of 99.9 n.m. It also shows the distance-to-go along the track up to a maximum of 999 n.m. The unit has a twin track facility which enables the navigator to pre-set the required track data for the next leg. The change-over from one leg to the other can be manual or automatic; if the latter is selected then the change-over will occur when the distance-to-go counters reach 000 nm. The system will operate in both forward and reverse directions according to the input signals, and the indicated position may be adjusted at any time, but there is no store to minimize the errors introduced while re-setting. The face of the 9478B display is shown in Fig 1.

14. The display unit weighs 4.5 lbs and measures $5\frac{3}{4}'' \times 4\frac{1}{2}'' \times 5\frac{1}{4}''$.

15. **Track Input to the Display.** The track input to the display unit is applied to the stators of a control differential transmitter (CDX), the rotor of which is turned manually by the operator to set required track. The output from the CDX, representing the resolution angle, is fed to the 9476 computer.

16. **Displays.** From Fig 3 the following displays can be identified:—

a. *Warning Lights.* The lights located in the top left hand and right hand corners of the unit indicate which of the twin track channels is in use. When the distance-to-go reaches 10 nm the illuminated lamp begins to flash.

b. *Veeder Counters.* There are three veeder counter displays. One for across track information and two for along track displays *ie* leg A and leg B. Each display is driven by an M motor, though only one

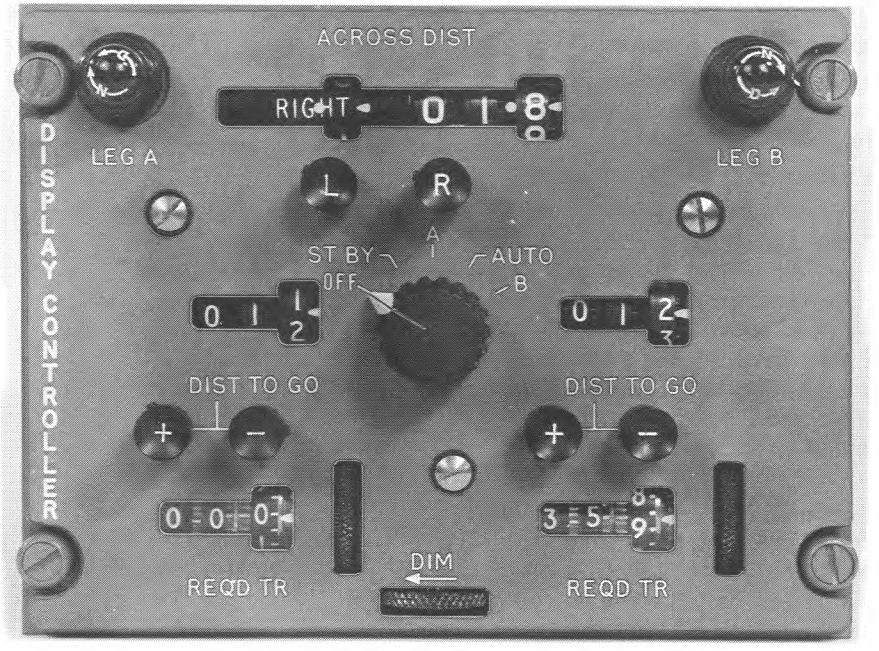


Fig 3 Display Unit Type 9478B

of the along track motors is driven at a time.

The face of the unit is covered by a perspex plate into which electro-luminescent panels are let to provide illumination for the controls and displays.

17. **Controls.** All the operating controls for the system are on the face of the unit, these are:—

a. *Facility Switch.* A five position switch marked OFF, ST/BY, A, AUTO and B is the main control for the display. The first two positions are self explanatory; positions A and B direct the distance gone information to the respective track channels while AUTO ensures automatic switching to the next channel once 000 nm is displayed on the channel in use. use.

b. *Set Track Controls.* Two thumb wheels enable the required tracks to be set up for channels A and B as required. The least significant counter is calibrated at intervals of 0.5°.

c. *Re-set Buttons.* Below each of the three counter displays are two reset buttons, marked L and R for the across display and + and − for the along displays. These buttons, which are used for setting up as well as resetting purpose, increase the displayed distance in the sense marked. Each button has a fast and slow reset facility; if the buttons are gently depressed the slow speed will be obtained while the fast speed is produced by pressing the switches firmly.

d. *Dimming Control.* The dimming control governs the intensity of the illumination provided by the electro-luminescent panels.

Operation

18. Prior to take-off the display unit should be set up as follows:—

a. Switch on the doppler, which in turn passes power to the computer.

b. Select ST/BY on facility switch.

c. Adjust dimming control as required.

d. Set required tracks for leg A and leg B on the appropriate counters.

e. Set distance-to-go for Leg A and Leg B on the appropriate distance-to-go counters.

f. Set across distance to 000 nm.

Just prior to take-off select A or B as required on the facility switch, this will direct the doppler information to the appropriate channel. If automatic change-over is required select AUTO; if not the next channel will have to be selected on the facility switch when 000 n.m.-to-go appears. When the second channel is operating the third leg can be set up on channel one.

19. **Resetting.** There is no store facility with the 9476B computer and resetting factors are introduced simply by pressing the appropriate re-set button while the computer is operating.

20. **Error Transfer at Turning Points.** In order to have an automatic change-over from one track to another without error it is advisable that the aircraft should be on track for the last few miles up to the turning point. The reason for this is that immediately after the automatic changeover, the reading on the ACROSS DIST counter refers to the new track; moreover with the 9487F display the appropriate demand signal will be sent to the autopilot. In Fig 4 the ACROSS DIST reading of 3 n.m. left will relate to the new track after the changeover and will of course be incorrect. If this information is actioned the aircraft will arrive at position D and not at the indicated position C. It will be seen that an along-track error also results after the turn. In a similar way in this example, an along track error at the turning point, *ie* turning early or late, will result in an across-track error on the next leg which will not be indicated on the ACROSS DIST counters. In general any errors in actual or indicated positions, or both, at the turning point will result in errors relative to the new track, the sense of which will depend on the change of track angle. These errors may be eliminated only by accurate resetting of the counters to read the correct position relative to the new track.

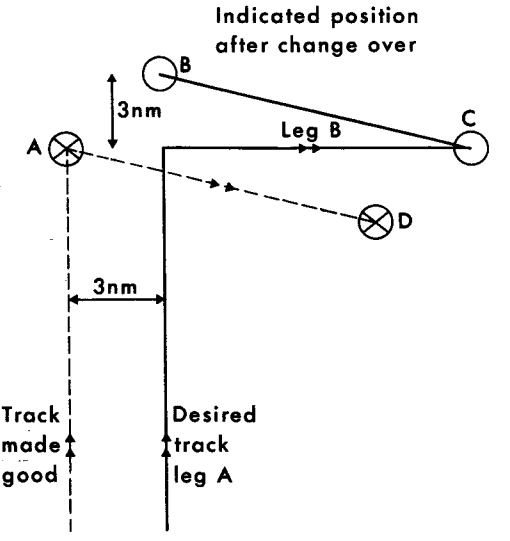


Fig 4 Turning Point Errors

21. **Turning Circles.** Accurate navigation of high-speed aircraft involves the use of precisely calculated turning circles at each turning point. The change-over from one track to another in this equipment is instantaneous and therefore an error very much akin to the across-track error at the turning point will arise. This error can be eliminated only by careful calculation of the distances involved and accurate resetting of the counters relative to the new track. It is desirable to reset the counters to a good fix as soon as possible after a turn, particularly if a large track change has been made.

22. **The Display Unit as a Grid Position Indicator.** One channel of the display unit may be used as a small area grid position indicator. The track counter should be set to read 000 which will be Grid North and the aircraft flown relative to Grid North by setting grivation on the VSC. The DIST TO GO counter will then indicate northings and the ACROSS DIST counter eastings. Distances right or left of the Grid North line

through any selected datum position will be shown directly on the ACROSS DIST counters, up to a maximum of 99.9 nm left or right. Distances north or south of the datum (relative to the grid) will be shown on the DIST TO GO counters, the value decreasing to the north and increasing to the south since the computer displays distance-to-go in relation to 000 Grid and not distance covered as the ACROSS DIST counters do. It will be found convenient to set a value on the DIST TO GO counters, *eg* 500 nm. The grid will then be as shown in Fig 5.

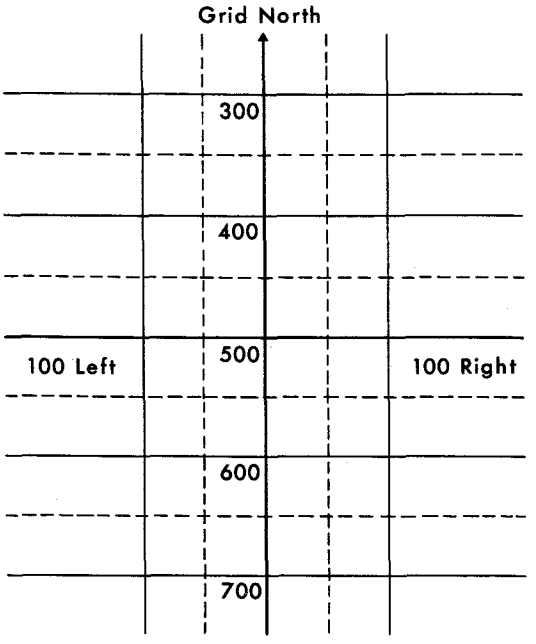


Fig 5 Small Area Grid

TYPE 9478F DISPLAY UNIT

Description

23. The 9478F display can be considered as a 9478B equipment which provides an output, proportional to the across track displacement, to an autopilot. Apart from this there are no differences which are of interest to the operator.

TYPE 9478G DISPLAY UNIT

Introduction

24. The auxiliary across track display type 9478G supplements the facilities of the 9478F by providing an expanded across track display. It also provides an output, proportional to the across track displacement, to the autopilot.

Description

25. The 9478G equipment displays the displacement across the track set on the 9478F display, distances of up to a maximum of 9.99 n.m. left or right are shown. No along track information is given as can be seen from Fig 6.

26. The display is inoperative until DROP is selected on the NORM/DROP switch. In this position the across track component of the doppler distance gone is switched from the 9478F to the 9478G display, and the across track display on the former is frozen.

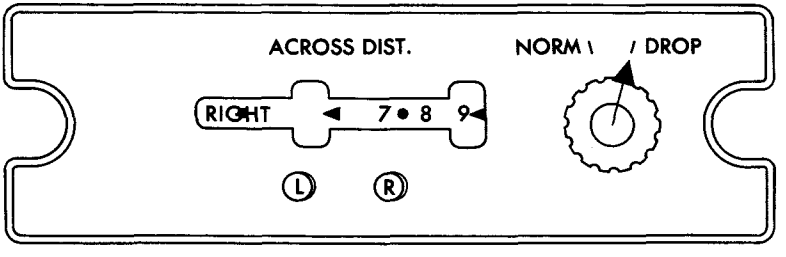


Fig 6 Type 9478G Display

The display on the 9478G can be changed using the reset buttons.

27. The unit weighs 2 lbs and measures $4\frac{3}{4}'' \times 1\frac{7}{8}'' \times 5\frac{1}{4}''$.

28. **Displays and Controls.** From Fig 6 the following display and controls can be identified:—

a. *Veeder Counters.* The veeder counter display and the associated potentionmeter providing the feed to the autopilot are identical with the corresponding components in the 9478F display. The potentiometer output from the 9478G is correctly scaled by reducing the voltage across it *via* a transformer in the 9476F computer.

b. *Reset Buttons.* The buttons marked L and R below the veeder counters enable the display to be adjusted. Each button has a two speed reset facility; if the buttons are gently depressed the slow speed will be obtained while the fast speed is produced by pressing the switches firmly.

c. NORM/DROP. When the facility switch is in the NORM position the 9478G unit is inoperative, and all doppler information is led to the 9478F. In the DROP position a times 10 factor is introduced into scaler 2 of the computer; the across track information is channelled to the 9478G display and the rate of change of the along track display on the 9478F is increased ten times.

The illumination of this unit is governed by the dimming control on the 9478F display.

Operation

29. The 9478G display will be used primarily in supply dropping operations, and it is beyond the scope of this chapter to detail drills for specialized tasks such as these. However there are certain points in the operation of this equipment which should be made here to prevent confusion.

30. Firstly, the across track display on the 9478G must be set to that on the 9478F before DROP is selected; there is no automatic synchronization of these displays. Alternatively, the 9478G can be set up to a fix or pinpoint.

31. Secondly, once DROP is selected the rate of change of the along track information is increased by ten. Thus if with 20 n.m. to go DROP is selected, the turning point will be indicated after travelling only 2 n.m. Furthermore, if the facility switch on the 9478F is to AUTO the doppler information will be resolved about the next track angle for the remaining 18 n.m. and the next leg distance will be reduced by 180 n.m. during this period. These dangers can be avoided either in any of the following ways:—

a. Increasing the distance-to-go along track by ten before selecting DROP (and, as there is no store, accepting the errors introduced during this reset).

b. Ensuring that AUTO is not selected and ignoring the along track display once DROP is selected.

c. Selecting DROP at the end of a leg, and setting ten times the distance-to-go on the next leg.

Printed in the UK for Her Majesty's Stationery Office by Hobbs the Printers
(1102/87) Dd8061612 C36 6/87 G3371

CHAPTER 7

THE FERRANTI NAVIGATION DISPLAY AND COMPUTER

CONTENTS

	Paras
Introduction	1
The Navigation System	2 – 5
THE NDC SYSTEM	
General	6 – 12
Destinations	13 – 22
Offsets	23
Offset Storage	24 – 26
Track Memory	27 – 29
Planned Fixing	30
Random Fixing	31
Weapon Aiming Data	32
Reversionary Modes	33
The Range and Bearing Indicator (RBI)	34
NDC Operating Controls	35
Indicator Lamps	36 – 39
Monitoring Facilities	40
THE PROJECTED MAP DISPLAY	
General	41
The Map and Map Drive	42 – 46
Map Changing	47
Setting up the Map	48 – 49
Optical System	50
Map Projection System	51 – 55
Counters Projection System	56
Map Orientation	57

Introduction

1. The Ferranti Navigation Display and Computer (NDC) forms part of the FE541 Inertial Navigation Attack System. This chapter deals with the NDC; however, as it is part of the FE541 system it cannot be considered in isolation. The main units of the FE541 are shown diagramatically in Fig 1. The inertial platform, present position computer and navigation controller are covered in Part 3, Sect 4, Chap 3.

The Navigation System

2. The navigation system is designed to provide steering signals to allow a pre-planned track to be flown from base to a planned target with facilities to allow an attack on an unplanned target, *eg* a target of opportunity. The track

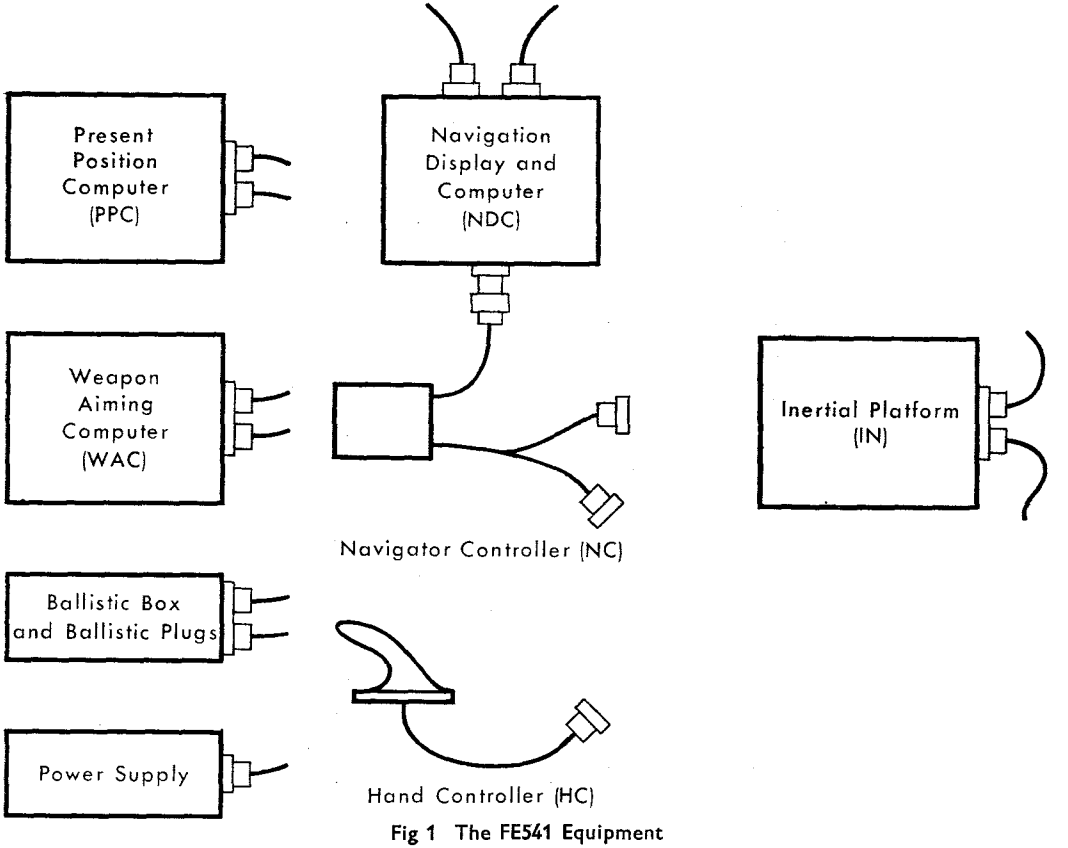


Fig 1 The FE541 Equipment

consists of a number of rhumb-line legs, each leg being a track between two planned destinations. At each stage of the flight the system computes and indicates heading from the present position to the next selected destination. The present position information is obtained from the PPC in terms of latitude and longitude.

3. The steering signals are displayed on the NDC range and bearing indicator (RBI) which indicates demanded heading and range-to-go to the next selected destination, true heading and drift. In addition a moving map display continuously indicates the aircraft's present position on a map of the ground over which the aircraft is flying. Superimposed on the map are latitude and longitude counters, as shown in Fig 2, giving the latitude and longitude of the present position and a look-up counter allowing certain other parameters to be displayed, *eg* ground speed and time to run to destination. The map display can be track or North orientated.

4. The navigation system can cover approximately a 1,000 nm square and includes facilities for taking fixes to check the accuracy of the system and if necessary to up-date the latitude and longitude information in the present position computer.

5. Eastings and northings from present position to the next destination and range information are supplied to the weapon aiming system.

THE NDC SYSTEM

General

6. The navigation computer is designed to provide the steering information necessary to fly a rhumb-line track between a number of points (destinations) whose co-ordinates are stored in terms of latitude and longitude on pairs of potentiometers. Generally the co-ordinates of the destinations are set in at base but there are stores suitable for setting in destination co-ordinates during flight. The destination potentiometers are set up so that when the wipers are at the mid-position of the potentiometers this corresponds to the required destination and the outputs are zero.

Fig 2 Navigation Display and Computer (NDC)

7. Fig 3 is a simplified block diagram showing the basic system. It can be considered in two sections. The first section consists of the latitude and longitude servos which drive the moving map, the latitude and longitude counters and the destination stores. The latitude and longitude servos position the map, counters and destination stores to correspond to the aircraft's present position. The second section consists of the navigation computer and the range, demanded heading, heading and drift servos. These drive the RBI to provide the steering information from the aircraft's present position to the next required destination.

8. The system is set up at base so that the latitude and longitude counters read the latitude and longitude of base and the base position appears under the present position circle on the moving map display. The destination potentiometer outputs are set up to read zero at the latitude and longitude of the destination so that at base the outputs correspond to the distance from base to the destination. The inputs to the latitude and longitude servos in flight are the latitude (ϕ) and longitude (λ) of the aircraft's present position obtained from PPC. These inputs are used to drive the latitude and longitude servos so that the latitude and longitude counters, the destination potentiometer wiper positions and the moving map all represent the aircraft's present position.

9. In order to fly the pre-planned track from base via the stored destinations to the target, each destination is selected as the aircraft overflies the previous one. The inputs to the navigation computer are the outputs of the selected destina-

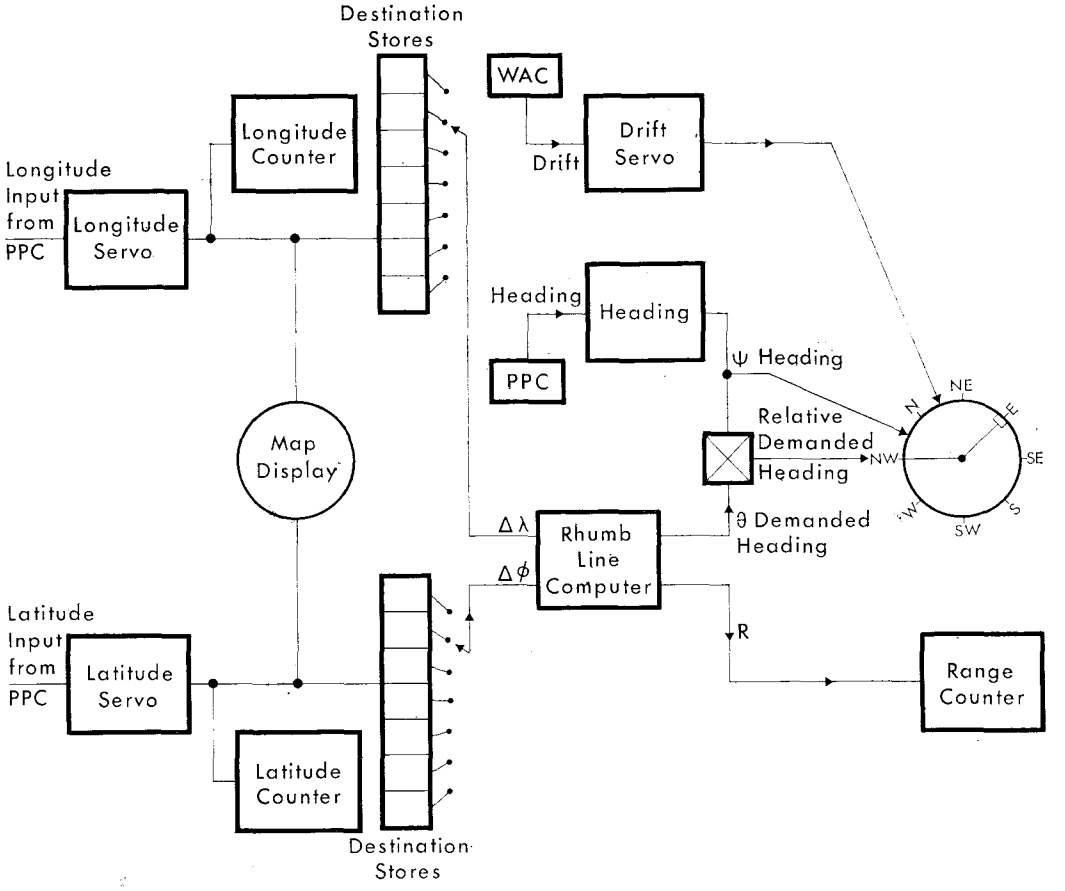


Fig 3 System Block Diagram

tion potentiometers and are, therefore, voltages proportional to the differences in latitude ($\triangle\phi$) and longitude ($\triangle\lambda$) between the aircraft's present position and the selected destination. From these inputs the computer calculates a heading to fly a rhumb-line track and range to the destination.

10. The control element in the computer is a resolver on the demanded heading servo-shaft which provides range as a voltage output and demanded heading as a shaft position. The range output operates the range servo which drives the range counter. The counter therefore reads range to the destination in miles. The demanded heading servo has the rhumb-line track input modified by an input from the drift so that the shaft rotation represents demanded heading and not demanded track.

11. True heading is obtained from the PPC via a servo in the power supply unit which drives the compass card. True heading and demanded heading are passed through a differential to give relative demanded heading which drives the forked pointer on the range and bearing indicator.

12. The input to the drift servo is obtained from the WAC. The drift servo-shaft drives the demanded heading pointer which moves round the rim of the compass card.

Destinations

13. The term "destination" is generally used to cover destinations, targets, identification points and the base. The co-ordinates of all these may be "stored" in the NDC and the range and heading to the selected destination can then

be displayed. A typical flight plan consists of a series of legs between successive desinations which collectively define the route to be followed. The following paragraphs describe how the destinations are stored.

14. The co-ordinates of the following destinations are stored in terms of latitude and longitude.:

a. Normal destinations which are usually points stored before take off:

(1) Destination 1.
(2) Destination 2.
(3) Destination 3.
(4) Destination 4.
(5) Base (the based may be regarded as a special case of a normal destination).

b. "In flight" destinations, which are more easily stored in flight:

(1) In flight 1 destination (IF 1).
(2) In flight 2 destination (IF 2).

c. Unplanned destination (U/PD), which is a point stored by the pilot as he flies over it for use later as a destination.

15. All these stores except IF 1 consist of potentiometers which are geared to the latitude servo-shaft to accommodate a range of ± 1,200 nm approximately from their zero (mid-point) position. The IF 1 latitude store, which also has a range of ± 1,200 nm, consists of potentiometers and a synchrotransformer. At ranges greater than 30nm the potentiometer provides a resolution of 0·5 nm, at 30 nm a resolution of 0·2 nm and at 6 nm an accuracy of 0·03 nm (180 ft). The appropriate part of the store is switched in as the range-to-go to the destination decreases through 30 nm and 6 nm.

16. The two remaining stores, which are only driven by the servo when required, are the unplanned destination (U/PD) store and an accurate store used for storing changes in plan range during unplanned attacks, a target of

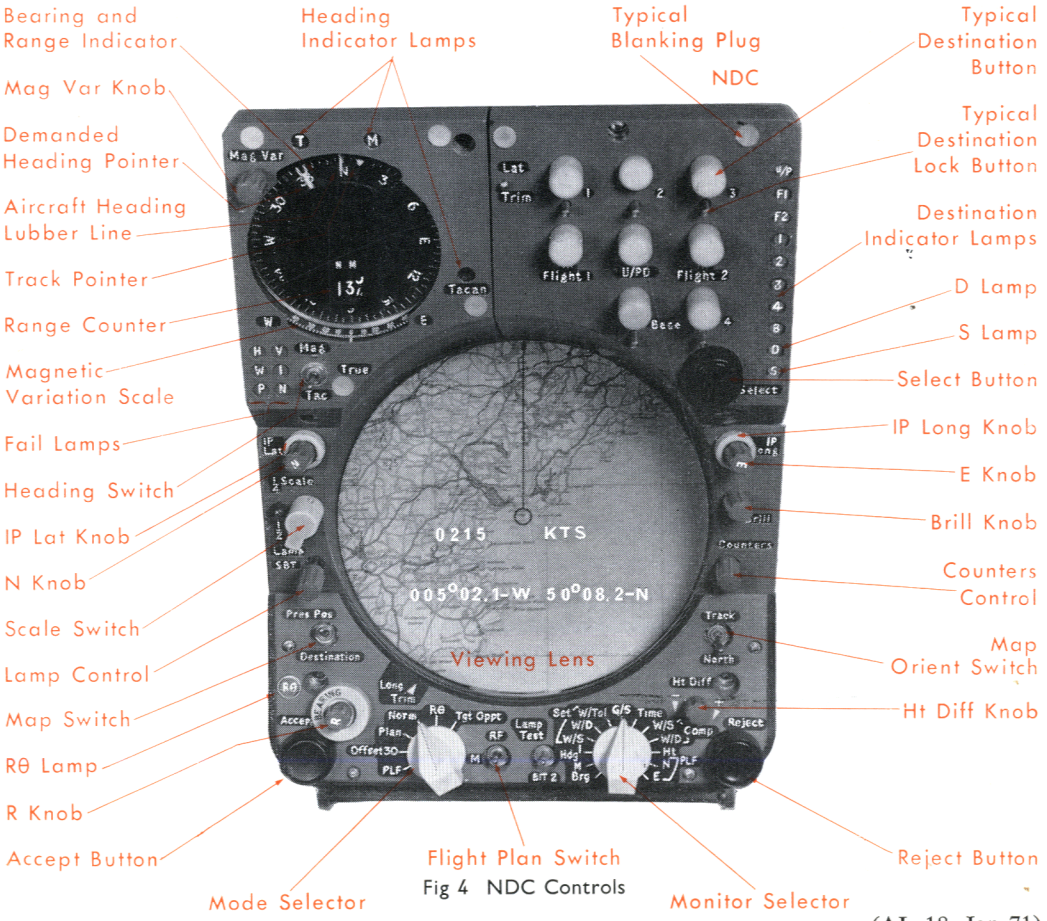


Fig 4 NDC Controls

opportunity or for taking a random fix. The accurate store is spring-loaded to zero and therefore can only be used for storing co-ordinates which are to be used immediately and it is referred to as the present position store in the following description.

17. The present position store consists of a potentiometer giving a coarse output and a synchrotransformer giving the fine output. The range of this store is 30 nm from the position where the store was set up.

18. The U/PD store is a potentiometer with a range of ± 1,200 nm from base.

19. **Normal and In Flight Destinations.** Normal destinations (base and 1 – 4) are normally stored on the ground but may be inserted during flight. The procedure is as follows (*see* Fig 4):

a. Select Map Switch to Destination.

b. Press desired destination button followed by the Select button, the map and counters will then run out to the previously-set destination and the RBI range counter will run to zero.

c. Press the Fix Take Button on the HC momentarily to stop any counter movement and then press the selected Destination Lock Button to hold the destination potentiometer at zero, (*see* para 6).

d. Using the HC, insert the latitude and longitude of the required destination.

e. Press another destination button which releases the Lock Button then press the Reject Button to unfreeze the system.

f. Repeat for other destinations.

IF1 and IF2 destinations are rather easier to store, particularly in flight, since only the actions at a, b and d above are required. These destinations may therefore be used alternately to give a simple leap-frogging technique, for example during ferry flights.

20. **Unplanned Destination.** To store an unplanned destination the HC fix button is pressed when directly over the point. The U/PD button is then pressed and when the RBI range has run to zero the Reject Button on the NDC is pressed and then the original destination reselected. Subsequently, if U/PD is selected, range and steering to the unplanned destination will be displayed.

21. **Target of Opportunity.** To store a present position for the purpose of returning to it immediately, the HC fix button is pressed when directly over the position and then the NDC mode selector is selected to Tgt Oppt. The range and steering to the chosen position are then displayed. The procedure may be used either to lead the pilot back to the stored position for the purpose of engaging a target on which a planned attack is to be made, or for some other purpose such as the return to a landing site which has been overflown.

22. **Destination Display.** Any destination once stored can be displayed on the map and on the latitude and longitude counters by selecting the destination and setting the map switch to Destination. With this switch set to Pres Posn, the range to the selected destination from present position is displayed on the range counter of the RBI to a maximum of 500 nm. Track and demanded heading are indicated against the compass card as already explained.

Offsets

23. In addition to the destination stores, three additional points, each in terms of an offset from one of the planned destinations, may be stored. The offsets are as follows:

a. *Rθ (Range and Bearing) From Any Stored Destination.* The range of this store is 500 nm from the stored destination.

b. *IP30 (Identification Point).* This is generally used with an IF 1 target but may be used with any destination. When used with an IF 1 target the IP30 would be an easily identifiable point within 30 nm of IF 1. A fix would be taken as the IP was overflown.

c. *PLF Target.* This is a point which is stored with respect to IF 1 and is within 6 nm of IF 1. In this case IF1 is the identification point at which a precise local fix is taken before proceeding to attack the PLF target.

Offset Storage

24. **Rθ Offset.** To store an Rθ offset, the mode selector is set to Rθ and the map switch to Destination. The map and counters then indicate the position of the Rθ point while the RBI shows the magnitude and direction of the offset from the associated destination. With the map switch set to Pres Posn the range and steering to the Rθ point from the present position are displayed. If, with the Rθ facility

in use, a new destination is selected, the Rθ offset will automatically be rejected, the Rθ lamp will extinguish and the range and steering to the selected destination will be displayed. Should the Rθ offset be required for the new destination it can be applied by setting the mode selector to Norm and then back to Rθ.

25. **IP30 Offset.** To store an IP30 offset the controls on the NDC are used as follows. The associated destination (usually IF 1) is selected. Then with the map switch to Destination and Offset 30 selected, the IP Lat and IP Long knobs are adjusted to set the co-ordinates of the IP30 identification point on the latitude and longitude counters. With the map switch returned to Pres Posn and the associated destination selected, the range and steering of the IP30 identification point are displayed.

26. **PLF Offset.** To store a PLF offset, the easterly, northerly and vertical components of the offset from IF 1 are set (in feet) using the E knob, N knob and Ht Diff knob as appropriate, and using the monitor counters as a read-out. When PLF is selected, the displays will refer to the associated IF 1 destination until the final stages of a PLF attack, when the PLF offsets will automatically be added and the displays will refer to the PLF target.

Track Memory

27. When a destination is selected, the NDC computes mid-latitude, rhumb-line track corrected for drift (*ie* demanded heading) and range directly to a position lying 14 nm ahead of the present position. This enables the planned track to be regained if deviation is made from it. If there is no deviation, demanded heading to the destination and to the point 14 nm ahead of present position are identical. If the aircraft deviates from the planned track however, a track memory potentiometer applies corrections to the demanded heading to regain the computed track.

28. Provided the angle between planned track and demanded track does not exceed 40° and track memory is not inhibited (*see* para 29), the steering instructions are continuously and exponentially directed at a point on the computed track which is 14 nm ahead of present position. When within 14 nm of the destination the track memory facility is switched out and demanded heading is given directly to the destination.

29. A new memorized track can be obtained by repressing the Select button. Track memory is inhibited in the following conditions:

a. When the mode selector is set to Rθ or Tgt Oppt.

b. During random fixing (*see* para 31).

c. If the map switch is set to Destination.

d. If TACAN is valid and is selected at the heading switch.

e. If range to destination is less than 14 nm.

Planned Fixing

30. Operation of the HC fix button disconnects the NDC from the PPC and effectively freezes all the displays at that moment. A planned fix is taken by pressing the fix button at the moment of overflying the selected destination. If the navigation is correct then the range to the destination as displayed on the range and bearing indicator will be zero and the next destination can be selected. However, any error in the navigation will be displayed as a range and bearing and the pilot can use this error to either update the PPC or not, by pressing the Accept or Reject button.

Random Fixing

31. A random fix is taken by pressing the fix button when a prominent feature which appears on the map is overflown. If, when the pilot has time to look at the map, the feature is not in the middle of the present position circle, he can enter the position of the fix as displayed on the map by momentarily operating the spring-loaded toggle flight plan switch to RF and then, using the HC, move the map and latitude/longitude counters until the feature lies in the centre of the present position circle. The distance and direction of this movement is displayed on the range and bearing indicator and is a measure of the navigational error in the system which may then be used for updating the PPC as in a planned fix if desired.

Weapon Aiming Data

32. The NDC supplies the following data defining the target's position to the WAC:

a. Plan range.

b. Range northings.

c. Range eastings.

d. Height offset when required.

Reversionary Modes

33. Details of the three reversionary modes available are given in Part 3, Sect 4, Chap 3, paras 36 – 39.

The Range and Bearing Indicator (RBI)

34. The range and bearing indicator is a three-inch panel mounted instrument at the top left-hand side of the NDC as shown in Fig 4. The display is essentially a heading display and provides the following information:

a. Heading, indicated by means of a servo-driven compass card read with respect to a fixed lubber line. Heading may be either true or magnetic.

b. Aircraft track, indicated by means of a servo-driven pointer, which moves round the rim of the compass card.

c. Drift angle, displayed as the angle between the track pointer and the lubber line.

d. Demanded heading to regain or maintain the required computed track to the next destination, displayed by means of a servo-driven forked pointer.

e. Range-to-go to the next destination, shown by means of a servo-driven three-digit counter reading up to 500 miles.

f. Magnetic variation, which can be set on the instrument by means of a knob which drives a dial with respect to a fixed index at the bottom of the indicator.

NDC Operating Controls

35. The controls for operating the system consist of:

a. Destination buttons, associated with a select button, enabling destinations to be stored and selected in flight as required. Destination lamps indicate the selected destination.

b. A flight plan switch which is spring loaded to M (Manual) and flicked to RF when a random fix is taken.

c. A switch allowing the map and latitude and longitude counters to show present position or the selected destination in the present position circle.

d. A fix take button, on the HC, enabling fixes to be taken with facilities for entering or rejecting them.

e. A mode switch enabling modes of navigation or attack to be selected. The modes of navigation which can be selected are navigation to a normal destination and to an IP30 or Rθ offset from a normal destination. The modes of attack which can be selected are attacks on a planned target, a target of opportunity or a precise local fix (PLF) target.

f. Controls allowing the Rθ, IP30 and PLF target offsets to be incorporated.

g. A scale change switch which optically changes the moving map from $\frac{1}{2}$ million to $\frac{1}{4}$ million scale, and vice versa.

h. A switch which allows the map to be track or North orientated.

j. A heading switch which allows true, magnetic or TACAN heading to be selected for display on the range and bearing indicator.

k. A knob, for setting in magnetic variation, which positions a variation scale with respect of a fixed pointer.

l. A height difference control which sets the height difference between the identification point and the target when an attack on a PLF target is to be carried out. This control is also used during a planned attack to store the difference between aircraft and target height when the aircraft enters the transition phase of the attack.

m. A monitor selector switch which allows certain parameters to be displayed on the monitor look-up counter superimposed on the moving map display.

n. Fail lamps which indicate failure of various parts of the complete navigation and weapon aiming system.

Indicator Lamps

36. **Fail Lamps.** Six fail lamps on the front of the NDC indicate the progress of the built-in-tests. The purpose of the built-in-tests is to detect the presence of faults in the equipment and to indicate which unit(s) are most likely to be faulty. The lamps, as shown in Table 1, are named with the initial letter of the function which they monitor:

FAIL LAMP	FUNCTION
I	Inertial
N	Navigation
W	Weapon Aiming
P	Power Supply
V	Velocity
H	Heading

Table 1 Fail Lamp Indicators

37. **Heading Lamps.** The heading lamps light to indicate the selection made at the heading switch.

38. **Rθ Lamp.** The Rθ lamp lights when Rθ is selected at the mode selector.

39. **Destination Lamps.** The destination lamps indicate which destination has been selected. They are labelled as shown in Table 2.

U/PD	..	Unplanned
F1	..	In Flight 1
F2	..	In Flight 2
1, 2, 3, 4	..	Normal Destinations
B	..	Base
D	..	Lights when the map switch is set to Destination
S	..	Lights when the Select Button is pressed and remains lit until the process of selection is complete

Table 2 Destination Lamp Indications

Monitoring Facilities

40. Depending on the setting of the NDC monitor selector, the following parameters will be displayed on the counters:

a. *BRG:* Displays the bearing indicated by the vertical aiming line of the head-up display during alignment.

b. *HDG* 1. Displays aircraft inertial heading.

c. *HDG M.* Displays aircraft magnetic heading from the C2G.

d. *SET W/S.* Displays the manually-inserted wind speed set in by the W/S control on the Navigation Controller (NC).

e. *SET W/D.* Displays the manually-inserted wind direction set in by the W/D control on the NC.

f. *W/TOL.* Displays the manually-inserted wind tolerance set in by the *W/TOL* control on the NC.

g. *G/S.* Displays inertially-computed ground speed.

h. *TIME.* Displays time-to-go to currently-selected destinations.

j. *W/S COMP.* Displays the inertially-computed wind speed.

k. *W/D COMP.* Displays the inertially-computed wind direction.

l. *HT.* Displays the height difference in feet between IF 1 and PLF target as set in by the HT DIFF control.

m. *PLF N.* Displays the offset distance in feet northings between IF 1 and the PLF target.

n. *PLF E.* Displays the offset distance in feet eastings between IF 1 and the PLF target.

THE PROJECTED MAP DISPLAY

General

41. The moving map display consists of a 35 mm coloured filmed map which is projected optically on to a screen and viewed through a 6 inch diameter lens.

The Map and Map Drive

42. The map is prepared from standard 1 : 250,000 Lambert conformal or Universal Transverse Mercator projections, processed to

convert the parallels and meridians into a rectangular grid. The resulting distortion is automatically resolved within the system during operation. Since only a small portion of the rectified map is used at any one time, pictorial inaccuracy is not noticeable.

43. The filmed maps are stored on reels in the form of strips, a strip covering approximately 60 nm N/S in width and 1,000 nm minimum E/W in length. Each reel of film covers an area approximately 1,200 nm N/S and 1000 nm E/W and consists of 20 strips of film joined together consecutively forming one continuous strip. Movement of the map image in longitude is thus obtained by using the longitude shaft output to drive the film sprocket. Movement of the map image in latitude is obtained by using the latitude servo-output to drive the light transfer prism across the width of the film. In practice the width of the film is made greater than 1° of latitude (60 nm) so that an overlap of approximately 7·5 nm is provided at the northern and southern edges of the film. Each of the strips of film is approximately 15 in long and the total length of the film is approximately 30 ft, allowing $2\frac{1}{2}$ ft at each end for a test graticule, title frame and leader.

44. To simplify the projection equipment for preparing the filmed map, the latitudes between 0° and 70° N or S have been divided into four bands as follows:

a. 0° – 45° N or S (band 1).

b. 35° – 55° N or S (band 2).

c. 42° – 62° N or S (band 3).

d. 51° – 71° N or S (band 4).

Because of convergence of the meridians at the N and S poles, each of these bands has its own E – W scale factor in terms of degrees of longitude per inch of film strip. At some latitude in each band there is a parallel of latitude which is neither compressed nor expanded in the rectification process and this latitude is known as the datum latitude ϕ_D for that particular band (*see* Fig 5).

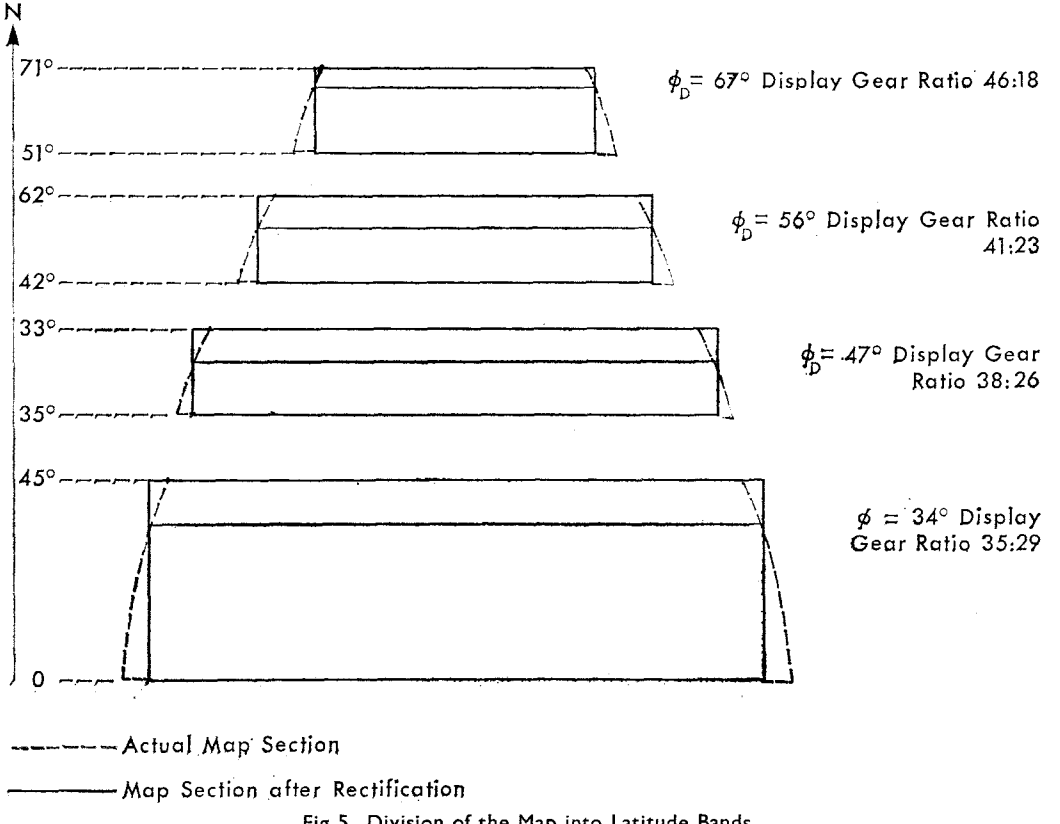


Fig 5 Division of the Map into Latitude Bands

45. To match the E – W factor of each band to the map drive, variable gearing providing four pre-selected gear ratios is interposed between the longitude servo and the longitude map drive. When a film is changed the appropriate gearing is selected corresponding to the scale factor of the film.

46. The longitude film drive is obtained from the longitude servo-shaft which drives a special sprocket designed to minimize lost motion along the length of the film. The latitude drive from the latitude servo drives the light transfer prism across the latitude axis of the film. Limit switches are operated by the light transfer prism drive at points which correspond to positions just outside the 1° latitude parallels. The limit switches initiate the automatic map change sequence.

Map Changing

47. When the light transfer prism moves outside the 1° parallels of the film strip, power is supplied to the map change servo. This servo drives the film sprocket and the light transfer prism through the appropriate gearing and differentials. This moves the longitude shaft through one strip length and the latitude shaft through 1° of latitude to an accuracy of better than 1/70 nm.

Setting up the Map

48. When a cassette is changed or a new map is loaded into the system, the latitude and longitude counters must be aligned with the latitude and longitude of the starting point of the map. If the latitude and longitude of the starting point of the map are known then the latitude and longitude counters can be set before the cassette is mounted in the system using the random fix procedure, *ie* the hand controller inputs are used to move the latitude and longitude counters. The cassette is then mounted and any slight misalignment in latitude due to the tolerance which may exist in the positioning of

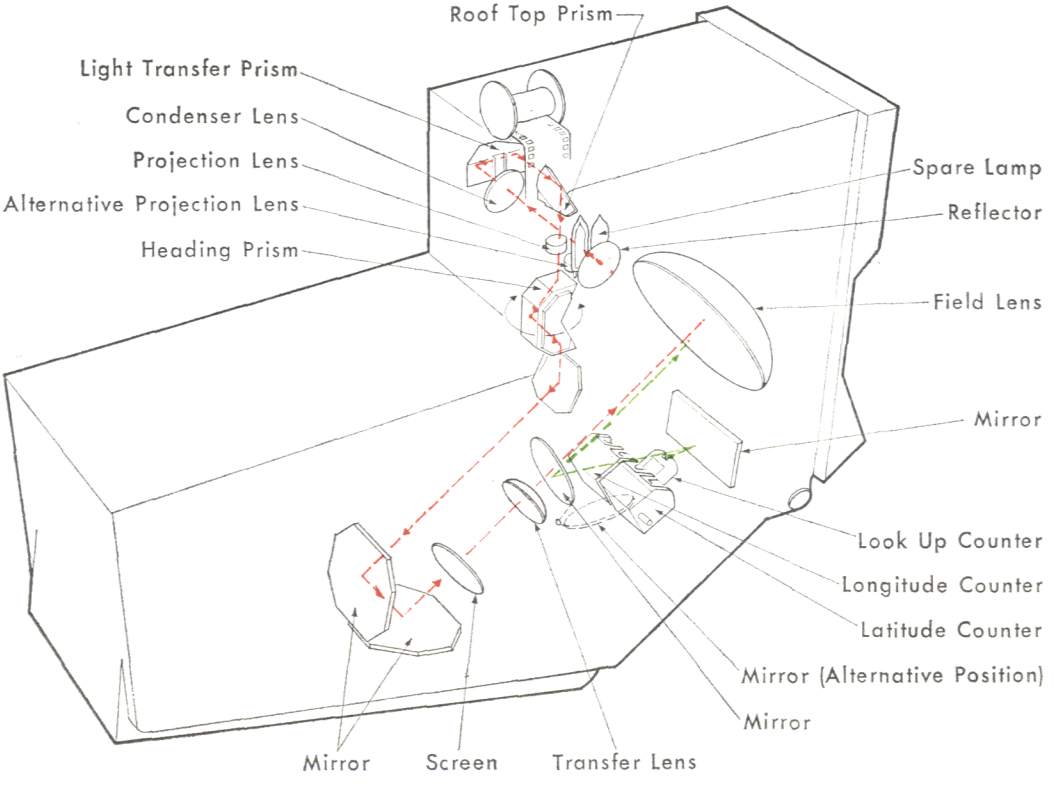


Fig 6 The Optical Projection System

the film in the film gate is taken out using the latitude trim knob. Operation of the trim knob changes the bias at the input to the map change servo and therefore changes the balance position of the latitude and longitude shafts. Since longitude is also affected by operation of the latitude trim knob, the cassette must then be removed and the longitude counter reset.

49. If the latitude and longitude of the film are not known these must first be found by mounting the cassette in the system and checking on the map display. The cassette is then removed and the procedure described above is followed.

Optical System

50. There are two separate optical systems included in the navigation display and computer. One is the projection system which displays the filmed map on the field lens, the other is a simpler system which projects the latitude, longitude and look-up counters on to the field lens.

Map Projection System

51. The map projection system, as shown in Fig 6, consists of a quartz iodine projector lamp which illuminates the film via the condenser lens and the light transfer prism. As stated earlier, the light transfer prism is driven by the latitude servo and this produces the apparent movement of the map in latitude. The film image is projected via the roof-top prism, the projection lens, the heading prism and mirrors on to an intermediate screen. The roof-top prism is driven in synchronism with the light transfer prism across the latitude axis of the film so that the section of the map being viewed is the section being illuminated.

52. The position of the heading prism determines whether the image of the map projected on to the screen is track or North stabilized. This prism is driven by the heading prism servo.

53. Engraved on the screen are the present position circle and a vertical line marker which represents aircraft track when the map image is track stabilized. The image produced on the screen is viewed via the transfer lens and the field lens, the final image being formed in the plane of the field lens.

54. The change of magnification from 1 : 250,000 to 1 : 500,000 is achieved by changing the projection lens. Operation of the $\frac{1}{4}/\frac{1}{2}$ scale switch on the front panel of the display mechanically moves one lens from the optical path and replaces it by the second lens. Similarly the operation of the lamp change switch replaces one lamp by the spare lamp which is mounted adjacent to it on the same carriage.

55. The viewing screen covers a map area of 14 nm diameter with $\frac{1}{4}$ (1 : 250,000) selected at the map scale switch, and 28 nm diameter with $\frac{1}{2}$ (1 : 500,000) selected. The central reference circle indicates aircraft present position with Pres Posn selected, or the destination co-ordinates with Destination selected; the diameter of the circle represents 1 nm at 1 : 250,000 scale or 2 nm at 1 : 500,000 scale.

Counters Projection System

56. The optical system for the projection of the counters on to the field lens consists of a lamp which illuminates the counters and two mirrors, one of which is curved, which project the image of the counters on to the field lens as shown in Fig 6. The curved mirror is moved away from the optical path of the map projection when the counters are selected off.

Map Orientation

57. The map may be North or track orientated by appropriate selection of the Map Orient switch on the NDC. Orientation is achieved by rotation of a heading prism which receives either a fixed voltage for North orientation, or a varying voltage from track information which continuously stabilizes the image to aircraft track.

DOVE PRISM

CONTENTS

	Paras.
Introduction	1
Image Inversion	2–5
Topographical Display Application	6–8

Introduction

1. The Dove Prism is a glass 45° prism used in optical instruments to rotate an image. It is illustrated in Fig. 1, and throughout the explanation which follows it will be assumed that the incident light is collimated and enters parallel to the base and normal to the edge AB.

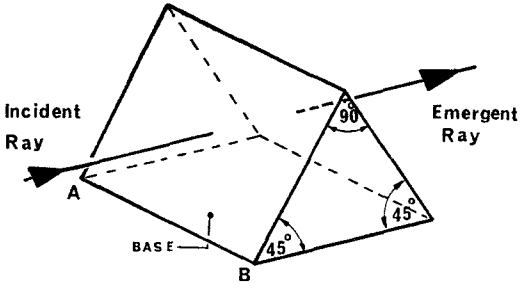


Fig. 1. Dove Prism

Image Inversion

2. Fig. 2a shows the path taken through the prism by rays of light when the image is perpendicular to the plane of the base. The image is inverted and emerges parallel to the incident ray. An image parallel to the base, Fig. 2b, emerges unchanged.

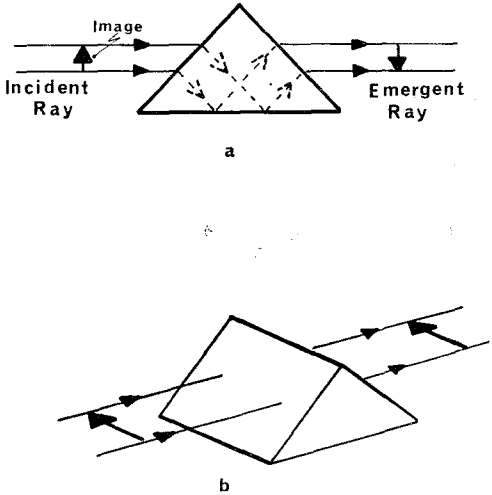


Fig. 2. Effect on Simple Images

3. If now a circular beam is passed through the prism, it too, will be inverted. The points 2 and 4 in Fig. 3 are in the plane of the base and will therefore retain their positions, but the points 1 and 3 will change places as will all points above and below the plane containing 2 and 4.

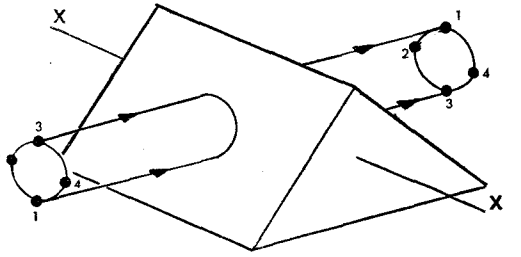
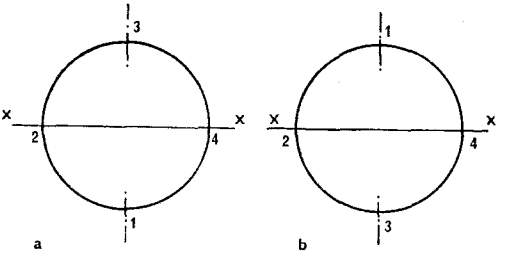


Fig. 3. Beam of Light Inverted

4. The plane about which the inversion takes place also contains the line XX in Fig. 3. All points on the image change places with their counterparts on the other side of this line. Thus we may use Figs. 4(*a*) and (*b*) to describe the action, looking along the direction in which the rays are moving in both diagrams. Fig. 4(*a*) shows the image on entry to the prism, while (*b*) shows it on emergence.



Incident Image Emergent Image

Fig. 4. Images Relative to XX

5. Fig. 5 shows the effect of rotating the prism by 45° about an axis along the beam direction. XX now lies along the quadrantal direction through points 5 and 7 and the image is therefore

inverted about that line. Points 3 and 4 change places, as do 1 and 2 and 6 and 8. Comparison of Fig. 5(*b*) with Fig. 4(*b*) shows that a 45° rotation of the prism has turned the emergent image through 90°. In general the image moves through twice the angle turned by the prism.

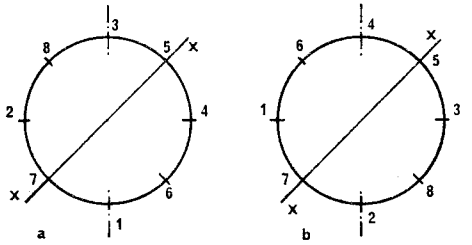


Fig. 5. Image Rotation

Topographical Display Application

6. In the Topographical Display Type C the image of the map enters the Dove Prism with west at the top of the beam. The other compass points are arranged as in Fig. 6.

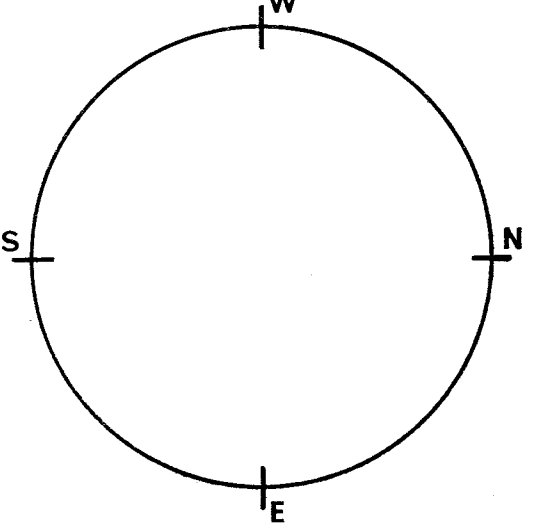


Fig. 6. Map Orientation on Incidence

7. **North Stabilized Mode.** In the north stabilized mode the prism is tilted to put XX through the image SE and NW points. The emergent image therefore appears as in Fig. 7(*b*) with north at the top of the screen. From the front of the screen the operator views the image in the conventional orientation of Fig. 7(*c*).

8. **Track Stabilized Mode.** In the track stabilized mode the prism is turned by half the track angle to produce a display with track angle at the top of the screen. Fig. 8 shows a rotation by the prism of $22\frac{1}{2}$° from the north stabilized mode position to indicate a track of 315°.

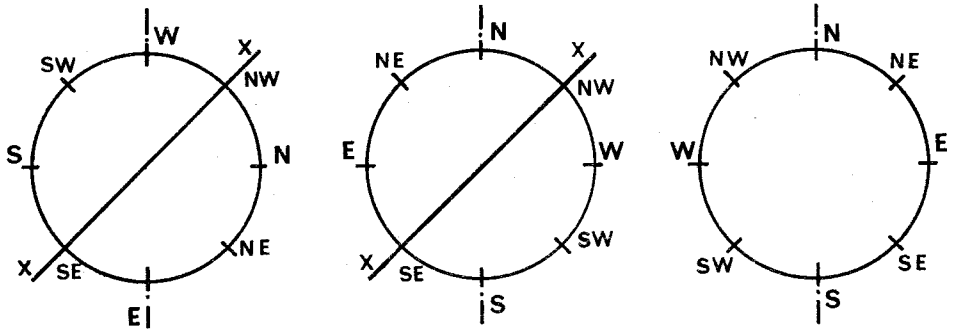


Fig. 7. North Stabilized Mode

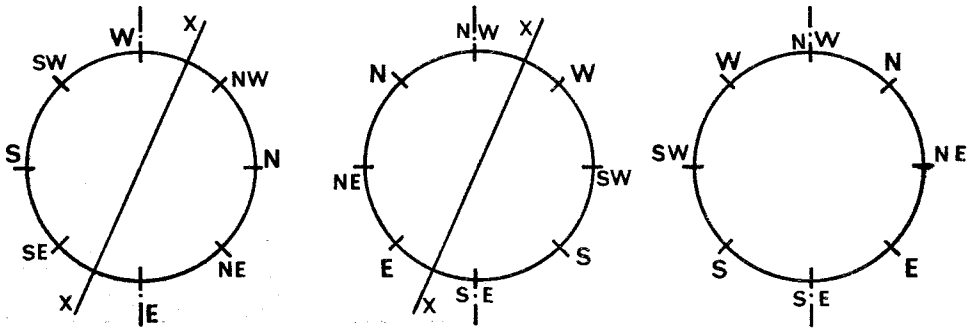


Fig. 8. Track Stabilized Mode

CHAPTER 8

GROUND POSITION INDICATOR MARK 7

CONTENTS

	Paras.
Introduction	1–2
PRINCIPLES OF OPERATION	
Basis of the Computation	3–6
Computing Mechanism	7
Heading, Drift and Track Mechanism	8–10
Groundspeed Mechanism	11–12
Co-ordinate Resolving Mechanism	13–14
Storage System	15–19
Northings Drive	20
Eastings Drive	21–23
Convergency	24
Lat/Long or Grid Display	25
Reset Facilities	26
Along/Across Track Mechanism	27–31
Wind Mechanism	32–34
True Airspeed	35
Wind Comparator...	36
EQUIPMENT	
General Description	37
Centre Display of Vector Information	38–43
Lat/Long or Grid Display	44–47
Along/Across Track Display	48–52
Control Section	53–56
Lighting	57
OPERATING PROCEDURES AND LIMITATIONS	
Switching On	58
Pre-flight Checks	59–60
In-flight Operation	61
Accuracy and Limitations	62–64

Introduction

1. The Ground Position Indicator (G.P.I.) Mk. 7 is an electro-mechanical analogue computer. From inputs of true heading, true airspeed, drift and groundspeed it computes and displays aircraft position (in lat/long or grid co-ordinates and in along/across track form) together with most of the vector information used in D.R. navigation. It is intended for installation in large transport aircraft where it will be the central element of a complete navigation system, a typical example of which is shown in Fig. 1. To provide a background, the instrument is here described in a Belfast aircraft environment. When fitted in other aircraft the input sources and output uses may differ slightly from those of the Belfast, but the principles of the instrument will not be affected.

2. In normal operation the G.P.I. Mk. 7 has inputs of groundspeed and drift from a Doppler radar, but provision is made for operation in a reversionary mode in which wind velocity is applied manually and combined with true airspeed for the display of ground position. The instrument itself is a self-contained unit mounted prominently on the navigation panel. All its automatic inputs are electrical and it requires power supplies of 200 volts a.c. at 400 c/s and 28 volts d.c. although provision is made for operation at 115 volts a.c. where required.

PRINCIPLES OF OPERATION

Basis of the Computation

3. The G.P.I. performs an analogue computation based on the navigation triangle of velocities. Given two sides of the vector triangle and the included angle, it solves the triangle for the third side. In the normal mode true airspeed, heading, groundspeed and drift are given and the wind velocity is solved. In the reversionary mode groundspeed and drift are found from the given true airspeed, heading, and wind velocity.

4. To perform the computation conveniently and to allow the reversionary mode of operation the G.P.I. resolves all the vectors into two, namely V1, which is the groundspeed along heading and V2, which is the groundspeed across heading. Fig. 2 shows how these two vectors are derived from the original triangle. One important angle not normally used in navigation computations is wind direction relative to heading, denoted by ø in Fig. 2 and derived simply by subtracting true heading from the true wind vector direction.

5. The velocities V1 and V2 may be derived in two different ways:—

(*a*) Given heading (θ), groundspeed (Vg), and drift (d), V1 is equal to Vg cos d and V2 is equal to Vg sin d.

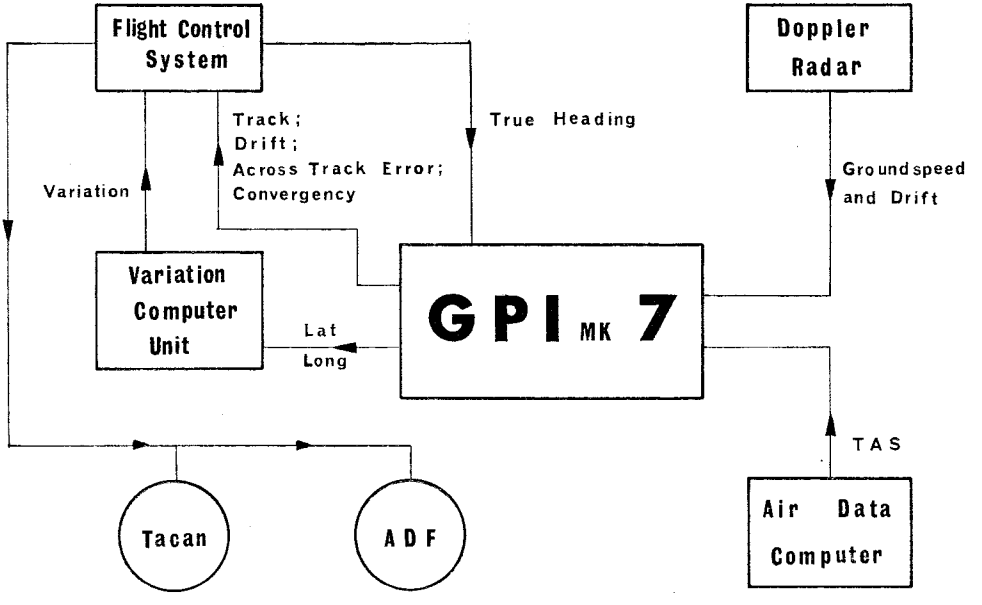
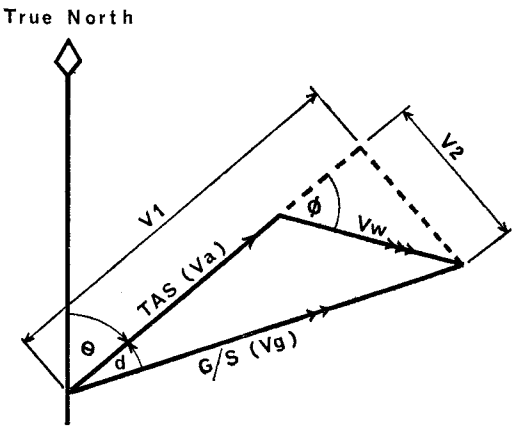


Fig. 1. Typical D.R. Navigation System Centred on G.P.I. Mk. 7



Θ	Heading
d	Drift
ϕ	Wind direction WRT hdg
V1	Component of Vg along hdg
V2	Component of Vg across hdg

Fig. 2. Quantities used in Computation

(*b*) Given heading (θ), T.A.S. (Va), and wind velocity (Vw), V1 equals Va plus the component of wind velocity along heading (Vw cosø) and V2 equals the component of wind velocity across heading (Vw sin ø). The relative directions of heading and wind will determine the signs of Vw cos ø and Vw sin ø.

6. The G.P.I. Mk. 7 computes V1 and V2 in both these ways and continuously compares the respective values from each source. If there is any difference between the two values of V1 or the two values of V2, then error signals are generated and used to correct the data in one or other source until they are equal. When the Doppler radar is functioning correctly then the values of V1 and V2 derived as in paragraph 5(*a*) (Vg cos d and Vg sin d) are assumed to be correct and the error signals are used to modify the wind velocity until the values of V1 and V2 (Va+Vw cos ø, and Vw sin ø) from the second source are the same as those derived from the Doppler. This is termed the normal mode of operation. When the Doppler radar is unserviceable or is in a memory mode then the G.P.I. Mk. 7 automatically switches to its reversionary or memory mode. In this event the wind velocity set on the instrument is assumed to be correct, and therefore the values of V1 and V2 determined from Va+Vw cos ø and Vw sin ø are assumed correct. The error signals are then used to change the values of groundspeed and drift accordingly until the two V1 and two V2 values are equal. In both modes of operation the Va vector is left with its input values of heading and T.A.S. unaltered. There is however a facility for setting a value of T.A.S. manually in the event of the air data computer being unserviceable.

Computing Mechanism

7. The schematic diagram in Fig. 3, on a folding leaf at the end of the chapter, shows how the values of V1 and V2 are derived, compared and then applied to the rest of the system to give the outputs and display. For purposes of explanation the mechanism will be divided into several small sections, but all are interconnected and work simultaneously. The sections which will be dealt with are as follows:—

(*a*) Heading, Drift and Track Mechanism.

(*b*) Groundspeed Mechanism.

(*c*) Co-ordinate Resolving Mechanism.

(*d*) Along/Across Track Mechanism.

(*e*) Wind Mechanism.

(*f*) Wind Comparator.

Heading, Drift and Track Mechanism

8. **Heading.** True heading is derived from the aircraft's compass system and appears in the form of signals on the stators of a control transformer (CT). The associated servomotor (SM1) turns the heading shaft, which has three functions as follows:—

(*a*) It operates the heading display on the face of the instrument.

(*b*) It provides an input to differential A where drift is added to give track.

(*c*) It provides an input into differential B where wind direction is applied to give wind vector direction relative to heading (ø).

9. **Drift.** Differential A enables drift to be added to heading to give track. The heading shaft is one input to the differential while the second input comes from a servomotor (SM2), which is driven by the error signal from a CT whose stators carry signals from a control transmitter in the Doppler radar equipment. The rotor of the CT is driven to the null by the differential output, which therefore turns through the angle of drift, while the servomotor shaft represents track. On the drift shaft are a CX for transmission of drift to the flight control system and a resolver synchro (RS1)

which is used to derive V1 and V2 (Vg cos d and Vg sin d) in conjunction with an input voltage representing groundspeed. SM2 is powered in two ways:

(*a*) In the normal mode, by the error signal from the CT rotor as described above.

(*b*) In the memory mode, by the error-in-V2 signal from the wind comparator.

10. **Track.** The track shaft has three functions:—

(*a*) It turns the pointer of the track dial on the face of the instrument.

(*b*) It sets the carriage of sphere resolver A in the co-ordinate resolving mechanism.

(*c*) It turns the rotor of a control differential transmitter (CDX 1) in the along/across track mechanism to compare track made good and desired track.

Groundspeed Mechanism

11. The groundspeed mechanism is a servo system which produces analogues of groundspeed (Vg) as a shaft rotation for sphere resolvers A and C, and as an a.c. voltage for RS1. In the normal mode the mechanism is controlled by the Doppler input, while in the memory mode it is driven by the error-in-V1 signal.

12. A servomotor (SM3) drives a shaft through an angle proportional to Vg. The shaft positions the wipers of two potentiometers; one provides an a.c. voltage to RS1 on the drift shaft, the other drives a velodyne system which incorporates a phonic wheel (*see* Appendix to Chap. 1, Sect. 5, A.P. 1234C). The frequency output of the phonic wheel is compared in the groundspeed frequency comparator with the Doppler frequency, which is a measure of the true groundspeed. Any error energizes SM3 which moves the shaft until the potentiometer settings cause the phonic wheel frequency to be the same as the Doppler frequency; the speed of the velodyne shaft is then proportional to groundspeed, as is the a.c. output to RS1. When the Doppler radar is on memory or unserviceable then SM3 is driven by the error-in-V1 signal instead of the error signal from the frequency comparator. When the G.P.I. Mk. 7 is used in conjunction with the Decca Doppler Type 62M (*see* A.P. 1234C, Sect. 5, Chap. 6) the setting of the LAND/SEA switch in that equipment determines the phonic wheel frequency produced in the G.P.I. Mk. 7 so that the same allowance (an increase of 1%) is made to the Doppler indicated groundspeed.

Co-ordinate Resolving Mechanism

13. **Functions.** The purpose of the co-ordinate resolving mechanism is to convert inputs of track and groundspeed into outputs either of change of latitude and longitude, or of grid northings and eastings. It also provides facilities for storing mileage whenever it is desired to reset the display, and in addition provides for fast or slow resets as required.

14. **Components.** The co-ordinate resolving mechanism includes:—

(*a*) Sphere resolver A, which, from inputs of groundspeed and track, produces northings and eastings components of groundspeed.

(*b*) Storage devices on both the northings and eastings shafts.

(*c*) Reset facilities, fast (motor-driven) and slow (manual) on both northings and eastings shafts.

(*d*) Sphere resolver B which converts eastings into change of longitude.

(*e*) Synchro transmitters of latitude and longitude.

Storage System

15. It is convenient to describe the storage mechanisms at this stage because they are integral parts of both the northings and eastings drives and not simply coupled into the main drives when required, as in the case of the G.P.I. Mk. 4. The storage mechanisms on both shafts are identical in principle, but there are certain refinements on the eastings side to cater for the inclusion of sphere resolver B, a secant gear used to give a longitude output. The mechanism located on the northings drive is described in detail; the differences in the eastings drive will be given in paragraphs 21—23.

16. The northings drive originates from the cosine output roller of sphere resolver A, and is fed into differential C. The second input to this differential comes from a servomotor (SM4) which is activated by an error signal from an inductive pick-off, via a high-gain amplifier. The rotor of the inductive pick-off is re-positioned into its null position by the differential output, thus making the differential a part of the servo loop. The inductive pick-off is a simple one consisting of an energized stator coil and a rotor coil which are at 90° to each other when the latter is in its null position. Any rotation of the rotor relative to the stator causes an e.m.f. (or error

signal) to be induced in it, the magnitude and phase of which depend on the amount and sense of the relative motion.

17. The action of the device is more easily understood by considering the chain of events immediately after taking off. When the ground-speed drive is received from the Doppler, sphere resolver A rotates and, unless the track is E/W, this provides an output from the cosine roller into differential C. The second input gear of the differential is stationary, since it depends on the servomotor and this is not yet powered. The input rotation therefore passes straight through the differential to turn the pick-off rotor out of its null position. The resulting error signal is phase detected, amplified and made to drive SM4. The servo drives into the differential in such a sense that the differential output turns the rotor back towards its null. Thus a continuous northings input produces a corresponding rotation of SM4, and this provides the latitude/northings output. The two inputs to differential C are equal and opposite, so the output shaft is stationary with the rotor held very slightly off its null position (otherwise there would be no e.m.f. to drive SM4).

18. When it is required to store mileage, the action of setting the OPERATE/STORE switch to STORE breaks the circuit from the rotor pick-off to the servomotor. The servo drive stops, and since this drive provides the output to the counters these too stop. The first input to the differential from the sphere resolver is maintained however, and causes the differential output to turn the pick-off rotor out of its null. Since there is no follow-up action the rotor is not re-positioned and the e.m.f. on its coil builds up. The mileage is now being stored as an error voltage in the pick-off rotor. When the follow-up circuit is made, *i.e.* the mechanism is set to normal operation, this stored voltage drives the servomotor rapidly to re-position the rotor to the null again thereby passing the stored mileage to the counters and resuming normal operation.

19. The limit of storage will evidently be a 180° rotation of the pick-off rotor. If the rotor is driven beyond this position then the error signal will change phase and on resumption of normal operation the servomotor will drive in the opposite direction to bring the rotor to the null the shorter way. In the G.P.I. Mk. 7 the gearing of the mechanism is such that a 180° rotation of the rotor represents 170 n.m. If mileage is stored beyond this the counters will be 340 n.m. in error when OPERATE is selected again. For example if one allowed 200 n.m. to be stored then, on selection of OPERATE, the system would unstore 140 n.m. in the opposite direction instead of 200 n.m. in the correct direction, thus indicating a position 340 n.m. in error. In practice the storage should be limited to 150 n.m., thus avoiding this occurrence. A flag indicator is used to warn the operator that this position has been reached.

Northings Drive

20. It has been shown in the previous paragraphs that the northings output to the rest of the system comes from SM4 in the storage mechanism. The drive is fed through another differential (D) and a clutch, which are part of the counter reset facility. This is described in more detail in paragraph 26. The drive is then applied to the counters through gearing to show either latitude or grid northings. The shaft also drives the rotor of a CX to give a synchro output of latitude for use in the Variation Computer Unit (V.C.U.), and in addition the shaft drive sets the carriage of sphere resolver B in the secant device.

Eastings Drive

21. The sine output roller of sphere resolver A gives eastings which is passed into the storage differential (E). As in the case of the northings drive the second input to the differential is taken from a servomotor (SM5), activated from an inductive pick-off whose rotor is controlled by the differential output. The mechanism is slightly more complicated by the fact that a secant multiplication must be included if it is desired to transform eastings into ch long. It is convenient to deal with the mechanism when grid eastings are required and then to treat the longitude case.

22. **Grid.** When the computer has grid mode selected, SM5 and differential E provide the required eastings drive, operating in exactly the same way as the northings drive. The eastings drive is passed to the counters via differential F and a clutch which provides a reset facility (*see* paragraph 26). The lat/long-grid switch on the instrument selects this routing for the information.

23. **Longitude.** When longitude is required another servomotor (SM6), a sphere resolver B, and a CX/CT team are included in the action to provide a secant multiplication. The error signal from the pick-off rotor is made to activate SM6 which provides the speed input, say X, to sphere resolver B. The collar of this resolver is set to the

angle of latitude by the northings shaft and therefore its outputs are X cos lat and X sin lat respectively. The cosine output is transmitted through the CX/CT team and SM5 to provide the second input to differential E. Thus the inputs to the differential are eastings and X cos lat. Any difference between these inputs results in movement of the pick-off rotor which therefore produces a change in the error signal to SM6. An increase or decrease in the speed X of SM6 reflects the change, which now continues until X cos lat equals eastings. When equality is reached the pick-off rotor becomes stationary and the error signal is stabilized. Since now X cos lat equals eastings so X equals eastings divided by cos lat, *i.e.* eastings sec lat or the rate of change of longitude. The output of SM6 is therefore passed through the reset mechanism and gearing to the longitude counter display. As in the case of the latitude mechanism the shaft is made to operate a CX for transmission of longitude to the V.C.U.

Convergency

24. The second output of sphere resolver B is X sin lat. But X equals the rate of change of longitude, so that this output is at a speed representing the rate of ch long multiplied by sin lat, *i.e.* the rate of meridian convergence. A CX transmits this value of convergency to the F.C.S. Mk. 29 (in the Belfast).

Lat/Long or Grid Display

25. The same counters are used for both lat/long and grid display. For the sake of simplicity a 60 n.m. grid is used; thus a reading of N00240 when grid function is selected is interpreted as $(2 \times 60) + 40$ *i.e.* 160 n.m. north on the more conventional 100 n.m. grid. The only difference therefore between the two displays is that when grid is selected the secant device is bypassed to give a direct readout of eastings instead of longitude.

Reset Facilities

26. A differential and clutch arrangement on both output shafts enables the operator to reset the counters rapidly by means of a d.c. motor, or slowly by manual control. The same control knob on the face of the instrument serves for both reset speeds; it must be pushed in before turning to give manual resetting. It is possible to reset whether the mechanism is set to store or not, although when using the fast reset the mechanism is effectively set to store for the period that the d.c. motor is operating.

Along/Across Track Mechanism

27. The track shaft is made to turn the rotor of CDX 1, whose stators carry signals corresponding to the selected desired track. Two desired tracks may be set in the along/across track computer by means of synchro transmitters, the selected one being that currently flown. The CDX rotor output signals, when used to activate a CT and servomotor (SM7), result in a shaft drive representing track error angle (e) *i.e.* the difference between desired track and track made good. This shaft is used to position the carriage of sphere resolver C, which also has a shaft rotation input of groundspeed.

28. Normally sphere resolver inputs of track error angle (e) and groundspeed (Vg) would give the required outputs of Vg cos e (along track speed) and Vg sin e (across track speed). Since, however, Vg sin e is usually very small this arrangement tends to produce uneven wear on the rotating sphere. To avoid this a constant angle of 45° is added to the error angle e causing both the output rollers to rotate more or less equally. The output quantities are then Vg cos (45+e) and Vg sin (45+e) and it is necessary to find their sum and difference in order to produce the required along and across track quantities. The results may be examined as follows:—

$$\begin{aligned}
&Vg \sin (45+e) + Vg \cos (45+e) \\
&= Vg (\sin 45 \cos e + \cos 45 \sin e + \\
&\qquad \cos 45 \cos e - \sin 45 \sin e) \\
&= Vg (\cos e + \sin e + \cos e - \sin e) \\
&\quad \frac{}{\sqrt{2}} \\
&= \sqrt{2}\, Vg \cos e
\end{aligned}$$

Similarly, $Vg \sin (45+e) - Vg \cos (45+e)$

$= \sqrt{2}\, Vg \sin e$

Thus the sum of the resolver outputs is proportional to the along track component of groundspeed and the difference to the across track component.

29. The along track part of the mechanism is provided with a storage facility. The across track part does not have this facility since movement across track is normally very slow; no appreciable errors, therefore, result from resetting the counter display without the assistance of a storage system.

30. **Across Track.** A difference between the two outputs of the sphere resolver, as shown, indicates an across track error; a comparison of these outputs is therefore made to determine the error. The sine output shaft from the sphere resolver

turns the rotor of a CX, and the cosine output shaft turns the rotor of CDX 2; the connections are such as to effect subtraction and therefore the output from the CDX represents the difference. This output is applied to a CT which activates servomotor SM8 to drive the across track counters. The across track distance computed is always that of the selected track, but the distance displayed will be the current computed distance plus any reading left on the counters from the previous stage. It is therefore advisable to reset the counters at the start of each stage. A clutch enables the counter indication to be reset manually. Provision is made for transmission of across track distance in suitable form to the compass system and autopilot.

31. **Along Track.** A similar type of synchro system is used to add the two output rotations of the sphere resolver to give the along track drive. The CX which is driven by the sine output of the sphere resolver is also coupled to CDX 3 whose rotor is again turned by the cosine output; the connections are such as to effect addition and therefore the output of SM9 is proportional to groundspeed along the desired track. This output is fed into a storage differential (G) to provide a memory facility; the system is exactly the same as in the case of the co-ordinate resolving mechanism, *i.e.* an inductive pick-off and servomotor (SM 10) which provides a second input to differential G and the further output of along track drive. Differential H allows the drive to be fed to the counters of the stage selected, and a clutch arrangement prior to each set of counters enables distance-to-go to be set on each. The sense of rotation of the along track drive is such as to reduce this distance-to-go down to zero.

Wind Mechanism

32. The wind mechanism consists basically of a motor (SM 11) whose shaft positions a potentiometer (C) which gives a voltage analogue of wind speed, and a second motor (SM 12) which gives a shaft rotation corresponding to wind direction. The motors may be operated either automatically by error signals from the wind comparator (G.P.I. in the normal mode) or the wind speed potentiometer and wind direction shaft may be set manually (G.P.I. in the reversionary mode).

33. **Wind Direction.** The method of computation used in the G.P.I. Mk. 7 requires a shaft angle of wind direction relative to heading. This is obtained by feeding the shaft angle of true wind direction into a differential (B) together with true heading in such a sense that the latter is subtracted from the former to give a shaft rotation corresponding to ø the wind vector direction relative to heading. The true wind direction shaft is either derived from SM 12 which is activated by an error signal from the wind comparator (through RS 2), or it may be set manually. In addition true wind direction is also displayed as a pointer indication. The differential output is made to turn the rotors of two resolvers (RS 2 and RS 3).

34. **Wind Speed.** An a.c. voltage proportional to windspeed (Vw) is obtained from potentiometer C. The potentiometer may be set automatically by the shaft of SM 11 which is activated by an error signal from the wind comparator (through RS 2), or it may be manually set. In both cases the shaft setting also turns counters which show the wind speed digitally. The potentiometer voltage is fed to the rotor coil of RS 3. Since the rotor is turned through ø the outputs from the stators of the resolver are a.c. voltages proportional to Vw cos ø and Vw sin ø, *i.e.* wind velocity along heading and wind velocity across heading. Vw sin ø is the second estimate of V2 mentioned in paragraph 6.

True Airspeed

35. True airspeed is received in the form of an a.c. voltage which controls a motor (SM 13) whose shaft positions the wiper of a potentiometer (D) giving an a.c. voltage proportional to T.A.S. In the event of the air data computer being unserviceable the potentiometer may be set manually. In both cases the T.A.S. is displayed by means of a pointer. The output of the potentiometer (Va) is combined with the a.c. voltage proportional to Vw cos ø, the component of wind velocity along heading, to provide the second estimate of V1.

Wind Comparator

36. It has been shown (paragraphs 9 and 11) that RS 1 on the drift shaft provides voltages proportional to Vg cos d and Vg sin d which are denoted by V1 and V2 (*see* Fig. 2). RS 3 in the wind mechanism provides Vw cos ø, which when added to Va from the T.A.S. mechanism gives V1, and Vw sin ø which is also, from Fig. 2, equal to V2. In the wind comparator the two V1 voltages are compared, as are the two V2 voltages. Any differences between the respective values are transmitted as error signals to various subsections of the G.P.I. which operate so that eventually the values of V1 and V2 fed into the com-

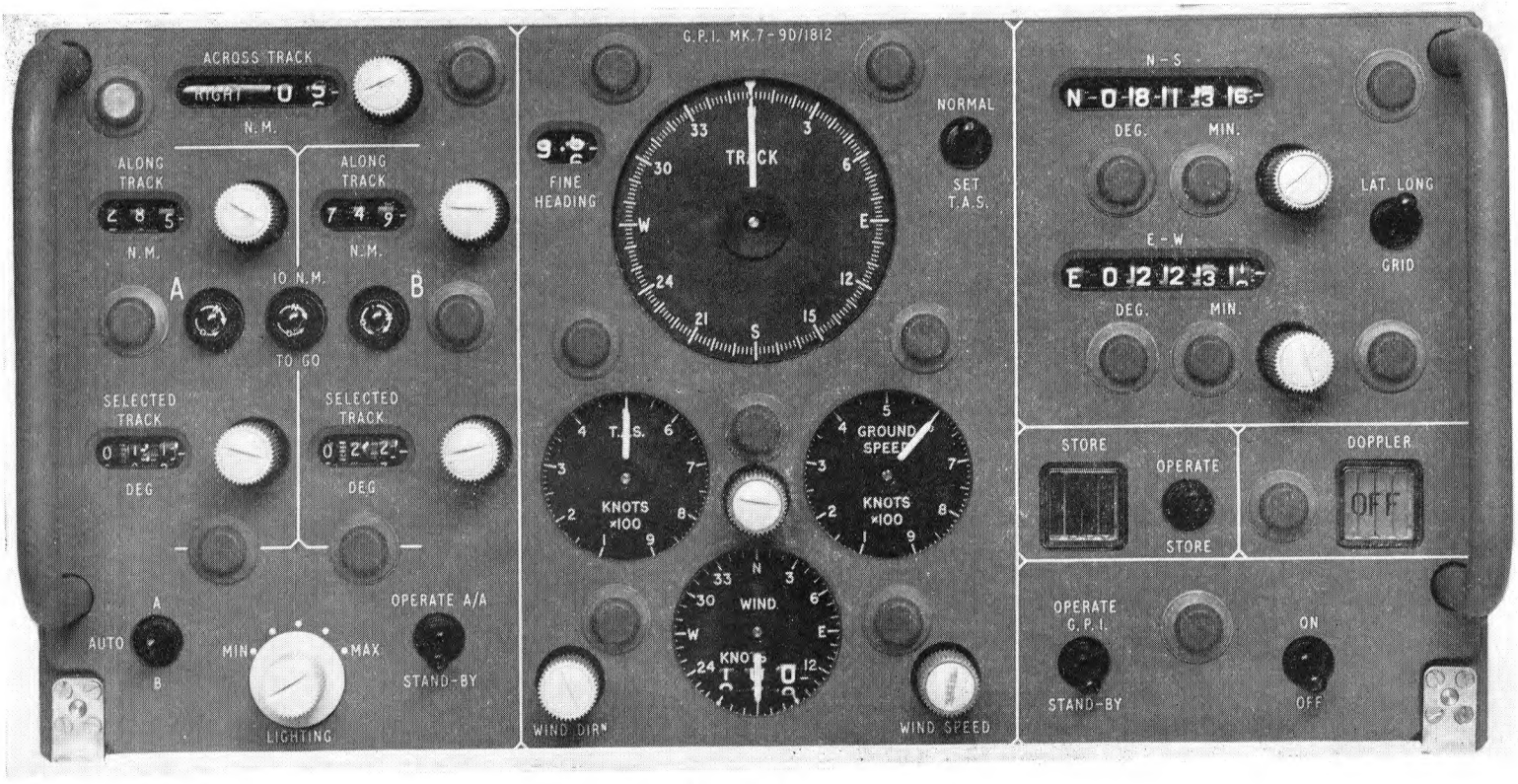

G.P.I. MK.7-9D/1812
ACROSS TRACK
RIGHT
N.M.
ALONG TRACK
N.M.
ALONG TRACK
N.M.
A
10 N.M.
B
TO GO
SELECTED TRACK
DEG
SELECTED TRACK
DEG
A
AUTO
B
MIN
MAX
LIGHTING
OPERATE A/A
STAND-BY
FINE HEADING
TRACK
NORMAL
SET T.A.S.
T.A.S.
KNOTS ×100
GROUND SPEED
KNOTS ×100
WIND
KNOTS
WIND DIR.
WIND SPEED
N-S
DEG.
MIN.
LAT. LONG
GRID
E-W
DEG.
MIN.
STORE
OPERATE
STORE
DOPPLER
OFF
OPERATE G.P.I.
STAND-BY
ON
OFF

Fig. 4. G.P.I. Mk. 7

parator agree and no error signals are transmitted. When the G.P.I. is in its normal mode the error signals are passed to the stator coils of RS 2 whose rotor is turned through ø. The outputs from the rotor coils are fed to the wind speed and wind direction motors respectively. Since one output from the rotor of RS 2 represents the error in the set wind speed, while the other represents that of the set wind direction, they energize the motors to drive the set values into coincidence with the computed values. When the G.P.I. is in its memory mode the error-in-V1 signal is passed to the groundspeed mechanism and the error-in-V2 to the drift servo. In short the system operates until:—

$$Vg \cos d = V1 = Vw \cos ø + Va$$

and

$$Vg \sin d = V2 = Vw \sin ø$$

EQUIPMENT

General Description

37. The G.P.I. Mk. 7 display, which is shown in Fig. 4 may be divided into four sections for purposes of explanation:—

(*a*) The centre display showing the vector information.

(*b*) The lat/long or grid display on the top right of the unit.

(*c*) The along/across track display on the left of the unit.

(*d*) The control sections.

Centre Display of Vector Information

38. The centre display consists of dials and counters showing heading, track, true airspeed, groundspeed and wind velocity. It also has controls for manual setting of true airspeed and wind velocity.

39. **Heading and Track Dial.** The uppermost dial, labelled TRACK has a pointer indication of track made good against a compass rose. In addition heading is shown by means of a small indicator (known as a bug) which travels round the outside of the compass rose. Drift angle is of course the difference between the two indicators (but there will normally be in addition a drift meter mounted separately on the navigation panel and fed directly from the Doppler equipment).

40. **Fine Heading Readout.** A digital counter to the left of the track dial and marked FINE HEADING gives the unit and first decimal figure of the heading. Thus the heading shown in Fig. 4 is 359.5 (T), any ambiguity being resolved by the bug. This counter is intended for use in compass swinging to ensure accurate alignment within the instrument; it is not anticipated that the aircraft will be flown to such fine limits.

41. **True Airspeed.** A dial marked T.A.S. and KNOTS × 100 displays the value of T.A.S. supplied by the air data computer or set on the instrument manually, the scale ranging from 90 to 900 knots. A switch just to the right of the track dial and marked NORMAL/SET T.A.S. feeds when in the NORMAL position, the automatic input from the air data computer to SM 13 and to the T.A.S. pointer. For manual operation the switch is put to SET T.A.S. and then any value of T.A.S. within the range may be set using a control immediately to the right of the T.A.S. dial. This control moves potentiometer D in addition to the pointer. It should not be used when the NORMAL/SET T.A.S. switch is to NORMAL, otherwise an incorrect value of T.A.S. will result for the period it is used.

42. **Groundspeed.** Adjacent to the T.A.S. dial is a similar one marked GROUNDSPEED and KNOTS × 100. The pointer operates over the same range of 90 to 900 knots. This meter will normally reproduce the reading on the separate Doppler groundspeed indicator. When the G.P.I. is in its memory mode it will indicate D.R. groundspeed.

43. **Wind Velocity.** A fourth dial at the bottom of this section of the display shows wind direction by means of a pointer and wind speed by means of a three-digit counter in the bottom half of the dial. In normal operation computed wind velocity will be displayed on this dial, but to enable the operator to set on D.R. wind velocity in the reversionary mode two controls at either side of the dial and marked WIND DIRN and WIND SPEED are provided. The WIND DIRN control turns the shaft input to differential B and the WIND SPEED control moves the wiper of potentiometer C. Slight oscillation of pointer and counter over a small range is normal, but if the wind speed is low the direction pointer will oscillate over a greater range. The maximum wind speed available is 250 knots.

Lat/Long or Grid Display

44. The lat/long or grid display is on the right-hand side of the unit and consists of two sets of

counters, each with a reset control and a lat/long or grid selector switch.

45. **Geographical Co-ordinates.** When the selector switch is set to the LAT/LONG position the degrees and minutes of latitude and longitude are displayed in the appropriate windows. The degree and minute figures differ in size for ease of reading and in addition there are DEG and MIN markings on the face of the unit below the appropriate digits. To the left of the series of digits N/S or E/W is displayed. Complete coverage in latitude and longitude is available, the counters changing in direction at the equator and 180° point.

46. **Grid Co-ordinates.** When GRID is selected the indications are in nautical miles again with N/S and E/W letters. The same counters are used as for the lat/long case, since the 60 n.m. grid system is used (*see* paragraph 25). The capacity of the counter system in the grid mode is therefore 10,800 (180 × 60) n.m. in each direction.

47. **Reset Controls.** Close to each set of counters is a reset control. Rotation of the control either way gives a fast, motor-driven reset. If the control is pushed in before rotation a fine, manual reset is available.

Along/Across Track Display

48. The along/across track display, which may be used at the same time as the co-ordinate display, occupies the left side of the unit. In operation it is very similar to the Marconi AD 2300 B computer described in Chapter 10 of this Section, but with the important addition of a storage facility to give more accurate resetting.

49. Two stages, A and B, are provided to enable the operator to change from one track to the other at a turning point. This change may be made manually or automatically. While only one stage is in use at a particular time it is possible to set the next track and distance-to-go on the inactive stage prior to the turn. A blue lamp associated with each stage is illuminated when that particular stage is active. An amber lamp, marked 10 NM TO GO is lit when the active stage has 10 n.m. to go to zero.

50. **Track and Distance-to-go.** Each stage has a control for selecting track, together with a counter readout of track in degrees, and a control for setting distance-to-go which appears again on a counter readout in nautical miles from 0 to 999. As the flight progresses the distance-to-go counters of the active stage reduce to zero. All the controls must be pushed in before turning, and no fast setting facility is available.

51. **Across Track.** A window at the top of the section marked ACROSS TRACK NM shows distance across track in nautical miles from 0 to 89. In addition the words RIGHT or LEFT appear before the digits to show the sense of the across track error. Care must be taken to ensure that the figure displayed always refers to the stage active at the time. At a turning point where the stages are changed there is no automatic change to relate the across track distance to the new track, and therefore it must be reset manually to give the correct indication, using the rotary control to the right of the window.

52. **Stage Selector Control.** A three-position switch, at the bottom left of the unit, marked A, AUTO and B enables the operator to select either A stage or B stage (A or B positions) or, when left in the mid-position (AUTO), engage the automatic along track changeover facility. With the switch in the AUTO position, as soon as the distance counters of the active stage reach zero the computer automatically switches to the other stage, *i.e.* desired track is taken from the other stage and the groundspeed drive now operates these counters. It is essential that the next distance-to-go is set on the inactive stage before the distance counters of the active stage reach zero, otherwise no automatic changeover will occur. If the counters of the inactive stage are left at zero then the active stage counters will run on backwards beyond zero, *i.e.* through 999, 998 . . . and the computer will still resolve relative to the same track. As long as the switch is left at AUTO, and the next track and distance set on in time, the computer will switch from one stage to the other indefinitely. The automatic change does not affect the across track presentation. The 10 n.m. warning lamp will glow at the correct point for each stage and will be extinguished when the counters reach zero.

Control Section

53. Located below the co-ordinate displays are certain switches and indicators concerned with the main power supply to the computer and its modes of operation.

54. **Energizing Switches.**

(*a*) *Main ON/OFF Switch.* A two-position switch at the bottom right, marked ON/OFF is the main power supply switch for the unit.

(*b*) *OPERATE G.P.I. / STAND-BY.* A two-position switch adjacent to the power supply switch and marked OPERATE G.P.I. STAND-BY energizes the lat/long-grid section of the computer and the centre display indicators when it is set to the OPERATE position. When set to STAND-BY there is no drive to the counters or pointers but the equipment is kept warmed up ready for immediate operation and counter setting is possible.

(*c*) *OPERATE A/A / STAND-BY.* A two-position switch below the along/across track display and marked OPERATE A/A / STAND-BY performs a similar function for the along/across section of the equipment. Before the along/across computer will work, however, it is necessary that the OPERATE G.P.I./STAND-BY, switch in sub-paragraph (*b*) above is set to OPERATE since this switch exerts an overall control.

55. **Storage Switch and Indicator.** The storage facility is selected by means of a two-position switch just below the longitude window marked OPERATE/STORE. For normal operation this switch will be set to OPERATE. On selection of STORE the counters of the co-ordinate display and the along track counters of the along/across display are frozen and the incoming mileage is stored. On re-selection of OPERATE after re-setting the stored mileage is rapidly unstored and normal operation is resumed. A flag indicator to the left of this switch and marked STORE warns the operator when the storage system is nearing its limit. The sequence of indications is as follows:—

(*a*) When the switch is set to STORE the flag immediately shows black and yellow stripes.

(*b*) If and when 150 n.m. are in store the flag shows the word FULL in black on a yellow background.

(*c*) When the switch is returned to OPERATE the black and yellow striped flag shows until the stored mileage has all been unstored, at which point the normal black indication is given.

56. **Doppler Indicator Flag.** A second flag to the right of the OPERATE/STORE switch and marked DOPPLER shows the mode of operation of the Doppler equipment, and therefore of the G.P.I. itself. When Doppler is operating normally, and so the G.P.I. is in its normal mode of operation, the flag shows black. When Doppler goes to memory and hence the G.P.I. switches to its reversionary mode, the flag shows a black M on a yellow background. When Doppler is not switched on the flag shows the word OFF in black on a red background.

Lighting

57. The face of the instrument is lit internally as soon as the main power supply switch is set to ON. A rotary control at the bottom of the along/across display allows the intensity of illumination to be controlled to suit the operator.

OPERATING PROCEDURES AND LIMITATIONS

Switching On

58. As soon as power is available the G.P.I. may be switched on by setting the main ON/OFF switch to ON. The lighting may then be adjusted to suit the operator. Since the instrument can accept manual inputs of T.A.S. and wind velocity its operation in the memory mode may be checked prior to take-off.

Pre-flight Checks

59. **In Dispersal.**

(*a*) Set the OPERATE G.P.I./STAND-BY switch to OPERATE and the OPERATE A/A/STAND-BY switch to STAND-BY.

(*b*) Set the OPERATE/STORE switch to OPERATE.

(*c*) Put the NORMAL/SET T.A.S. switch on SET T.A.S. and put in a value of T.A.S. using the control adjacent to the T.A.S. display.

(*d*) The lat/long counters should now move in accordance with the heading and wind velocity registered, and appropriate groundspeed and drift values indicated.

(*e*) The A/A section may then be checked by setting the OPERATE A/A/STAND-BY switch to OPERATE, selecting A and B tracks in turn and observing the appropriate counter movement.

(*f*) After checking the instrument in this way the computer switches should both be set to STAND-BY again and the NORMAL/SET T.A.S. switch to NORMAL.

(*g*) Check the synchronization of the heading bug and counters with the main compass system.

60. **Prior to Take-off.**

(*a*) Set on the correct latitude and longitude (or grid co-ordinates) of base, and the tracks and

distances of the first two legs of the flight plan, if desired.

(*b*) Set the stage selector of the along/across section to AUTO, and check that the across distance counter reads zero.

(*c*) On take-off set both OPERATE/STAND-BY switches to the OPERATE position.

In-flight Operation

61. In flight the G.P.I. will register the position of the aircraft and the remainder of the navigational data to within the limits of its own instrumental accuracy and that of the heading and Doppler inputs. Resetting to a fix obtained from another source is facilitated by the storage system. The Doppler indicator flag always shows the mode in which the instrument is operating.

Accuracy and Limitations

62. The instrumental accuracy is of the order of $\frac{1}{2}\%$ of the distance flown, but account must also be taken of possible input errors, of secant gear slip, and of variations in the nautical mile with latitude and altitude.

63. **Secant Gear Slip.** Secant gear slip occurs when the rate of change of longitude exceeds 5,190 minutes per hour. This condition is achieved at, for example, 80°N or S when the eastings component of groundspeed is 900 kts, or at 88°N or S when the eastings component is 180 kts. Considerations of heading definition would probably cause the use of the Grid mode in these higher latitudes but a lat/long presentation is possible provided the limitation is not exceeded.

64. **Latitude and Altitude Error.** The computation is correct for a nautical mile of 6080 ft. At latitudes other than 47°—48° N or S, or at significantly great altitudes, the appropriate corrections detailed in paragraphs 83—89 of A.P. 1234C, Sect. 5, Chap. 2 may be applied.

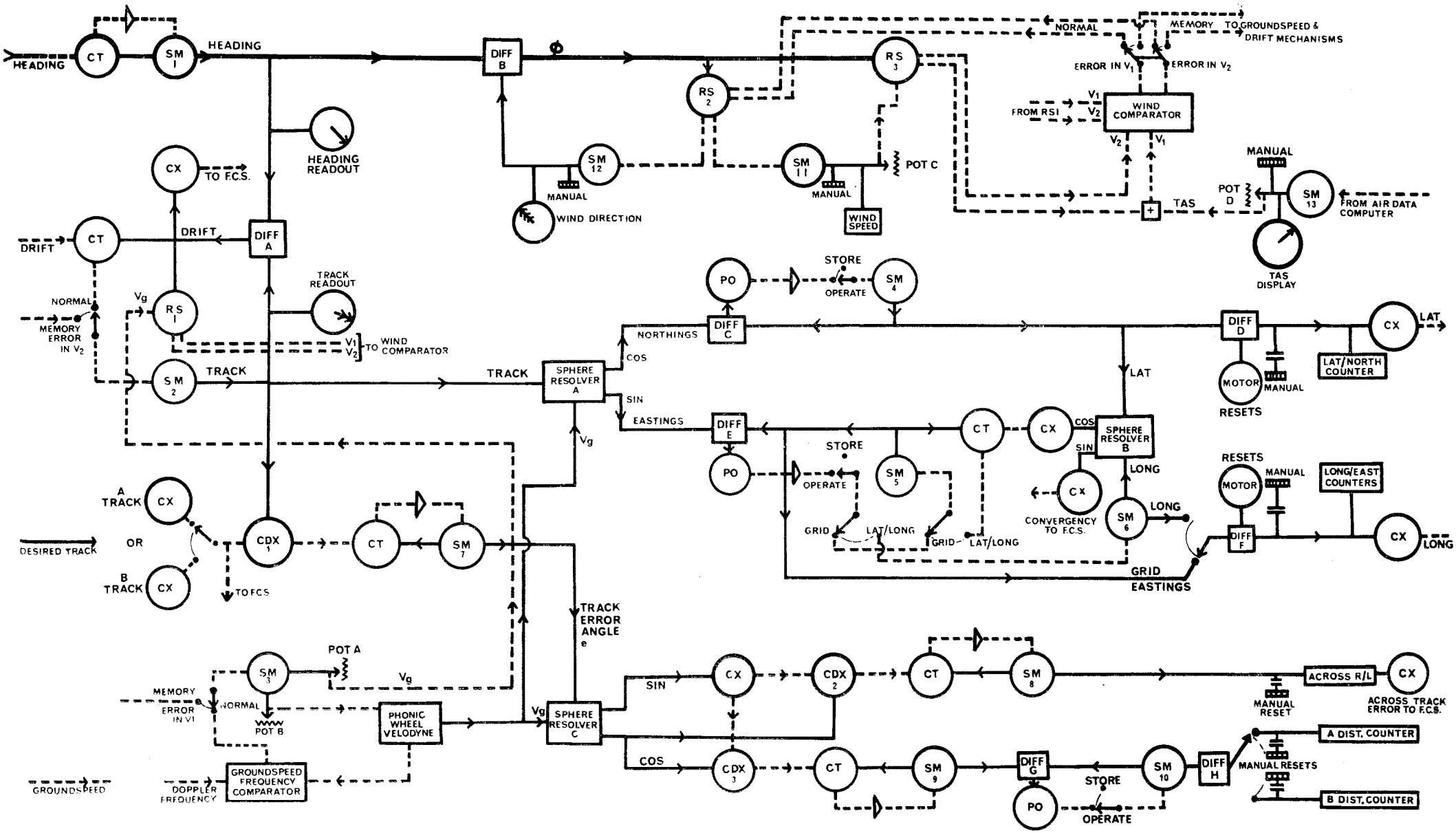

HEADING
CT
SM 1
HEADING
DIFF B
Φ
RS 3
RS 2
NORMAL
MEMORY TO GROUNDSPEED & DRIFT MECHANISMS
ERROR IN V1
ERROR IN V2
FROM RSI
V1
V2
WIND COMPARATOR
V2
V1
MANUAL
POT D
SM 13
FROM AIR DATA COMPUTER
TAS
TAS DISPLAY
CX
TO F.C.S.
HEADING READOUT
SM 12
MANUAL
WIND DIRECTION
SM 11
MANUAL
POT C
WIND SPEED
DRIFT
CT
DRIFT
DIFF A
TRACK READOUT
NORMAL
MEMORY ERROR IN V2
Vg
RS 1
V1
V2
TO WIND COMPARATOR
SM 2
TRACK
TRACK
SPHERE RESOLVER A
PO
STORE
OPERATE
SM 4
DIFF C
NORTHINGS
COS
SIN
EASTINGS
DIFF D
CX
LAT
LAT/NORTH COUNTER
MOTOR
MANUAL
RESETS
LAT
DIFF E
PO
STORE
OPERATE
SM 5
CT
CX
COS
SIN
SPHERE RESOLVER B
LONG
RESETS
MOTOR
MANUAL
LONG/EAST COUNTERS
CX
LONG
CONVERGENCY TO F.C.S.
CX
SM 6
LONG
DIFF F
GRID
LAT/LONG
GRID
LAT/LONG
GRID EASTINGS
A TRACK
CX
OR
B TRACK
CX
DESIRED TRACK
CDX 1
TO F.C.S.
CT
SM 7
Vg
TRACK ERROR ANGLE e
POT A
SM 3
MEMORY ERROR IN V1
NORMAL
POT B
Vg
PHONIC WHEEL VELODYNE
Vg
SPHERE RESOLVER C
SIN
CX
CDX 2
CT
SM 8
ACROSS R/L
CX
MANUAL RESET
ACROSS TRACK ERROR TO F.C.S.
A DIST. COUNTER
MANUAL RESETS
B DIST. COUNTER
COS
CDX 3
CT
SM 9
DIFF G
PO
STORE
OPERATE
SM 10
DIFF H
GROUNDSPEED
DOPPLER FREQUENCY
GROUNDSPEED FREQUENCY COMPARATOR

Fig. 3. G.P.I. Mk. 7

CHAPTER 9

THE TACTICAL AIR NAVIGATION SYSTEM

CONTENTS

	Paras
Introduction	1
The 9447 Series Computer	2–3
System Configuration	4
TANS 9447D	
Introduction	5
Description	6–18
Performance	19
TANS 9447F	
General	20
Description	21–28
Performance	29
OPERATING PROCEDURES	
General	30
Setting up Procedure	31–32
Inserting Information	33–42
Data Setting	43–45
Present Position Update	46–50
Information Display	51–56
Selection and Display of Steering Modes	57–61

Introduction

1. **General.** The Tactical Air Navigation System (TANS) comprises a digital computer (Decca T9447 Series) and a display and control unit which are combined in a single module that can be fitted in an instrument panel or control console. The computer is basically a general purpose computer with a fixed programme designed to resolve specific navigation problems. With sensor inputs from the Doppler, Heading, Attitude, Air Data and Decca Navigator (T9447–F only) and data manually inserted at the control unit, the TANS calculates, and displays on selection by the operator, the following information:

a. Continuously updated Present Position in the following formats:

(1) Latitude and Longitude.

(2) Grid co-ordinates.

(3) Bearing and Distance to an inserted location.

(4) Decca Navigator Hyperbolic co-ordinates (9447–F only).

b. Continuously predicted Steering information from present position:

(1) Direct to a previously inserted static or moving waypoint.

(2) To rhumb line routes between previously inserted waypoints.

Note: For the purpose of this chapter the term 'waypoint' (WP) is used to describe a target, destination, turning point or any geographic location previously inserted into the computer.

c. Sensor input values and computed navigation variables (eg Heading and Wind Velocity).

d. Outputs from the system to external equipment are also available as follows:

(1) Heading, Distance and Time to a selected WP, for a Remote Indicator.

(2) Along and Across drives to an automatic chart display.

(3) Correction rates for gyroscopes.

The 9447 Series Computer

2. **Operating Principle.** The aircraft's present position in Latitude and Longitude is inserted manually into the computer via the keyboard to provide the initial datum. Inputs from the Decca Navigator and/or the Doppler radar together with Heading from the compass system provide data for the computer to continually update present position in increments of Latitude and Longitude. Further computations, using previously inserted data eg waypoint positions, are then made to provide the information and outputs given in para 1. A computing cycle is completed every 400 to 800 ms depending on the outputs required and the display format selected by the operator. With a TAS input the computer can calculate wind velocities at flight level and provide a back up mode of operating should the Decca Navigator and Doppler inputs become invalid. Technical details of the 9447 Series computer are given in AP 112 0203.

3. **Self Test.** The computer has an automatic test facility whereby a test sequence is performed at each computation cycle. If this sequence detects a malfunction the Computer Fail lamp is illuminated (*see* para 9).

System Configuration

4. Variants and sub-variants of the 9447 computer differ in the interface with different combinations and types of sensor inputs and with the scaling and number of outputs. A system configuration showing possible inputs and outputs is illustrated in Fig 1. This chapter will describe the purely Doppler version, the 9447D, and then the developments made in the 9447F with its additional inputs from the Decca Navigator.

TANS 9447D

Introduction

5. The D variant of the 9447 Series computer is designed to interface signals from the Doppler radar, Air Data sensors, True or Magnetic Heading reference and Pitch and Roll attitude sensors. Sub-variants differ in the type and number of inputs, eg the D28 fitted in the Dominie aircraft has no pitch or roll inputs and inputs from the Doppler 62 are in Distance Gone and Drift Angle whilst inputs from the Doppler 70 Series are in velocities Along and Across Heading and may have a vertical Up/Down velocity input.

Description

6. **Display and Control Panel.** The display, control switches, fail lamps and push-button keyboard are all mounted on the front panel of the TANS unit (*see* Fig 2).

7. **TANS Display.** The display comprises two lines of nine alpha-numeric digits. The decimal point is not indicated on the display and varies with each display format (*see* Figs 4 and 5 at the end of this chapter). When the TANS is not in the OP mode the extreme left-hand digit(s) flash. These digits indicate the last selected waypoint number(s) for

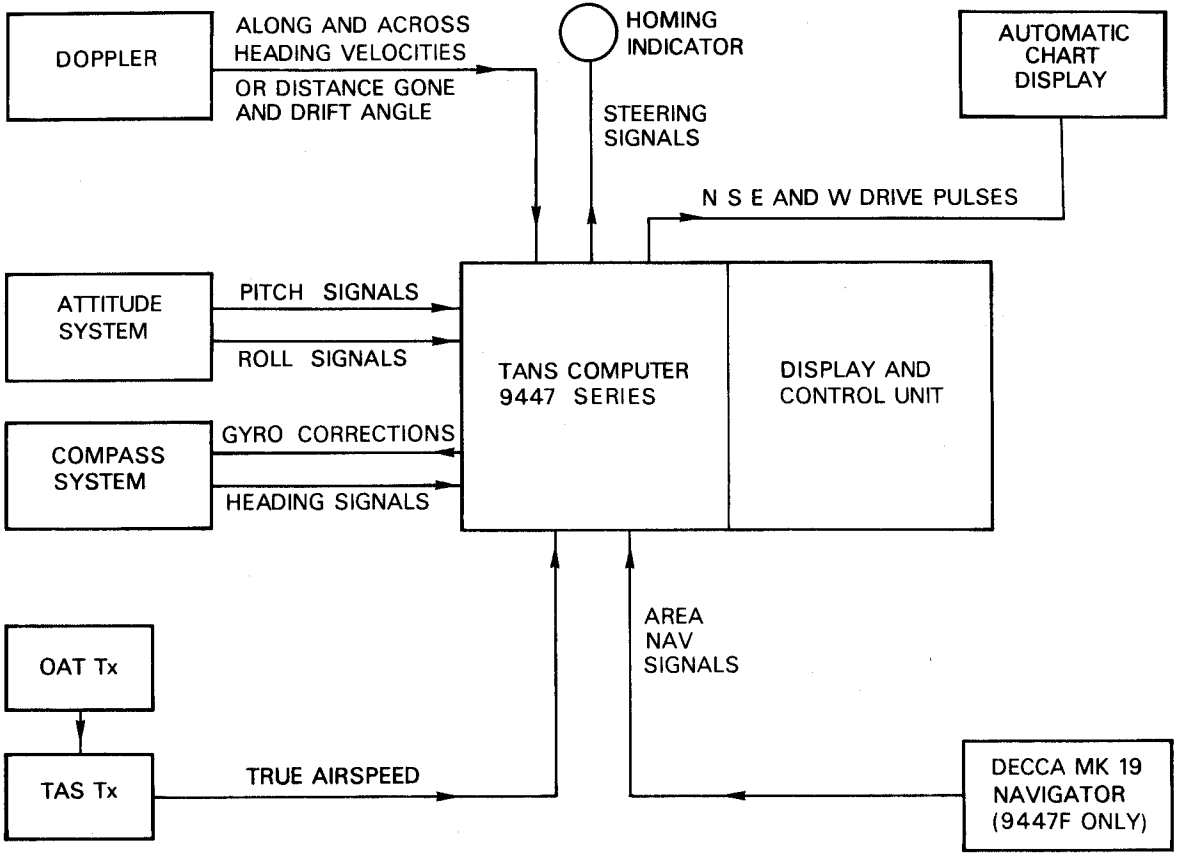


Fig 1 A TANS Configuration

steering purposes, except in the Data mode when they indicate the Data mode selected.

8. **Control Switches.** The purpose of the control switches is as follows:

a. *ON/OFF Switch.* This switches 28v power to the computer power unit. It is a spring loaded toggle switch which must be pulled outwards and downwards to switch off.

b. *Navigation Mode Switch.* This is a four position click switch marked:

(1) *L/L.* This selects Latitude and Longitude format for position displays.

(2) *GRID.* This selects grid co-ordinates format for position displays.

(3) *B–D.* This selects Bearing and Distances for position displays.

(4) *INT.* This selects interception data. Display indicates heading-to-steer and time to intercept. Also used to insert or display bearing and distance from one waypoint to another.

c. *Sensor Switch.* This is a three position switch that is used in conjunction with the Sensor Fail lamp (*see* para 9b) and controls the sensor inputs used by the computer as follows:

(1) *DOP.* In this position all valid sensor inputs are used by the computer.

(2) *S/Y.* The stand-by position is used when no inputs are fed to the computer, eg start-up and taxying.

(3) *ADS.* This position isolates the Doppler from the computer. The TANS will operate using the remaining sensor inputs and either the last wind velocity stored by the computer or a wind manually entered by the operator.

d. *Land/Sea Switch.* This three position switch is used to modify Doppler inputs for differing sea conditions (*see* AP 3456C, P4, S1, Ch 1) as follows:

(1) *SEA-R.* Used when over rough sea.

(2) *LAND.* Used over land (Doppler inputs unmodified)

(3) *SEA-S* Used over smooth sea.

There is a constant comparison by the TANS of Doppler and Air Data velocities when SEA-S is selected to guard against the possibility of excessive Doppler spectrum distortion. If the result of the comparison is invalid the Sensor Fail lamp is illuminated and the TANS automatically reverts to using ADS as in c(3) above. This may occur inadvertently if the wind at flight level is strong while the wind at sea level is light. In this case select SEA-R; if the lamp extinguishes the Doppler input may be considered valid and the switch left in the SEA-R position.

Note: If the Land/Sea switch is left in a SEA position, when flying over land, or vice versa, the system will operate with an error of approximately 2% of the total distance flown in that condition.

e. *Display Dimmer Switch.* This controls the brilliance of display, illuminated keys and Fail lamps.

9. **Fail Lamps.** Two lamps are used as follows to indicate failures in the system:

a. *Computer Fail Lamp.* This is situated above the Dimmer switch and marked FAIL. It illuminates in the event of computer malfunction. It also illuminates approximately ten seconds after switch on to check the lamp circuit. If no fault exists in the circuit the lamp will extinguish after present position (Latitude and Longitude) has been entered in the computer (*see* Switching on Procedure, para 32).

b. *Sensor Fail Lamp.* This is situated to the right of the Sensor Switch and it operates as follows:

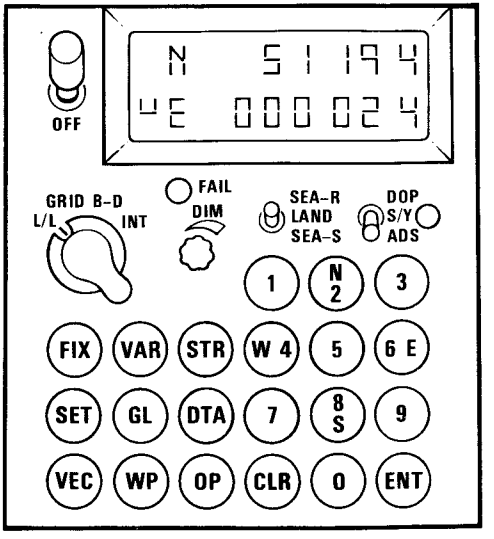


Fig 2 TANS Computer T9447D Front Panel

(1) *FLASHING.* – Switch in S/Y position.
– No Heading input.
– No valid velocity input.

(2) *STEADY ON. Selected Sensor input is invalid.*

With DOP selected on the Sensor switch the lamp will illuminate if the Doppler signal is lost, (eg when flying over smooth water or with excessive movement of the aircraft in pitch and/or roll at high altitude). The TANS will then automatically revert to using the Air Data System (ADS) as in para 8c(3) above. When adequate signal strength is restored the Doppler will re-acquire correct velocity values, the TANS will function normally again and the lamp will go out.

Note: With pitch and roll inputs to the TANS the Doppler velocities are refined for aircraft pitch and/or roll. If pitch and/or roll angle exceeds 60° from the horizontal the Doppler velocities are ignored, the Sensor Fail lamp lights and the TANS reverts to ADS until the pitch and/or roll is less than 60 degrees.

10. **Computer Keyboard.** The keyboard (*see* Fig 2) has 21 press-button keys; 11 provide functional control and 10 are used to insert numerical data.

11. **Functional Keys.** The eleven functional keys operate as follows:

a. *OP.* May be pressed at any time to restore normal operation of TANS when continuously updated present position is displayed in the format selected on the Navigation Mode switch.

b. *WP.* Used in conjunction with numeric keys to insert and/or display information relating to waypoints. Navigation Mode switch must be set to appropriate position when displaying or entering waypoint information.

c. *FIX.* Freezes present position display only. Aircraft movement is stored throughout its use. Used when inserting data concerning present position. Data is valid for instant of pressing FIX.

d. *GL.* Grid Lock. Used when present position is to be entered into the computer in Grid co-ordinates.

e. *SET.* Used when data, other than present position, is to be entered into the computer. Data willl be valid for the instant of pressing ENT. Aircraft movement is not stored.

f. *CLR.* Used to clear displays after pressing FIX or SET prior to entering new data. It may be used in two ways:

(1) One short press will clear the last entered digit, or the 1st digit on the right if no digits have been entered.

(2) If the key is held depressed for two seconds the display will clear completely.

g. *ENT.* Enters the data which has been keyed in via the numeric keys into the computer. When present position is to be entered or updated the ENT key is

ineffective unless held depressed until the display blanks momentarily (after 2 seconds). This safeguards against accidental insertion of wrong data.

h. *VEC.* Used to enter or display the vector of a moving waypoint.

i. *STR.* Permits selection of steering facility to or between selected waypoints.

j. *VAR.* Used to enter or display Magnetic Variation which appears on the top line of the display. The bottom line indicates the Variant and sub-variant coded number of the T9447 computer.

k. *DTA.* Used in conjunction with the numeric keys to select the Data Mode required. (*See* para 16).

12. **Numeric Keys.** Numbered 0 to 9, keys 2, 4, 6 and 8 are additionally marked N, W, E and S respectively. When used to enter Latitude and Longitude or Variation the appropriate letter is pressed before the numbers.

13. All keys have integral lighting. The functional keys, with the exception of the OP, ENT and CLR keys, increase their brightness when pressed to indicate the function selected.

14. **Computer Data Modes.** Display formats (*see* Fig 5) selected by the DTA and respective numeric keys are used to display the values of sensor inputs and calculated navigation variables.

15. **Sub-variants.** Sub-variants of the T9447D computer differ in their number and type of sensor inputs and therefore differ in the data available for display. For example the sub-variant D–28 has Doppler inputs of Distance Gone and Drift Angle, not Along and Across velocities, and no attitude inputs of Pitch, Roll or Angle of Attack angles. Where no input is available a will appear in the top right hand corner of the display.

16. **Data Displays.** The following describes the information displayed using the DTA and numeric keys:

a. *DTA 1.* The computed Track and Groundspeed derived by the computer from the available sensor inputs.

b. *DTA 2. Wind Direction and Speed.* With valid Doppler inputs the wind displayed is the average wind found over the preceding 2 minutes. If Doppler inputs become invalid, the display freezes and the computer use the indicated wind, which can be manually updated via the keyboard, for computations (*see* para 43). When valid Doppler inputs are again available (Sensor switch to DOP, Sensor Fail lamp out) the computer resumes the averaging process. The instantaneous Doppler wind may be displayed and/or entered into the computer by the keying sequence 'SET–DTA–2–ENT'.

c. *DTA 3. Heading and Drift Angle.* Displays the inputs from the Heading system and the Drift angle input from the Doppler in the D–28 sub-variant. Other versions display the calculated Drift angle.

d. *DTA 4. Roll Angle and True Air speed.* The bottom line of the display indicates the True Air speed computed from the Air Data system inputs or the direct input from a TAS Unit.

e. DTA 5. Pitch Angle and Mach Number. Mach Number is derived from the Air Data System inputs. When an Automatic Chart Display is fitted this display indicates chart scale.

f. *DTA 6. Along and Across Doppler velocities.* In the D–28 version the top line of the display indicates the Groundspeed input from the Doppler regardless of its validity.

g. *DTA 7. Angle of Attack and Vertical Speed* (if available).

h. *DTA 8. Surface Motion Compensation.* This is an operator set value of direction and speed (*see* para 17).

i. *DTA 9. Time and Static Outside Air Temperature.* The top line indicates Clock Time when the initial time is entered by the operator, otherwise, Elapsed Time from switch on.

j. *DTO O.* Allows a check on the display and lamps. Key DTA–0–8 all numeric displays show 8. Press OP to restore normal operations.

Notes:

1. All angular displays have 360° as the datum, eg 355° represents 5° left drift, roll with left wing low, nose down pitch or negative angle of attack.

2. Doppler velocity sense:
Blank = Right, Forwards and Up.
— = Left, Backwards and Down.

3. Where Data Modes are required in sequence it is only necessary to press DTA once, after which the required mode can be selected with numeric keys.

17. **Surface Motion Compensation.** It may be necessary to compensate the Doppler inputs for movement of the water surface, caused by wind and/or tide and current (*see* AP3456C P4 S1 Ch 1). For this the TANS uses a keyboard procedure as shown in para 44. The compensation is only applied when the Land/Sea switch is in a Sea position.

18. **Grid Display.** The TANS can indicate Grid positions in one of two forms dependent on the particular variant:

a. The Naval Tactical grid based on the Data Mile (2000 yards) where position can be in any quadrant. The numeric origin is effectively defined when Grid position is inserted using the Grid Lock (GL) key.

b. The Transverse Mercator grid based on kilometres where positions are indicated in Easting and Northing coordinates scaled to 999.99 Kilometres. The Grid Zone is determined by the Latitude and Longitude of present position to which the Grid must be locked using the procedure in para 34. The central meridian of each Zone is assumed to be at Eastings 500.00 except for certain variants where the central meridian of the British National grid is at Eastings 400.00.

Note: Some TANS variants have the facility to change Grid Zone automatically when crossing zone borders. At the N/S borders this occurs exactly on the border. When transitting the E/W border the change does not occur until ½° Longitude into the adjacent zone, when automatically the coordinates of the new zone are displayed.

Performance

19. Table 1 gives the quoted permissible errors of the 9447 D computer operating between 89N and 89S Latitudes in the temperature range –35°C to +70°C. They do not include any error in the sensor inputs.

MODE	ERROR
Lat and Long	0.25% of the distance flown +0.1 minute N/S and E/W initial setting error.
GRID	As Lat and Long, +0.02 grid units E and N + 0.08% of the distance from Grid Lock position.
Bearing and Distance	1.5° bearing 0.6% range
Intercept	2.0° heading, 0.1 minute of time, + 0.5% of time.

Table 1 Performance Summary of 9447D Computer

TANS 9447F

General

20. In the 9447 F version of the TANS the computer is programmed to accept and process hyperbolic inputs from the Decca Navigator to update present position in addition to Doppler velocity inputs. The computer thus performs independent and concurrent computations based on input data from two main sensors. In normal operation the Sensor switch is set to HYP (hyperbolic) where the Decca Navigator position is rate-aided by Doppler velocity information and is used for display and steering computations. In the event of either of the hyperbolic or Doppler inputs becoming invalid, reversion to the alternative sensor is automatic. If both the main sensors fail the air data back-up is used. When the Sensor switch is set to DR (eg when outside Decca Navigator cover) the Doppler inputs are used with air data as an automatic back-up in the event of Doppler failure.

Description

21. **Control/Display Panel.** The 9447 F version differs externally only in that the Navigation Mode switch has an additional position marked HYP, which allows positions to be inserted and/or displayed in Decca Navigator hyperbolic co-ordinates, and that the Sensor switch is marked HYP, S/Y, and DR (*see* Fig 3).

22. **Sensor Fail Lamp.** With the Sensor Switch in the HYP position the lamp lights when the Decca Navigator inputs fail and automatic reversion to the DR mode occurs. In the DR position the lamp lights when both the Doppler and Air Data inputs fail, the Decca Navigator inputs are valid and automatic reversion to HYP occurs. The lamp flashes on the S/Y position and when the Decca Navigator, Doppler and Air Data inputs are unusable. It may also flash when the Decca Navigator inputs are 'noisy'. There is no lamp indication of Doppler input failure.

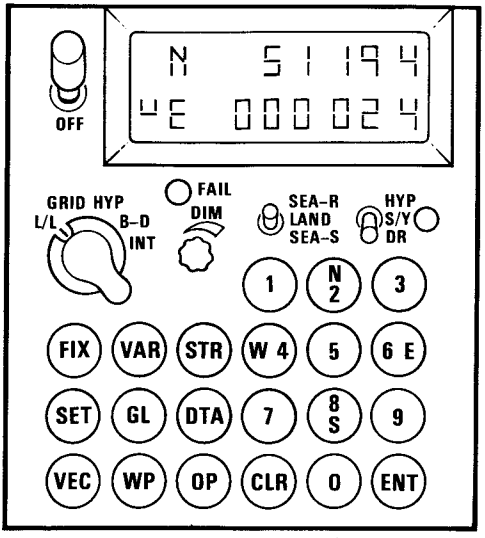


Fig 3 TANS Computer T9447F front Panel

23. **Display of Decca Co-ordinates.** The unprocessed Decca Navigator inputs are displayed by pressing the VAR key when the lower display line will indicate, in sequence, the Chain number and letter, and the Red, Green and Purple co-ordinates (*see* Fig 4 Format 11). The computer compares all three inputs for angle of cut, lane expansion factor and signal quality and selects the optimum two for computations. These are displayed when the Navigation Mode Switch is on HYP and the OP key is pressed.

Note: Zone references in the Decca Navigator system are in alphabetic order and must be converted to a numerical sequence (A = 0, B = 1, C = 2, etc).

24. **Data Displays.** The 9447 F has an additional Data Display that enables the operator to compare the two positions being continuously calculated by the computer. When the DTA key is pressed the display shows the Bearing (top line) and the Distance (bottom line) from the HYP position to the DR position.

25. **Data Retention Facility.** When the computer is switched off, information cur-

rently held in the computer data store can be retained by the use of an external d.c. supply (eg battery). This means that after an overnight stop for example, the operator does not need to re-insert present position data, waypoints etc. It should be noted that if the aircraft is moved with the computer off the stored present position data will be in error.

26. **Inserting and/or Displaying Information.** The HYP position on the Navigation Mode Switch enables present position or the position of Waypoints to be inserted or displayed in Decca co-ordinates. The same keying sequences as given in paras 36 and 37 are used and co-ordinates entered as follows:

a. Key in Zone number, Lane number and Lane fraction (Red or Green) – Co-ordinates displayed on the top line of the display.

b. Key in Zone number, Lane number and Lane fraction (Green or Purple) – Co-ordinates displayed on the bottom line.

27. **To Insert Waypoints in Decca Co-ordinates.** Waypoints may be entered in Decca co-ordinates (Zone, Lane and Lane Fractions) of the Decca chain in use. If a waypoint is required to be inserted which is outside the coverage of the chain in use, the waypoint must be entered in Lat/Long. When chain is changed TANS will indicate the Decca co-ordinates of the waypoint appropriate to the new chain. It is not possible to insert a waypoint in Decca co-ordinates of a chain not actually selected on the Decca Navigator.

28. **Updating on Decca Fringe Cover.** When flying out of Decca cover it is advisable to update DR position to HYP position while still in good cover (Sensor Switch to HYP, Sensor light out) and conversely when approaching Decca coverage.

MODE	ERROR
L/L (HYP)	0.05% of the distance to the farthest station +0.1 min Lat/Long initial setting error.
L/L (DR) Doppler inputs	0.25% of the distance flown +0.1 min Lat/Long initial setting error.
GRID	As L/L +0.02 grid units E and N +0.08% of the distance from Grid Lock position.
B and D	1.0° bearing 0.5% range
INT	1.5° bearing 0.1 minute of time +0.5% of time to Intercept

Table 2 Performance Summary of 9447F Computer

Performance

29. Table 2 gives the quoted permissible errors of the 9447 F computer operating between 89N and 89S Latitudes in the temperature range –35°C to +70°C. They do not include any error in the sensor inputs.

OPERATING PROCEDURES

General

30. Paras 31 to 61 give the operating procedures for both the D and F versions of the TANS, with explanatory notes where necessary. Formats referred to are detailed in Figs 4 and 5 at end of chapter. A summary of the switch and push-button sequences used is shown diagramatically in Fig 6.

Setting Up Procedure

31. **Power Supply.** The TANS power unit is designed to absorb transient

changes of power. During start up and taxi, when these changes are most likely, the display may go out. This conserves available power to maintain the data store. If the DC voltage is too low the data store may be corrupted. If the Computer Fail lamp illuminates, switch off, switch on and start procedure again.

32. **Initial Setting Up.** With aircraft power, Doppler and Heading systems on and the Decca Navigator to OP (F version, see Mk 19 operating instructions):

a. Select L/L on Navigation Mode Switch.	–
b. Turn DIM control fully clockwise.	–
c. Land/Sea switch as required.	–
d. Select S/Y on Sensor switch.	–
e. TANS power switch to ON.	– The Sensor Fail lamp flashes and the display shows N/E and all zeros. The lower waypoint digit shows steady ⌊⌋ . Computer Fail lamp on after 10 seconds.
f. Key in Present Position as shown in para 33.	– Computer Fail lamp out.
g. Insert other information as required (see paras 33–45).	
h. At Take Off point set Sensor Switch to DOP or HYP as applicable.	– Sensor Fail lamp out. Normal operation commenced.

Note: Initial Present Position must always be entered in Latitude and Longitude as the computer does all its computations in increments of Lat and Long before converting the results to other formats.

Inserting Information.

33. **To insert Present Position in Latitude and Longitude:**

a. Navigation Mode switch	– Select L/L.
b. Press FIX key	– FIX brightens.
c. Key in Lat/Long	– Lat/Long displayed (Format 1).
d. Press ENT until displays blank momentarily	– FIX dims. Present Position displayed and automatically updated.

34. **To insert Datum Grid Position:**

a. Navigation Mode switch	– Select GRID. E and N displayed automatically.
b. Press GL key	– Display freezes, GL and FIX brighten.
c. Insert Eastings/ Northings	– Eastings and Northings grid displayed (Format 2) – note Northings on bottom row.
d. Press ENT key	– GL and FIX key dim. Present position displayed and automatically updated. Grid co-ordinates are now associated with latitude and longitude.

35. **To insert or change Magnetic Variation** (if required, *see* Note 2):

a. Press SET key	– SET key brightens.
b. Press VAR key	– VAR key brightens.
c. Press CLR key (if required)	– Clear digits as required.

d. Insert variation	– Variation displayed (Format 8) (see Note 2).
e. Press ENT key (twice)	– SET and VAR keys dim. Normal operation restored (*see* Note 3).

Notes:

1. If function keys are pressed in an incorrect sequence it is recommended that the OP key is pressed and keying sequence is recommenced.

2. When variation is set on the compass master indicator True heading is fed to the computer. If zero variation is set on the compass Magnetic heading is fed to the computer and variation must be keyed in. VARIATION MUST NEVER BE SET AT BOTH THE COMPASS AND THE COMPUTER or large errors in position will result.

3. It is necessary to press ENT twice if only part of the variation has been changed.

36. **To insert the Position of a Waypoint:**

a. Navigation Mode switch	– Select L/L, GRID, HYP or B–D as required.
b. SET key	– Press, SET brightens.
c. WP key	– Press, WP brightens.
d. Waypoint number key	– Press, waypoint number and its current position is displayed.
e. Lat/Eastings/ Bearing/ Red or Green co-ordinates	– Insert, co-ordinates displayed (Format 1 to 4).
f. Long/ Northings/Distance/ Green or Purple co-ordinates	– Insert, co-ordinates displayed (Format 1 to 4).
g. ENT key	– Press, SET and WP dim, position stored. Normal operation restored.

37. **To Insert Present Position in Decca Co-ordinates:**

a. Press FIX	– FIX brightens. Display freezes.
b. Set Mode switch to HYP.	
c. Key in Zone number Lane number and Lane fraction (Red or Green)	– Co-ordinates displayed on top line.
d. Key in Zone number Lane number and Lane fraction (Green or Purple)	– Co-ordinates displayed on bottom line.
e. Press ENT until displays blank momentarily (two seconds approx.)	– FIX dims. Present position displayed and automatically updated (Format 3).
f. Sensor switch on HYP	– Sensor Fail lamp extinguished after a short period.

Note: When Decca co-ordinates are entered, displayed Latitude and Longitude will adjust to the received and computed position.

38. **To insert Present Position as a Waypoint:**

a. Navigation mode switch	– Select L/L, GRID or HYP.
b. FIX key	– Press, FIX brightens, display freezes.
c. WP key	– Press, WP brightens.
d. Waypoint number required	– Press, number displayed.
e. ENT key	– Press, FIX and WP dim, present position stored as a waypoint.

Normal operation restored.

39. **To insert a Waypoint as Bearing and Distance from another Waypoint already entered:**

a. Navigation mode switch	– Select INT.
b. SET key	– Press, SET brightens.
c. WP key	– Press, WP brightens.
d. New waypoint number (to)	– Insert, number displayed on lower row.
e. Reference waypoint number (from)	– Insert, number displayed on upper row.
f. Bearing and distance from reference Waypoint to new Waypoint.	– Insert, check readout (Format 5).
g. ENT key.	– Press, SET and WP dim. Waypoint position entered as bearing and distance from reference waypoint. Normal operation restored.

40. **To insert a Waypoint to be Vectored:**

a. Navigation mode switch	– Select L/L, GRID, B–D or HYP.
b. SET key	– Press, SET brightens.
c. VEC key	– Press, VEC brightens.
d. WP key	– Press WP brightens.
e. Waypoint number required	– Press, waypoint number and its current position displayed (Format 1, 2 and 3).
f. CLR key (if required)	– Press, display clears.
g. Lat/Eastings/ Bearings	– Insert, co-ordinates displayed (Format 1, 2 and 3).
h. Long/ Northings/ Distance	– Insert, co-ordinates displayed (Format 1, 2 and 3).
i. ENT key	– Press, vector of waypoint displayed (Format 7).

41. **To insert the Vector of a Waypoint** (after completing procedure in para 40):

a. Waypoint track	– Insert, track displayed.
b. Waypoint speed	– Insert, speed displayed.
c. ENT Key	– Press, SET, VEC and WP dim.

Note: If it is necessary to change a waypoint's position and retain the existing vector, the above procedure must be adhered to. If VEC is not pressed after SET, any vector entered is deleted.

42. **To Change the Vector of a Waypoint:**

a. SET key	– Press, SET brightens.
b. VEC key	– Press, VEC brightens.
c. Waypoint number required	– Press, number displayed. Vector displayed (Format 7).
d. CLR key (if required)	– Press, display clears.
e. Waypoint new track	– Insert, track displayed.
f. Waypoint new speed	– Insert, speed displayed.
g. ENT key	– Press, SET and VEC dim. Normal operation restored.

Data Setting

43. **Wind Velocity (Data Mode 2):**

a. SET key	– Press, SET brightens.
b. DTA key	– Press, DTA brightens.
c. Numeric key 2	– Press, 2 and present wind vector displayed (Format DTA 2).
d. Wind direction	– Insert, direction displayed.
e. Wind speed	– Insert, speed displayed.
f. ENT key	– Press, SET and DTA dim. Normal operation restored.

44. **Surface Motion Compensation (Data Mode 8):**

a. SET Key	– Press, SET brightens.
b. DTA key	– Press, DTA brightens.
c. Numeric key 8	– Press, 8 and previous SMC displayed.
d. Surface motion – direction	– Insert, direction displayed (Format DTA 8).
e. Surface motion – speed	– Insert, speed displayed. (Format DTA 8).
f. ENT key	– Press SET and DTA dim. Normal operation restored.

45. **Time (Data Mode 9):**

a. SET key	– Press, SET brightens.
b. DTA key	– Press, DTA brightens.
c. Numeric key 9	– Press, 9 and elapsed time displayed (Format DTA 9).
d. Time – hours, minutes and tenths of minutes	– Insert, precise time displayed (*see* Note).
e. ENT key	– Press when precise time is reached. SET and DTA dim. Normal operation restored.

Note: The time keyed into the computer is valid the instant ENT is pressed. The display reverts to 0000.1 after 2400.0.

Present Position Update

46. **To Update Present Position by overflying a pre-set Waypoint:**

a. Navigation mode switch	– Select B–D.
b. WP key	– Press, WP brightens
c. Waypoint number required	– Press, Waypoint number and its computed bearing and distance displayed.
d. FIX key	– Press when overhead waypoint. FIX brightens, display freezes.
e. CLR key	– Press and hold, FIX dims. Present position updated to the waypoint position. Normal operation restored.

47. **To Update Present Position:**

a. Navigation mode switch	– As required.

b. Press FIX key	– FIX key brightens, display freezes.
c. Press CLR key	– Display clears.
d. Lat/Eastings Decca co-ordinates	– Insert, co-ordinates displayed (Format 1, 2 or 3).
e. Press ENT key	– Information temporarily stored.
f. Press CLR key	– Display clears
g. Long/ Northings/ Decca co-ordinates	– Insert, co-ordinates displayed (Format 1, 2 or 3).
h. Press ENT key (hold for two seconds)	– Display blinks, data entered. FIX key dims. Normal operation restored.

48. **To Update Present Position as a Bearing and Distance to a pre-set Waypoint** (eg co-located VOR/DME beacons):

a. Navigation mode switch	– Select B–D.
b. WP key	– Press, WP brightens.
c. Waypoint number required	– Press, waypoint number and its computed bearing and distance displayed.
d. FIX key	– Press, display freezes. On VOR/DME indicators, note bearing and distance to pre-set waypoint (*see* Note).
e. Bearing and distance to pre-set waypoint	– Insert, updated bearing and distance displayed.
f. ENT key	– Press and hold, FIX dims. Present position updated to known ground position. Normal operation restored.

Note: Headings and bearings displayed on the computer are magnetic if zero variation is set on the compass, even though variation is keyed into the computer. The guide is: true in – true out; magnetic in – magnetic out. Therefore, when updating to a VOR/DME fix with variation set on the compass the observed magnetic bearing must be corrected to true before entering. If however, the variation is set in the computer and not at the compass, the bearing must be entered without correction.

49. **To Update DR Position to Decca Position:**

a. Press FIX	– FIX brightens. Displays freeze.
b. HYP selected on Sensor Switch. Check Sensor Fail Light out	– Position calculated by the computer using Decca inputs is displayed.
c. Press ENT until displays blank momentarily (two seconds approx)	– FIX dims. DR position updated to Decca position. Normal operation restored.

50. **To Update Decca Position to DR Position:**

a. Press FIX	– FIX brightens. Displays freeze.
b. Select DR on Sensor switch. Check Sensor Fail Light out	– Position calculated by the computer using Doppler and/or Air Data is displayed.
c. Press ENT until displays blank momentarily (two seconds approx)	– FIX dims. Decca position updated to DR position. Normal operation restored.

Information Displays

51. **To display Waypoint Position or Bearing and Distance of a waypoint from Present Position:**

a. Navigation mode switch	– Select L/L, GRID or B–D, as required.
b. WP key	– Press, WP brightens.
c. Waypoint number required	– Press, waypoint number and its position displayed (Format 1, 2 or 3).
d. OP key	– Press, WP dims. Normal operation restored.

52. **To display 'Heading-to-steer' and 'Time-to-go' to Intercept a Waypoint selected using a steering procedure (see para 57):**

a. Navigation Mode switch	- Select INT.
b. OP key (if necessary)	– Press. Heading-to-steer and time-to-go displayed (Format 4); maximum time-to-go display capability is 5 hours 16.4 minutes.

53. **To display Data using Data Modes:**

a. DTA key	– Press, DTA brightens.
b. Mode numbers required	– Press, corresponding mode displayed (Fig 5).
c. OP key	– Press, DTA dims. Data mode terminated. Normal operation restored.

54. **To display Magnetic Variation and computer sub-variant type:**

a. VAR key	– Press, VAR brightens, variation displayed on upper line and sub-variant type (eg 28) displayed on lower line (Format 8).
b. OP key	– Press, VAR dims. Normal operation restored.

55. **To display Position of a Waypoint as a Bearing and Distance from another Waypoint:**

a. Navigation Mode switch.	– Select INT.
b. WP key	– Press, WP brightens.
c. Waypoint number (to)	– Press, number displayed on lower row (display indicates bearing and distance from present position).
d. Waypoint number (from)	– Press, number displayed on upper row. Display indicates bearing and distance of (to) waypoint from (from) waypoint (Format 5).
e. OP key	– Press, WP dims. Normal operation restored.

56. **To display the Position and Vector of a Waypoint:**

a. Navigation Mode switch	– Select L/L, GRID or B–D.
b. WP key	– Press, WP brightens.
c. Waypoint number required	– Press, waypoint number and position displayed.
d. VEC key	– Press, VEC brightens. Vector displayed (Format 7).
e. OP key	– Press, WP and VEC dim. Normal operation restored.

Selection and Display of Steering Modes

57. Two steering modes are available; a Route mode where an automatic sequence of waypoints can be followed for routine navigation, and a Tactical mode using a single waypoint, static or moving, as the steering objective.

58. **Route Mode.** In the Route mode the computer defines the rhumb line track between the two current waypoints and combines speed, heading error and across track error in the steering signal such that there is smooth guidance to the destination waypoint via that track. The track maximum attack angle is either 45° (fixed wing) or 90° (rotary) depending on the sub-variant of the T9447 computer. As the destination waypoint is approached a smooth transfer on to the next track occurs at a certain time ahead of the waypoint, depending on the angle of turn, providing the position of the next waypoint has been entered. If no further waypoint has been entered guidance will continue on an extrapolation of the existing track. The waypoints inserted are followed automatically in an ascending or descending order, determined by the order in which they were selected, eg if waypoints 6 (to) and 5 (from) are selected in para 60, subsequent steering will be to waypoints 7, 8, and 9; if waypoints 5 (to) and 6 (from) are selected subsequent steering will be to waypoints 4, 3, 2 and 1. The computer does not sequence through 9 to 0 (ascending) or 0 to 9 (descending). If a random sequence of waypoint numbers is required for steering, eg 5, 2, 9, 4 etc, waypoints 2 (to) and 5 (from) must be selected as in para 60. Waypoint 9 must then be selected shortly before waypoint 2 is overflown and waypoint 4 shortly before 9 is overflown etc, otherwise the steering information will relate to the next consecutive numbered (ascending or descending) waypoint.

59. **Tactical Mode.** In the Tactical mode guidance is directly from present position to the selected waypoint if it is static, or to the predicted point of interception if moving.

60. **To Select Guidance to Waypoints in the Route Mode:**

a. STR key	– Press, STR brightens.
b. Waypoint number (TO)	– Press, SET brightens, waypoint number TO appears as bottom left hand digit.
c. Waypoint number (FROM)	– Press, waypoint number FROM appears as top left hand digit.
d. ENT key	– Press, STR and SET dim. Guidance is sequential from waypoint to waypoint. Next waypoint is selected automatically. Computer in OP mode, display controlled by the Navigation Mode switch (Formats 1, 2, 3 or 4).

A secondary mode of Route steering display is now available:

e. STR key	– Press again, display shows across track error and distance to run to the 'TO' waypoint (Format 7). The Navigation Mode switch will not affect the display.

61. **To Select Guidance to a Waypoint in the Tactical Mode:**

a. STR key	– Press, STR brightens.
b. Waypoint number (TO)	– Press, SET brightens, waypoint number is bottom left hand digit (upper blank).

c. ENT key – Press, STR and SET dim, computer in OP mode. Display controlled by the Navigation Mode switch.

d. STR key – Press again, display shows closing speed and range to intercept the waypoint selected (Format 8). Navigation Mode switch will not affect the display.

A secondary mode of Tactical steering display is now available:

FORMAT 1 (L/L) DECIMAL POINT

Latitude		N			5	1	3	9	5	Degrees, mins
Longitude	5	W		1	2	3	2	4	6	

Current WP

FORMAT 2 (GRID)

Eastings			E	4	3	2	7	9	Kilometres or
Northings	5		N	6	2	1	8	7	Data Miles

FORMAT 3 (HYP)

				Zones	Lanes		Lane Fraction		
Decca				0	1	1	2	5	6	Red or Green
Hyperbolic	5			0	3	3	3	4	7	Purple or Green

Zone letters are decoded as follows,
A=0, B=1, C=2, etc.
␣ 's displayed if hyperbolic data is not correctly entered.

FORMAT 4 (B-D)

Bearing				0	9	3	1			Degrees
Distance	5			1	4	8	1	8		Nm or Km

FORMAT 5 (INT)

Heading to steer				0	3	0	1			Degrees
Time-to-Go	5				0	0	4	3	2	Hrs/Mins.

Maximum "Time-to-Go" display capability is 5hours 16.4 minutes

FORMAT 6 (B-D with INT)

From

Bearing	3			0	9	3	1			Degrees
Distance	5			1	4	8	1	8		Nm or Km

To

Either or both left hand digits in top and bottom row flash when TANS not in OP mode

From

FORMAT 7 ROUTE MODE (STR)

Across Track Error	2			—	0	5	2	3		Nm or Km
Distance-to-Go	3				0	2	1	4	7	Nm or Km

Blank= Right
— = Left

TO

FORMAT 8 TACTICAL MODE (STR)

Closing Speed						1	2	7		Knots or Km/hr.
Range to Int.	3				1	3	9	4	2	Nm or Km

FORMAT 9 (VEC)

Course				1	2	3	0			Degrees
Speed	3					0	1	6		Knots or Km/hr.

FORMAT 10 (VAR)

Mag. Var.			E	0	1	0	5			Degrees
Sub-Variant No	3			3		1	1			

D Variant Sub-variant

FORMAT 11 (VAR)

Magnetic Variation			E	0	1	0	5			Degrees
Chain/Lanes	3			0	5		2			Chain code or Lanes & Fractions for R,G or P (Changes every cycle)

FORMAT 11 Lower line cyclic variations

(a)

3			0	5		2		

0 5 refers to the chain number and can be between 00 and 10. 2 refers to the chain letter. and is decoded as follows: 0=A: 1=B: 2=C: 3=D: 4=E: 5=F: ␣ 's=unusable chain.

(b)

3			1	7	2	3		

1 7 2 3 refers to the Red pattern. 17 can be between 00 and 23 and is the lane number, 23 is the lane fraction.

(c)

3			3	5	3	6		

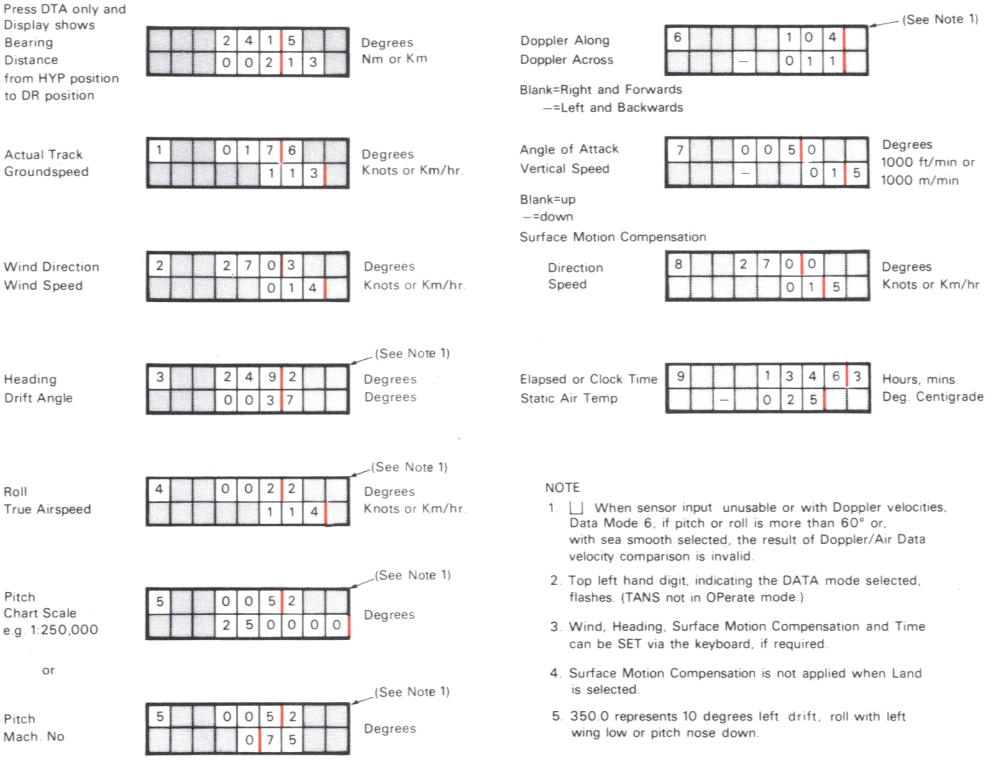
3 5 3 6 refers to the Green pattern. 35 can be between 30 and 47 and is the lane number, 36 is the fraction.

(d)

3			7	1	4	7		

7 1 4 7 refers to the Purple pattern. 71 can be between 50 and 79 and is the lane number. 47 is the lane fraction

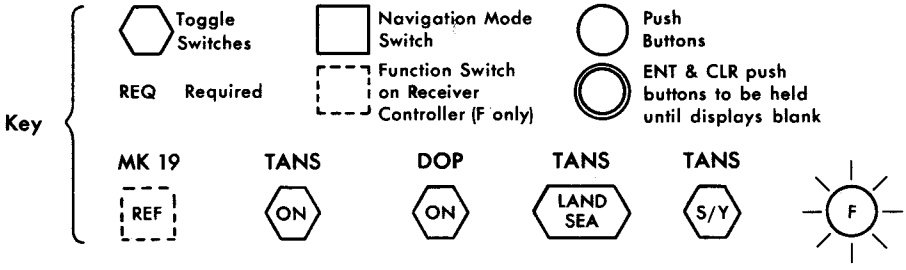
Fig 4 Display Formats



NOTE

1. ␣ When sensor input unusable or with Doppler velocities, Data Mode 6, if pitch or roll is more than 60° or, with sea smooth selected, the result of Doppler/Air Data velocity comparison is invalid.
2. Top left hand digit, indicating the DATA mode selected, flashes. (TANS not in OPerate mode.)
3. Wind, Heading, Surface Motion Compensation and Time can be SET via the keyboard, if required.
4. Surface Motion Compensation is not applied when Land is selected.
5. 350.0 represents 10 degrees left drift, roll with left wing low or pitch nose down.

Fig 5 Display Formats – Data Mode



To FIX Present Position in Latitude/Longitude

L/L — FIX — CO-ORDS — ENT HOLD FOR 3 SECS — F

To Define GRID Position

GL — GRID — CO-ORDS E/W N/S — ENT

To SET, or Change, Magnetic Variation

SET — VAR — E or W — DEGS — 0.1 DEGS — ENT

To SET, Change, or Display, position of Waypoints

AS REQ — SET IF REQ — WP — WP NO — CO-ORDS IF REQ — ENT IF REQ — OP TO END DISPLAY MODE

To FIX Present Position as a Waypoint

L/L GRID or HYP — FIX — WP — WP NO — ENT

To SET, Change, or Display , a Waypoint as a BRG/DIST from another

INT — SET IF REQ — WP 'TO' — LOWER WP NO — 'FROM' — UPPER WP NO

Waypoint

CO-ORDS IF REQ — ENT IF REQ — OP TO END DISPLAY MODE

To SET, Change, or Display, the Position of a Vectored Waypoint

AS REQ — SET — VEC — WP — WP NO — PRE-SET CO-ORDS SHOWN — CLR IF REQ

and/or its Vector

CO-ORDS — ENT PRE-SET VECTOR SHOWN — WP COURSE SPEED IF REQ — ENT IF REQ — OP TO END DIPLAY MODE

To SET, Change, or Display, the Vector of a Waypoint

SET IF REQ — VEC — WP NO — VECTOR SHOWN — CLR IF REQ — WP COURSE SPEED

Fig 6a Keying Sequences

To SET, or Display, Surface Motion Compensation (SMC)

(SET) IF REQ (DTA) (8) (SMC) DIRECTION SPEED (ENT) IF REQ (OP) TO END DISPLAY MODE

To SET CLOCK TIME on DTA 9

(SET) (DTA) (9) ( ) HR ( ) MIN ( ) 0.1 MIN (ENT) AT SET TIME

To Update Co-ordinates of Present Position (Co-ords)

[L/L] (FIX) (CLR) CORRECT TOP LINE CO-ORDS (ENT) (CLR) CORRECT BOTTOM LINE CO-ORDS (ENT) HOLD FOR 3 SECS

To Update Present Position by over-flying a Pre-set WP

[B/D] (WP) ( ) WP NO (FIX) WHEN OVER-HEAD (CLR) HOLD FOR 3 SECS (ENT) HOLD FOR 3 SECS

To Update Present Position as a BRG/DIST to Pre-set WP

[B/D] (WP) ( ) PRE-SET WP NO (FIX) (BRG/DIST WAYPOINT TO PRE-SET) (ENT) HOLD FOR 3 SECS

To Display Heading to Steer and Time to Go

[INT] (STR) ( ) WP NO (ENT)

STEERING MODES

Direct to a Pre-set Waypoint (Tactical)

(STR) ( ) WP NO (ENT) PRESS (STR) AGAIN DISPLAY SHOWS CLOSING SPEED DISTANCE TO WP (OP) TO END DISPLAY MODE

Sequentially between Pre-set Waypoints (Route)

(STR) 'TO' ( ) LOWER WP NO (ENT) 'FROM' ( ) UPPER WP NO PRESS (STR) AGAIN

DISPLAY SHOWS ACROSS TRACK ERROR DISTANCE TO WP (OP) TO END DISPLAY MODE

To Update DR Position to HYP Position

(HYP) (FIX) (ENT) HOLD FOR 3 SECS

To check Hyperbolic Co-ordinates

(VAR) Chain Number, Red, Green, Purple Lanes and Fractions shown in sequence on Lower Line

Fig 6b Keying Sequences

PART 3

SECTION 4

INERTIAL NAVIGATION SYSTEMS

Chapter

1 **Principles of Inertial Navigation**
Annex: **Alignment of Inertial Systems**

2 **INS Errors and Mixed Systems**
Annex: **Derivation of some INS Errors**

3 **The Ferranti Inertial Platform**
Annex: **Position Computing**

4 **The Elliott E3 Heading Reference System**
Annex: **Runway Alignment**

5 **The Elliott E3R Heading Reference System**
(*To be issued later*)

CHAPTER 1

PRINCIPLES OF INERTIAL NAVIGATION

CONTENTS

	Paras
Introduction	1
BASIC PRINCIPLES	
Acceleration	2–8
ACCELEROMETERS	
Basic Principle	9–15
Accelerometer Stabilization	16–17
INTEGRATORS	
Function of the Integrator	18
Analogue Integrators	19–21
Digital Integrators	22
GYROSCOPES	
Terms and Types	23–24
Single Degree of Freedom Gyros	25
Two Degree of Freedom Gyros	26
Comparison of Single and Two Degree of Freedom Gyros	27
Pick-Offs and Torquers	28
PLATFORM STABILIZATION	
Gyro Control of the Platform	29–38
PLATFORM MOUNTING	
Gimballed Systems	39–45
Non-Gimballed Systems	46
DRIFTS AND ACCELERATIONS SENSED BY AN INERTIAL PLATFORM	
Gyro Corrections	47–50
Accelerometer Corrections	51–53
Summary	54

Annex: Alignment of Inertial Platforms

Introduction

1. In an inertial navigation system, velocity and position are obtained by continuously measuring and integrating vehicle acceleration. Inertial navigation systems, are self-contained and are capable of all-weather operation.

BASIC PRINCIPLES

Acceleration

2. The basis of inertial navigation is the measurement of a vehicle's (aircraft's) acceleration along known directions. Accelerometers detect and

measure accelerations along their sensitive axes (input axes). The accelerometer outputs are integrated, once to obtain velocity along the sensitive axis, and again to obtain distance travelled along the sensitive axis.

3. **Relationship between Acceleration, Velocity and Distance.** The velocity achieved and the distance travelled by a vehicle accelerating from rest at a constant rate are obtained from the following equations:

$$v = at, \text{ and } s = \tfrac{1}{2}at^2$$

where, a = acceleration
v = velocity
s = distance
t = time

Aircraft accelerations are not constant, and must be integrated to obtain velocity and distance:

$$v = \int a.dt, \text{ and}$$

$$s = \iint a.dt, \text{ or } s = \int v.dt$$

The basic principle of inertial navigation is, therefore, the double integration of acceleration with respect to time (Fig 1).

4. **Measurement Axes.** Acceleration must be measured along two axes, usually orthogonal, if vehicle velocity and displacement are to be defined in a given plane. Since most accelerometers are designed to measure acceleration along one axis only, two accelerometers are required for inertial navigation in a two dimensional plane. In aircraft systems the accelerometers are usually mounted with their input axes aligned with north and east, and this alignment must be maintained if the correct accelerations are to be measured. Moreover, the sensitive axes must be kept perpendicular to the gravity vertical, otherwise, the accelerometers sense part of the gravity acceleration. The reference frame defined by these directions, *ie* local North, local East and local Vertical, is called the Local Vertical Reference Frame. Other reference frames can be used, but the local vertical is the most commonly used in aircraft systems, and is the only one considered in this chapter.

5. **Gyro Stabilization.** Once the accelerometers have been aligned in the chosen reference frame, they must be capable of maintaining that orientation during aircraft manoeuvres. The accelerometers are therefore mounted on a platform which is suspended in a gimbal system that isolates the accelerometers from aircraft manoeuvres. However, this platform is not inherently stable, and any tendency for the platform to rotate with the aircraft must be detected and opposed. Gyros are therefore mounted on the platform to detect platform rotation and control platform attitude. Three single degree of freedom gyros are normally used; one gyro detects rotation about the North axis, another rotation about East, and the third rotation about the vertical. The platform rotations detected by the gyros are used to generate error signals, proportional to change in platform attitude, which are used to motor the platform back to its correct orientation.

6. **Effect of Earth Rotation and Vehicle Movement.** An INS operating in the local vertical reference frame must maintain its alignment relative to Earth directions. The gyros used to stabilize the platform are rigid in space and must therefore be corrected for Earth rate and transport wander to make them "Earth stable". Additionally, the accelerometers must be corrected for the effects of coriolis acceleration and the central acceleration caused by rotating the platform to maintain alignment with the local vertical reference frame.

7. **Platform Control.** The platform control unit computes and applies the gyro and accelerometer

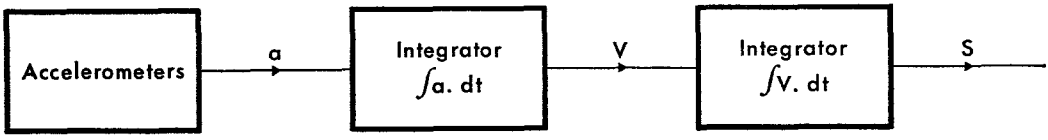


Fig 1 Principle of Inertial Navigation

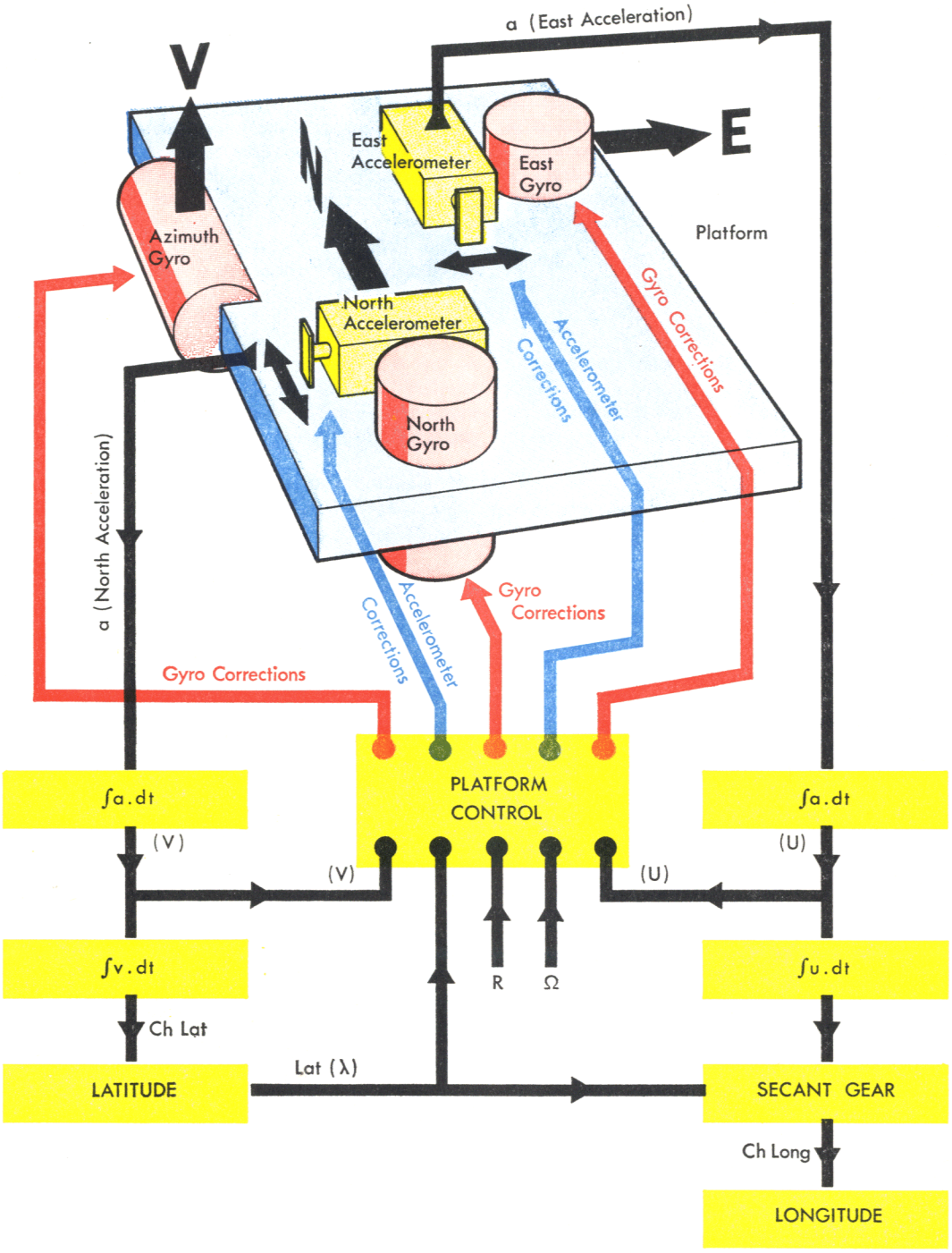


Fig 2 A Simple Inertial Navigation System

correction terms from calculated values of ground-speed and latitude and stored values of Earth radius and Earth rotation rate.

8. **Simple INS.** A simple INS, capable of solving the navigation problem, is illustrated in Fig 2. A third vertically mounted accelerometer must be added if vertical velocity is required, *eg* in weapon aiming applications. The individual INS components are discussed in detail in the following paragraphs.

ACCELEROMETERS

Basic principle

9. The basic INS measuring instrument is the accelerometer, on whose accuracy the INS output accuracy depends. Accelerometers utilizing the pendulum principle are normally used in INS applications.

10. **Basic Pendulous Force Balance Accelerometer.** A basic pendulous force balance accelerometer is shown schematically in Fig 3. With the case horizontal and the instrument at rest or moving at a constant velocity, the pendulum bob lies central and no pick-off current flows. When the instrument is accelerated along its sensitive axis, the bob is deflected and the deflection is sensed by the pick-off. A current flows through the restorer coils such that a force is exerted on the bob to restore it to the central position. The initial deflecting force is proportional to the acceleration experienced (a), since the mass of the bob is constant (F = ma). The restoring force is proportional to the current through the restorer coil and is equal and opposite to the initial force, *ie* the restorer current is proportional to the acceleration. The pendulum is free to swing only along the sensitive axis, and accelerations perpendicular to this axis have no effect on the bob.

11. **Restorer Coil Gain.** Several manufacturers have adapted the force balance principle to

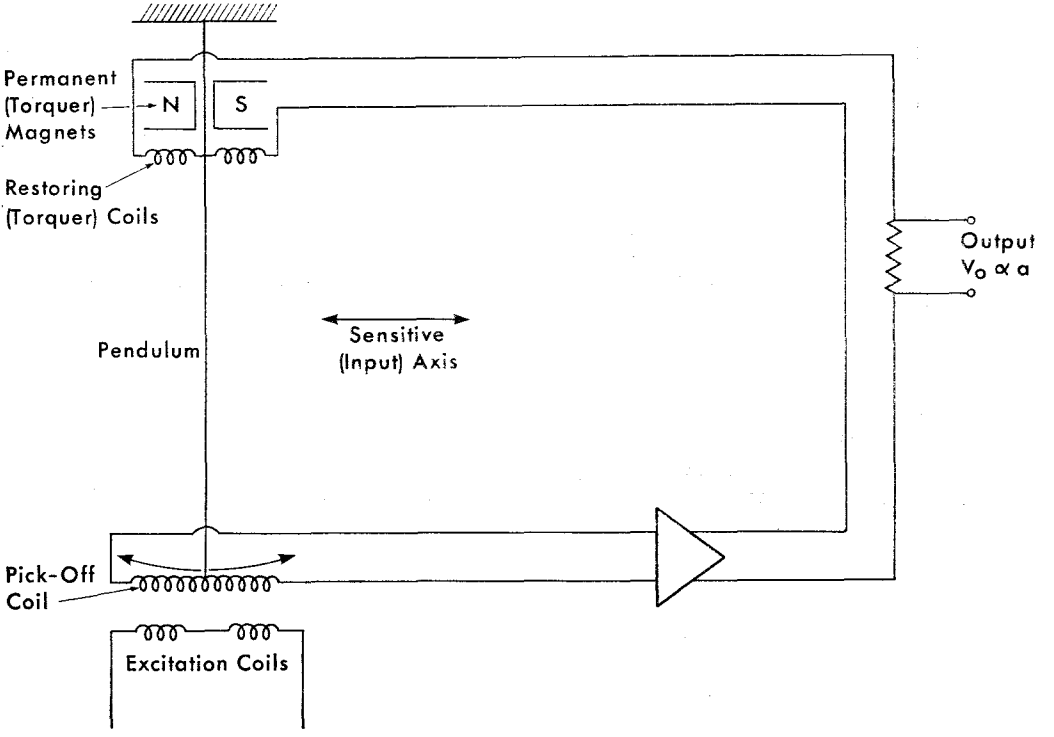


Fig 3 Basic Pendulous Force Balance Accelerometer

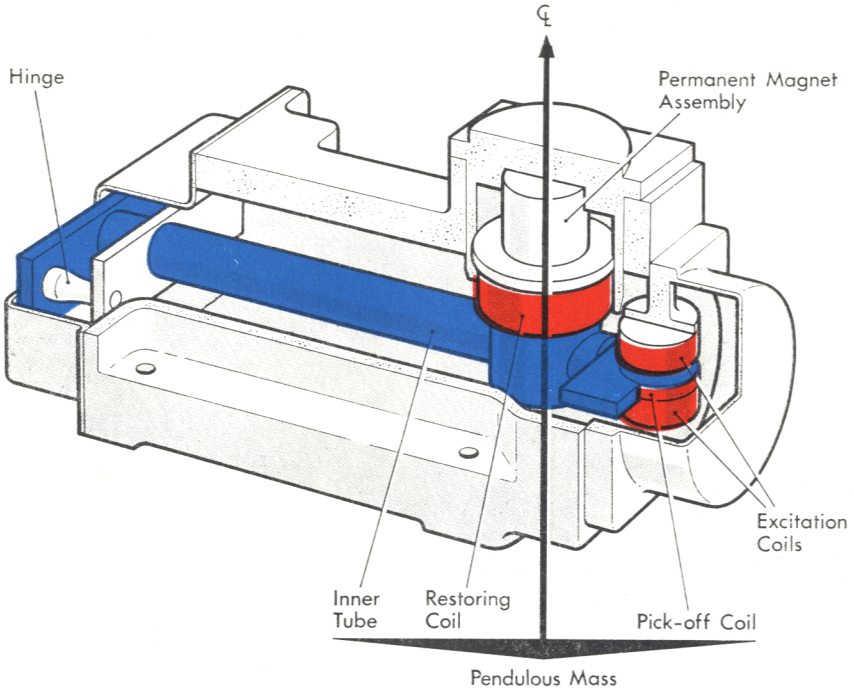


Fig 4 End Pivot Pendulous Accelerometer

produce accelerometers of inertial quality. A common feature of the pendulous force balance accelerometers is their very high restorer coil gain which severely limits the bob displacement. Two typical force balance accelerometers are described below.

12. **Ferranti End Pivot Accelerometer.** The Ferranti end pivot accelerometer is shown in Fig 4. The inner tube is the pendulum arm, and the restorer coil and the pick-off arm form the bob.

13. **Litton Centre Pivot Accelerometer.** The Litton centre pivot accelerometer is shown in Fig 5. The case is fluid-filled to float the pendulous mass thereby reducing pivot friction. Pendulosity is achieved by offsetting the pivot from the centre of gravity.

14. **Accelerometer Outputs.** An analogue output of acceleration is obtained from a resistive tapping in the accelerometer restorer circuit. Alternatively, in digital systems the output may be obtained in pulse form, thereby eliminating the conversion from analogue voltage to a suitable digital signal. The pulse generator produces a train of constant frequency DC pulses of equal amplitude and width. The pulse polarity and the restorer current polarity are both determined by the accelerometer pick-off. At the null position, alternative positive and negative pulses are generated in the torque rebalance circuitry. When the pendulous element is displaced, the pick-off changes the polarity of a sufficient number of pulses to restore the pendulous element, through the restorer coil, to the null position. The output pulse rate is then proportional to the acceleration, and each pulse represents an increment of velocity. By summing the pulses algebraically, the pulsed torque accelerometer gives a direct output of velocity in digital form.

15. **General Characteristics.** Accelerometers used in inertial applications should have the following characteristics:

a. *Low Sensitivity Threshold.* Accelerometers should be capable of detecting accelerations of the order of $10^{-6}g$.

b. *Wide Sensititivy Range.* Accelerometers should be able to detect acceleration over the

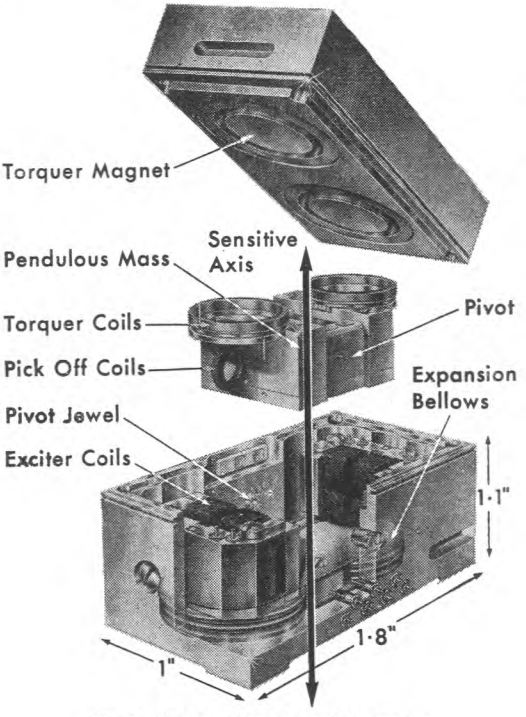


Fig 5 Centre Pivot Accelerometer

specified operating range with the same degree of accuracy; the range is normally from –10 to + 10*g*, depending on the amplifier gain used.

c. *Accelerometer Output.* This should vary in direct proportion to the input over the complete range of measurement of the instrument; a typical tolerance is 0·001% of acceleration applied.

d. *High Scale Factor.* The restorer current is proportional to the acceleration; actual current strengths depend on the amplification (scale factor) used. A typical scale factor is 5 ma/*g*.

e. *Good Zero Stability or Null Uncertainty.* Instrument error may result in the presence of an output for zero input. The null should, however, be defined within $\pm$ 0·00002*g*.

f. *Small Dimensions.* Accelerometers should be light (3 to 8 ounces) and small ($2\frac{1}{2}$ in $\times$ 1 in).

g. *High Shock Load Resistance.* Accelerometers should be able to withstand shock loads in the order of 60*g*, and have low response to vehicle vibration frequencies.

Accelerometer Stabilization

16. **Gravity Effect.** If the accelerometer is tilted, it detects the algebraic sum of the components of both gravity and horizontal acceleration acting along the sensitive axis (*see* Fig 6). The integrated accelerometer output, therefore, produces incorrect velocity and distance.

17. **Displaced Orientation.** When the accelero-

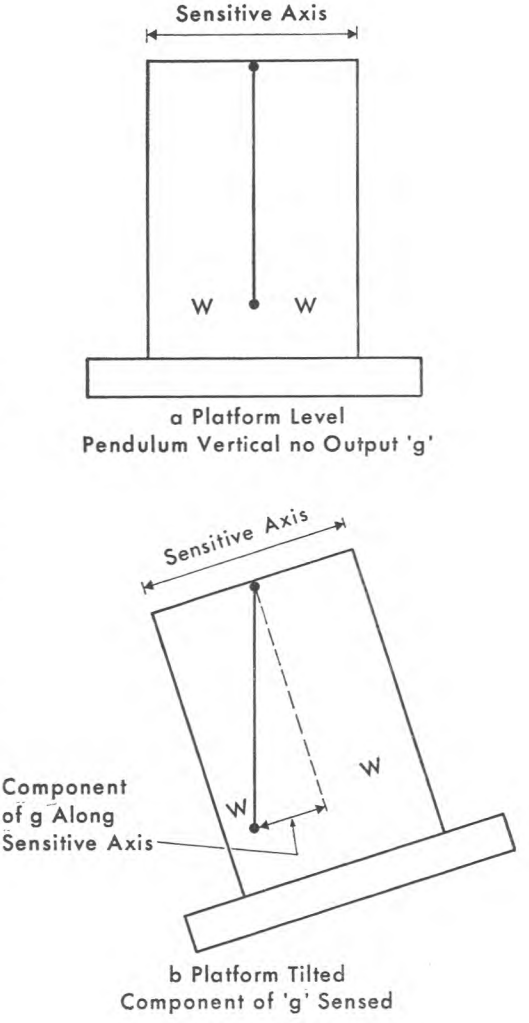


Fig 6 Accelerometer Tilted

meters' sensitive axes are displaced from their correct orientation, errors are produced in both the latitude and longitude channels. The platform in Fig 7 is displaced in azimuth and accelerated in a south/north direction. The north accelerometer does not detect the full acceleration, and part of the acceleration is detected by the east accelerometer.

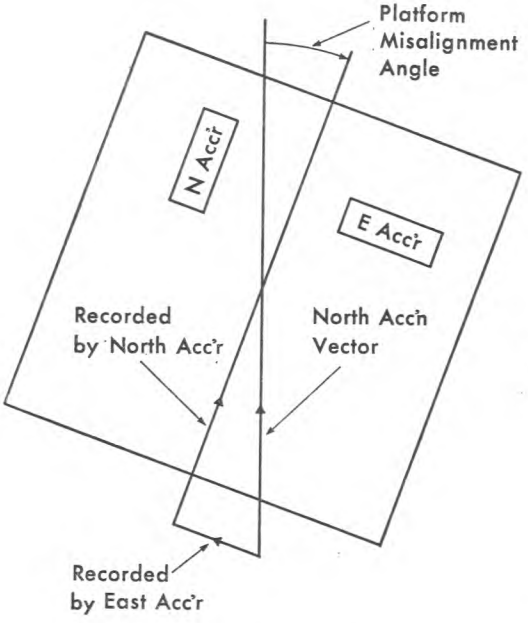


Fig 7 Accelerometer Misaligned

INTEGRATORS

Function of the Integrator

18. The accelerometer outputs are integrated to obtain velocity and again to obtain distance. The initial integration may be carried out within the accelerometer or by a separate integrating device. The accelerometer output may be in voltage analogue form if analogue techniques are used, or pulse form if digital techniques are used.

Analogue Integrators

19. Analogue integrators are normally electronic or electro-mechanical. Electronic integrators are more accurate, but are capable of integrating continuously for only limited periods of time. The electro-mechanical integrators are less accurate, but can integrate indefinitely.

20. **Miller Integrator.** A Miller amplifier circuit is an electronic integrating device, providing a voltage which is the integral of a varying input voltage (*ie* $V_0 = -\frac{1}{CR}\int V.dt$), the process being fast and accurate. The Miller integrator can only be used to integrate for limited periods of time, since the integrator must be allowed to regain its stable state periodically. The Miller integrator is therefore used to integrate spasmodic inputs, *eg* accelerations: it is not used to integrate continuous inputs, *eg* velocity. The circuit diagram of the Miller integrator is shown in Fig 8. A more detailed description of the Miller integrator is contained in AP 3302.

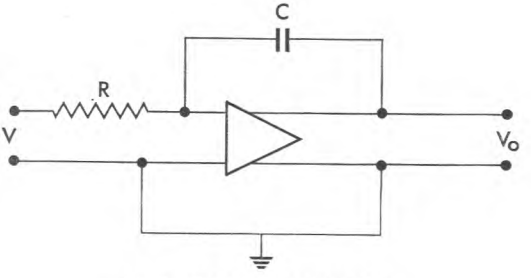


Fig 8 The Miller Integrator

21. **The Velodyne.** The velodyne is an electro-mechanical device which converts a voltage input into an output of shaft rotation proportional to the integral of the applied voltage. The operation

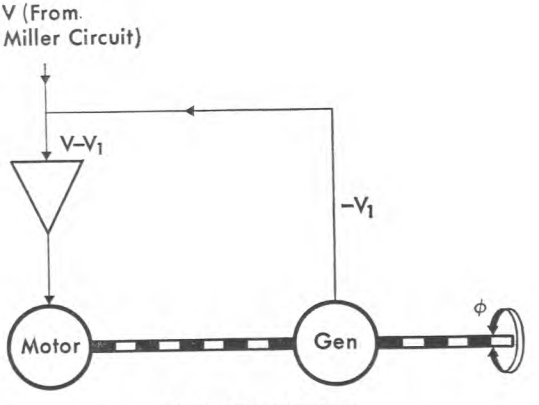


Fig 9 The Velodyne

of the velodyne (Fig 9) is described below:

a. As voltage V is applied, the motor starts to turn and drives a generator. As the motor speed increases, the output (V_1) from the generator increases, and is fed back to reduce the input voltage.
b. The process continues until the input is steady and is balanced by the feedback voltage. At this stage the output shaft is rotating at a constant speed.

The speed of both the motor and the output shaft $d\Phi/dt$ is directly proportional to the input voltage ($V - V_1 \propto d\Phi/dt$). The angle through which the shaft turns (Φ) is therefore proportional to the time integral of the input voltage

(*ie* $\Phi \propto \int (V - V_1).dt$). The velodyne can integrate continuously, and is therefore used for the second integration of acceleration to obtain distance travelled. Fig 9 shows a schematic diagram of the velodyne.

Digital Integrators

22. In modern systems digital integration may be performed in a central general purpose (GP) digital computer, or in a digital differential analyser (DDA) which solves the platform integrations in a digital manner.

GYROSCOPES

Terms and Types

23. The following discussion on gyroscopes assumes an understanding of basic gyrodynamics (Part 2, Sect 2). The following terms are included for clarification:

a. *Degrees of Freedom.* In the convention used throughout this chapter, the gyro rotor axis is not counted as a degree of freedom, since it cannot be a sensitive axis. A free or space gyro is therefore defined as a two degree of freedom gyro.
b. *Gyro Drift.* The term gyro drift describes any movement of the gyro spin axis away from its datum direction.
c. *Levelling Gyros.* Gyros which control the platform about the horizontal axes are called levelling or vertical gyros, irrespective of the direction of their spin axes.

24. **Types of Gyro.** A gyro is described as being of inertial quality when the real drift rate is $0{\cdot}01°$ per hour or less. Such low drift rates were first achieved with single degree of freedom rate integrating gyros.

Single Degree of Freedom Gyros

25. **Rate Integrating Gyro.** The rate integrating gyro achieves its accuracy by reducing gimbal friction: the gimbal and rotor assemblies are floated in a fluid. A typical floated rate integrating gyro is illustrated in Fig 10; the rotor is pivoted in an inner can (gimbal), which in turn is floated in an outer can. The outer can contains all the controls, pick-offs, torquers and heaters, *etc.* Rotation of the gyro about the input (sensitive) axis causes the gyro inner can to precess about the output axis, *ie* there is relative motion between the inner and outer cans. This precession is sensed by the pick-offs which measure the angular displacement of the inner can relative to the outer can. Thus, the pick-off output is proportional to the time integral of the input turning rate. This output signal is used to drive the platform gimbals to maintain the platform in the required orientation. The ratio of output to input (gimbal gain) is a function of rotor mass, gimbal size and fluid viscosity. A high ratio enables the gyro to detect small input rates. However, the fluid viscosity varies with temperature. Temperature must therefore be controlled to ensure a constant gimbal gain. With this type of gyro, it is also important to limit the inner can precession: as the inner can precesses, the rotor and the input axes are also precessed. Unless this precession is rapidly detected and opposed (the gimbal drives the platform and the gyro in opposition to the input), cross coupling errors are likely to occur. A cross coupling error is caused by the gyro sensing a rotation about a displaced input axis.

Two Degree of Freedom Gyros

26. Two degree of freedom gyros are used in some IN applications. SDF and TDF gyros have comparable performances, but the TDF gyro has the advantage of being able to detect movement about two axes. Since the INS monitors motion about three axes, two TDF gyros are not only sufficient, but also supply a redundant axis; the spare axis is normally utilized to monitor azimuth. The two TDF gyros must have their spin axes at right angles to each other; both axes may be horizontal, or alternatively one horizontal and the other vertical.

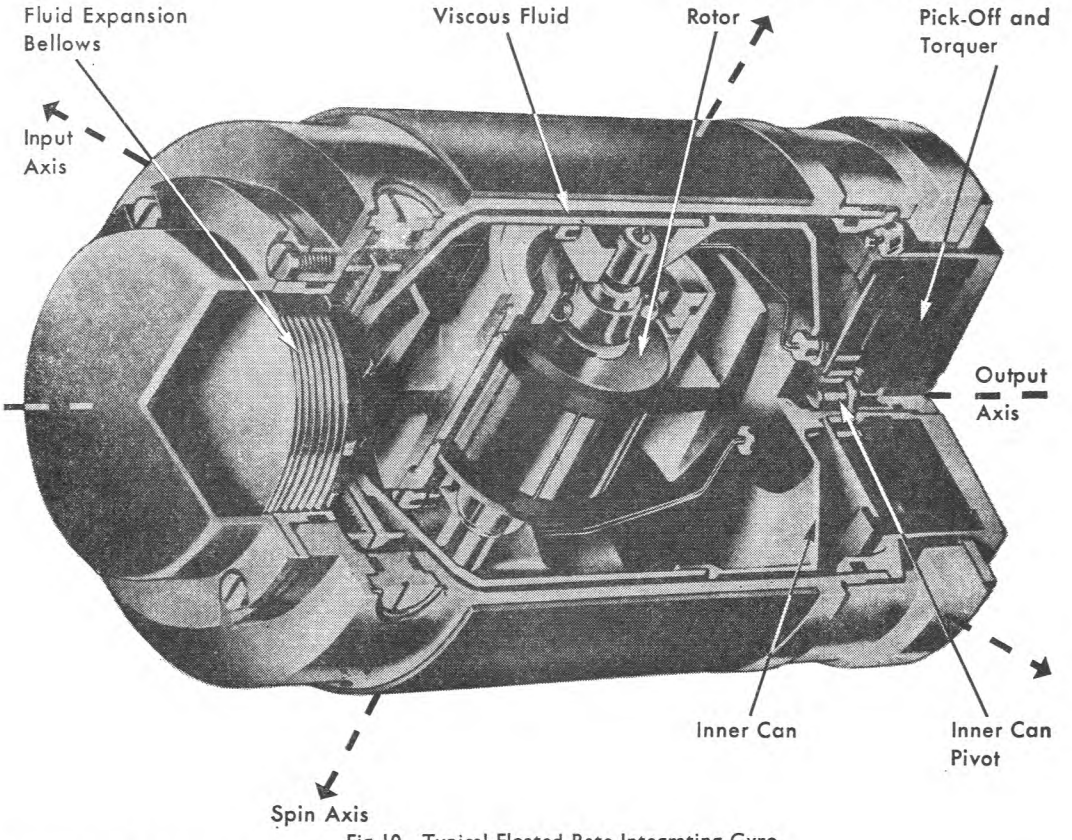


Fig 10 Typical Floated Rate Integrating Gyro

Comparison of Single and Two Degree of Freedom Gyros

27. The single and two degree of freedom gyros are compared in Table 1.

Pick-Offs and Torquers

28. Angular movement about a gyro's sensitive axis is detected by pick-offs which generate electrical signals proportional to the movement.

PROPERTY	SDF	TDF
Number Required in IN platform	Three	Two (one redundant axis)
Gyro Gain	Normally controlled by fluid viscosity	Output = input
Cross Coupling	Limited rotor axis movement minimizes cross coupling	No cross coupling – angular displacement is measured against fixed input axis
Vehicle Movement detection capability	Detected by rotor axis movement	Detected by gimbal axis movement
Accuracy	0·003°/hr to 0·1°/hr	As for SDF

Table 1 Comparison of SDF and TDF Gyros

aligned in attitude and azimuth using one of the techniques described in the Annex. Any platform misalignment causes errors, which are discussed fully in Chap 2.

32. **Use of Gyros.** The desired platform orientation is maintained by mounting reference gyros on the platform to detect changes in platform alignment. The gyro outputs are used to drive gimbal motors which return the platform to its correct orientation.

33. **Platform Arrangement.** The platform may be arranged as shown in Fig 12. The three gyros have their input axes mutually at right angles and aligned with the local vertical reference frame. The error pick-offs and torquers are built into the gyro cases and are not shown in the diagram. The platform is gimbal mounted to permit the aircraft freedom of manoeuvre without disturbing the platform away from its alignment with the local vertical reference frame. Each gimbal is driven by a servo motor controlled by the error signals from the gyros.

34. **Control on North.** The gyros in Fig 12 are arranged with their sensitive axes pointing in the directions about which rotation is to be detected. The East gyro has its sensitive axis pointing East, and is therefore capable of detecting rotation about East. On northerly headings, pitch manoeuvres are detected by the East gyro which generates an error signal. This error signal activates the pitch gimbal, thereby maintaining the platform's alignment with the reference frame. Similarly, roll is detected by the North gyro, and yaw by the azimuth gyro: the North gyro activates the roll gimbal motor, and the azimuth gyro the

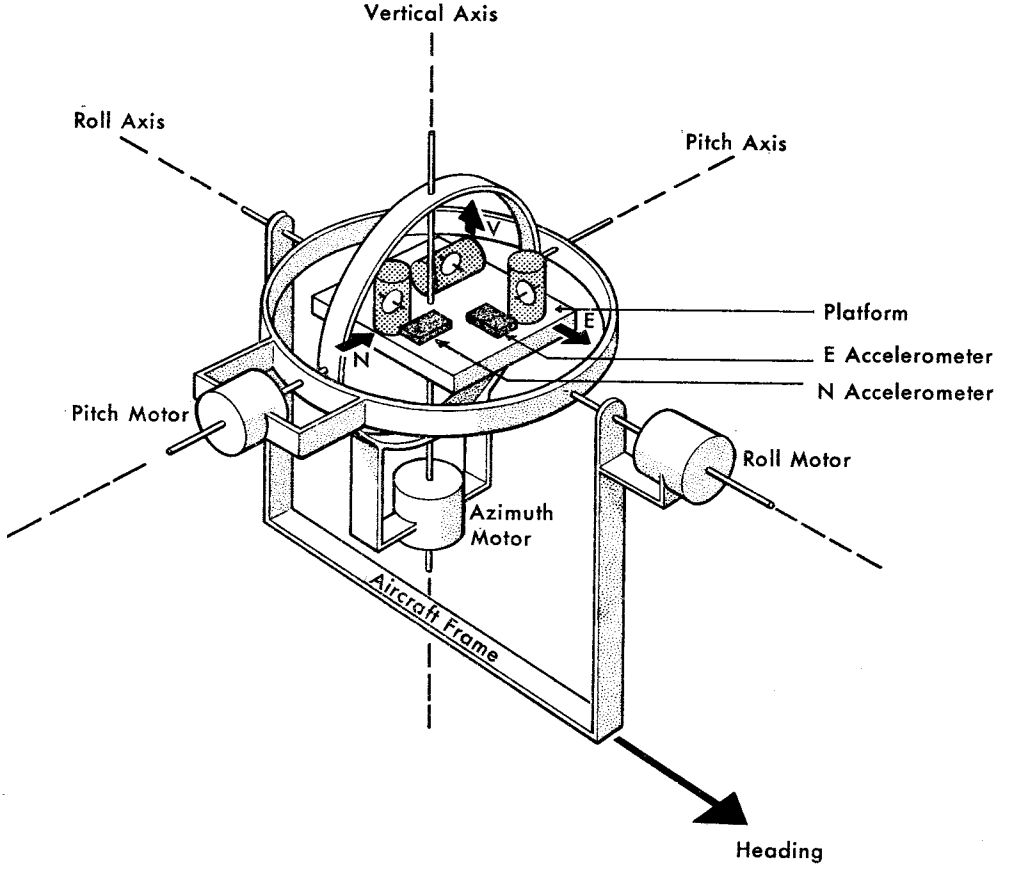


Fig 13 Platform Arrangement (Aircraft Heading East)

yaw gimbal motor. The action is summarized in Table 2.

Heading	Manoeuvre	Sensing Gyro	Correcting Servo-Motor
North	Yaw	Azimuth	Azimuth
	Pitch	East	Pitch
	Roll	North	Roll

Table 2 Action on North

35. **Control on East.** In Fig 13, the same platform is again shown, this time heading East. The action on East is summarized in Table 3.

Heading	Manoeuvre	Sensing Gyro	Correcting Servo-Motor
East	Yaw	Azimuth	Azimuth
	Pitch	North	Pitch
	Roll	East	Roll

Table 3 Action on East

36. **Conclusions.** Two main conclusions may be drawn from Tables 2 and 3:

a. Yaw, or change of heading, is corrected by the azimuth servo-motor which is always controlled by the azimuth gyro.

b. Pitch and roll are corrected by the pitch and roll servo-motors respectively. However, the control may be exercised by either the North or the East gyros or both, dependent upon aircraft heading.

37. **Change of Heading.** The action of the azimuth gyro and servo-motor keeps the platform aligned with the north datum. However, the pitch and roll gimbals remain oriented to the aircraft pitch and roll axes (Figs 12 and 13). Relative motion about the vertical between the platform and the pitch and roll gimbals is yaw, and angular displacement is change of heading. A pick-off of the angular displacement relative to true North as defined by the platform, produces an output of heading.

38. **Control During Manoeuvres.** On northerly headings, the North gyro senses roll and wholly controls the roll servo-motor; on easterly headings the East gyro controls the roll servo-motor. On intermediate headings, the control is shared between the North and East gyros, the amount of control exercised being determined by the heading. A sine-cosine resolver, set by the azimuth servo-motor, determines the amount of control and transmits the error signal to the appropriate servo-motor. The action is shown in Fig 14.

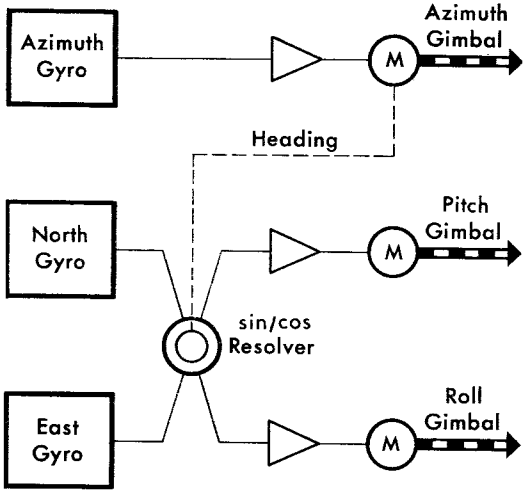


Fig 14 Gimbal Control Signals

PLATFORM MOUNTING

Gimballed Systems

39. The stable element of the inertial platform is mounted in gimbals to isolate the platform from vehicle manoeuvres. Three types of gimbal system are in common use.

40. **Three-Gimbal System.** Figs 12 and 13 are diagrams of a three-gimbal system. In such a system there are three input/output axes, azimuth, pitch and roll. Each gimbal imparts freedom about one particular axis, the particular gimbal being named after that axis.

a. *Azimuth Gimbal.* The stable element is rigidly attached to the azimuth, or first, gimbal. In allowing relative motion between the stable element and the pitch gimbal, the platform is

isolated from vehicle movement about the vertical axis.

b. *Pitch Gimbal.* The pitch gimbal isolates the platform from pitch manoeuvres.

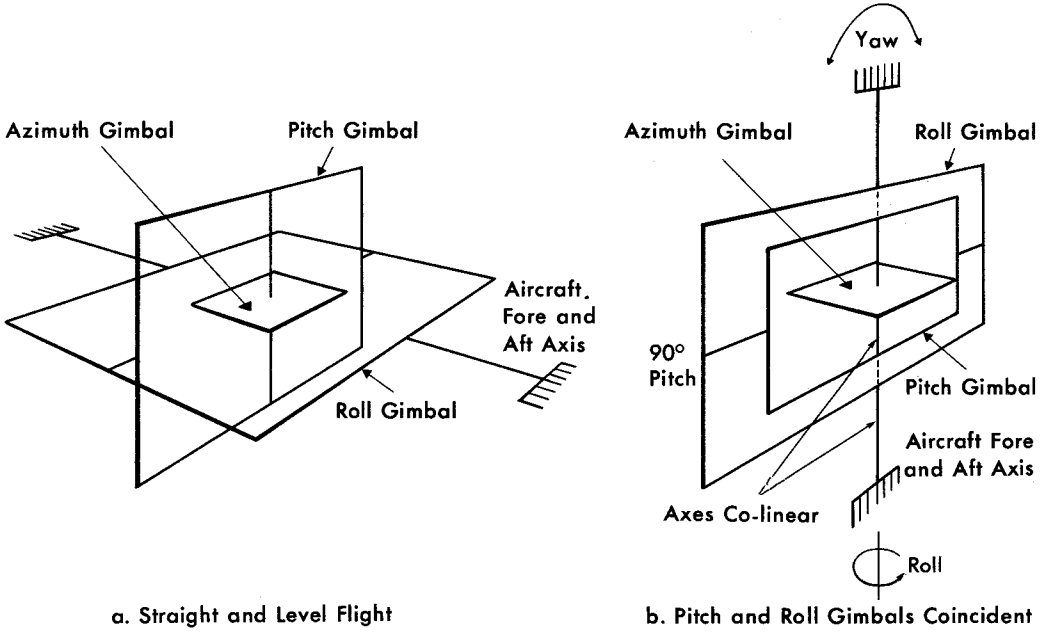
c. *Roll Gimbal.* The roll gimbal isolates the platform from roll manoeuvres.

In some installations, the pitch and roll gimbals are reversed in order of position.

41. **Gimbal Lock.** Gimbal lock occurs when two axes of rotation become co-linear and, as a result, one degree of freedom is lost. Fig 15 illustrates how gimbal lock can occur in a three-gimbal system. If the vehicle pitches through 90°, the first and third gimbal axes become coincident, and the platform stable element is no longer isolated from yaw.

42. **Gimbal Error.** In a three-gimbal system (gimbal order; Azimuth, Pitch and Roll) the roll gimbal axis, which is parallel to the aircraft roll axis, assumes an angle relative to the plane of the platform stable element whenever the aircraft pitches through large angles. When this occurs, the gimbal roll axis and the plane of the levelling gyros' input axes are no longer parallel. Should the aircraft now roll, the gyros sense only a component of roll angle (roll × cos pitch angle), and the roll servo displaces the roll gimbal by an amount (roll × cos pitch angle) instead of the full value of roll angle.

43. **Four-Gimbal System.** In a four-gimbal system the order of the gimbals is azimuth, inner roll, pitch and outer roll. The fourth gimbal is introduced to keep the second and third gimbals at right angles, thereby avoiding both gimbal lock and gimbal error. The fourth gimbal is controlled by a pick-off which detects changes in the angle between the second and third gimbals.



a. Straight and Level Flight

b. Pitch and Roll Gimbals Coincident

Fig 15 Gimbal Lock

44. **Gimbal Flip.** With a four-gimbal system, heading change is picked off from the relative motion between the azimuth and inner roll gimbals. If, however, the aircraft

completes a half loop and roll-out manoeuvre, the aircraft heading changes by 180°; but there is no motion between the azimuth and inner roll gimbals, and the indicated heading remains unchanged. This problem is overcome by employing gimbal flip.

As the pitch angle passes through 90°, the outer gimbal is driven through 180° (*ie* flips), tending to drive the platform through 180° about the vertical. This tendency is detected by the azimuth gyro which provides an appropriate output signal. This signal keeps the platform correctly orientated by driving the platform in opposition to the flip. 180° relative motion is produced between the azimuth and inner roll gimbals and the heading output remains correct.

45. **Comparison of Three- and Four-Gimbal Systems.** A four-gimbal system is heavier, larger and costs more than a three-gimbal system. However, since the second and the third gimbals of the four-gimbal system are kept at right angles, the aircraft has full freedom of manoeuvre without disturbing the platform.

46. **Inside-Out System.** In the inside-out system, the azimuth cluster containing the gyros and accelerometers lies outside the gimbals. The basic inside-out system (Fig 16a) has three gimbals and is not fully manoeuverable but the addition of a fourth gimbal permits unrestricted manoeuvre. A special case of this type of four-gimbal system (*see* Fig 16b) uses two TDF gyros instead of the more normal three SDF gyros. The following advantages are claimed for the inside-out arrangement:

a. The replaceable parts are more accessible.

b. The gimbals have less mass than in a conventional gimbal system and are more responsive to control.

Non-Gimballed System

47. **Strapped-Down Systems.** Strapped down systems dispense with the gimbal mounted stable element. The accelerometers are attached to the airframe and their sensitive axes are aligned with the aircraft axes. A gyro attitude reference passes the aircraft's orientation, relative to the chosen reference frame, to a digital computer which processes the accelerometer outputs and calculates aircraft displacement within the reference frame. Although gyros are not required to stabilize the accelerometers, very accurate gyros are required to

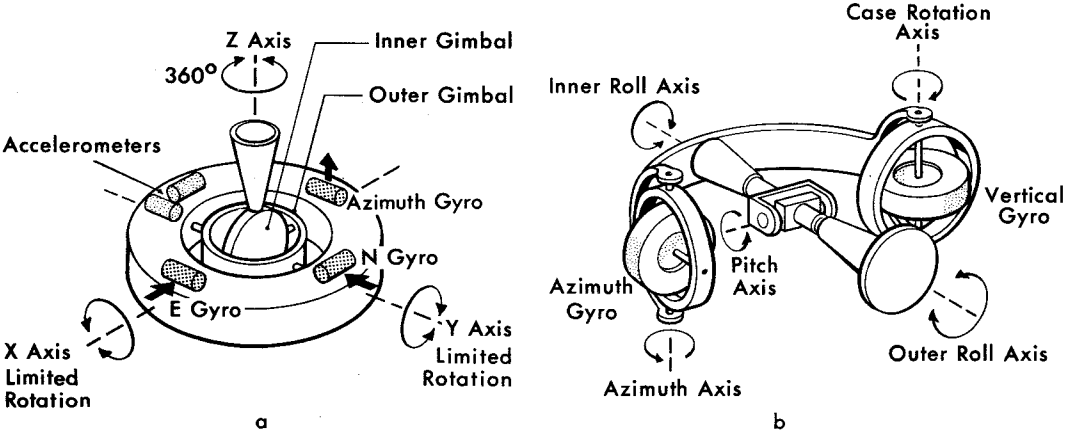


Fig 16a Inside-out Platform Arrangement

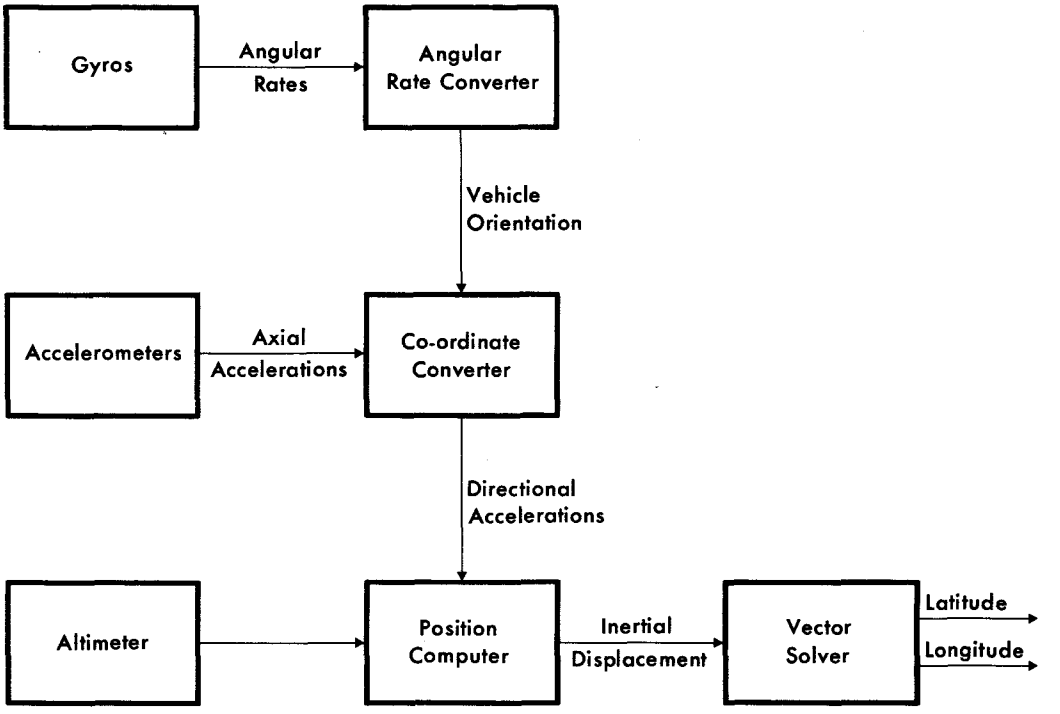


Fig 17 Strapped-down System Block Diagram

define the reference directions. Strapped-down systems became feasible for aircraft use with the development of the lightweight digital computer. The block diagram of a strapped-down system is shown in Fig 17.

CORRECTIONS TO INERTIAL SENSORS

Introduction

48. It is normal to navigate aircraft with reference to the local Earth co-ordinates of latitude, longitude and height. Aircraft INS are therefore normally aligned as described in para 30; each of the 3 axes of the local vertical reference frame has an accelerometer to detect movement along it and a gyro to provide stabilisation against rotation around it. Accelerometers and gyros are both inertial devices in that their sensitive axes extend infinitely in straight lines; in other words they operate with reference to the constant axes of inertial space. Local vertical axes however are not constant. It is therefore necessary to change the orientation of the platform axes relative to inertial space in order that the accelerometers are kept aligned with the local vertical axes. This means that the stabilising effect of the gyros must be adjusted by the rates at which local vertical axes diverge from inertial axes. These rates are due to Earth rotation and vehicle movement as shown in Table 4. (*See* P2, S2, C1, para 42). The changing orientation of the platform also makes corrections to the accelerometer outputs necessary.

49. It is now necessary to analyse Earth and transport rates into components affecting the local vertical axes. These are the rates which are applied to the platforms axes to correct it from inertial space stabilisation

	Earth Rate	Vehicle Movement
North Gyro	$\Omega \cos \lambda$	$\frac{U}{R}$
East Gyro	zero	$-\frac{V}{R}$
Aximuth Gyro	$\Omega \sin \lambda$	$\frac{U}{R} \tan \lambda$

Table 4 Platform Correction Terms

to local vertical stabilisation. The method used in the following discussion is that of vector analysis. A rate of rotation is represented by a vector shown parallel to the axis of the rotation. Its length is proportional to the rate of rotation and its direction is the direction an ordinary right hand threaded screw would move if subjected to the rotation in question. This is shown in Fig 18 which shows the Earth's rotation vector. The vector is parallel to the Earth's spin axis, its length represents 15:04°/hr (Ω) and its direction is from South to North.

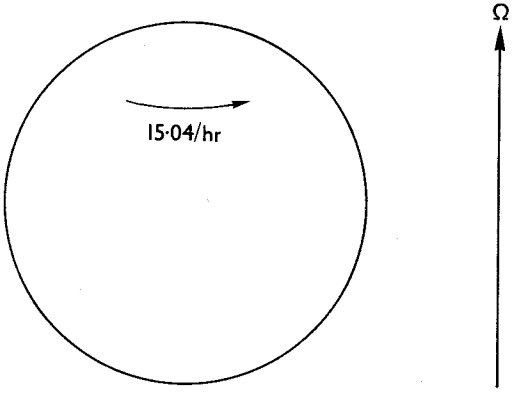


Fig 18 Earth's Rotation Vector

Gyro Corrections

50. **Earth Rate.** The Earth's rotation vector may be analysed into components acting about the local vertical axes at any point on the Earth's surface. The component acting about local East is always zero because local East is always at 90° to the rotation vector. At the poles the rotation vector coincides with the local vertical axis, and at the equator it coincides with the local North axis. This means that an INS not corrected for Earth rotation will appear to drift, but not topple, at the pole; whereas at the Equator it will topple about local North but not drift. Fig 19 shows how the Earth rotation rate is resolved into vector components acting about local North and local vertical axes at intermediate latitudes.

51. **Transport Rates.** Fig 20 shows that any movement around the circumference of a circle equates to a rotation about the centre of the circle. The angle θ, in radians, is found by dividing the circumferential distance A–B by the radius of the circle. Similarly, the rate of rotation may be found by

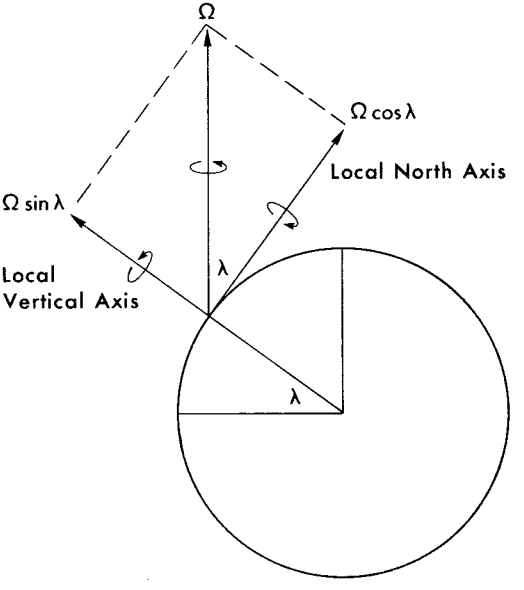


Fig 19 Earth Rate Vector Components

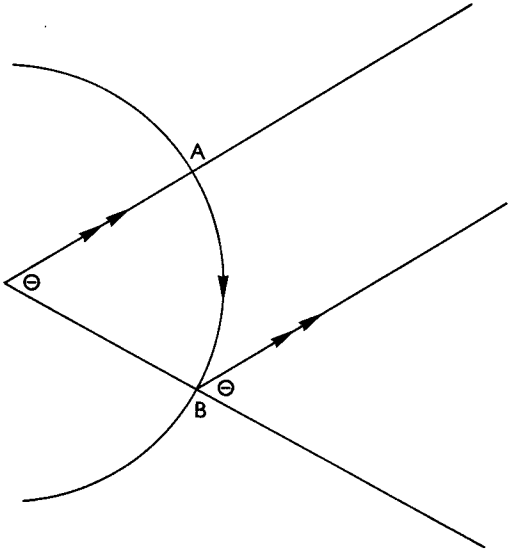


Fig 20 Circular Movement

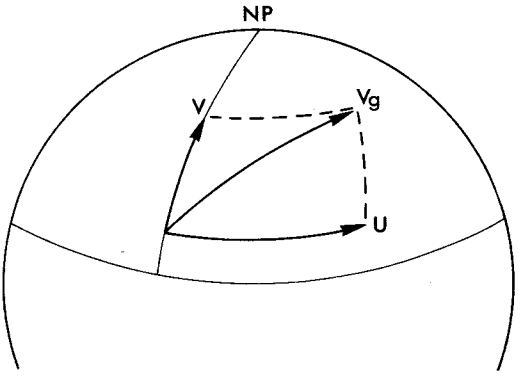


Fig 21 Components of Velocity Vector

dividing the rate of movement from A to B by the radius. The axis of the rotation is perpendicular to both the radius and the tangent, ie, normal to the surface of the page.

Fig 21 shows how a total aircraft velocity vector Vg may be resolved with North and East components. Conventionally, velocity north is annotated 'V' and velocity east 'U'. Component V produces a rotation rate of V/R radians/hr about an axis parallel to the local East axis and through the centre of the earth; (where V is in knots and R is the radius of the Earth in nm. Component U, however, acts along a parallel of latitude, ie a small circle; U therefore produces a rotation rate of $\frac{U}{R \text{ Cos } \lambda}$ about the Earth's polar axis as shown in Fig 22. This rate must be resolved into rates about the local North and local vertical axes, $\frac{U}{R}$ and $\frac{U \tan \lambda}{R}$ respectively, before it can be applied to the IN platform. This is achieved using the same analysis by vectors as was used for Earth rate, for the axis of rotation is the same: the Earth's spin axis. The quantities arrived at by this analy-

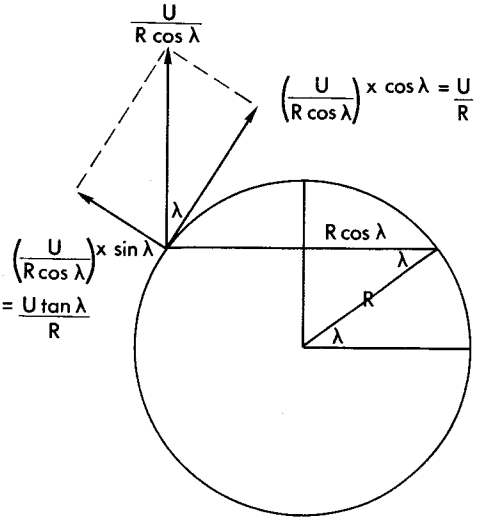


Fig 22 Rotation Rates

sis are in radians per hour; they may be approximated to degrees per hour by substituting 60 for R in the final expressions.

52. **Correction Method.** The drift due to the error rate is eliminated by applying an equal and opposite correction to the gyro output axis. The correction is applied through a torque motor on the gyro output axis, which turns the gyro about its output axis at the same rate but in the opposite direction to the precession caused by the error rate.

Accelerometer Corrections

53. Stabilising a platform to local Earth axes requires that it be rotated relative to a spatial reference in order to compensate for the effects of Earth rotations and vehicle movement. The resulting change in the local axes relative to spatial references makes 2 types of accelerometer corrections necessary:

a. *Central or Centripetal Acceleration.* AP3456K Part 2 Sect 4 Ch 2 paras 14–17 show that a body moving at a constant speed v in a circle radius r has a constant acceleration of v^2/r directed towards the centre of the circle. This is a central or centripetal acceleration and affects a local vertical INS because as the platform is transported over a spherical surface it is rotated to maintain its alignment with local North and the local vertical.

b. *Coriolis Acceleration.* Coriolis acceleration results from the combination of aircraft velocity and the rotation of the earth over which it flies. A lateral acceleration relative to inertial references is necessary to make good a desired track measured against meridians which are themselves in motion.

54. **Central Accelerations.** At any instant when an INS is moving over the Earth's surface it is moving along an arc of a great circle. An acceleration of $\frac{Vg^2}{R}$ therefore affects the vertical accelerometer where Vg is along track velocity and R is the radius of the Earth. Pythagoras' theorem enables us to convert this term to its component form $\frac{V^2 + U^2}{R}$ and thus make use of the 1st integrals of the North and East channels accelerations. This quantity as a correction must be added to the output of the vertical accelerometer. (*See* note at end of para 59).

55. Central acceleration corrections must also be applied to the horizontal accelerometers because of meridian convergence. Any East component of velocity acts along a small circle of latitude whose radius is R cos λ. There is thus a central acceleration of

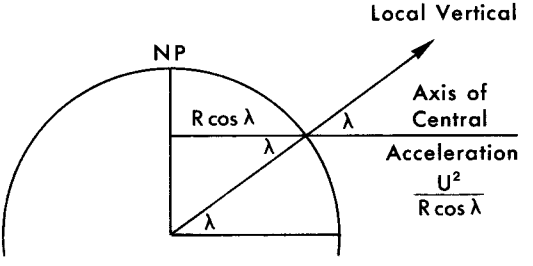


Fig 23 Axis of Central Acceleration

$\frac{U^2}{R \cos \lambda}$ along this radius, that is, along an axis inclined at λ to the local vertical. This is shown in Fig 23. Because of this inclination, the total acceleration may be resolved by vector analysis into 2 components, one affecting the North accelerometer and the other the vertical accelerometer. These are shown in Fig 24.

The component $\frac{U^2}{R}$ is contained within the vertical accelerometer correction already discussed. The acceleration component $\frac{U^2 \tan \lambda}{R}$ however, must be subtracted from the output of the North accelerometer be-

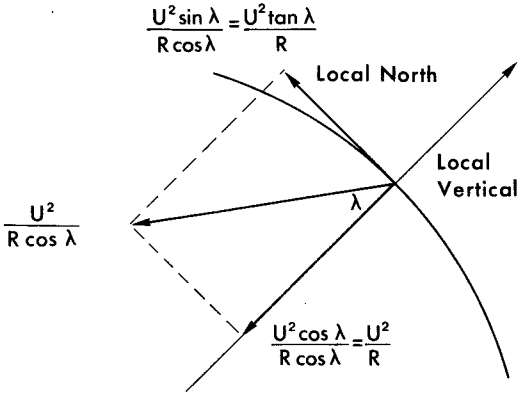


Fig 24 Components of Central Acceleration

cause it is caused entirely by an Eastward motion. This apparent contradiction arises because while "East" is a constant direction in terms of navigation over the surface of the Earth, it is a direction which constantly changes with respect to the fixed axes of inertial space. We thus have an Eastward velocity component producing an output from the North accelerometer, this must be removed for purposes of navigation.

56. Now consider an aircraft flying a great circle track at a constant groundspeed. The track angle is constantly changing and, therefore, so are U and the North accelerometer correction. In order that a constant total velocity vector results, the output of the East accelerometer must be adjusted in inverse proportion to the North accelerometer correction. The horizontal accelerometer central corrections thus produce varying V and U components of total velocity as track angle changes relative to the converging meridians. Table 5 shows that if there is no East component of velocity there is no central correction to either horizontal axis. If there is no North component, only the North accelerometer correction is applied, as discussed earlier. Also the magnitude of the corrections to the horizontal accelerometers increases as latitude increases, ie as meridian convergence increases.

57. **Coriolis Acceleration.** An aircraft flying a constant track over a spherical rotating Earth follows a path which is curved relative to the constant axes of inertial space. As shown in para 50 there is a component of Earth rotation which acts about the local vertical axis, this component $\Omega \sin \lambda$ varies with latitude. An observer may thus be regarded as being at the centre of a rotating disc of Earth's surface; the direction of rotation being anti clockwise when viewed from above, in the Northern hemisphere. An aircraft flying towards a given point on the horizon is therefore flying to a destination which is moving constantly to the left. A straight track over the ground thus produces a track which is curved relative to a constant spatial direction; this can only be achieved if there is a sideways acceleration. This acceleration is the Coriolis effect and is detected by the horizontal accelerometers. It must, however, be removed if the system is to produce navigation information which is correct relative to Earth co-ordinates. A similar correction is applied to the vertical accelerometer because of the component of Earth rotation acting about the local North horizontal axis. The corrections are given below:

a. $2\Omega V \sin\lambda$ applied to the East accelerometer.

b. $-2\Omega U \sin\lambda$ applied to the North accelerometer.

c. $3\Omega U \cos\lambda$ applied to the Vertical accelerometer.

58. **Gravity corrections.** When a third accelerometer is used in the vertical channel to measure vertical acceleration for weapon aiming purposes its sensitive axis will necessarily be in line with the gravity vector; the accelerometer will sense the acceleration due to gravity as well as aircraft vertical acceleration. Its output must therefore, be corrected for gravity, in addition to coriolis

and centripetal accelerations. Because the gravity acceleration decreases as the distance from the centre of the Earth increases, the correction is dependent on aircraft altitude. The correction is given by:

$$go \left\{ \frac{2h}{R} - 1 \right\}$$

where go is the gravity at the surface of the Earth and h is the aircraft altitude.

Summary

59. The gyro and accelerometer correction terms are summarized in Table 5.

Note: In the Southern Hemisphere, the signs of the azimuth gyro correction terms are reversed. That is, earth rate (Ω) and velocity East (U) are negative.

Axis	Gyros		Accelerometers		
	Earth Rate	Transport Wander	Central	Coriolis	Gravity
North	$\Omega \cos \lambda$	$\frac{U}{R}$	$\frac{-U^2 \tan \lambda}{R}$	-2Ω U Sin λ	nil
East	nil	$\frac{-V}{R}$	$\frac{UV \tan \lambda}{R}$	2Ω V sin λ	nil
Azimuth/ Vertical	$\Omega \sin \lambda$	$\frac{U}{R} \tan \lambda$	$\frac{U^2 + V^2}{R}$	2Ω U cos λ	$go \left\{ \frac{2h}{R} - 1 \right\}$

Table 5. Gyro and Accelerometor Terms

ALIGNMENT OF INERTIAL SYSTEMS

CONTENTS

	Paras
Introduction	1–2
Self-Alignment	3–8
External Reference Alignment	9–14
Airborne Alignment	15

Introduction

1. The stable element in an INS must be accurately aligned in both azimuth and attitude to allow the accelerometers to measure accelerations along their chosen axes. Three methods of system alignment are commonly used:

a. Self-alignment
b. External reference alignment
c. Airborne alignment

2. **Warm-Up Period.** Many of the gyros and accelerometers used in modern inertial systems are fluid-filled, the components achieving optimum accuracy only when the fluid is at the correct operating temperature. The first stage in any alignment sequence is to bring the fluid-filled components to the correct operating temperature; during this warm-up period the gyros are also run up to their operating speeds.

Self-Alignment

3. The process of self-alignment takes place in four stages:

a. Warm-up period
b. Coarse alignment
c. Fine levelling
d. Gyro-compassing

4. **Warm-Up.** During the warm-up period the fluid-filled accelerometers and gyros are brought to the correct operating temperature and the gyros are run up. This phase normally takes between 3 and 4 minutes.

5. **Coarse Alignment.** During the coarse alignment phase the platform is roughly levelled and aligned in azimuth, thereby removing gross alignment errors and cutting the overall alignment time to a minimum.

a. *Coarse Levelling.* The pitch and roll gimbals are driven until they are at 90° to each other. The platform is then roughly levelled using either the aircraft frame as the reference, or using the outputs from gravity switches or the horizontal accelerometers.
b. *Coarse Azimuth Alignment.* Coarse azimuth alignment is achieved by turning the platform in azimuth until the heading output agrees with the aircraft's best known true heading; normally obtained from the GM compass or standby heading reference system.

Coarse alignment levels and aligns the platform within 1° – 2° in a few seconds.

6. **Fine Levelling.** Fine levelling is achieved using the accelerometer null technique; the accelerometer outputs should be zero for a stationary aircraft, but if the platform is tilted there is an output due to gravity (*see* Chap 1, Para 16). Using this output, the platform is moved until zero acceleration is sensed, *ie* $g\phi$ is zero. In practice the accelerometer outputs are zero only when the acceleration due to gravity is balanced by the accelerometer bias. Fine levelling normally takes 1 to 1½ minutes, levelling the platform to within

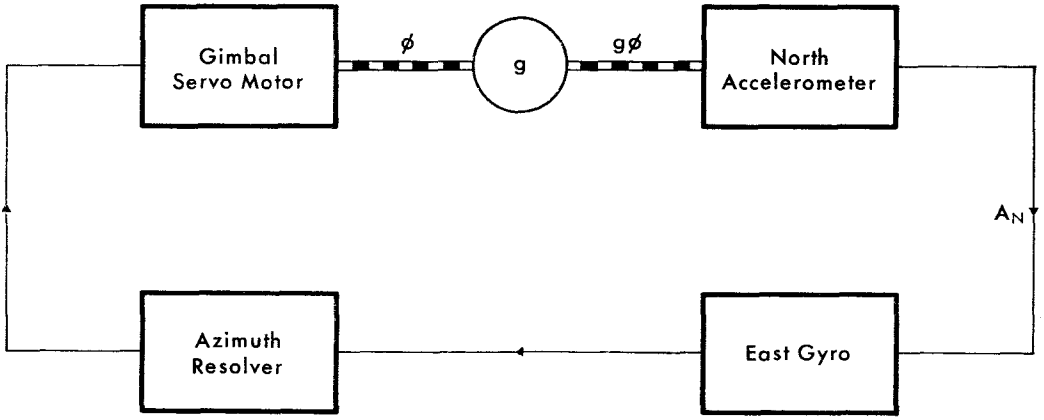


Fig 1 Fine Levelling Loop

approximately 6 seconds of arc. The accuracy achieved depends on the accuracy of the accelerometers used. Fig 1 shows one of the two INS levelling loops.

7. **Gyro-Compassing.** Gyro-compassing or fine azimuth alignment is the final stage of self-alignment.

a. *Effect of Misalignment in Azimuth.* When the platform is misaligned in azimuth, the east gyro detects a component of Earth-rate which causes an apparent wander of the east gyro ($-\Omega \cos \lambda \sin \Psi$) where Ψ is the misalignment angle as shown in Fig 2. This output of apparent wander causes the platform to tilt about East, resulting in the north accelerometer detecting a component of the acceleration due to gravity. The output of the north accelerometer is fed through a high gain amplifier (K2 in Fig 3) and used to torque the azimuth gyro, and hence the platform. When the output from the east gyro is zero, it is assumed that zero Earth-rate is detected; the east gyro, and therefore the platform, is correctly aligned. In addition to feeding the accelerometer output to the azimuth gyro, it must be fed through the levelling loop to return the platform to the level.

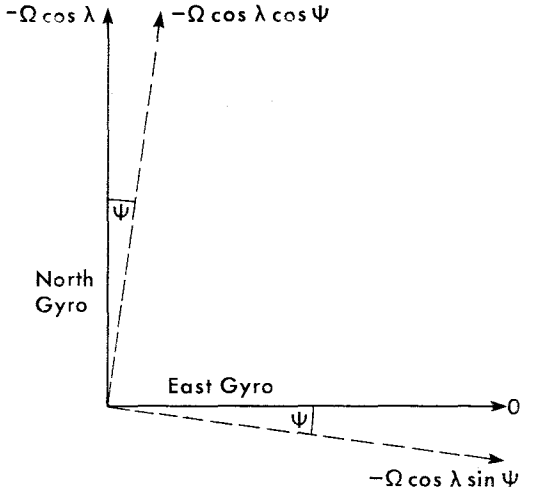


Fig 2 Misalignment in Azimuth

b. *Residual Alignment Error.* Since the output from the east gyro is only zero when the apparent wander due to Earth-rate is balanced by the drift of the east gyro, the accuracy achieved by gyro-compassing depends on the real drift rate of the east gyro.

c. *Gyro-Compassing Limitations.* The Earth-rate sensed by a misaligned east gyro depends on the cosine of the latitude, and therefore gyro-compassing cannot be achieved close to the North or South pole ($\cos 90° = 0$). At mid-latitudes a gyro-compassing accuracy of a few minutes of arc is readily achieved but this accuracy decreases as latitude increases.

8. **Self-Alignment Time.** The time taken in carrying out the full self-alignment sequence

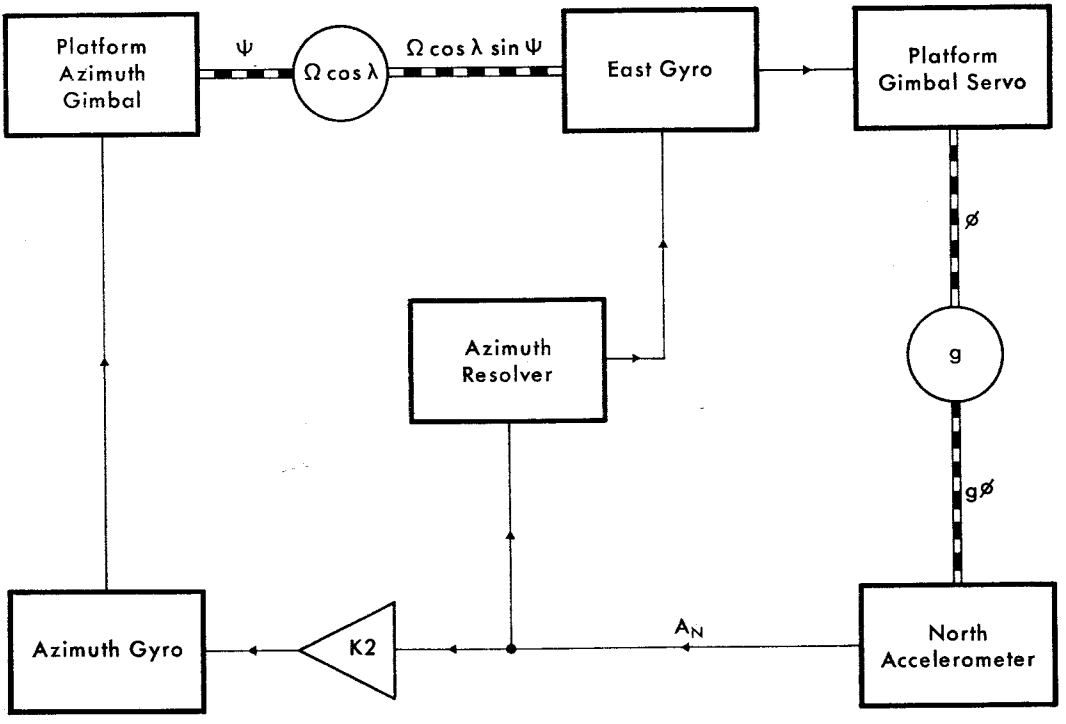


Fig 3 Gyro-Compassing Using a Closed Loop

depends on the accuracy required, the latitude and the ambient temperature. In UK latitudes a typical INS self-aligns in 12 – 17 minutes; between 6 and 9 minutes being required for the gyro-compassing phase.

External Reference Alignment

9. Most INS are self-levelling and external reference methods are usually confined to azimuth alignment. The following external reference alignment methods are used:

a. Transfer gyro method.
b. Synchro memory.
c. HUD alignment.
d. Optical slave system.
e. Runway alignment.

10. **Transfer Gyro Method.** The transfer gyro method permits rapid alignment and is particularly useful for aircraft operating from forward airfields or from ships. Two main components are used in the transfer gyro system:

a. *Datum Gyro.* The datum gyro, which is mounted on a firm protected base, establishes an accurate north datum by gyro-compassing.

b. *Transfer Gyro.* The transfer gyro is usually an azimuth gyro which is located on the base plate by dowels. The transfer gyro rotor is aligned with true North as indicated by the datum gyro. Once aligned, the transfer gyro is carried to the aircraft and used to align the aircraft azimuth gyro; during the transfer the unit is self-powered. Accuracies of about 10′ of arc are obtained using this method.

11. **Synchro Memory Method.** The platform may be gyro-compassed at any convenient time and the azimuth obtained stored in a synchro memory system. Provided the aircraft is not moved, the platform may be realigned subsequently by torquing it to the stored azimuth. This method is accurate within a few minutes of arc of the original bearing.

12. **HUD Alignment.** If the true bearing of a distant object is known, the HUD can be used to measure the relative bearing of the object from the aircraft and the true heading can then be calculated. The INS is then slewed until the heading output agrees with the calculated heading. This method can be extended for use when the true bearing of the object is not known, providing that a gyro-compassing alignment can first be carried out. After gyro-compassing, the HUD is used to measure the relative bearing of the object. As the true heading of the aircraft is known, the true bearing of the object can be calculated. The INS can then be shut down and, when a rapid alignment is required, the procedure outlined above may be used. This method allows the aircraft to be moved between gyro-compassing and subsequent rapid alignment provided that it is returned to within a few feet of its original position and the object is still within the field of view of the HUD. Accuracies of 0·2° can be achieved using HUD alignment.

13. **Optical Slave System.** A suitable system of mirrors set on the stable element can be used to reflect a light source from a datum telescope; when correctly aligned the reflected light falls on a reference within the datum telescope. A misaligned platform may be realigned by driving the platform until the reflected beam returns to the reference. This alignment method requires an accurately surveyed site for mounting the datum telescope, and an unobstructed sight-line between the telescope and the platform mirrors. The optical slave system may be used to align platforms with reference frames other than the local vertical.

14. **Runway Alignment.** Azimuth errors may be corrected during the take-off run if precise runway heading and take-off distance are known; the specific values are stored in a small computer. At lift-off the aircraft is held as close to the runway centre line as possible. The accelerometer outputs are processed in the computer to provide along and across runway distances; these distances are also stored. An across runway distance is assumed to be the result of azimuth gyro misalignment, and the gyro is therefore torqued until the error is removed. An accuracy approaching 0·2° is claimed for this technique.

Airborne Alignment

15. A pure INS cannot self-align in flight, an external reference is required to distinguish movement-induced and misalignment-induced accelerations. Fix monitored INS may be mechanized to perform airborne alignment automatically, normally with computer assistance. The normal method involves the comparison of INS TMG and the TMG obtained from fixing; any across track discrepancy is attributed to azimuth misalignment. Airborne alignment can also be achieved in doppler/inertial mixed systems by comparing the doppler and inertial velocity outputs.

CHAPTER 2

INS ERRORS AND MIXED SYSTEMS

CONTENTS

	Paras
Schuler Tuning	1–3
Errors	
Error Sources	4–5
Initial Levelling Error	6–7
Accelerometer Errors	8
Integrator Errors	9
Levelling Gyro Drift	10
Initial Azimuth Misalignment	11–14
Azimuth Gyro Drift	15
Mixed Inertial Systems	
Introduction	16–17
Velocity Damped Inertial Systems	18–19
Fix Monitored Inertial Systems	20
Stellar Monitored Inertial Systems	21
Summary of Mixed Systems	22

Annex: Derivation of Some INS Errors

Schuler Tuning

1. When a pendulum is accelerated, the bob lags behind the suspension point in the opposite direction to the acceleration (Newton's First Law). When the acceleration stops, the pendulum oscillates with a period (T) equal to:

$$T = 2\pi\sqrt{\frac{l}{g}}$$

where: T is in seconds,
l is the length of the pendulum in feet,
and g is the gravity acceleration in feet/second^2.

2. **The Earth Pendulum.** Imagine a pendulum whose bob lies at the Earth's centre. If the suspension point were accelerated around the Earth, the bob would remain vertically below the suspension point because it is at the Earth's centre of gravity. A platform mounted on the suspension point tangential to the Earth's surface, *ie* horizontal, would therefore remain horizontal irrespective of the acceleration experienced. The vertical defined by the normal to the platform is therefore unaffected by acceleration. If, for any reason, the bob on the Earth pendulum became displaced from the Earth's centre, the pendulum would start to oscillate. The oscillation period would be 84·4 minutes (obtained by substituting the Earth radius in feet for l in the equation above).

3. **The Platform Pendulum.** The INS stable element is maintained normal to the local vertical

by feeding back the aircraft's radial velocity as levelling gyro control signals, and in this way the north and east accelerometers are prevented from detecting components of the gravity acceleration.

The control signals are the $\frac{V}{R}$ and $\frac{U}{R}$ terms for vehicle movement (transport wander) applied as shown in Fig 1. By mechanizing the platform to remain horizontal, an analogue of the Earth pendulum of period 84·4 minutes is produced. Should the platform be displaced from the horizontal it would oscillate with a period of 84·4 minutes. This period is known as the Schuler period after Dr Maximilian Schuler who discovered the properties of the Earth pendulum. A platform is said to be "Schuler Tuned" if its oscillation period is 84·4 minutes.

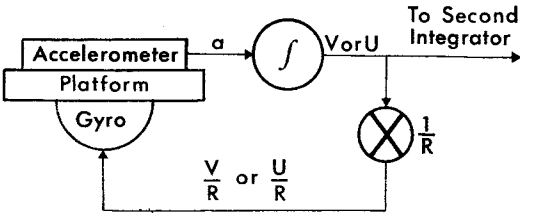


Fig 1 Schuler Tuning

ERRORS

Error Sources

4. The following errors affect inertial navigation systems:

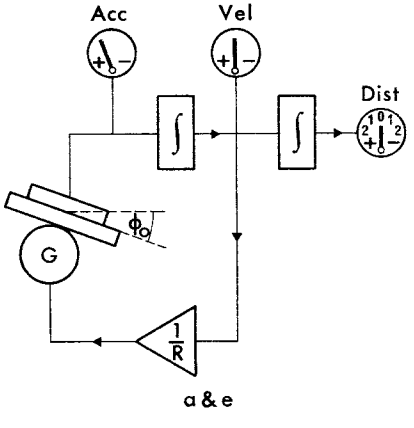
a. Initial levelling error (Initial tilt).
b. Acceleration error (Accelerometer bias).
c. Integrator error (Velocity error).
d. Levelling gyro drift.
e. Initial alignment error (Azimuth misalignment).
f. Azimuth gyro drift.

5. **Bounded Errors.** Errors originating in, or effective within, the Schuler loops, are oscillatory and propagate at the Schuler frequency. These errors, which oscillate about a constant mean and therefore do not grow continuously with time, are termed bounded errors.

Initial Levelling Error

6. **Oscillation.** No matter how carefully the stable element (platform) and its sensors are aligned, there is always some residual error in the vertical, *ie* the platform is not completely level. When the "navigate" mode is selected (at the conclusion of the alignment phase) the following sequence takes place.
(*Note:* The lettering of the sub-paragraphs corresponds with the lettering in Fig 2):

a. The accelerometer detects the component of gravity, $g\phi_0$ (strictly $g \sin\phi_0$, but the approximation is correct for small angles and ϕ_0 expressed in radians). Following the convention that clockwise tilts produce positive



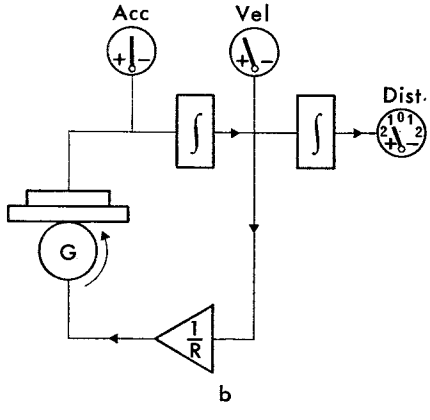


Fig 2 Initial Levelling Misalignment—Oscillation

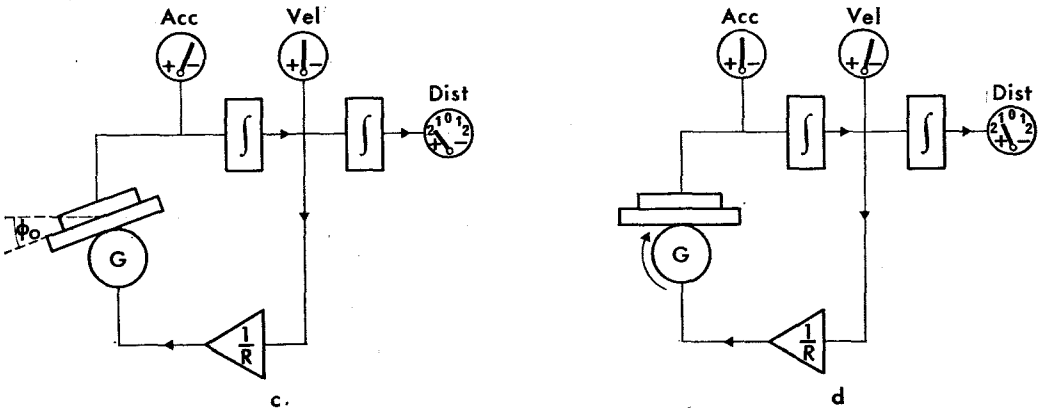


Fig. 2 (cont.) Initial Levelling Misalignment—Oscillation

acceleration, $g\phi_0$ is sensed as a positive acceleration. The integration of the accelerometer output takes a finite time, and therefore, velocity and distance are zero at the instant the "navigate" mode is selected.

b. The integration of the detected acceleration produces a positive velocity which drives the platform anti-clockwise to the horizontal. The accelerometer now detects zero acceleration, but the positive velocity continues to drive the platform.

c. After the platform passes the horizontal the accelerometer detects the opposite gravity effect, sensed as a negative acceleration. The positive velocity reduces to zero at angle ϕ_0. (the original tilt error) and for an instant the platform drive stops. However, the negative acceleration is integrated into negative velocity which drives the platform clockwise.

d. The clockwise drive brings the platform once again to the level position, resulting in zero output from the accelerometer. However, the negative velocity continues to drive the platform clockwise.

e. After the platform passes the horizontal the accelerometer detects the gravity effect, sensed as a positive acceleration. This reduces the negative velocity to zero at angle ϕ_0. The cycle is then repeated.

7. **Initial Tilt Errors.** The errors caused by an initial tilt are shown in Fig 3. *Note:* the errors are bounded and do not increase with time. An initial levelling error of 6 seconds of arc is shown to cause a velocity error bounded by $\pm$ 0·75

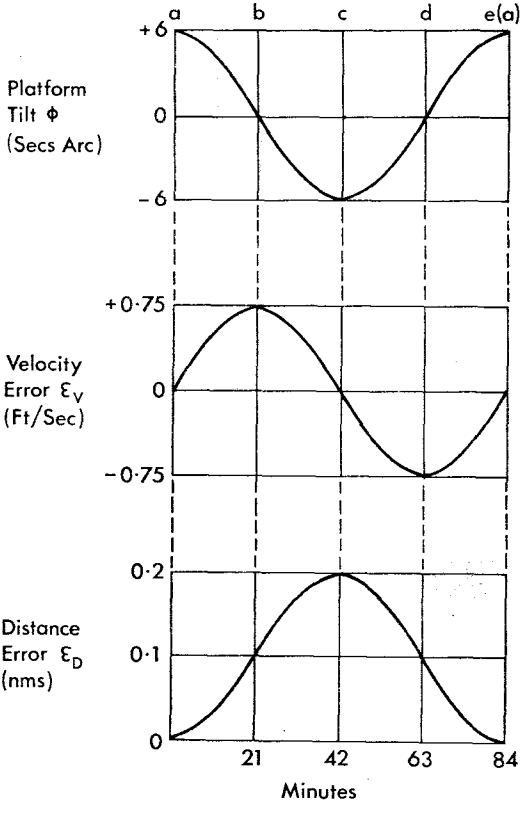


Fig 3 Initial Levelling Misalignment—Errors

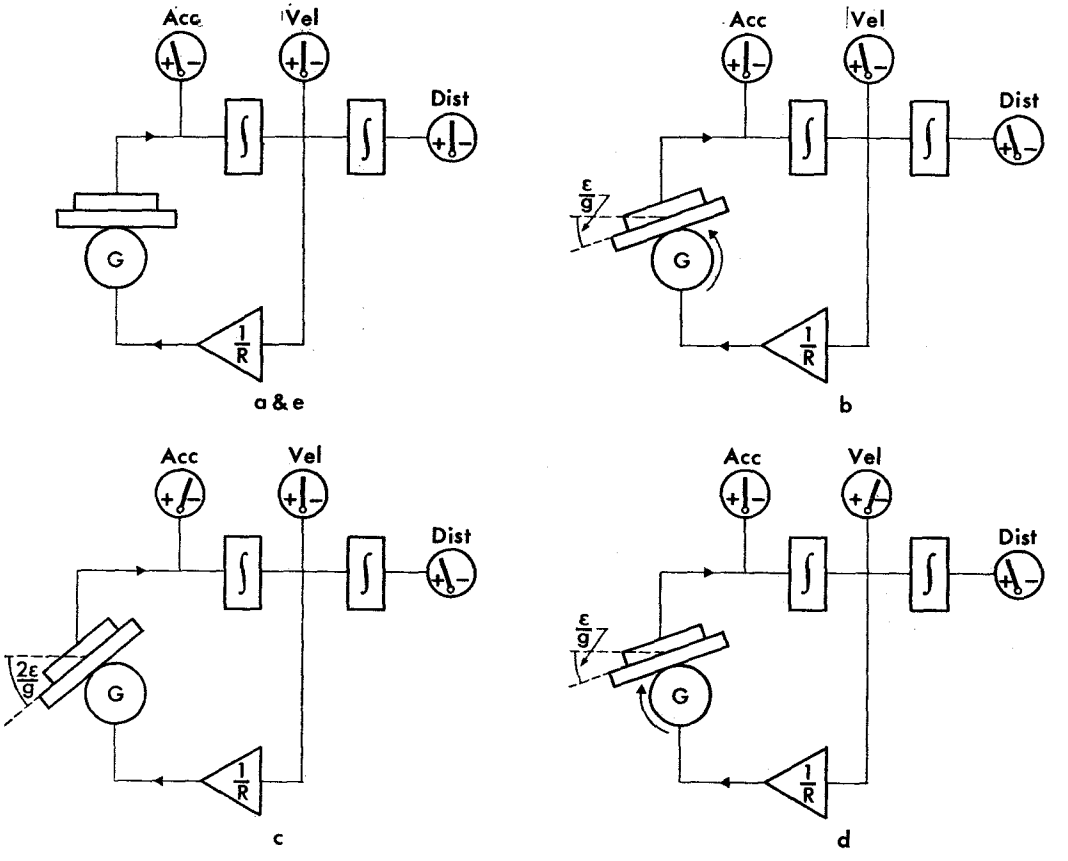


Fig 4 Accelerometer Bias—Oscillation

feet per second (0·45 knots) and a mean distance error of 0·1 nm. After one complete Schuler period both the velocity and distance errors have returned to zero. The a, b, c, d and e positions of the error curve are labelled to correspond to the sub-figure lettering of Fig 2.

Accelerometer Error

8. Some common accelerometer characteristics are described in Chap 1. The errors caused by bias – a term used to describe the condition when the accelerometer gives an output when there is zero input – are explained. After integration they produce the proportional velocity error which rotates the platform at an incorrect rate. As with levelling errors, an oscillation is set up because the velocity error is fed back through the Schuler loop. Fig 4 shows the oscillation caused by an accelerometer bias. A bias error of 0·00003 g (0·001 ft/sec^2) results in the accelerometer producing no output acceleration when the platform is tilted through 6 seconds of arc. The platform therefore oscillates between 0 and 12 arc seconds. Fig 5 shows the error curves for such a bias. There is the same correspondence between Figs 4 and 5 as between Figs 2 and 3.

Integrator Errors

9. The first integrator (or integration process) is within the Schuler loop. Any error in the integrator results in an incorrect velocity output which produces a platform oscillation and associated error curves similar to those previously discussed. A first integrator error of 0·1 ft/sec

drift of 0·01°/hr, the ramp grows at approximately 0·6 nm/hr and has an oscillation of ± 0·13 nm.

Initial Azimuth Misalignment

11. If an INS is properly aligned in azimuth the east gyro senses zero component of Earth-rate and the north gyro outputs a signal proportional to – $\Omega \cos \lambda$. If the INS is misaligned in azimuth by an angle Ψ, the east gyro will output – $\Omega \cos \lambda \sin \Psi$, and the north gyro – $\Omega \cos \lambda \cos \Psi$.

12. The north gyro is torqued for Earth-rate by $\Omega \cos \lambda$ and therefore the torquing error will be:

$$\Omega \cos \lambda - \Omega \cos \lambda \cos \Psi$$

$$= \Omega \cos \lambda (1 - \cos \Psi)°/\text{hr}$$

Since the magnitude of the misalignment angle is unlikely to exceed 0·5°, this error may be disregarded.

eg at Latitude 55° and $\Psi = 0{\cdot}5°$ the error is:

$$15{\cdot}05 \cos 55° (1 - 0{\cdot}99996)°/\text{hr}$$

$$= 0{\cdot}0003°/\text{hr}$$

13. The error for the east gyro, given by–$\Omega \cos \lambda \sin \Psi$°/hr, is appreciable even when Ψ is small. At Latitude 55° and Ψ of 0·1°,the error is 0·015°/hr. This error appears as east levelling gyro drift which causes the platform to oscillate about east and affects the north accelerometer, northern velocity and latitude determination. The error curves produced in the latitude channel by an initial azimuth misalignment are similar to those caused by levelling gyro drift. The unbounded nature of the resulting distance error makes it essential to keep the initial azimuth alignment error as small as possible, and preferably less than 0·1°. The effect of various misalignment angles is shown in Table 1.

14. Azimuth misalignment also results in slightly incorrect accelerations being sensed by the misaligned accelerometers. The resultant errors may become significant under prolonged accelerations, *eg* during long accelerated climbs.

Azimuth Gyro Drift

15. Azimuth gyro drift ($\delta\Psi$), like azimuth misalignment, registers as east levelling gyro drift and produces an increasing azimuth alignment error. The errors produced oscillate about means which increase with time. The increasing mean velocity error produces an unbounded distance error which follows a parabolic growth rate (illustrated in Fig 7).

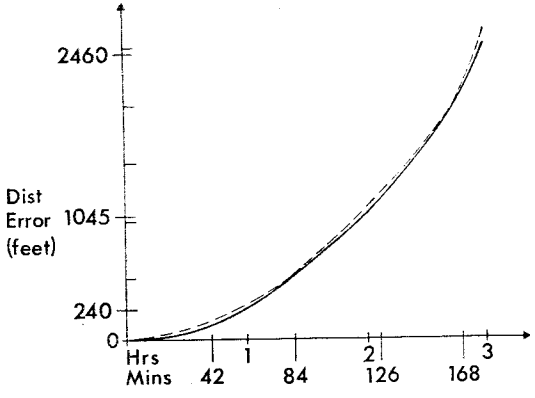


Fig 7 Azimuth Gyro Drift—Distance Error

AZIMUTH ERROR	RESULTANT EAST GYRO DRIFT	DISTANCE ERROR AT 1 HOUR	MAX VELOCITY ERROR
~~0·003~~° 0·03°	0·005°/hr	2,220 ft	1 ft/sec
0·1°	0·015°/hr	6,600 ft	3 ft/sec
0·2°	0·03°/hr	13,310 ft	6 ft/sec
0·3°	0·045°/hr	19,970 ft	9 ft/sec

Table 1 Effect of Initial Azimuth Misalignment (North Channel)—LAT 55°N

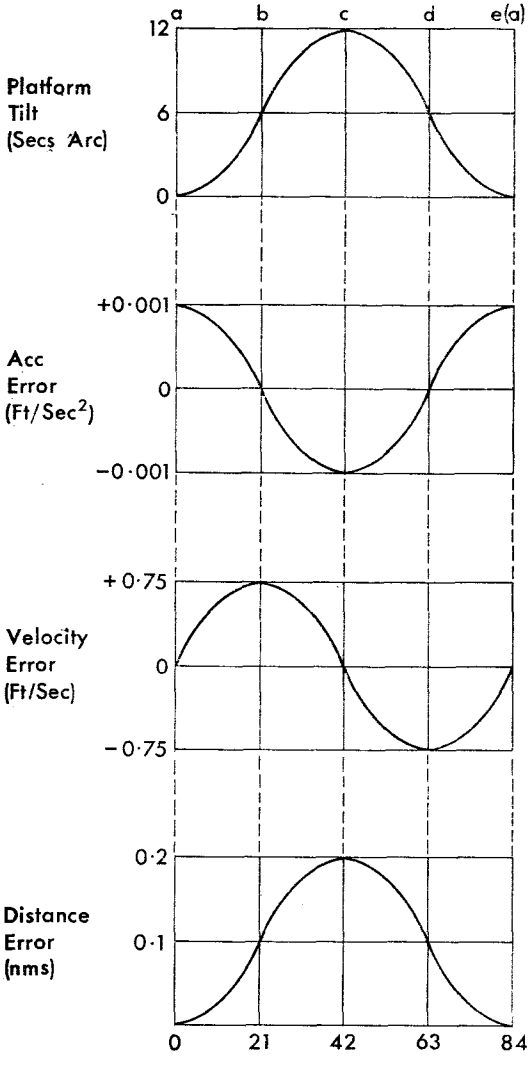


Fig 5 Accelerometer Bias—Errors

gives rise to an oscillation of ± 6 seconds of arc, a mean velocity error of zero (maximum of ± 0·1 ft/sec) and a mean distance of zero with a maximum of ± 80 feet. The second integrator is outside the Schuler loop and any errors caused by it produce a position error that increases linearly with time.

Levelling Gyro Drift

10. Although the desired drift for an IN gyro is of the order of 0·001°/hr it is probable that the drift rate in flight will be greater. A typical figure of 0·01°/hr is used to illustrate the effect of gyro drift on the platform:

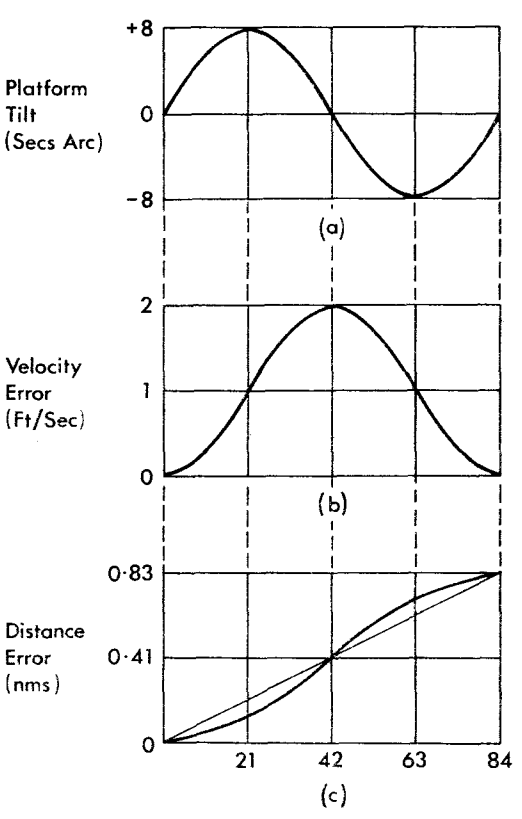


Fig 6 Errors Caused by Gyro Drift

a. *Oscillation.* The stable element is turned away from the horizontal at the rate of 0·01°/hr. The rotation is bounded by the Schuler loop, and the platform tilt curve is shown at Fig 6a.

b. *Velocity Error.* The acceleration error follows the same curve as that shown for platform tilt (Fig 6a). After integration the velocity curve at Fig 6b is obtained, which shows that a mean velocity error develops over the Schuler period.

c. *Distance Error.* The second integration results in the distance error which grows with time because of the mean velocity error. The growth rate is oscillatory about a mean ramp increase (Fig 6c). The distance error due to levelling gyro drift is unbounded, and for a

MIXED INERTIAL SYSTEMS

Introduction

16. In the preceding paragraphs it was shown that some INS errors are bounded and others increase continuously with time. In addition, INS are extremely accurate in the short term, while most other navigation aids have errors which, although relatively large, do not increase with time. The mixed inertial system aims to combine the short term accuracy of a pure INS with the long term accuracy of some other aid thereby enhancing the overall accuracy of both systems.

17. The improvement is achieved by updating the velocity channels or the azimuth channel. Three types of systems are used:

a. Velocity damped systems.
b. Fix monitored systems.
c. Stellar monitored systems.

Velocity Damped Inertial Systems

18. Velocity damping makes the Schuler oscillation decay by comparing the output of the first integrator (V_i) with a reference velocity (V_r) and feeding back the error term ($V_e = V_i - V_r$) to the integrator. There are several possible sources of reference velocity, *eg* velocity between fixes, another INS or, more commonly, a doppler radar.

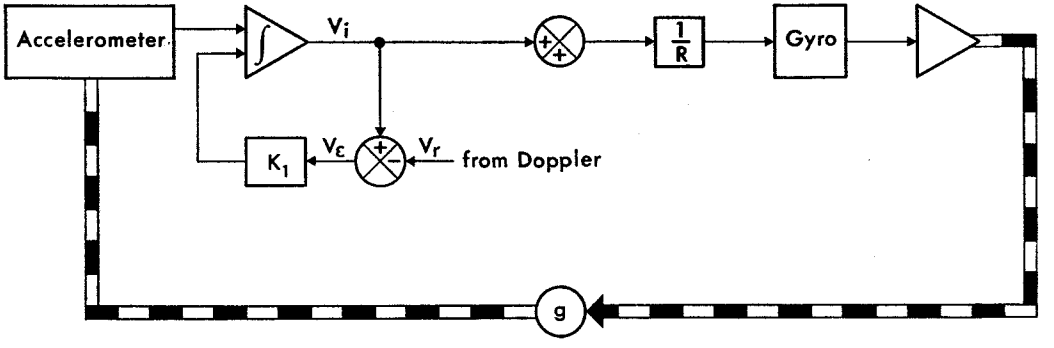


Fig 8 A Simple Doppler Inertial System

19. **Damping by Doppler Velocity.** A simple doppler inertial system is able to damp the oscillations by the method shown in Fig 8.

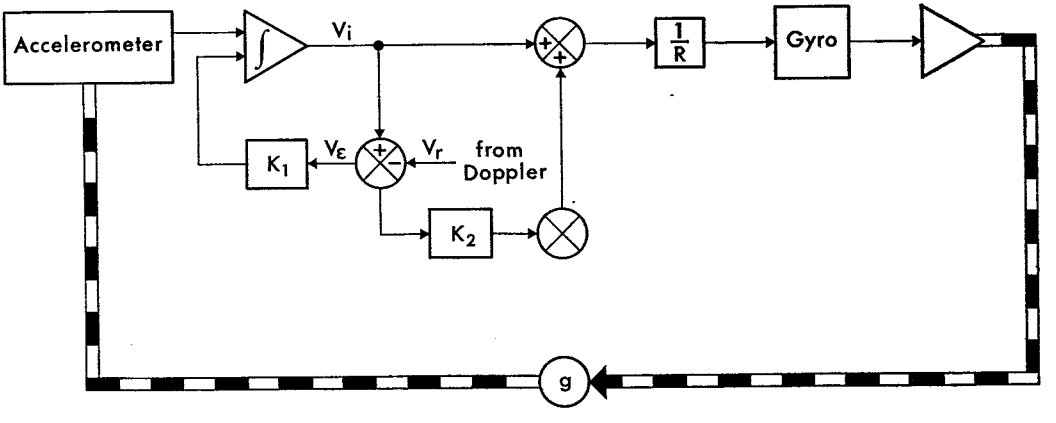


Fig 9 A Tuned Doppler Inertial System

However, this system must take a long time to reduce the error because only the long term doppler velocity changes are of interest. To reduce large errors more quickly, the error signal may be fed forward directly to the gyro through the K_2 loop (*see* Fig 9).

Fix Monitored Inertial Systems

20. Accurate radio fixing is available in areas of high traffic density, but not worldwide. Fix monitored inertial systems make use of the fixes, when available, to enhance system accuracy. The fixes may be used to:

a. Damp the Schuler oscillations (as with the doppler velocity) by providing a comparison of inertial velocity and rate of change of position (*ie* velocity).

b. Correct and drift trim the gyros.

c. Provide azimuth alignment checks by comparing the two tracks.

d. Correct the INS position output.

Stellar Monitored Inertial Systems

21. The accuracy of any astro-navigation system depends on the accuracy of the vertical datum. A pure INS provides an extremely accurate vertical either directly, or remotely, for automatic astro systems. An astro tracker may be used to monitor platform azimuth, or its output mixed with the INS output to provide:

a. Monitoring of platform tilt and hence gyro drift rates.

b. Fix monitoring using two stars.

Fig 10 compares the accuracy of a system with and without stellar monitoring.

Summary of Mixed Systems

22. Table 2 summarizes the properties of the various forms of IN mixing.

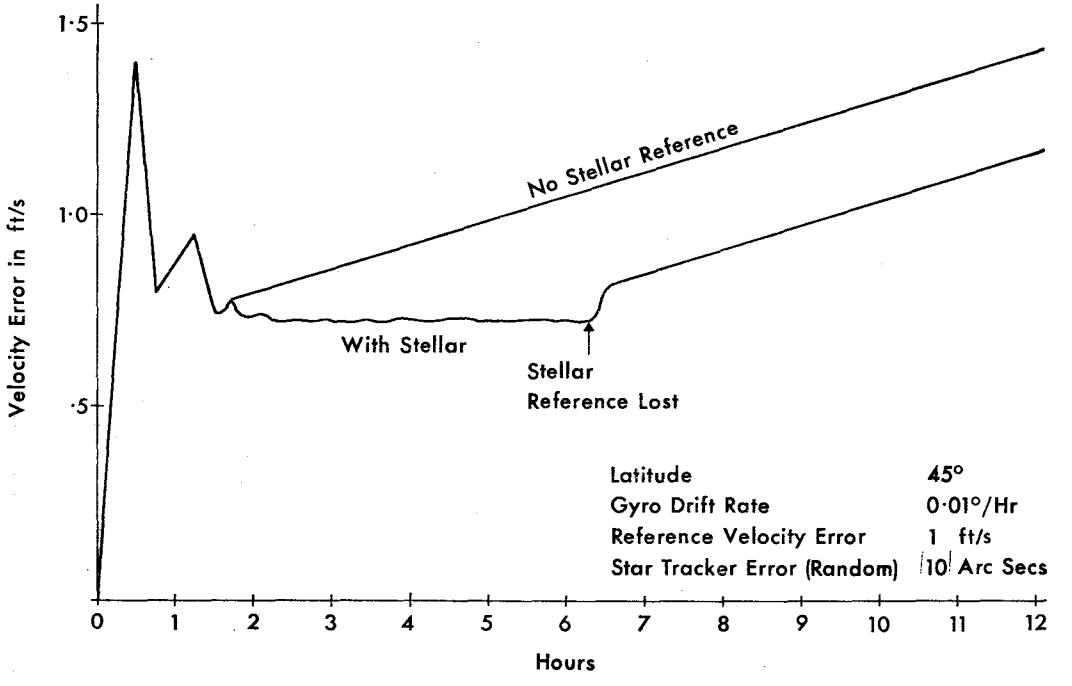


Fig 10 Accuracy Comparison with/without Stellar Monitoring

	POSITION ERROR	AIRBORNE GYRO TRIMMING	SELF-CONTAINED
PURE IN	Unbounded	No	Yes
DOPPLER/IN	Unbounded but reduced	Yes	Yes
LR FIX/IN	Bounded	Yes	No
STELLER FIX/IN	Bounded	Yes	Yes

Table 2 Properties of Mixed Systems

DERIVATION OF SOME INS ERRORS

CONTENTS

	Paras
Introduction	1
Initial Levelling Errors	2–6
Accelerometer Errors	7–11
Errors Due to Gyro Drift	12–16
Errors After One Hour	17–23
Errors Caused by Incorrect Position	24–25

LIST OF SYMBOLS USED

Symbols	26

Introduction

1. In this annex, some INS error equations are examined to deduce the probable magnitude of the errors. In most of the error equations time is measured in seconds and angles in radians; simple integration is shown in full, but complicated calculations are not derived. Unless stated otherwise, the figures referred to, are those of the main chapter.

Initial Levelling Errors

2. The effect of initial levelling error is to cause the accelerometer to be tilted out of the horizontal thereby giving a false output of acceleration. Assuming a platform tilt of $\Phi°$, the accelerometer detects an acceleration proportional to $g\,\Phi$, which gives an oscillatory error since it occurs within the Schuler loop (*see* Figs 2 and 3). The platform tilt oscillation is a cosine function of the Schuler frequency, ω_S, and is a maximum at time t = 0.

3. **Acceleration Error.** The platform tilt at time t is given by $\Phi \cos \omega_S t$, and therefore the acceleration error, E_A, at time t is:

$$E_A = g\,\Phi \cos \omega_S t$$

4. **Velocity Error.** The velocity error (E_V) is the time integral of E_A, *ie*

$$E_V = \int E_A .\, dt$$

$$= g\,\Phi \int_0^t \cos \omega_S t . dt$$

$$E_V = \frac{g\,\Phi}{\omega_S} \sin \omega_S t$$

5. **Distance Error.** The distance error is the time integral of E_V, *ie*

$$E_D = \int E_V.dt$$

$$= \frac{g\Phi}{\omega_s}\int_0^t \sin \omega_s t.\, dt$$

$$E_D = \frac{g\Phi}{\omega_s^2}(1 - \cos \omega_s t)$$

6. **Magnitude of Errors.** The magnitude of the errors may be determined from the preceding equations. Taking ω_s as $12{\cdot}4 \times 10^{-4}$ rads/sec and assuming an initial tilt (Φ) of 0·000029 rads:

a. *Acceleration Error.* The maximum acceleration error occurs when cos $\omega_s t$ is a maximum, which is at time t = 0, when cos $\omega_s t = 1$;

$$\text{Max } E_A = g\Phi \cos \omega_s t$$

$$= 32{\cdot}2 \times 0{\cdot}000029$$

$$= 0{\cdot}0009 \text{ ft/sec}^2$$

b. *Velocity Error.* The maximum velocity occurs when sin $\omega_s t$ is a maximum, which occurs at time t = 21·1 mins (position b in Fig 3):

$$\text{Max } E_V = \frac{g\Phi}{\omega_s} \sin \omega_s t$$

$$= \frac{32{\cdot}2 \times 29 \times 10^{-6}}{12{\cdot}4 \times 10^{-4}}$$

$$= 0{\cdot}75 \text{ ft/sec, or } 0{\cdot}45 \text{ knots}$$

c. *Distance Error.* The velocity error decays after time t = 21·1 mins, but the distance error continues to increase to time t = 42·2 mins. The quantity (1–cos$\omega_s t$) will be a maximum when cos $\omega_s t$ is a minimum, *ie* when cos $\omega_s t$ equals – 1, which occurs at time t = 42·2 mins (position c of Fig 3):

$$\text{Max } E_D = \frac{g\Phi}{\omega_s^2}(1 - \cos \omega_s t)$$

$$= \frac{32{\cdot}2 \times 29 \times 10^{-6} \times 2}{12{\cdot}4 \times 12{\cdot}4 \times 10^{-8}}$$

$$E_D = 1214 \text{ ft, or } 0{\cdot}2 \text{ nm}$$

Accelerometer Errors

7. A bias error (E_b) of 0·00003 g is used for illustration. The accelerometer gives an output proportional to E_b under zero acceleration. The error lies within the Schuler loop, is oscillatory, oscillating as the cosine of the Schuler frequency and is a maximum at time t = 0 (*see* Figs 4 and 5).

8. **Acceleration Error.** The acceleration error at time t = 0 is given by E_b, and at time t = t by:

$$E_A = E_b \cos \omega_s t$$

9. **Velocity Error.** The velocity error (E_V) is the time integral of E_A:

$$E_V = \int E_A.\, dt$$

$$= E_b \int_0^t \cos \omega_s t.dt$$

$$E_V = \frac{E_b}{\omega_s} .\, \sin \omega_s t$$

10. **Distance Error.** The distance error (E_D) is the time integral of E_V:

$$E_D = \int E_V.dt$$

$$= \frac{E_b}{\omega_s}\int_0^t \sin \omega_s t.dt$$

$$E_D = \frac{E_b}{\omega_s^2}(1 - \cos \omega_s t)$$

11. **Magnitude of Errors.** The maximum error produced by a bias error of 0·00003 g can be deduced from the preceding equations:

a. *Acceleration Error.* The maximum acceleration error occurs at time t = 0:

$$\text{Max } E_A = E_b$$

$$= 0{\cdot}00003\ g, \text{ or } 0{\cdot}001 \text{ ft/sec}^2$$

b. *Velocity Error.* The maximum velocity error occurs when the quantity sin $\omega_S t$ is a maximum, which occurs at time t = 21·1 mins, when sin $\omega_S t = 1$.

$$\therefore \text{Max } E_V = \frac{E_b}{\omega_S} \sin \omega_S t$$

$$= \frac{1 \times 10^{-3}}{12{\cdot}4 \times 10^{-4}}$$

$$= 0{\cdot}8 \text{ ft/sec, or } 0{\cdot}47 \text{ kts}$$

c. *Distance Error.* The maximum distance error occurs when cos $\omega_S t$ is a minimum, which occurs at time t = 42·2 mins, when cos $\omega_S t = -1$:

$$\therefore \text{Max } E_D = \frac{E_b}{\omega_S^2} (1 - \cos \omega_S t)$$

$$= \frac{1 \times 10^{-3} \times 2}{12{\cdot}4 \times 12{\cdot}4 \times 10^{-8}}$$

$$= 1300 \text{ ft, or } 0{\cdot}21 \text{ nm}$$

Errors Due to Gyro Drift

12. Gyro drift causes serious errors in INS. Although gyros can be bench tested to accuracies of 0·001°/hr there are indications from flight tests that the drift rate degrades in operational use.

13. **Levelling Gyro Drift Errors.** It may be assumed that the platform will be exactly level (*ie* $\Phi = 0$) at time t = 0 until the levelling gyro drift rate causes it to tilt.

a. *Platform Tilt.* Because the levelling gyro drift error is within the Schuler loop the platform tilt angle is bounded by the action of the Schuler loop (*see* Fig 6). Therefore, the maximum platform tilt is $\frac{\delta_1}{\omega_S}$, where δ_1 is in radians per second, and the tilt at time t is given by:

$$\Phi = \frac{\delta_1}{\omega_S} \sin \omega_S t$$

b. *Acceleration Error.* The accelerometer detects the acceleration error (E_A) which is proportional to $\frac{g\delta_1}{\omega_S}$:

$$\therefore E_A = \frac{g\delta_1}{\omega_S} \sin \omega_S t$$

c. *Velocity Error.* The velocity error (E_V) is the time integral of E_A:

$$\therefore E_V = \int E_A .dt$$

$$= \frac{g\delta_1}{\omega_S} \int_0^t \sin \omega_S t.dt$$

$$= \frac{g\delta_1}{\omega_S^2} (1 - \cos \omega_S t)$$

d. *Distance Error.* The distance error (E_D) will be the time integral of E_V:

$$E_D = \int E_V .dt$$

$$= \frac{g\delta_1}{\omega_S} \int_0^t 1 - \cos \omega_S t.dt$$

$$= \frac{g\delta_1}{\omega_S^2} \left(t - \frac{\sin \omega_S t}{\omega_S} \right)$$

14. **Magnitude of Errors.** From the error equation in sub-para 13d and Fig 6 it can be seen that the distance error is unbounded, the magnitude of the error being time dependent. The maximum errors for tilt, acceleration and velocity are calculated using a drift rate of 0·01°/hr (4·848 × 10^{-8} rads/sec).

a. *Platform Tilt.* The maximum platform tilt occurs at time t = 21·1 mins, when sin $\omega_S t$ = 1:

$$\Phi_{max} = \frac{\delta_1}{\omega_S}$$

$$= \frac{4{\cdot}848 \times 10^{-8} \times 57{\cdot}3 \times 3600}{12{\cdot}4 \times 10^{-4}}$$

$$= 8 \text{ arc secs}$$

b. *Acceleration Error.* The maximum acceleration error occurs at time t = 21·1 mins, when $\sin \omega_S t = 1$:

$$E_{A\,max} = \frac{g\delta_1}{\omega_S} \sin\omega_S t$$

$$= \frac{32{\cdot}2 \times 4{\cdot}848 \times 10^{-8}}{12{\cdot}4 \times 10^{-4}}$$

$$= {\cdot}00126 \text{ ft/sec}^2,$$

c. *Velocity Error.* The maximum velocity error will occur at time t = 42·2 mins, when $\cos \omega_S t = -1$.

$$E_{V\,max} = \frac{g\delta_1}{\omega_S{}^2} (1 - \cos \omega_S t)$$

$$= \frac{32{\cdot}2 \times 4{\cdot}848 \times 10^{-8}}{12{\cdot}4 \times 12{\cdot}4 \times 10^{-8}} \times 2$$

$$= 2{\cdot}032 \text{ ft /sec or } 1{\cdot}2 \text{ kt}$$

15. **Azimuth Misalignment Errors.** Chap 2, paras 11–14 show that for all practical misalignment angles (Ψ) only the north channel is affected. The error appears as a drift of the east levelling gyro, the magnitude being given by:

$$\Omega \cos\lambda \sin \Psi°/\text{hr}$$

Therefore, the errors can be determined using the equations for levelling gyro drift, but substituting $\Omega \cos\lambda \sin\Psi$ (rads/sec) for δ_1:

a. *Platform Tilt.* The platform tilt at time t is given by:

$$\Phi = \frac{\Omega \cos\lambda \sin\Psi}{\omega_S} \sin \omega_S t$$

b. *Acceleration Error.* The acceleration error at time t is given by:

$$E_A = \frac{g\,\Omega \cos\lambda \sin\Psi}{\omega_S} \sin \omega_S t$$

c. *Velocity Error.* The velocity error at time t is given by:

$$E_V = \frac{g\,\Omega \cos\lambda \sin\Psi}{\omega_S{}^2} (1 - \cos \omega_S t)$$

d. *Distance Error.* The distance error at time t is given by:

$$E_D = \frac{g\,\Omega \cos\lambda \sin\Psi}{\omega_S{}^2} \left(t - \frac{\sin \omega_S t}{\omega_S} \right)$$

16. **Azimuth Gyro Drift Error.** Chap 2, para 15 shows that azimuth gyro drift rate ($\delta\Psi$) only affects the north channel, except when the drift rate is large. The platform tilt angles at time t can be shown to be:

$$\Phi = \frac{\delta\Psi\,\Omega \cos\lambda}{\omega_S{}^2} (1 - \cos \omega_S t)$$

The remaining errors are derived below:

a. *Acceleration Error.* The acceleration error depends on the tilt error and is given by:

$$E_A = \frac{g\delta\Psi\,\Omega \cos\lambda}{\omega_S{}^2} (1 - \cos \omega_S t)$$

b. *Velocity Error.* The velocity error is the time integral of the acceleration error:

$$E_V = \int E_A .dt$$

$$\text{Let } K = \frac{g\delta\Psi\,\Omega \cos\lambda}{\omega_S{}^2}$$

$$\text{then } E_V = K \int_0^t 1 - \cos\omega_S t.dt$$

$$= K\left(t - \frac{\sin \omega_S t}{\omega_S}\right)$$

$$E_V = \frac{g\delta\Psi\Omega\cos\lambda}{\omega_S^2}\left(t - \frac{\sin\omega_S t}{\omega_S^2}\right)$$

c. *Distance Error.* The distance error is the time integral of E_V:

$$E_D = \int E_V.dt$$

$$= K\int_0^t t - \frac{\sin\omega_S t}{\omega_S}.dt$$

$$= K\left(\frac{t^2}{2} - \frac{(1-\cos\omega_S t)}{\omega_S^2}\right)$$

$$= \frac{g\delta\Psi\Omega\cos\lambda}{\omega_S^2}\left(\frac{t^2}{2} - \frac{(1-\cos\omega_S t)}{\omega_S^2}\right)$$

Errors After One Hour

17. The distance errors due to initial levelling error (tilt), accelerometer error (bias), levelling gyro drift, azimuth misalignment and azimuth gyro drift ($E_{D\Phi}$, E_{Db}, $E_{D\delta_1}$, $E_{D\Psi}$, and $E_{\delta\Psi}$ respectively) are calculated at 1 hour (t = 3,600 seconds) to enable comparisons to be made.

18. **Tilt Error.** The tilt error could appear in both INS channels and therefore the equations apply equally for both north and east errors:

$$E_{D\Phi} = \frac{g\Phi}{\omega_S^2}(1 - \cos\omega_S t)$$

$$= \frac{32{\cdot}2 \times 29 \times 10^{-6}}{12{\cdot}4 \times 12{\cdot}4 \times 10^{-8}}(1 + \cos 75°\,47'\,14'')$$

$$= 757 \text{ ft}$$

19. **Accelerometer Error.** Accelerometer bias could be present in both accelerometers. The distance error could therefore appear in both north and east channels:

$$E_{Db} = \frac{E_b}{\omega_S^2}(1 - \cos\omega_S t)$$

$$= \frac{1 \times 10^{-3}}{12{\cdot}4 \times 12{\cdot}4 \times 10^{-8}}(1 + \cos 75°\,47'\,14'')$$

$$= 810 \text{ ft}$$

20. **Levelling Gyro Drift Error.** Levelling gyro drift error is likely to be present in both channels:

$$E_{D\delta_1} = \frac{g\delta_1}{\omega_S^2}\left(t - \frac{\sin\omega_S t}{\omega_S}\right)$$

$$= \frac{32{\cdot}2 \times 4{\cdot}848 \times 10^{-8}}{12{\cdot}4 \times 12{\cdot}4 \times 10^{-8}}\left(3600 + \frac{\sin 75°\,47'\,14''}{12{\cdot}4 \times 10^{-4}}\right)$$

$$= 4449 \text{ ft}$$

21. **Azimuth Misalignment Error.** Using an azimuth misalignment error of 0·1°, at a latitude of 55°, the east channel error can be ignored. The error in the north channel is:

$$E_{DN\Psi} = \frac{g\,\Omega\cos\lambda\sin\Psi}{\omega_S^2}\left(t - \frac{\sin\omega_S t}{\omega_S}\right)$$

$$= \frac{32{\cdot}2 \times 15{\cdot}04\cos 55°\sin 0{\cdot}1°}{12{\cdot}4 \times 12{\cdot}4 \times 10^{-8} \times 57{\cdot}3 \times 3600} \times \left(3600 + \frac{\sin 75°\,47'\,14''}{12{\cdot}4 \times 10^{-4}}\right)$$

$$= 6698 \text{ ft}$$

22. **Azimuth Gyro Drift Error.** The error in distance caused by drift of the azimuth gyro appears in the north channel. However, if $\delta\Psi$ is large, or a long time period is considered, a significant error also appears in the east channel. In the following example a drift rate of 0·01°/hr is used:

$$E_{DN\delta\Psi} = \frac{g\,\delta\Psi\Omega\cos\lambda}{\omega_S^2}\left(\frac{t^2}{2} - \frac{1 - \cos\omega_S t}{\omega_S^2}\right)$$

$$= \frac{32{\cdot}2 \times 4{\cdot}848 \times 10^{-8} \times 15{\cdot}04 \cos 55°}{12{\cdot}4 \times 12{\cdot}4 \times 10^{-8} \times 57{\cdot}3 \times 3600} \times \left(\frac{3600^2}{2} - \frac{1 + \cos 75° 47' 14''}{12{\cdot}4 \times 12{\cdot}4 \times 10^{-8}} \right)$$

$$= 240 \text{ ft}$$

23. **Error Totals.** The total channel error is the root sum square of the individual errors. However, it should be noted that:

a. No attempt has been made to feed in the incorrect positions resulting from any of the distance errors.

b. No statistical limits have been shown for the errors. It is assumed that the errors have the same level of probability.

Therefore, the north channel error is:

$$E_{DN} = \sqrt{\left(E^2_{D\Phi} + E^2_{Db} + E^2_{D\delta_1} + E^2_{D\Psi} + E^2_{D\delta\Psi}\right)}$$

$$= \sqrt{(757^2 + 810^2 + 4449^2 + 6698^2 + 240^2)}$$

$$= 8120 \text{ ft}$$

The east channel error is:

$$E_{DE} = \sqrt{\left(E^2_{D\Phi} + E^2_{Db} + E^2_{D\delta_1}\right)}$$

$$= \sqrt{(757^2 + 810^2 + 4449^2)}$$

$$= 4585 \text{ ft}$$

The total error is:

$$E_{DT} = \sqrt{\left(E^2_{DN} + E^2_{DE}\right)}$$

$$= \sqrt{(8120^2 + 4585^2)}$$

$$= 9325 \text{ ft, or } 1{\cdot}53 \text{ nm}$$

Errors Caused by Incorrect Position

24. It is possible that an incorrect geographic position could be set into the INS because of an incorrect fix, or alternatively a map error. In either case, the INS computer would compute for the false position, producing incorrect terms throughout the system. The gyros and accelerometers are corrected for Earth rate and vehicle movement, the appropriate correction terms involving latitude and north and east velocity. For simplification, only the gyro correction terms listed below are considered:

a. East gyro : $-\frac{V}{R}$

b. North gyro : $\Omega \cos\lambda + \frac{U}{R}$

c. Azimuth gyro : $\Omega \sin\lambda + \frac{U}{R} \tan\lambda$

25. **Possible Case of Incorrect Position.** The graph in Fig 1 below was calculated using error equations based on the following:

True position: 50°N 00° E/W

False position: 49° 55′N 00° E/W

North groundspeed: 0 kt

East groundspeed: 600 kt

The graph was calculated at five minute increments.

LIST OF SYMBOLS USED

26. **Symbols.** The following symbols are used in inertial navigation:

Acceleration Error	E_A
Velocity Error	E_V
Distance Error	E_D
Tilt Error	E_Φ
Accelerometer Bias	E_b
Azimuth Misalignment	Ψ
Levelling Gyro Drift	δ_1

Azimuth Gyro Drift	$\delta\Psi$	Time	T
Schuler Frequency	ω_S	Gravity	g
Earth Radius	R	Groundspeed	
Earth Rate	Ω	North	V
Latitude	λ	East	U

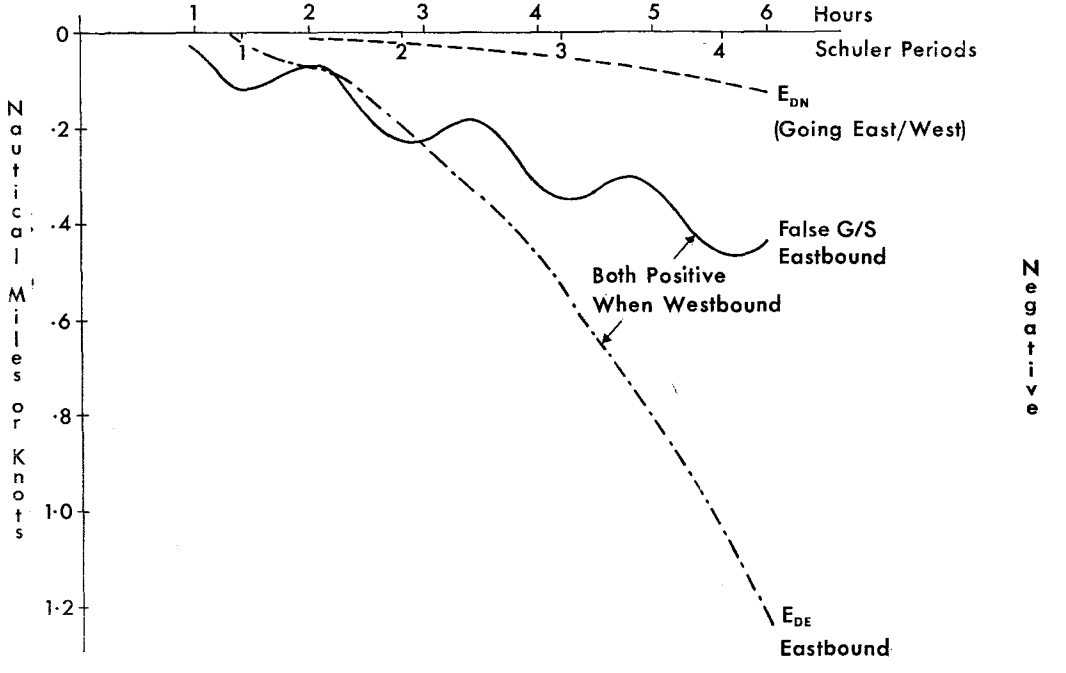


Fig 1 Plot of Errors Due to Incorrect Position

CHAPTER 3

THE FERRANTI INERTIAL PLATFORM

CONTENTS

	Paras
Introduction	1–3
THE PLATFORM AND COMPONENTS	
General	4
The Gimbal Assembly	5–7
The Gyroscopes	8
The Accelerometers	9
Platform Outputs	10–12
Platform Inputs	13
Platform Temperature Control	14–15
THE PRESENT POSITION COMPUTER	
Computing Principles	16–20
THE NAVIGATION CONTROLLER	
General	21
Controls	22
ALIGNMENT, CALIBRATION AND REVERSIONARY FUNCTIONS	
Introduction	23–25
Alignment Methods	26
Normal Alignment	27–31
Rapid Alignment	32–33
Calibration Alignments	34–35
Reversionary Conditions	36–39

Annex: Position Computing

Introduction

1. The Ferranti inertial platform and its associated units which together form the FE 541 system are illustrated at Fig 1. This chapter is concerned with the inertial platform only, and emphasis is therefore placed on the functions of the inertial platform, the present position computer (PPC), the navigation controller (NC), and the power supply unit. However, the platform is installed as part of an inertial navigation attack system, and cannot be considered in isolation. Reference will be made to other components of the system, particularly the navigation computer (which incorporates a navigation display and displays some of the parameters set at the navigation controller), the weapon aiming computer (WAC—which operates in conjunction with the PPC to perform some computations), and the head-up display (HUD) which is not a part of the FE 541 system.

2. The platform is a conventional four-gimballed system, the instrument cluster being contained within the inner or azimuth gimbal. It has three accelerometers: two horizontal and one vertical. The vertical accelerometer forms part of a height channel, the output of vertical velocity being fed to the weapon aiming computer.

3. The PPC is an analogue device, and is semi-automatic in operation. The PPC and the platform are controlled by the navigation controller. The platform, the PPC and the NC form part of both the FE 541 and FE 570 navigation and attack systems.

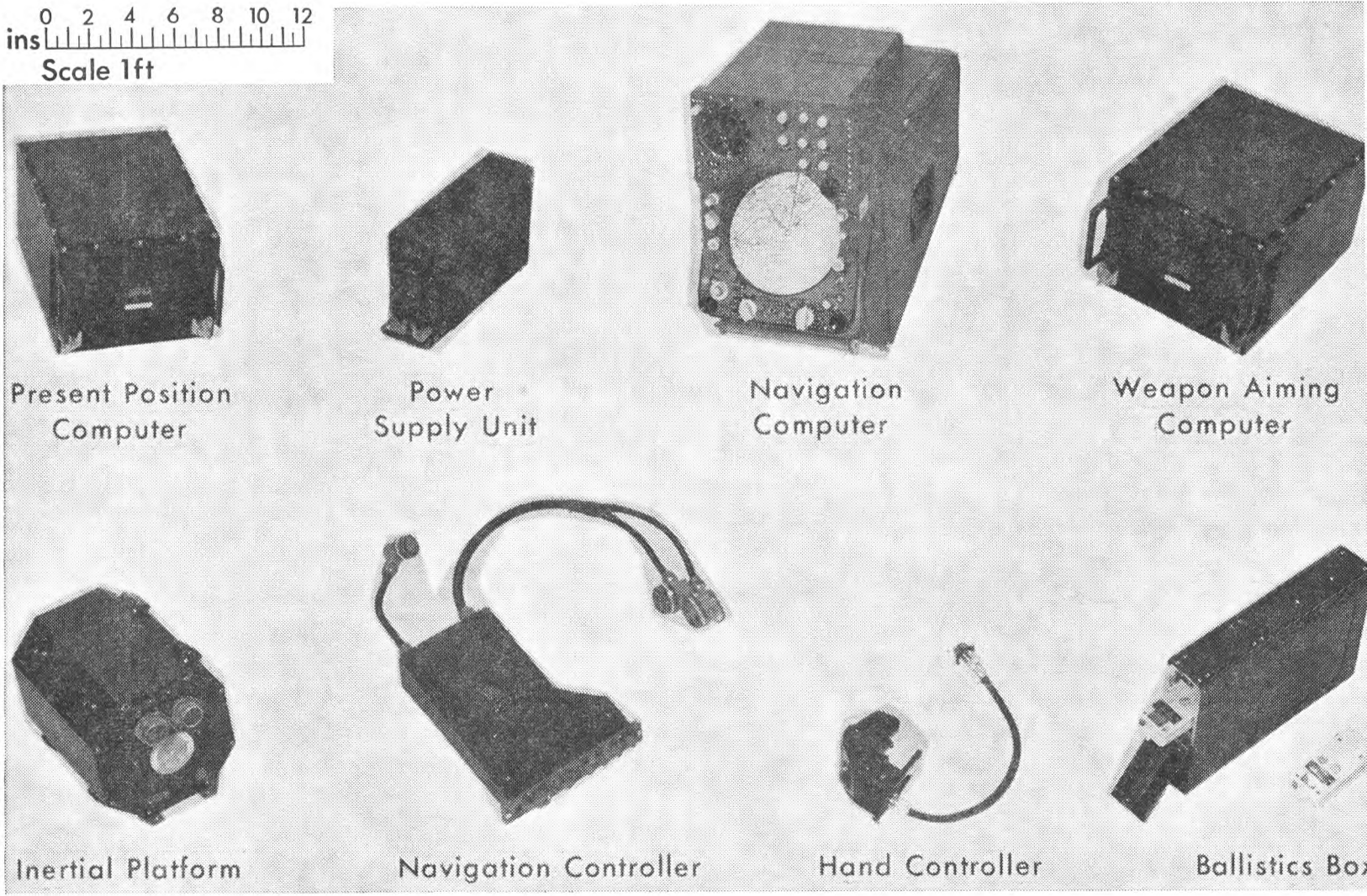


Fig 1 The Inertial Platform and Associated Units of the FE 541 System

THE PLATFORM AND COMPONENTS

General

4. The inertial platform is fully aerobatic and has four gimbals. The central cluster contains the three accelerometers, the three gyros and their associated circuits and wiring.

The Gimbal Assembly

5. The gimbal arrangement is, from the inside outwards, azimuth, inner roll, pitch and outer roll. The angular freedoms of the gimbals are:

a. Azimuth, pitch and outer roll gimbals are unlimited.

b. Inner roll gimbal is limited to $\pm$ 12°.

6. **The Gimbal Drives.** The three inner gimbals are driven by direct drive AC two-phase induction motors. Each motor, together with the slip rings and brushes, is an integral part of the gimbal bearing assembly. The outer roll gimbal is driven by two geared motors. The two motors are needed to cope with the high inertia of the system and the rapid reversal of direction needed to prevent gimbal lock (*see* Chap 1, paras 26 to 28).

7. **The Gimbal Pick-Offs.** On the gimbal bearing, at the opposite end to the drive motors, are the gimbal synchros which detect and transmit the position of the gimbals. The outer roll gimbal has several synchros, resolvers, and potentiometers. The output of the detector on the inner roll gimbal is used by the computer to prevent gimbal lock.

The Gyroscopes

8. The gyros used to stabilize the platform are of the single axis, rate integrating, floated type. The North and East gyros are mounted with their output axes vertical to avoid gravity sensitive drifts. The azimuth gyro has to be mounted with its input axis vertical, its drift rates therefore being larger. The gyro characteristics are:

a. *Drift Rates.*

(1) The long-term drift rates are:
Input axis vertical (IAV)—0·04° per hr (1σ).
Output axis vertical (OAV)
—0·004° per hr (1σ).

(2) The short-term drift rates are:
IAV — 0·01° per hr per g (1σ).
OAV—0·001° per hr (1σ).

b. *Size and Weight.* Each gyro is a cylinder 2·8″ long and 2·1″ in diameter, weighing one pound.

c. *Gyro Motor.* The spin motor is a hysteresis synchronous motor which runs up with a power input of 26V 350Hz, and is maintained at normal running speed by 20V 350Hz $\pm$35mHz.

d. *Pick-offs and Torquers.* The pick-off is an AC differential transformer, its output being amplified. The torquer is a permanent magnet DC motor which can achieve a maximum continuous torquing rate of 2,000° per hour; higher rates can be achieved intermittently. When used in the system the torquing rate is limited to about 200° per hr.

The Accelerometers

9. The accelerometers used are of the single axis, viscous damped, DC torquer restrained, pendulum type. The pick-off is an AC differential transformer. The accelerometer has the following characteristics:

Forcer current scale factor	0·155 mA per ft per sec^2.
Measurement range	$\pm 20g$.
Threshold	less than $2 \times 10^{-7}g$.

Platform Outputs

10. The principal outputs from the inertial platform are those concerned with aircraft accelerations and attitude.

11. **Acceleration Outputs.** The outputs of the three accelerometers are passed to the present position computer (PPC). All three outputs are amplified in the PPC, the height acceleration being passed to the weapon aiming computer (WAC) before integration.

12. **Attitude Information.** The platform supplies information on the aircraft attitude about all three axes:

a. *Heading.* A synchro on the azimuth gimbal supplies the aircraft heading to the PPC, and WAC, the head-up display (HUD or LCOS) and the navigation computer. It also drives the heading repeater shaft in the power supply unit which resolves the outputs from the North and East gyros for gimbal control.

b. *Bank Angle.* Four synchros, mounted on the outer roll gimbal, supply the bank angle to various parts of the weapons system.

c. *Pitch Angle.* A synchro on the pitch gimbal supplies the aircraft pitch angle to the WAC.

Platform Inputs

13. The principal inputs to the platform are those for platform stabilization, accelerometer feedback currents, gyro corrections and power supplies.

Platform Temperature Control

14. As the gyros are of the floated type the fluid supporting the rotor assembly has to be closely controlled in temperature to maintain the same viscosity and density. Operating the gyros at a constant temperature means that corrections for drift rates have a good repeatability.

15. Heat is distributed internally around the platform by a circulating mixture of helium and nitrogen. A controlled supply of cooling air is blown around the outer canister. Rapid heaters, capable of raising the platform temperature by 17 °C per minute, are used to attain the platform operating temperature of + 70 °C. When that temperature is reached the rapid heaters switch off automatically, thereafter the temperature being maintained by fine heaters.

THE PRESENT POSITION COMPUTER

Computing Principles

16. The double integration functions, the corrections and control signals described in Chap 1 (Principles of Inertial Navigation) are all derived from the PPC. To ensure extreme accuracy in computation many of the basic equations are transformed to suit the analogue devices used in the computer; the method of computing position is described in the Annex.

17. **Latitude Computation.** The present latitude (λ) of the aircraft may be expressed as latitude increments ($\delta\lambda$) from a defined mid-latitude (λ_M), thus:

$$\lambda = \lambda_M + \delta\lambda.$$

Providing the increments $\delta\lambda$ are small any trigonometrical function such as sin $\delta\lambda$ and cos $\delta\lambda$ may be accurately expressed as a power series of two or three terms. The mid-latitudes used are: –90°, –60°, –30°, 0°, +30°, +60° and +90°, therefore $\delta\lambda$ never exceeds 15°. Thus:

$$\cos\lambda = \cos(\lambda_M + \delta\lambda)$$
$$= \cos\lambda_M \cos\delta\lambda - \sin\lambda_M \sin\delta\lambda.$$

This is expressed as a power series expansion for analogue computing.

18. **Height Corrections.** Height information is derived from an air data computer via the WAC. It is summed with the radius (R) of the Earth in the PPC to correct velocity for altitude.

19. **Radius and Ellipticity Corrections.** Corrections for the Earth's ellipticity as a function of latitude and for the Earth's radius as a function of height are calculated in the PPC.

20. **The Wind Velocity Comparator.** A wind velocity (meteorological) and a tolerance figure are set at the navigation controller (NC). The PPC compares the inertial /ADC dead reckoning vectors with the meteorological wind vectors and indicates on the navigation computer when the dead reckoning value is outside the tolerance limits.

THE NAVIGATION CONTROLLER

General

21. The navigation controller (NC) shown at Fig 2 provides the controlling signals for the various alignment modes and switching signals for reversionary operation.

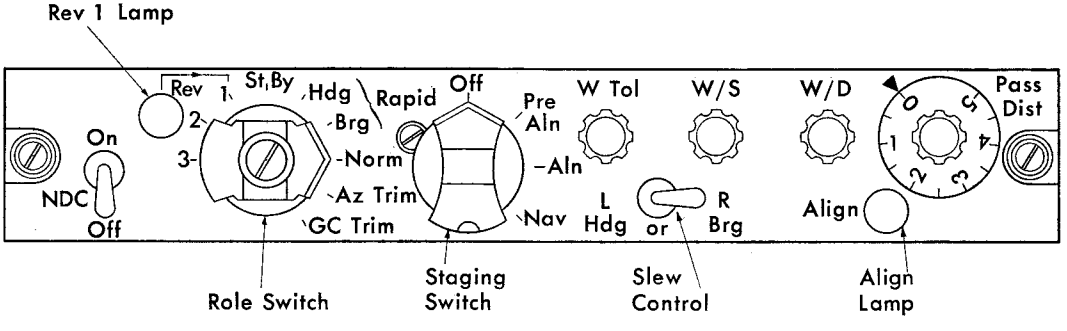


Fig 2 Controls on the Navigation Computer

Controls

22. The following controls are on the navigation controller:

a. *NDC Switch.* This switches the associated navigation computer on or off.

b. *Role Switch.* This is a nine-position switch which makes the following selections:

(1) *Norm.* This selects normal alignment mode.

(2) *Az Trim.* Azimuth gyro drift is measured and corrected.

(3) *GC Trim.* The East gyro is trimmed in place of the North gyro.

(4) *Hdg.* This selection permits the platform to be slewed so that it is aligned to aircraft heading.

(5) *Brg.* This selection permits the platform to be slewed such that when the bearing marker (HUD/LCOS) is on a known bearing the platform is aligned to aircraft heading.

(6) *Rev* 1, 2 *and* 3. These are selections for reversionary modes (*see* paras 37–39).

(7) *St By.* This selection applies aircraft power to the standby heaters.

c. *Staging Switch.* This is a four-position switch for making the manual selections required during platform alignment.

d. *Slew Control.* This control is used in conjunction with the *Hdg* or *Brg* selection on the role switch to slew the platform to the required alignment.

e. *W/D, W/S and W Tol.* These controls are used in conjunction with the navigation computer to set the meteorological wind velocity and a tolerance value into the PPC (*see* para 20).

f. *Pass Dist.* This control sets the target pass distance into the WAC.

ALIGNMENT, CALIBRATION AND REVERSIONARY FUNCTIONS

Introduction

23. In Chap 1 it was demonstrated that the platform must be accurately aligned so that the accelerometers point along the reference axes. As the Ferranti platform is an Earth referenced system these axes are North, East and the local vertical.

24. The gyros that hold the platform to the reference axes must be carefully calibrated so that their drift rates are known and can be corrected. The random drift of the gyros may be considered in two parts: one part which is repeatable over several runs and one which is peculiar to the current sortie, *ie* long and short-term drift rates. All three gyros must be calibrated and corrected.

25. So that all facilities from the platform and computer are not lost due to a partial unserviceability there are reversionary functions available. These functions may be selected manually or, in one case, automatically.

Alignment Methods

26. The method used to align the platform is largely determined by the time available and the accuracy required; greater accuracy is achieved through a normal alignment, which takes about 11 minutes, than through a rapid alignment, which takes $2\frac{1}{2}$ to 3 minutes (including engine start). The normal alignment should give accuracies to better than 2 nm per hour (CEP) after alignments at UK latitudes. The navigation accuracy after rapid alignments is worse than that after normal alignment by a factor of about two.

Normal Alignment

27. The manual selections made on the navigation controller (*see* Fig 2) and the automatic sequence for normal alignment are described in the following paragraphs.

28. **Initial Manual Selections.** The following selections are made on the NC:

a. NDC switch to *On* (this switches on the navigation computer).

b. Role switch to *NORM.*

c. Staging switch to *Pre Aln.*

29. The following automatic sequences now take place:

a. *Step Oa.* The heaters raise the platform temperature and the built-in test (BIT) sequence operates. A fault lamp on the navigation computer will light; when it goes out (after about ten seconds) the system goes automatically to step Ob.

b. *Step Ob.* The heaters continue to heat the platform and the present position is set into the navigation computer by the operator.

When the *Align* lamp flashes at a constant rate indicating that the platform temperature is above 35 °C, the system is ready to be manually switched to *Aln*.

30. The staging switch is set to *Aln* and the following automatic sequences now take place:

a. *Step* 1*a*. The platform is slewed to the North indicated by the aircraft compass system. When the platform temperature reaches 70 °C the heaters switch off. The platform is erected approximately to the vertical by the North and East accelerometer outputs driving the gimbal motors directly.

b. *Step* 1*b*. The gyros are run up to about quarter speed on 26V AC 350 Hz, at which speed the next step becomes operative. Erection by the accelerometers continues.

c. *Step* 2*b*. The gyros continue to run up until they reach full speed (after about 45 seconds). The platform is now fully stabilized. At the beginning of this step the levelling loops are operating at 1/60 of the normal Schuler period (about 1·4 minutes). As the gyros run up the gain of the loops is reduced until it is about 1/30 the Schuler period when the gyros are at full speed.

d. *Step* 2*c*. The gyro drive voltage is reduced to 20V AC 350 Hz to maintain a constant speed. The azimuth gyro, fed with a torquing input to compensate for earth rate, takes over from the aircraft compass system.

e. *Steps* 3 *and* 4. The gain of the loops is reduced to 1/15 the Schuler period. The outputs from the accelerometers continue to be used to level the platform, and gyro-compassing starts.

f. *Step* 5. The drift rate of the north gyro is measured and corrected. When the align lamp gives a steady light *Nav* is selected at the staging switch. It should be noted that the align lamp may give a steady light during step 4.

31. The staging switch is set to *Nav* and the equipment is ready for flight. A summary of the normal alignment sequence is as follows:

a. Heating and BIT—Steps Oa and Ob.

b. Coarse azimuth alignment and vertical erection—Steps 1a and 1b.

c. Vertical erection—Steps 2b and 2c.

d. Gyro-compassing and fine levelling—Steps 3 and 4.

e. North gyro drift measurement and correction, gyro-compassing and fine levelling—Step 5.

Rapid Alignment

32. The platform is rapidly aligned by cutting short the normal alignment sequence when the platform is erected to the local vertical. The platform is then manually slewed until it is aligned in azimuth in relation either to a known heading or bearing, or a stored heading or bearing (*see* Chap 1, Annex A, para 10).

33. The normal alignment procedure is followed until 1½ minutes after the staging switch has been set to *Aln* (this must be timed by the operator). At this stage the role switch is set to *Hdg* or *Brg* as appropriate and the platform is slewed manually to the required alignment using the slew control. The function of the HUD/LCOS and the navigation computer in this operation are described in the relevant Aircrew Manual. After 2 minutes at the *Aln* selection the staging switch is set to *Nav*.

Calibration Alignments

34. **Calibration Alignment by Azimuth Trim.** The normal alignment and switching sequence is followed as far as the end of step 5, except that the Role switch is placed to *Az Trim* instead of *Norm*. The platform is then switched automatically to Step 6 for 48 minutes during which the azimuth gyro is allowed to drift. Any heading error results in the east gyro detecting an Earth rate component. This causes the platform to tilt and produce an output from the North channel first integrator, which is proportional to the angle through which the platform has drifted in azimuth. Because the platform drifts for a fixed time the drift rate can be computed, and it is stored on a potentiometer. After 48 minutes in Step 6, the system switches to Step 7 where a gyro-compass sequence is completed, and simultaneously the stored drift rate is applied to the azimuth gyro in such a manner that it cancels out the gyro drift. *Nav* may be selected at the end of Step 7.

35. **Calibration Alignment By Gyro-Compassing Trim.** So that the East gyro may be corrected, a gyro-compassing sequence can be set up by selecting the role switch to *GC Trim*. The platform is driven round through 90° so that the East gyro becomes the North gyro and vice versa. The correction terms are also switched. A normal alignment automatically proceeds, the East gyro

being drift trimmed at Step 5 instead of the north gyro. The system must be switched off and allowed to run down, then re-aligned using a normal sequence, before *Nav* can be selected.

Reversionary Conditions

36. **General.** There are three reversionary conditions which can be selected by the operator, one of which is also an automatic reversion in the event of a failure in the platform, the PPC, the HUD/LCOS or the navigation computers.They are known as *Rev* 1, *Rev* 2 and *Rev* 3.

37. **Reversionary Condition 1** (*Rev* 1). A failure in the platform or the PPC is indicated by an apparent malfunction of the navigation computer. Thus the operator would not know the source of the fault. Switching to *Rev* 1 may be manual or automatic:

a. *Manual Switching to Rev* 1. When *Rev* 1 is selected the navigation computer uses air data, the set wind velocity and compass heading to compute present position. If the navigation computer is still unusable *Rev* 2 should be selected.

b. *Automatic Switching to Rev* 1. An excessive output from the gyros, a failure of the 15kHz supply, an excessive output from the height channel capture amplifier or a reduction in the gyro and accelerometer pre-amplifiers' HT current will light the *Rev* 1 lamp in the NC. From the beginning of Step 2c onwards during alignment and at any time in *Nav* these faults will initiate *Rev* 1, switching off the power supply to the IN sub-system and the WAC.

38. **Reversionary Condition 2** (*Rev* 2). If the NDC fails *Rev* 2 should be selected. In this mode a steering signal and the distance to go to base are shown in the HUD.

39. **Reversionary Condition 3** (*Rev 3*). When a comparison of the set wind velocity and the computed wind velocity with the inertial velocities shows a platform failure *Rev* 3 should be selected. The platform is slaved to the compass and becomes, in effect, a velocity damped DR computer.

POSITION COMPUTING

CONTENTS

	Paras
Reference Frame	1
Rotational Accelerations	2–3
Measured Accelerations	4
Rotational Accelerations Derived	5
Total Accelerations	6–8
PPC COMPUTING	
Introduction	9–10
Gravity and Acceleration Corrections	11
First Integrator Outputs	12
Velocity Corrections	13
Gyro Correction Terms	14
Height	15

Reference Frame

1. The Ferranti platform is stabilized to a local vertical and local North reference frame, and is therefore rotated about all three axes to compensate for Earth rate and transport wander. Thus in addition to aircraft accelerations, the accelerometers also sense the accelerations caused by platform rotation.

Rotational Accelerations

2. It can be shown that these rotational accelerations are equivalent to the expressions for coriolis and central acceleration derived in Chap 1 (paras 32-39).

3. Let: ω_Z = Platform rotation about local vertical

$$= \frac{U}{R} \tan \lambda + \Omega \sin \lambda.$$

ω_Y = Platform rotation about local North

$$= \frac{U}{R} + \Omega \cos \lambda.$$

ω_X = Platform rotation about local East

$$= -\frac{V}{R}.$$

V_S = Spatial velocity in the local North direction

$= V$.

U_S = Spatial velocity in the local East direction

$= U + \Omega R \cos \lambda$, where $\Omega R \cos \lambda$ is the circumferential velocity of the Earth at latitude λ.

Measured Accelerations

4. The accelerations A_Y and A_X sensed in the local North and East directions respectively are the sum of the spatial accelerations ($\dot{V}_S$ and $\dot{U}_S$) and the rotational accelerations coupled with them. These rotational accelerations are now derived.

Rotational Accelerations Derived

5. It can be seen in Fig 1 that because of rotation ω_Z the velocity vectors U_S and V_S will each, in a small time δt, be turned through a small angle $\omega_Z \delta t$. Thus after time δt a component δV will appear normal to the vector U_S, and similarly, a small component δU will appear normal to the vector V_S. Since δt is small the angle $\omega_Z \delta t$ will be small and thus:

$$\delta V = \omega_Z \delta t U_S, \text{ and}$$
$$\delta U = \omega_Z \delta t V_S.$$

Therefore change in velocity during δt in directions N and E

$$= +\omega_Z \delta t U_S \text{ and } -\omega_Z \delta t V_S.$$

Therefore acceleration North

$$= \frac{\omega_Z \delta t U_S}{\delta t} = \omega_Z U_S,$$

and acceleration East

$$= \frac{\omega_Z \delta t V_S}{\delta t} = -\omega_Z V_S.$$

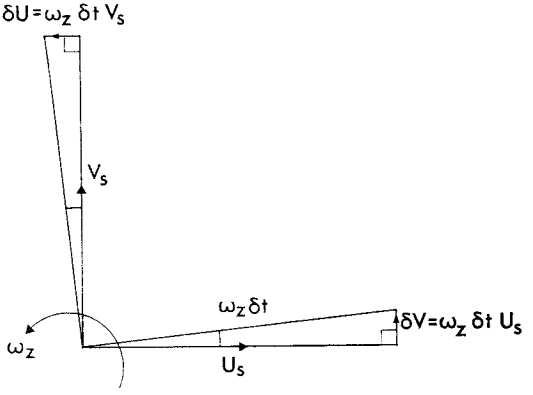


Fig 1 Generation of δU and δV

Total Accelerations

6. Substituting in para 4 for the rotational accelerations, we have:

$A_Y = \dot{V}_S + \omega_z U_s$ .....(1)

$A_X = \dot{U}_S - \omega_z V_s$ .....(2)

7. Substituting in (1) the expressions in para 3, we have:

$$A_Y = \dot{V} + \left(\frac{U}{R}\tan\lambda + \Omega\sin\lambda\right)x \left(U + \Omega R \cos\lambda\right)$$

$$= \dot{V} + \frac{U^2}{R}\tan\lambda + 2\Omega U \sin\lambda + \Omega^2 R x \sin\lambda\cos\lambda,$$ whence:

$\dot{V}$ = aircraft acceleration North.

$\frac{U^2}{R}\tan\lambda$ = central acceleration due to aircraft movement.

$2\Omega U \sin\lambda$ = coriolis acceleration.

$\Omega^2 R \sin\lambda\cos\lambda$ = northerly component of centrifugal force due to Earth rotation.

This latter term is automatically compensated for in the north accelerometer by erecting to the local vertical. Thus when subtracting rotational accelerations from accelerometer outputs, this term must be subtracted from the corrections.

8. Substituting in (2),

$A_x = \dot{U}_s - \omega_z V_s$

$= (U + \Omega R \cos\lambda) - \omega_z V_s$

$$= \dot{U} - \Omega R \sin\lambda.\dot{\lambda} - \left(\frac{U}{R}\tan\lambda + \Omega\sin\lambda\right)V$$

$$= \dot{U} - \Omega V \sin\lambda - \frac{UV}{R}\tan\lambda - \Omega V \sin\lambda$$

$$= \dot{U} - 2\Omega V \sin\lambda - \frac{UV}{R}\tan\lambda,$$

where:

$\dot{U}$ = aircraft acceleration East.

$2\Omega V \sin\lambda$ = Coriolis acceleration.

$\frac{UV}{R}\tan\lambda$ = central accleration due to aircraft motion.

PPC COMPUTING

Introduction

9. For the PPC to give outputs of position and velocity in Earth co-ordinates, unwanted accelerations must be removed either as accelerations, or as velocities or displacements after the appropriate integrations have been performed. The PPC in the Ferranti system removes the errors progressively.

10. Fig 2 is a representation of the computing method used in the PPC. It should be noted that only the output of position in latitude and longitude is derived. Outputs of velocity are also available for display by the navigation computer but are omitted for simplicity.

Gravity and Acceleration Corrections

11. The output of the North accelerometer is fed with a gravity correction to compensate for the later correction (*see* para 7) which is redundant as the platform is erected to the local vertical. The terms $\omega_Z U_S$ and $-\omega_Z V_S$ are subtracted to give an input to the first integrator of $\dot{V}_S$ and $\dot{U}_S$.

First Integrator Outputs

12. The outputs from the first integrators are the spatial velocities V_S and U_S along the North and East local axes.

Velocity Corrections

13. The spatial velocities are corrected to provide local velocities. Since $V_S = V$ no correction is in fact required; however, U_S is converted to

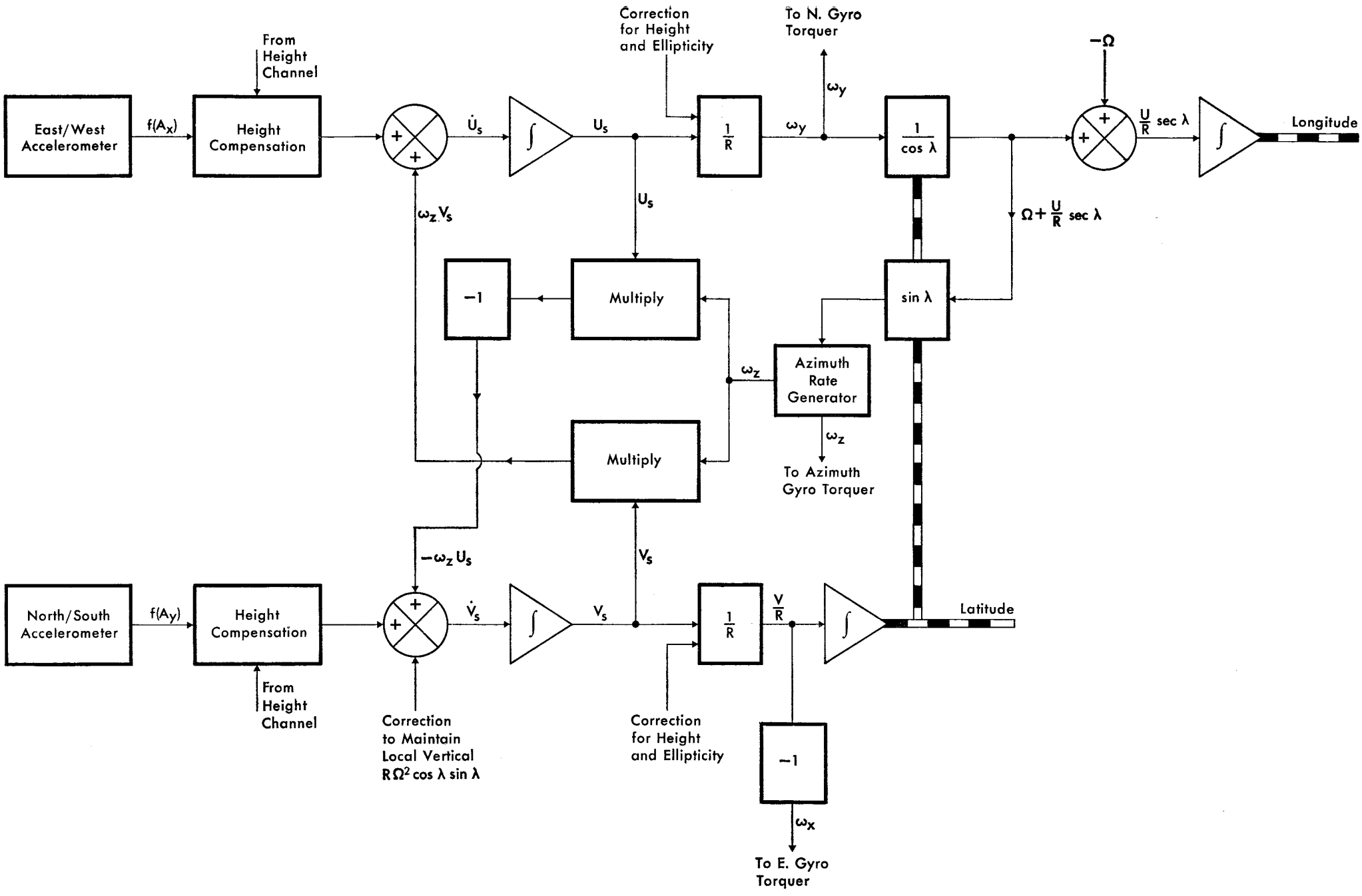


Fig 2 PPC Computing Chain

U by removal of the Earth rate component (Ω R cos λ) before the second integrator. The output of the second integrator is aircraft displacement N/S and E/W which is displayed as latitude and longitude by the navigation computer.

Gyro Correction Terms

14. The gyro correction terms ω_X, ω_Y, ω_Z are derived and fed to the gyros after compensation for Earth radius and ellipticity factors.

Height

15. The height channel is similar in principle to the horizontal channels.

Printed for Her Majesty's Stationery Office by Wilfred Edmunds Ltd., Station Road, Chesterfield.
445/802375/3,750/12-70/468/2

CHAPTER 4

THE ELLIOTT E3 HEADING REFERENCE SYSTEM

CONTENTS

	Paras
Introduction	1
System Components	2
COMPONENT DESCRIPTION	
E3 Stable Platform	3 – 7
IN Control Panel	8 – 9
Ground Speed Resolver	10
Platform Computer No 1	11
Platform Computer No 2	12
IN Power Supply Unit	13
ALIGNMENT	
Alignment Phases	14
Erection and Gyro-Compassing	15 – 17
Fine Alignment	18
Runway Alignment	19 – 26
Survey Alignment	27
Bearing Alignment	28 – 29
OPERATION	
Platform Stabilization	30 – 31
Heading	32
Velocities	33 – 34
Roll and Pitch	35
Annex: Runway Alignment	

Introduction

1. The primary purpose of the E3 inertial reference system is to provide high quality true heading information. Additionally, the system provides northerly and easterly velocities, and pitch and roll information.

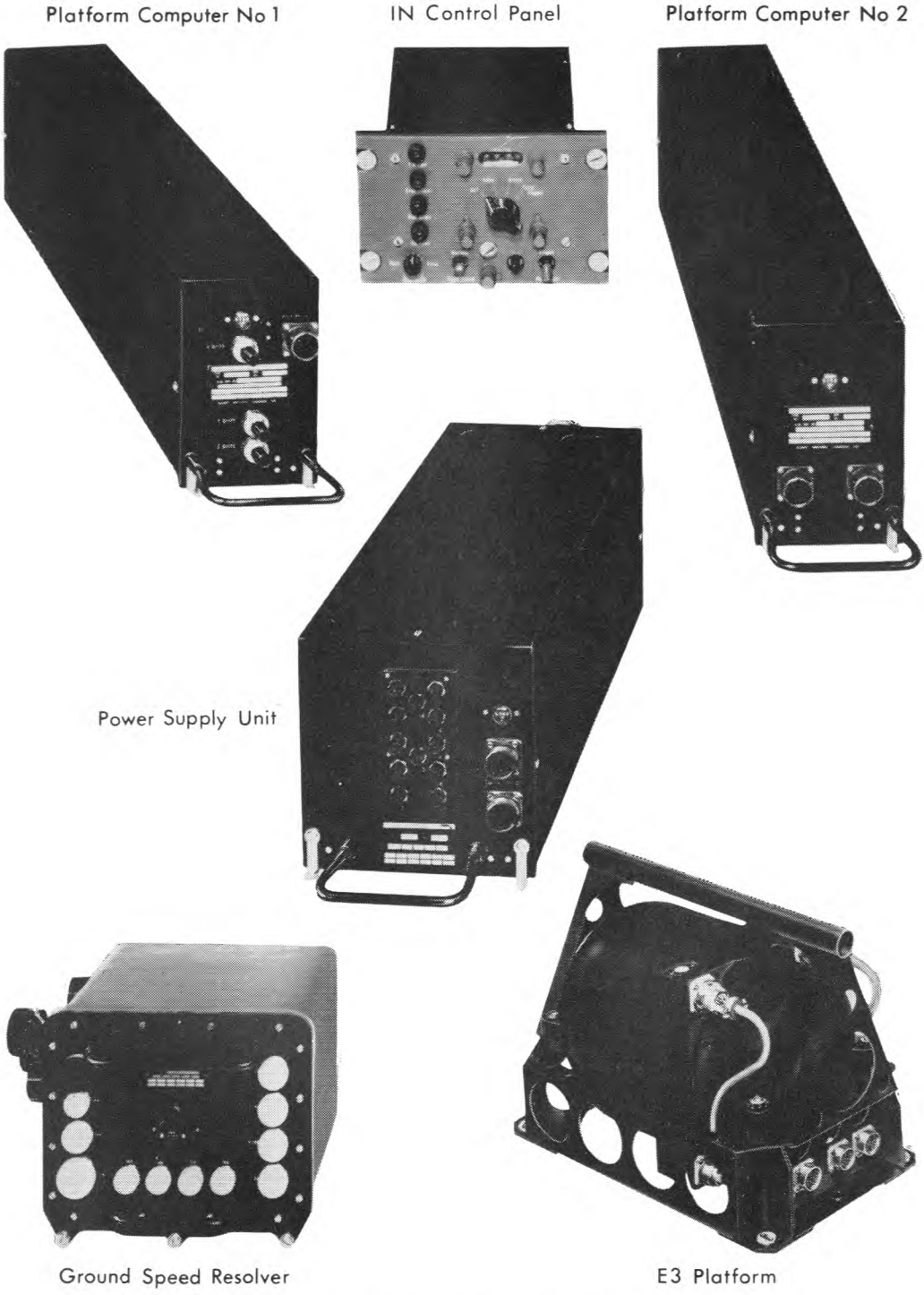


Fig 1 Components of the E 3 System

System Components

2. The inertial system consists of the following components which are illustrated at Fig 1:

a. Elliott E3 stable platform.

b. Inertial navigation control panel.

c. Ground speed resolver.

d. Platform computer No 1.

e. Platform computer No 2.

f. IN power supply unit.

COMPONENT DESCRIPTION

E3 Stable Platform

3. **General.** The heart of the inertial system is the three gimbal unit illustrated at Fig 2. It has complete freedom in roll and azimuth but is restricted in pitch. The platform has schuler-tuned vertical channels and works in a North-pointing local Earth axis system of co-ordinates.

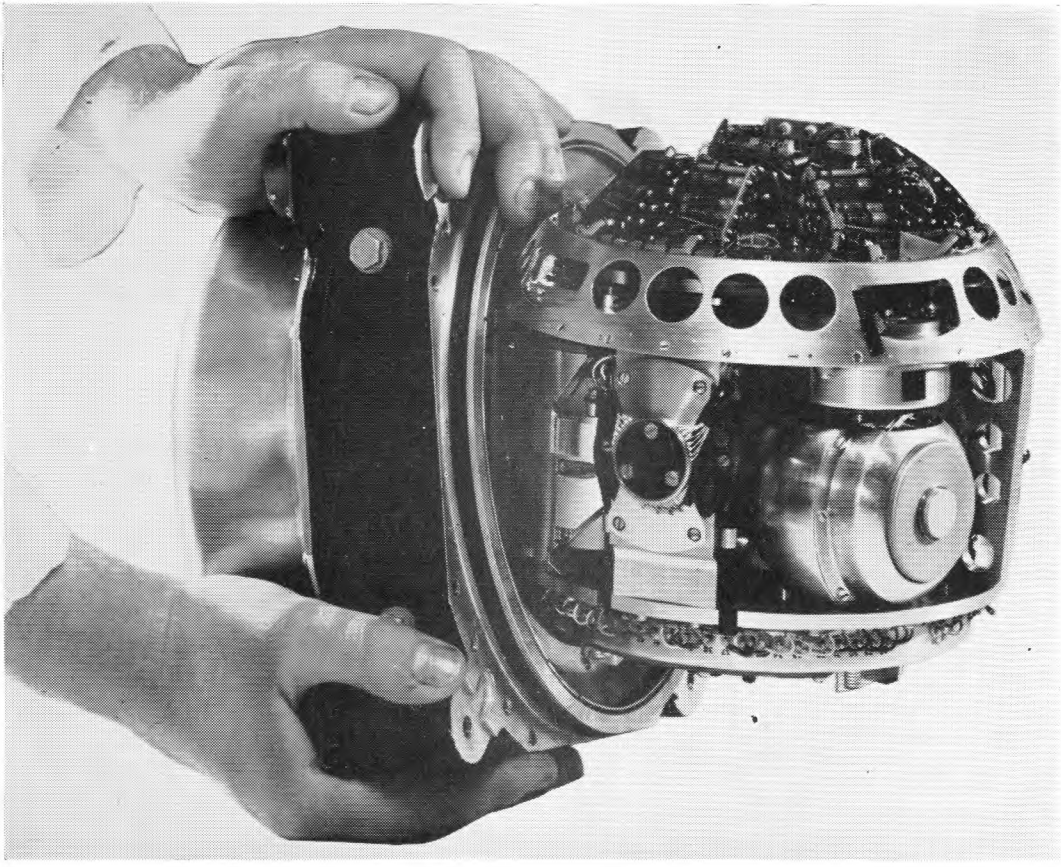


Fig 2 Elliott E 3 Stable Platform

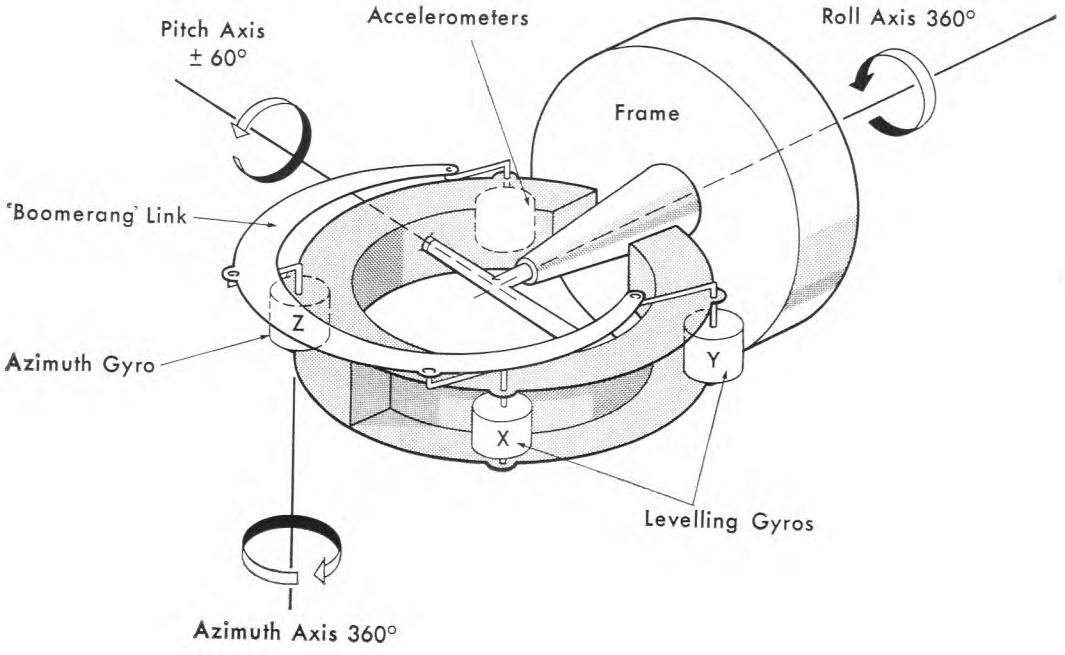


Fig 3 Inside-Out Gimbal System

4. **Gimbal Description.** The construction of the platform embodies an inside-out configuration as illustrated at Fig 3. Four packages, three containing gyroscopes and the fourth two accelerometers, make up the stable element of the unit. The gyroscopes and accelerometers are coupled by a precision linkage "boomerang link", which ensures that they are precisely aligned in azimuth. The main advantage of the inside-out layout is the accessibility of all gyroscopes, accelerometers, major components and platform electronics, without the need to dismantle the gimbals. This permits easier servicing and lower manufacturing costs.

5. **Accelerometers.** The accelerometers fitted are the Ferranti Type FA2G. These are single-axis pendulous accelerometers capable of sensing a maximum of 20*g*.

6. **Gyroscopes.** The gyroscopes fitted are BAC GG49 miniature rate integrating gyroscopes. To maintain their specified performances the gyroscope temperatures are controlled to within 0·1°C of the operating temperature which is 73°C ± 3°C. Rapid heating is provided whereby the gyroscope will reach the operating temperature within 2 minutes from an ambient temperature of + 20°C. The rotors of the gyroscopes are maintained at a constant speed using a drive supply derived from a crystal-controlled oscillator, having a frequency stability of 0·01%, housed in the power supply unit.

7. **Mechanical Details.** The unit is contained within a canister 8½ in in diameter and 13 in long; it weighs 31 lb. The degrees of freedom permitted by the gimbals are:

a. In azimuth — unlimited.
b. In roll — unlimited.
c. In pitch — ± 60°.

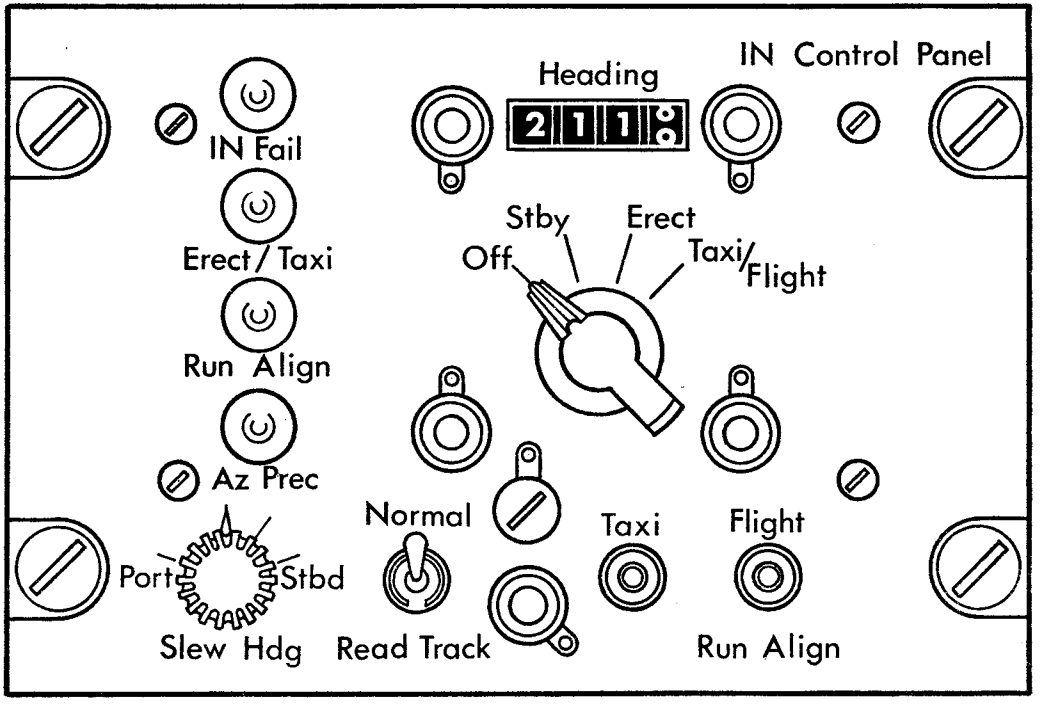


Fig 4 IN Control Panel

IN Control Panel

8. The inertial navigation control panel is illustrated at Fig 4. It carries the following controls:

a. *Main Selector Switch.* The main selector switch has four positions: OFF, STBY, ERECT and TAXI/FLIGHT. The STBY position allows the platform to be coarsely erected and aligned. ERECT switches to fine alignment to the local vertical. TAXI/FLIGHT sets the integrators to zero, allows the platform to be finely aligned in azimuth by the use of runway alignment computation, permits the aircraft to be moved and is also the position used during flight.

b. *Flight/Run Align Switch.* This is a three-position toggle switch, biased to centre OFF. The UP position activates the flight mode, DOWN position the runway alignment mode.

c. *Taxi.* The TAXI button is used to re-enter the system to the taxi mode from the flight mode. It may be used during a hold in the take-off sequence due to unserviceabilities or after flight. Its effect is to zero the integrators and hold them at zero.

d. *Heading/Track Counters.* The counters give an indication of either platform heading or the ground speed resolver computed track, dependent on the setting of the NORMAL/READ TRACK switch. TRACK may be selected either when flight has been selected prior to take-off or after the Runway Alignment lamp has extinguished following a runway alignment selected take-off. The two-speed slew heading control permits platform headings to be set when the main selector switch is in the STBY position, and runway heading to be set when in the TAXI/FLIGHT position.

9. The following indicators are also included on the inertial navigation control panel:

a. *IN FAIL.* This light comes on when the IN system fails and when power is supplied to the system with the main selector switch selected OFF.

b. *ERECT/TAXI.* When the erection phase is complete or when the TAXI button is pressed this light will come on. It goes out when the FLIGHT/RUN ALIGN switch is operated.

c. *RUN/ALIGN.* This light comes on when RUN/ALIGN is selected and goes out after the preset distance along the runway; at that time the inertial system enters the flight mode.

d. *AZ PREC.* After the runway alignment computation, this light comes on when the "Z" gyroscope is precessed (*see* Fig 3). It also comes on when the heading is precessed from the synchronous astro compass. The light goes out as soon as the azimuth precession is complete.

Ground Speed Resolver

10. In the ground speed resolver, doppler ground speed is resolved about track to give northerly and easterly velocity components by means of a ball resolver. The outputs are compared with the corresponding inertial components and the resultant error signals are fed back to the first stage integrators. When doppler information is not available the ground speed resolver supplies inertially-derived ground speed and drift, and northerly and easterly increments of ground distance gone.

Platform Computor No 1

11. This computer contains:

a. The first stage integrators which produce the northerly and easterly velocities from the platform outputs of acceleration in these axes.

b. The servo, positioned by latitude inputs, to provide earth rate and transport wander correction terms to the gyro precession amplifiers.

c. The gyroscope precession amplifiers which supply the torquing currents to maintain the stability of the platform.

Platform Computer No 2

12. This computer contains:

a. The platform heading repeater unit which provides primary heading to the ground speed resolver and to the central heading system.

b. The runway alignment computer and the circuitry necessary to update the platform to a synchronous astro compass heading.

IN Power Supply Unit

13. The IN power supply unit supplies stabilized DC voltages for all the components of the inertial system. It embodies protection circuits to isolate the platform if the gyroscopes operate outside their temperature limits. The protection circuitry breaks the AC supply to the IN Power Supply Unit and hence supplies produced by it to all other units are cut.

ALIGNMENT

Alignment Phases

14. Alignment falls broadly into two phases: the erection/gyro-compassing phase, which is carried out while stationary, and a fine alignment phase in which any one of three fine aligning methods may be used.

Erection and Gyro-Compassing

15. The erection and gyro-compassing process consists of the erection phase, in which the platform is coarsely erected to a position suitable for the fine erection, and the gyro-compassing phase which follows. If the platform is misaligned by 5° in all axes, the complete alignment sequence may take 16 min or longer depending upon the initial gyroscope temperatures.

16. **Coarse Erection.** Erection of the platform commences when STANDBY is selected. The platform is coarsely erected by the nulling of the pitch and roll outputs to level the platform to the aircraft frame axes. Coarse azimuth alignment is achieved by torquing the platform until the heading counter reads the best available true heading.

17. **Fine Erection and Gyro-Compassing.** In this phase the platform erects to the vertical by nulling any outputs of the accelerometers due to gravity and then enters a gyro-compassing phase

which removes any component of Earth rotation in the output of the North/South accelerometer. During this phase the aircraft must not be moved.

Fine Alignment

18. After the erection and gyro-compassing, a fine alignment should be carried out using one of the following methods:

a. A runway alignment.

b. A survey alignment.

c. A bearing method.

Runway Alignment

19. **Introduction.** The alignment system is based on the assumption that, during take-off, an aircraft makes good a track parallel to the centre-line of the runway of known bearing. During take-off the equipment resolves the horizontal components of its velocity outputs about the preset runway true bearing and uses any across-track component so determined to align the platform in azimuth.

20. **Components.** The runway alignment equipment is part of the Platform Computer No 2 and is illustrated at Fig 5.

21. **Operation.** The platform is erected in the normal manner and a best estimate of the aircraft's true heading, normally from a magnetic source, is inserted using the slew heading control on the IN control panel, when the aircraft is in the taxi/flight mode. The surveyed runway true bearing, corrected by a calibration graph for a particular runway alignment computer, is set in the resolver. A distance S^1 is set by the manufacturer in comparator No 1. This distance is calculated to be slightly less than the take-off run.

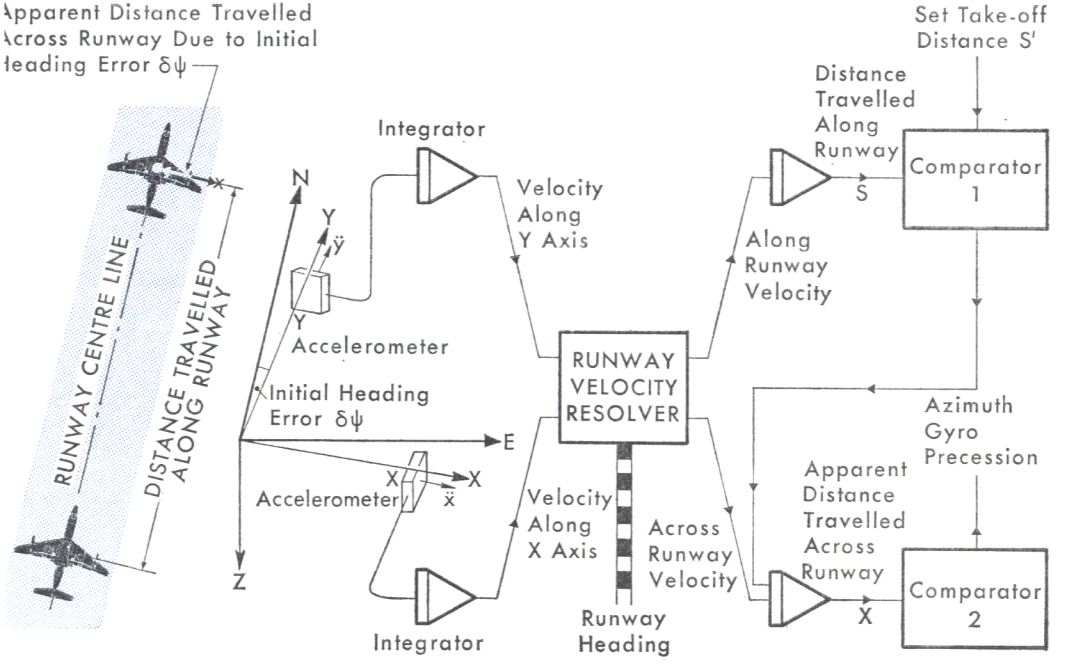


Fig 5 Runway Alignment System

22. The aircraft is positioned on the runway so that the platform is as near as possible to the centre-line. The system is released into the runway alignment mode just prior to the start of the take-off run. The aircraft carries out a normal take-off, aiming only to be on the centre-line just prior to rotation. Position at intermediate times is immaterial.

23. During the take-off normal inertial outputs of acceleration converted to East-West and North-South velocities are fed to the resolver where they become along and across-runway velocities. These velocities are then fed to integrators where they become along and across-runway distances.

24. When S, the distance travelled along the runway, is equal to S^1, the preset take-off distance, comparator 1 disconnects the inputs to the two integrators. The quantity S^1 is now fed into the across-runway integrator to unstore X, the distance travelled across the runway. The time for this operation will be proportional to X and inversely proportional to a known S^1, and will therefore be proportional to the tangent of the mis-alignment angle. While the unstoring is in progress the azimuth gyro is precessed at a constant rate. When comparator 2 detects the end of the unstore process it disconnects the precession signal from the azimuth gyro. The amount that this gyro has been precessed is thus proportional to the tangent of the angular misalignment and, if the misalignment is small, to the angular misalignment itself. The assumption that the angle is equal to its tangent will not introduce significant errors until misalignment angles greater than 7° are encountered.

25. **Advantages.** With the E3 stable platform, accurate fine gyro-compassing is not possible. Runway alignment is the easiest and quickest of the alignment techniques available. This technique is particularly suited to situations where rapid alignment of an inertial system is required.

26. **Accuracy.** Using S^1 of 2,000 ft the system is capable of aligning the E3 system to 0·17° at the 1 sigma level.

Survey Alignment

27. In the survey alignment method the heading of the aircraft is measured by an accurate external means whilst the aircraft is parked. This may be done at any time before flight provided that the system is not then moved until the platform is aligned. After platform erection, when FLIGHT has been selected, the surveyed true heading is set on the heading counters. In the Nimrod, synchronous astro compass controls are used to achieve this. The system should then be left in the FLIGHT mode until the completion of the flight, remembering that the accuracy of the heading will decay with time.

Survey Alignment

28. After platform erection and the selection of the FLIGHT mode, a sextant may be used to sight an object of known true bearing from the aircraft. Any discrepancy between the known bearing and that observed can be applied as a correction to the heading readout in the same way as is described in the previous paragraphs.

29. The known bearing used in this method may be either that of a previously surveyed distant object or the calculated azimuth of a celestial body.

OPERATION

Platform Stabilization

30. The output pick-off signal of the azimuth gyroscope is passed through a pre-amplifier and a servo-amplifier to the azimuth servo-motor to complete the azimuth servo-loop. Similarly the X and Y axis gyroscopes together with two pre-amplifiers, a resolver, and the pitch and roll servo-amplifiers, control the vertical axis. All pre-amplifiers, and servo-amplifiers are contained within the platform canister. The servos control the platform under normal flight conditions to movements which will not exceed 0·1 min of arc in the vertical plane and 0·25 min of arc in the azimuth axis. The platform will accept turning rates of 600° per sec in any axis.

31. The system requires a separate computer to supply it with latitude so that the Platform Computer No 1 can compute the necessary correction terms for the Y and Z gyroscopes which require to be precessed by earth rate ($\Omega \cos \lambda$ and $\Omega \sin \lambda$ respectively). The Platform Computer No 1, using incremental movement East/West from the ground speed resolver and latitude, also computes the transport wander term $\left(-\frac{U}{R} \tan \lambda\right)$ to be fed to the Z gyroscope.

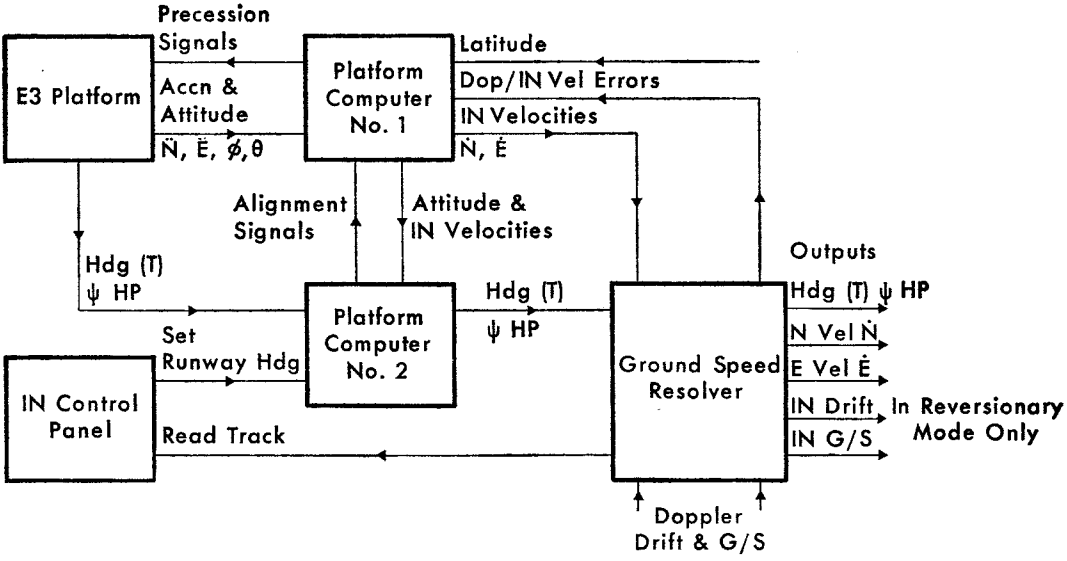


Fig 6 Signal Flow

Heading

32. The following paragraphs should be read in conjunction with Fig 6 which illustrates the signal flow of the system. Once the system is aligned, transmission of the platform angle relative to its mounting is by synchro. This heading, ψ HP, is fed via platform computer No 2 to the ground speed resolver and to the aircraft heading system. Heading counters are provided on the IN control panel. By selecting read track, the aircraft track as computed by the ground speed resolver may also be displayed on these counters.

Velocities

33. The aircraft accelerations sensed by the platform are fed to the Platform Computer No 1 where integration is carried out producing the velocities (Ṅ and Ė) which are fed to the ground speed resolver and to the aircraft central computer. The ground speed resolver also has inputs of doppler drift and ground speed which it converts into doppler velocities (Ṅ and Ė). The track about which the doppler ground speed is resolved is derived from corrected doppler drift and true heading from the platform. The ground speed resolver compares the inertial and doppler velocities passing the difference, or velocity error signals, back through the Platform Computer No 1 to update inertial velocity outputs during straight and level flight.

34. In periods of doppler unlock, the ground speed resolver uses inertial velocities to compute incremental ground movement, ΔN and ΔE. When doppler is locked on it computes ΔN and ΔE on doppler velocities only.

Roll and Pitch

35. The platform outputs of roll and pitch are fed to the Platform Computer No 1 and also directly to the aircraft attitude system.

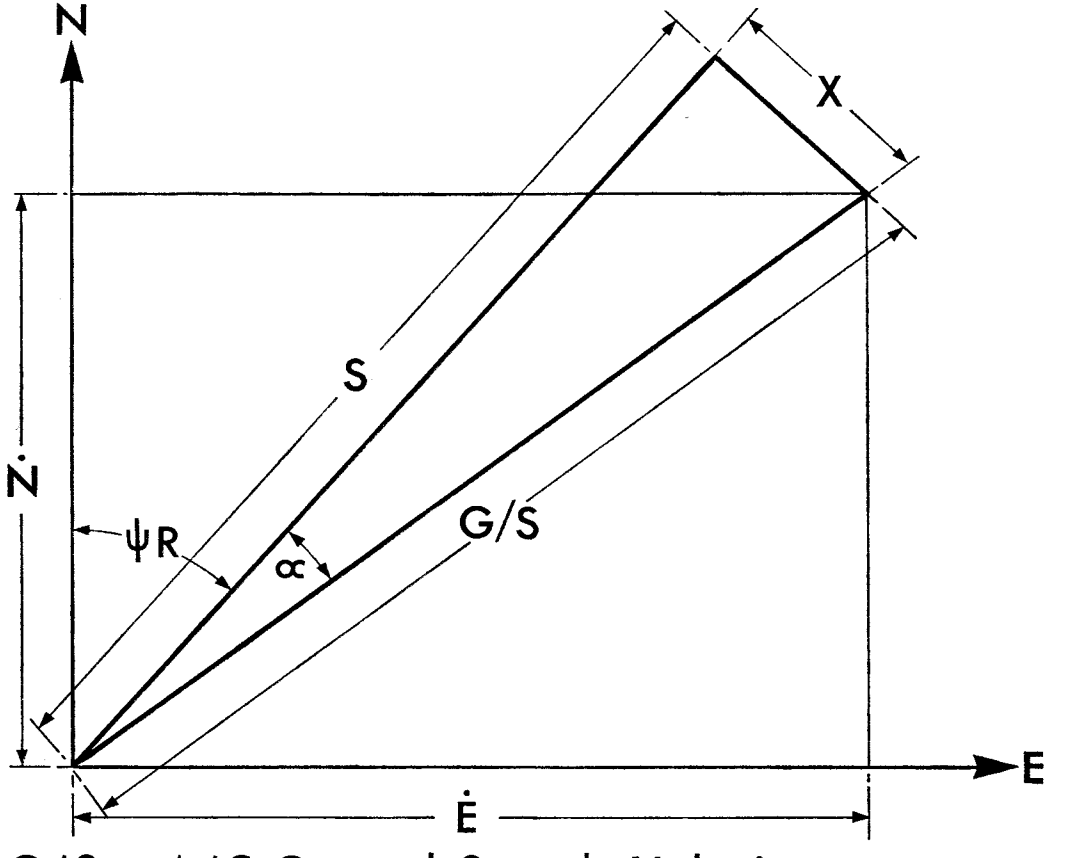
RUNWAY ALIGNMENT

CONTENTS

Paras
Runway Alignment Calculations ... ... 1 – 7

Runway Alignment Calculations

1. The surveyed heading of the runway (ψ R) is set on the heading counter indicator. In the runway alignment phase this heading is the datum for all alignment calculations.



G/S = A/C Ground Speed Velocity
Ṅ = Northerly Component of G/S
Ė = Easterly Component of G/S
ψR = True Runway Heading
∝ = Platform Misalignment
S = Along Track Distance
X = Across Track Distance

Fig 1 Vector Diagram

2. Referring to Fig 1, if the aircraft travels along the runway with a platform misalignment of ∞ degrees then:

$$\dot{N} = G/S \cos(\psi R + \alpha) \quad \ldots\ldots\ldots (A)$$
$$\dot{E} = G/S \sin(\psi R + \alpha) \quad \ldots\ldots\ldots (B)$$

3. In Fig 2, the runway velocity resolver, which is a double sin/cos potentiometer with a shaft input of ψ R, produces outputs of:

$\dot{N} \cos \psi R$ ..... (1)
$\dot{N} \sin \psi R$ ..... (2)
$\dot{E} \cos \psi R$ ..... (3)
$\dot{E} \sin \psi R$ ..... (4)

4. By adding the outputs (1) and (4) and substituting values of $\dot{N}$ and $\dot{E}$ from (A) and (B):

(1) + (4) = Along track velocity

$= \dot{N} \cos \psi R + \dot{E} \sin \psi R$

$= G/S \cos(\psi R + \alpha) \cos \psi R$
$G/S \sin(\psi R + \alpha) \sin \psi R$

$= G/S \cos(\psi R + \alpha - \psi R)$

$= G/S \cos \alpha$

5. By Subtracting (3) from (2) and substituting from (A) and (B):

(2) – (3) = Across track velocity

$= \dot{N} \sin \psi R - E \cos \psi R$

$= G/S \cos(\psi R + \alpha) \sin \psi R - G/S \sin(\psi R + \alpha) \cos \psi R$

$= G/S \sin(\psi R + \alpha - \psi R)$

$= G/S \sin \alpha$

6. As α is always small, it can be assumed that $\cos \alpha = 1$, and $\sin \alpha = \alpha$, then:

Along track velocity = $G/S \cos \alpha$
$= G/S$

Across track velocity = $G/S \sin \alpha$
$= G/S\ \alpha$

7. Thus by torquing the platform in azimuth for a time proportional to $\int G/S\ \alpha |dt| / \int G/S\ dt$ over the time from line-up to arriving at S^1, *ie* for X/S^1, the misalignment angle α is removed.

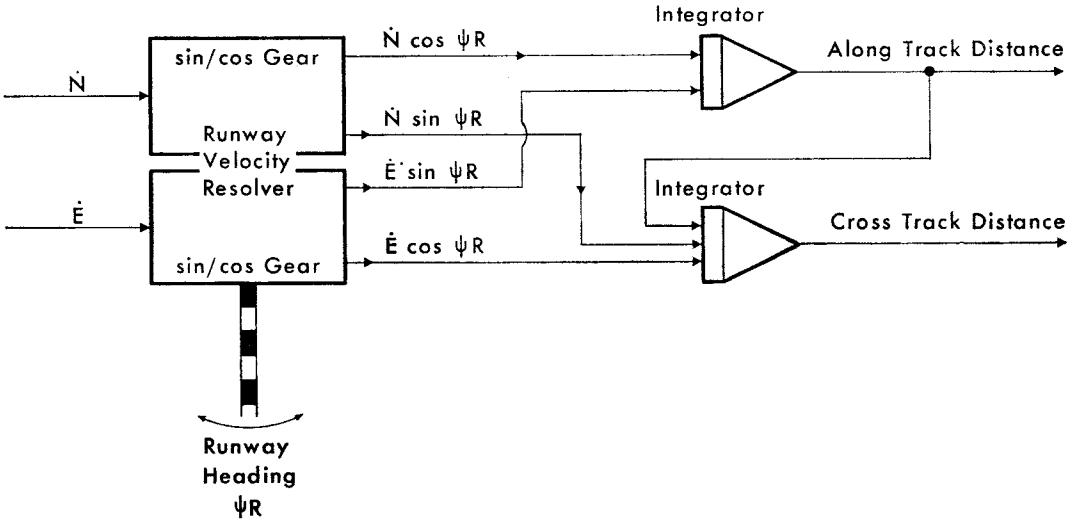


Fig 2 Signal Flow

CHAPTER 5

THE ELLIOTT E3R HEADING REFERENCE SYSTEM

(To be issued later.)

PART 4

ASTRONOMICAL INSTRUMENTS

Section

1 Sextants

2 Astro Compasses

PART 4

SECTION 1

SEXTANTS

Chapter

1 **Bubble Sextants, Marks 9AM, 9BM**

2 **Periscopic Sextant, Mark 2**

3 **Periscopic Sextants, Marks 2A, 2B**

4 **The Kollsman Periscopic Sextant**

CHAPTER 1

BUBBLE SEXTANTS, MARKS 9AM, 9BM

CONTENTS

	Paras.
Introduction	1– 4
Principles	5
Measuring Mirrors	6– 8
Bubble System	9–14
General Description	15–16
Bubble Unit	17–18
Bubble Illumination	19–23
Sun Shades...	24
Automatic Averaging Attachment	25–26
Telescopic Eyepiece	27–29
Carrying Case	30
Instrument Errors	31–32
Tilt and Acceleration Errors	33–35
Analysis of Tilt Errors	36–38
Analysis of Acceleration Errors	39–43
Operation	44–49
Depression Sights	50

Introduction

1. A sextant is an instrument for measuring the altitude of a celestial body, *i.e.* the arc of the vertical circle through the body intercepted between the body and the celestial horizon.

2. The bubble sextant uses a bubble level to give an artificial horizon from which the altitude can be measured. When the body is viewed through the sextant an image of the bubble can also be seen, the two being made to coincide by adjustment of the sextant mirrors. The altitude so measured is then indicated on the scales of the sextant.

3. In flight the bubble moves continually, owing to the instability of the aircraft. Consequently an accurate altitude cannot be measured from a single observation. This difficulty is overcome by the provision of an averaging device which automatically takes 60 shots over a period of one or two minutes according to the Mark of sextant. One-sixtieth of the altitude is added to the scale counters for each shot taken so that the final altitude shown is an average of the 60 measured and consequently of reasonable accuracy.

4. No provision is made for observing the natural horizon as an alternative to the bubble horizon.

Principles

5. Two main principles are involved in the construction of the bubble sextant:—

(*a*) The angular measurement of the altitude of the body, by means of measuring mirrors.

(*b*) The establishment of an artificial horizon from which the altitude may be measured by means of a bubble system.

These are discussed separately below.

Measuring Mirrors

6. The corollary to the Laws of Reflection of Light (*see* A.P. 1234B, Sect. 2, Optics) states "If a ray of light is reflected by two mirrors, whose normals are in the same plane as the ray, then the angle between the initial and final paths of the ray is equal to twice the angle between the mirrors."

7. This corollary is used in the measuring system of the Mk. 9 series sextants. Two mirrors

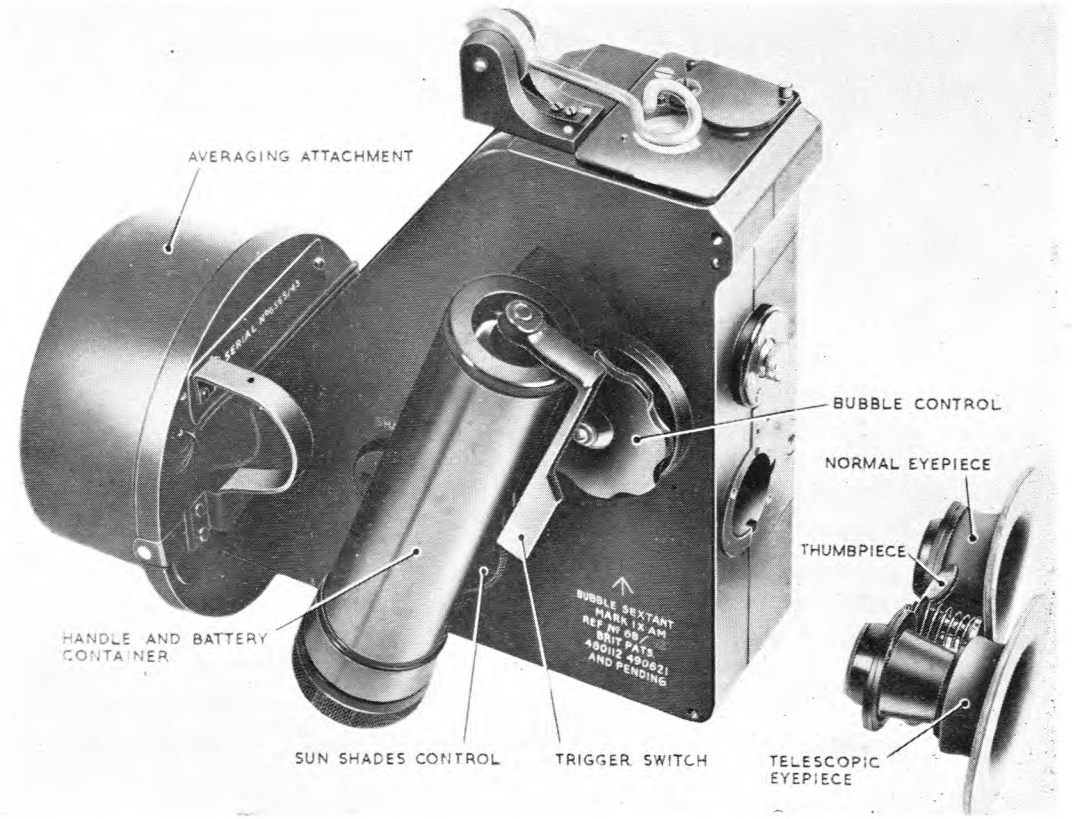


Fig. 1. Bubble Sextant, Mk. 9AM

known as the Index Mirror and the 5° Increase Mirror are arranged as shown in Fig. 3. The path of the original ray AB can be assumed to be the line of sight from the star to the eye. If the final path of the ray, CD, is in the plane of the sensible horizon, then the angle between AB produced and CD will be the altitude of the star. This angle, BDC, will be twice the angle between the mirrors.

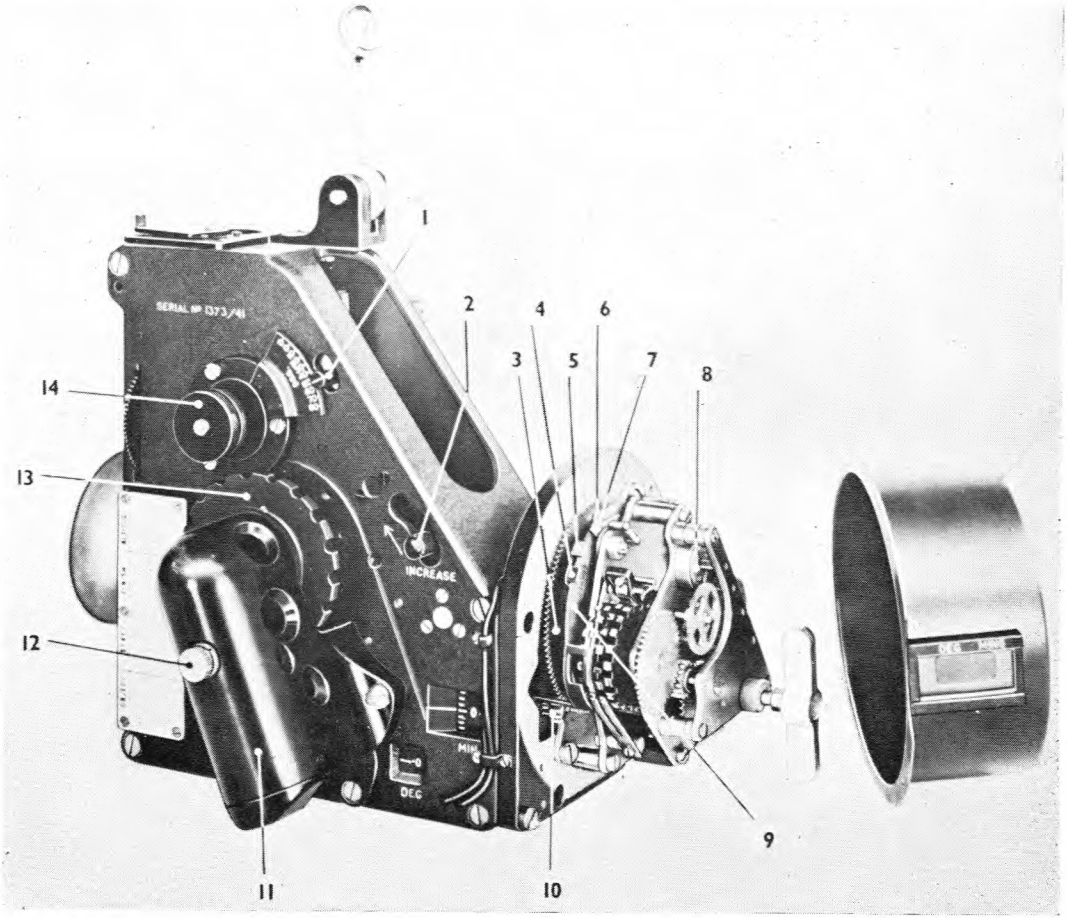
8. The sensible horizon in this case is represented by the image of a bubble appearing at infinity (*see* paras. 9 to 14). The index mirror can be adjusted by means of coarse and fine setting controls to bring the final reflected ray coincident with the ray from the bubble and therefore into the plane of the sensible horizon. The resulting angle between the mirrors is doubled and calibrated as altitude. The 5° increase mirror moves through $2\frac{1}{2}$° and can be placed in one of two positions, NORMAL or INCREASE. In the latter position 5° is automatically added on to the fine setting and averaging counters.

Bubble System

9. Since the force of gravity is always vertical, an air bubble in a sphere filled with liquid will come to rest vertically above the centre of the sphere. A ray of light from the bubble centre to the centre of the sphere will thus be a true vertical (BC in Fig. 4).

10. The same result is obtained in the Mk. 9 series sextants by enclosing the bubble and liquid in a small chamber, the upper surface of which is part of a sphere (Fig. 5). The bubble is then vertically above the centre of curvature of the spherical top of the bubble chamber and BC is still the true vertical.

11. A lens or combination of lenses of focal length equal to the radius of curvature of the bubble chamber top is placed with its optical centre at C, the centre of curvature (Fig. 6). Under these conditions rays of light from the bubble emerging from the lenses will be parallel to BC



1. 10 Deg. Index
2. 5 Deg. Increase Lever
3. Setting Wheel
4. Rotating Arm
5. Carry Pawl
6. Setting Wheel Roller
7. Carry Pawl Trip
8. Centrifugal Governor
9. Totalizing Wheel
10. Setting Wheel Pinion
11. Handle
12. Lamp
13. Fine Setting Knob
14. Coarse Setting Knob

Fig. 2. Bubble Sextant, Mk. 9AM—Right-hand Side

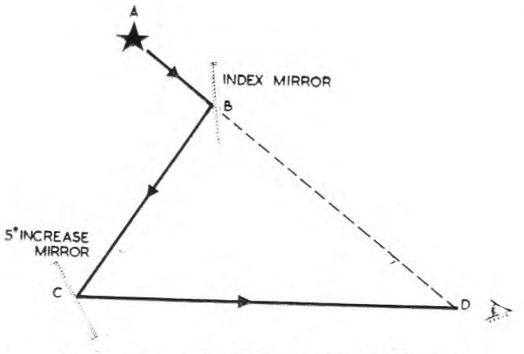


Fig. 3. Principle of the Measuring Mirrors

provided that BC coincides, or nearly coincides, with the principal axis of the lenses. The combination of lenses used to produce a parallel pencil of rays from a diverging source is known as the *collimating* lens.

12. Rays of light from an object an infinite distance away are assumed to be parallel. Conversely, the production of parallel rays of light from a given object will have the effect of making the object appear at infinity. Thus in this case the bubble image has been made to appear at infinity. Since the body whose altitude is being measured also appears at infinity, there will be no errors in alignment due to parallax.

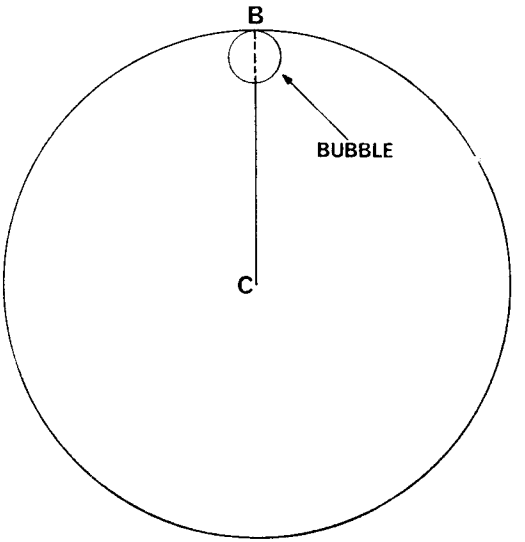


Fig. 4. Air Bubble in a Sphere

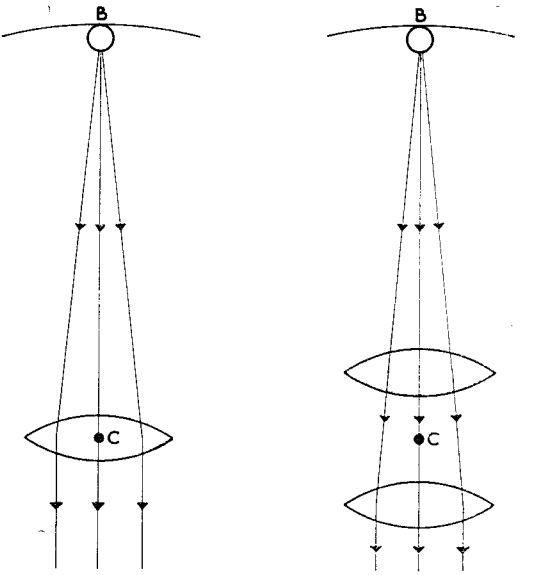


Fig. 6. Production of Parallel Rays

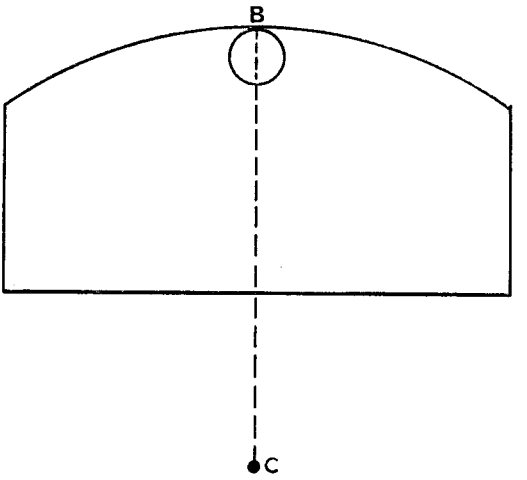


Fig. 5. Bubble Chamber Principle

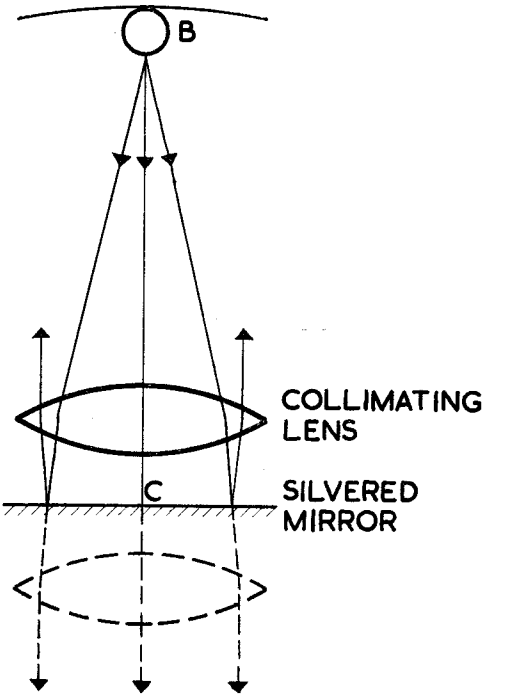


Fig. 7. Production of an Upward Pencil of Rays

13. Through the optical centre C (Fig. 7) is placed a silvered mirror parallel to the lenses. This reflects back the original rays so that they meet the upper lens a second time in the same manner as they would have met the lower lens. They thus emerge as an upward pencil of rays parallel to the principal axis of the lens, and the second lens may be dispensed with.

14. A plain glass, known as the 45° Glass, is placed above the lens at 45° to the silvered mirror. The rays emerging from the lens will be reflected by this glass. Since the plane to which both the mirror and the glass are perpendicular, is the vertical plane, BC will be parallel to it and the central ray from the bubble, which is initially vertical, will, after reflection at the glass, be hori-

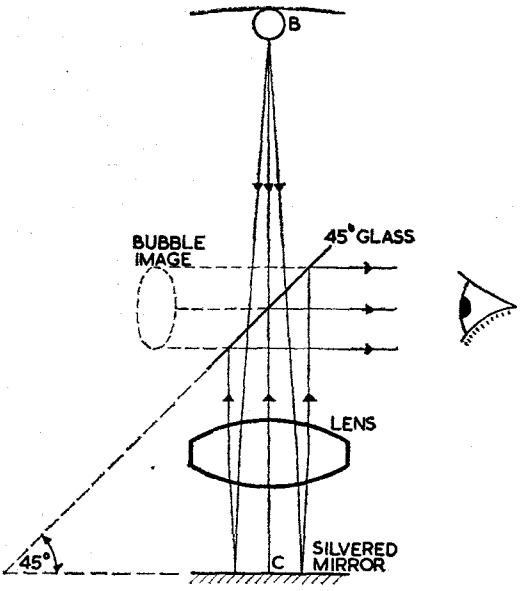
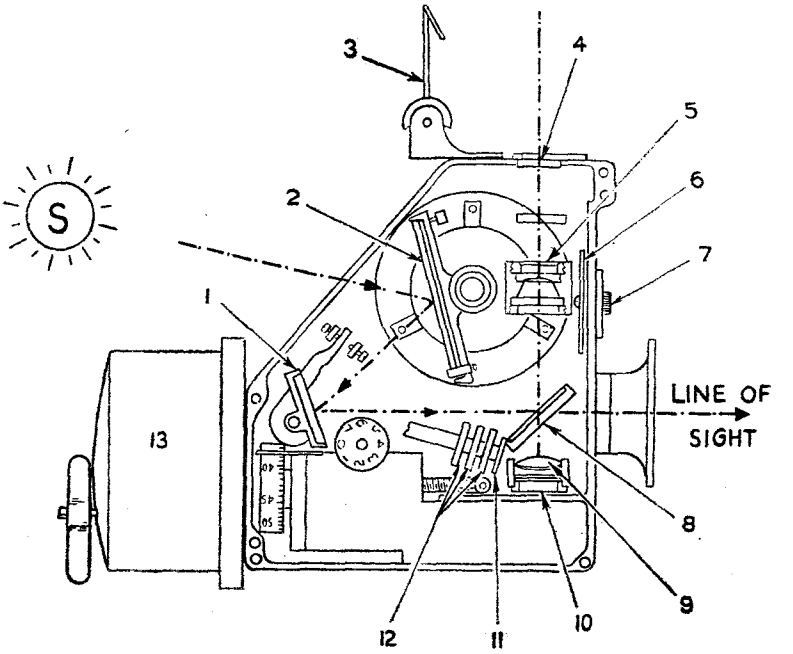


Fig. 8. Production of Horizontal Pencil of Rays

zontal, and together with the other rays which emerge from the lens, will form a horizontal pencil of light. If the eye is placed in the path of this horizontal pencil of light, an image of the bubble will be seen at an infinite distance in the plane of the sensible horizon.

General Description

15. In Fig. 9, light from the illuminated bubble contained in the bubble chamber shines down through the unsilvered mirror and is reflected back as a parallel beam by the mirror at the base of the collimating lens. A proportion of the light is reflected horizontally by the unsilvered mirror, so that an image of the bubble is seen when the observer looks through the eyepiece. The lens system, formed by the lens and the mirror, is adjusted so that its focal length is equal to the radius of curvature of the spherical surface against which the bubble rests, and the bubble is therefore always in focus. Provided that the bubble is not touching the edge of its container, it will always be vertically above the centre of the lens and its image will appear in the horizontal plane when viewed through the eyepiece.



1. 5° increase mirror
2. Index mirror
3. Suspension hook
4. Matt perspex window
5. Bubble chamber
6. Aperture disc.
7. Bubble lamp
8. Unsilvered mirror
9. Collimating lens
10. Mirror
11. Shutter
12. Sun shades
13. Automatic attachment

Fig. 9. Line of Sight and Bubble Image Path

16. The setting knobs of the sextant enable the angle between the mirrors to be varied until the images of the celestial body and the bubble coincide. The altitude of the body is then equal to twice the angle between the mirrors, and is so shown on the sextant scales.

Bubble Unit

17. The bubble chamber is connected by a thin capillary tube to the air chamber and by a fine bore tube to the diaphragm chamber (*see* Fig. 10). When no bubble is visible the diaphragm chamber and bubble chamber are filled, and the air chamber partly filled with liquid. When the control knob is rotated to draw back the diaphragm and hence increase the volume of the diaphragm chamber, liquid will be drawn through the capillary from the air chamber until none remains in it. Then air will pass into the bubble chamber to form a bubble. If the sextant is tilted so that this bubble is away from the capillary tube, the bubble will remain trapped in the chamber when tension on the control knob is released.

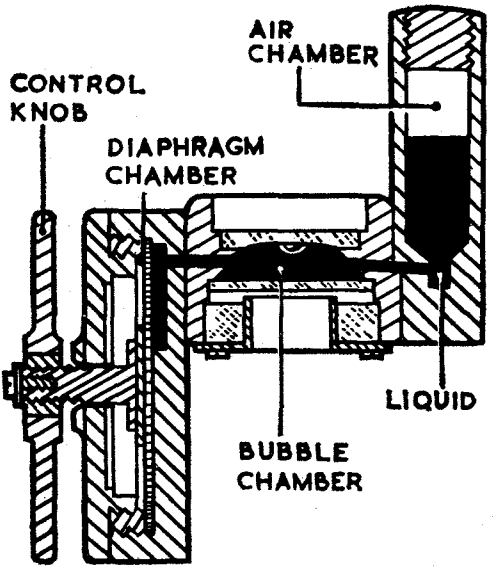


Fig. 10. Section of Bubble Chamber

18. To remove the bubble it is necessary to increase the tension on the control knob and to hold the sextant so that the bubble rests against the entrance to the capillary tube. If the tension on the diaphragm is then released the bubble will be drawn back into the air chamber. The surface tension of the liquid is such that the bubble will pass through the capillary tube only when the system is under reduced pressure.

Bubble Illumination

19. In daylight, a frosted window above the bubble chamber provides light for the illumination of the bubble. The light can be shut off when desired by closing the shutter above the perspex.

20. At night, the light from a small electric bulb is carried round the base of the bubble chamber and is concentrated by a small reflector on to the

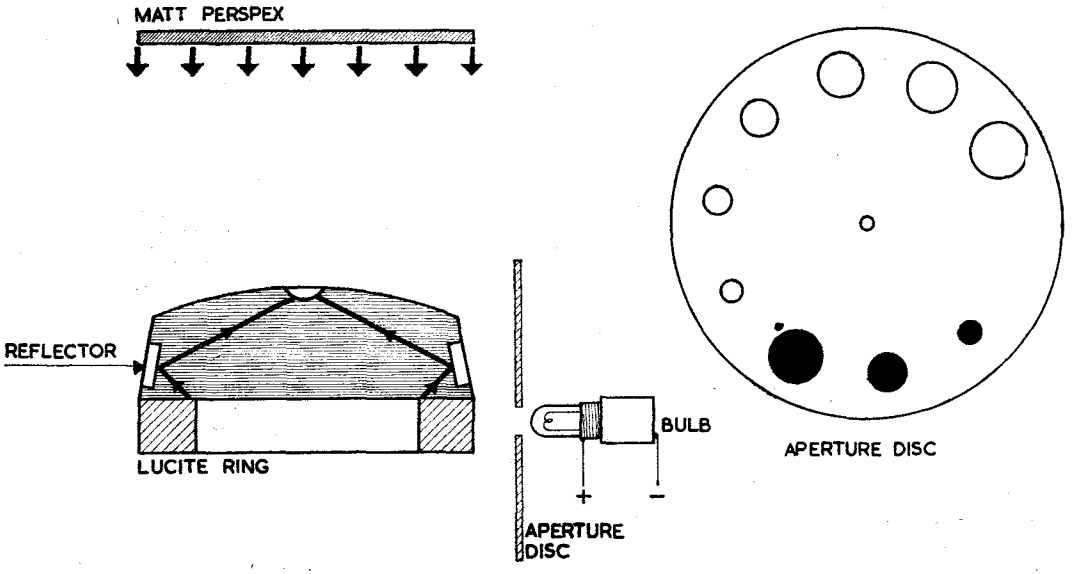


Fig. 11. Bubble Illumination System

bubble (*see* Fig. 11 (*a*)). The amount of light from the bulb entering the bubble chamber can be controlled by an aperture disc containing nine apertures of varying sizes, three of which are fitted with red filters (*see* Fig. 11 (*b*)).

21. The 3-volt battery for bubble lighting is contained in the left handle of the sextant, and also supplies lighting for the outside scales. A two-way self-centring switch on the handle determines whether bubble or scale lighting is in operation.

22. If it is desired to use the aircraft power supply, a plug adaptor is supplied which can be inserted in place of the battery. Plug adaptors are issued ready for use on 12-volt supply, but may be altered to 24 volts by removing the screw and spring washer seen through a hole in the side, close to an arrow marked FOR 24-V CIRCUIT, REMOVE SCREW AND WASHER.

23. The bubble lighting is cut off automatically at the end of the sight by the averaging mechanism

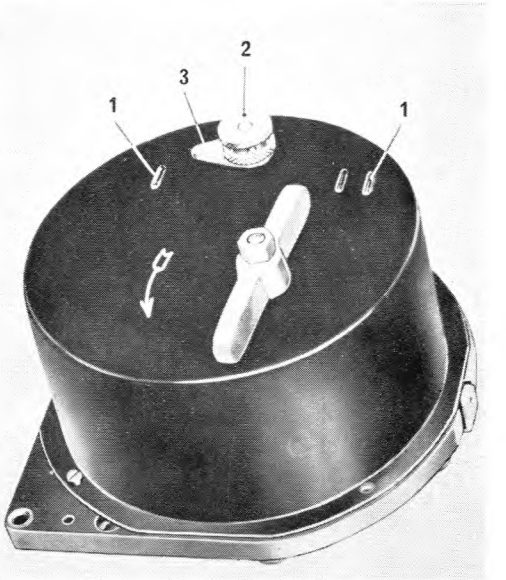
Sun Shades

24. To protect the eyes of the operator when the sun is being observed, sun shades of varying intensities may be set in the line of sight. By rotation of the lower hand-wheel on the left-hand side of the sextant, one or more of three shades may be raised into the line of sight, until the desired intensity is obtained. An indication of the density is given by numbers between 0 and 7 inclusive, visible through an aperture adjacent to the control wheel.

Automatic Averaging Attachment

25. Because of acceleration errors in flight (*see* paras. 39-43) the altitude of a body cannot be determined from a single observation. The averaging attachment automatically records 1/60th of the altitude as indicated by the index mirror setting every two seconds for a two-minute period or every one second for a one-minute period, and gives the total of the 60 recordings on a counter. The clockwork mechanism is linked to the bubble lighting circuit so that the bubble lights only when the clockwork has been wound. The winding of the clockwork sets the averaging counters to zero.

26. The averaging attachment on the Mk. 9AM sextant gives the average of 60 shots taken over a two-minute period. The Mk. 9BM incorporates a two-speed gear to give an average of 60 shots taken over either a one-minute or two-minute period. The required time of run is selected by the change train knob (*see* Fig. 12), the pointer attached to the knob indicating the selected time of run against the markings I and II on the cover for the one-and two-minute runs respectively.



1. Indicator marks
2. Change train knob
3. Change train knob pointer

Fig. 12. Mk. 9BM—Two-Speed Attachment

Telescopic Eyepiece

27. The letter M in the nomenclature of the Mk. 9AM and BM sextants indicates that they are fitted with an adjustable double eyepiece. This consists of a swivel bracket pivoted about a central bolt which is mounted on the sextant body. At one end of the bracket is a standard eyepiece and at the other end a simple twin-lens telescope, having a magnification of two diameters. This means that the diametrical field of view is halved and through it stars will appear four times as bright. Thus Polaris, with a magnitude of 2.1, appears as bright through the telescopic eyepiece as Procyon, with a magnitude of 0.5, appears normally.

28. To change the eyepieces, the bracket is held by means of the two thumb-pieces, pulled outwards from the sextant body and rotated through 180°.

29. It has been found that the operator sometimes makes the bubble too small when the telescopic eyepiece is used during the formation of the bubble. It is better therefore to adjust the bubble to the correct size and obtain an approximate altitude of the star with the standard

eyepiece and then to change to the telescopic eyepiece before starting the actual sight. The double eyepiece is illustrated in Fig. 1.

Carrying Case

30. A plastic carrying case (Fig. 13) is provided, fitted with two locking clasps and a leather carrying handle. The interior is padded to fit the shape of the sextant. Two spare batteries can be carried in vulcanite tubes clipping into the back of the case, while spare bulbs are contained in a tin box under the tubes. The adaptor (*see* para. 22) clips in the back together with its plug and 7½ft. of cable.

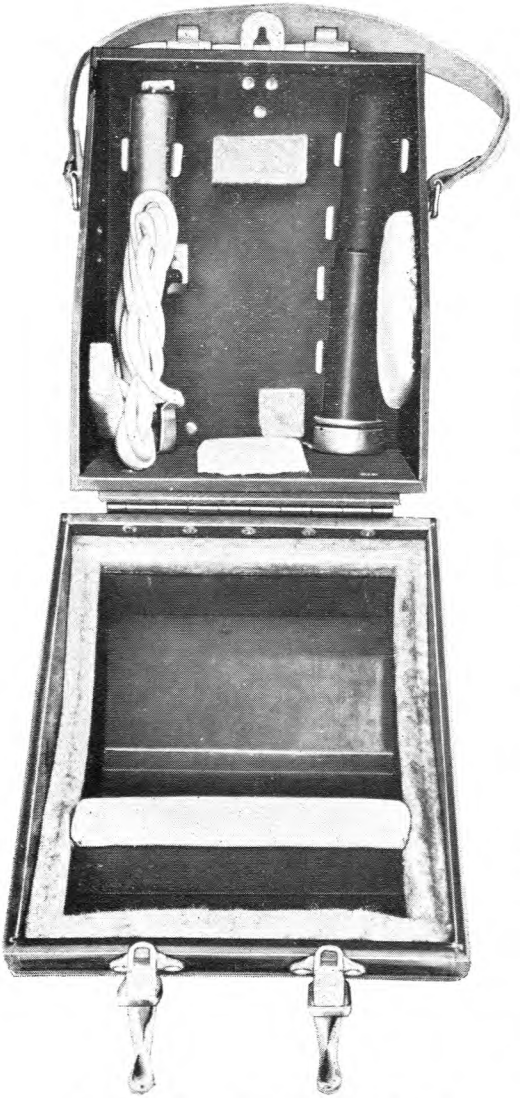


Fig. 13. Interior of the Carrying Case

Instrument Errors

31. The bubble sextant is a delicate instrument and is liable to error if any of its component parts need adjustment, *e.g.* if the index and 5° increase mirrors are not perpendicular to the frame of the sextant, or if the plain glass is not at 45° to the base mirror. These errors can be detected by use of the Portable Collimator, described in Chapter 2 of this section.

32. The correction of errors of this type is not to be undertaken by the navigator but only by the instrument repairer or the manufacturer.

Tilt and Acceleration Errors

33. The accurate measurement of altitude by means of the measuring mirrors, and the correct indication of the horizon by means of the bubble system, depend upon certain optical conditions. When these conditions are not fulfilled, errors are to be expected. If, for instance, the frame of the sextant is not held parallel to the plane of the vertical circle, then the angle between the measuring mirrors will not equal half the altitude. In the case of the bubble chamber, it is essential that the force of gravity be the only force acting on the bubble, otherwise it will not remain vertically above the optical centre of the lens system, the first requirement in the production of the horizontal ray.

34. **Tilt.** Tilt will affect both the measuring mirrors and the mirrors of the bubble system. The corollary of the two laws of light employed in the sextant states that the mirrors shall both be perpendicular to the same plane (in this case the plane of the side of the sextant) and the ray of light falling upon them shall be parallel to that plane. If the ray of light is not parallel to that plane, the error introduced will affect both the measuring mirrors and the 45° glass and base mirror.

35. **Acceleration.** Acceleration, on the other hand, will affect the liquid and the bubble. The fact that another force, other than that of gravity, is acting in another direction on the liquid, will mean that the ray of light from the bubble does not represent a true vertical and therefore will not produce an indication of the true horizontal.

Analysis of Tilt Error

36. **Fore-and-Aft Tilt.** No error will arise in the case of fore-and-aft tilt because:—

(*a*) When the sextant is tilted in the fore-and-aft plane, provided that this tilt does not exceed

3°, the bubble will still define the true vertical. More than 3° introduces the effect of surface tension, drawing the bubble to the sides of the chamber.

(*b*) In the bubble system the 45° glass and the base mirror will remain at right angles to the vertical plane indicated by the bubble and therefore the horizon will be defined correctly.

(*c*) The index mirror and the 5° increase mirror will still be at right angles to the plane of the vertical circle and therefore the measuring system will function correctly.

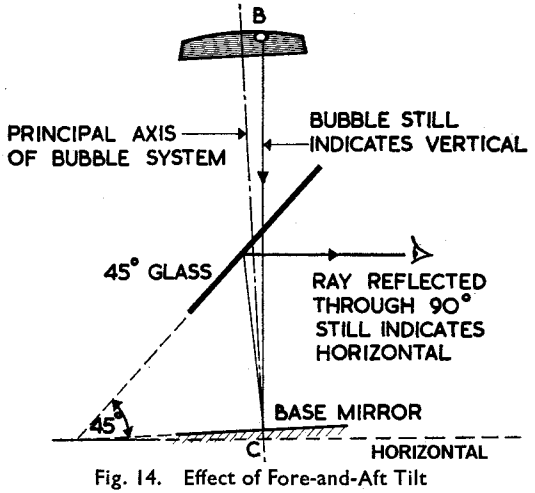


Fig. 14. Effect of Fore-and-Aft Tilt

37. **Lateral Tilt.** When the sextant is tilted laterally errors will occur because:—

(*a*) The index and 5° increase mirrors are no longer perpendicular to the plane of the vertical circle. The effect on the measuring mirrors will be to cause them to measure an arc inclined to the vertical by the angle of tilt, which is obviously greater than the arc of the vertical circle representing true altitude.

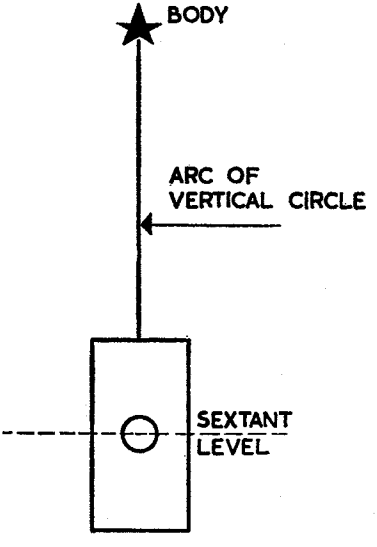
(*b*) In the bubble system the 45° glass and the base mirror are no longer perpendicular to the vertical defined by the bubble. The bubble will appear above the horizon and this will cause the measured altitude to be too small.

(*c*) The combined effects of (*a*) and (*b*) are such that the errors tend to cancel out. However, since the error in the bubble system is the greater, the final altitude will be too small.

38. Because lateral tilt causes an error in the sextant readings while fore-and-aft tilt does not, two parallel lines are drawn on the surface of the bubble chamber in a fore-and-aft direction. If the bubble and star are kept between those parallel lines, lateral tilt is confined to ½°, with a resultant error of less than 1 min. of arc.

Analysis of Acceleration Errors

39. The effect of an acceleration caused either by variation of aircraft speed or change of heading is to displace the bubble in the direction of the acceleration. The observer, who may be unaware of the presence of an acceleration, can hold the sextant in such a way as to place the bubble in the centre of the field of view of the sextant which



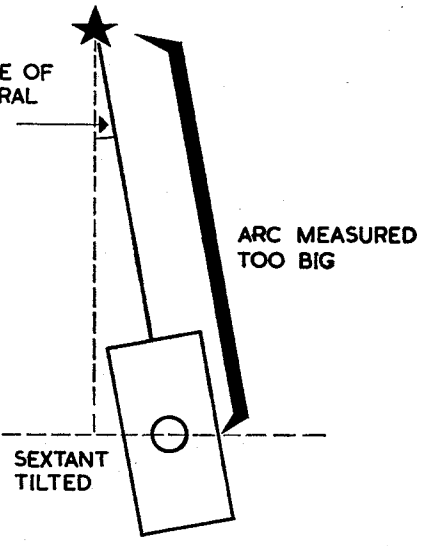


Fig. 15. Lateral Tilt—Effect on the Index and 5° Increase Mirrors

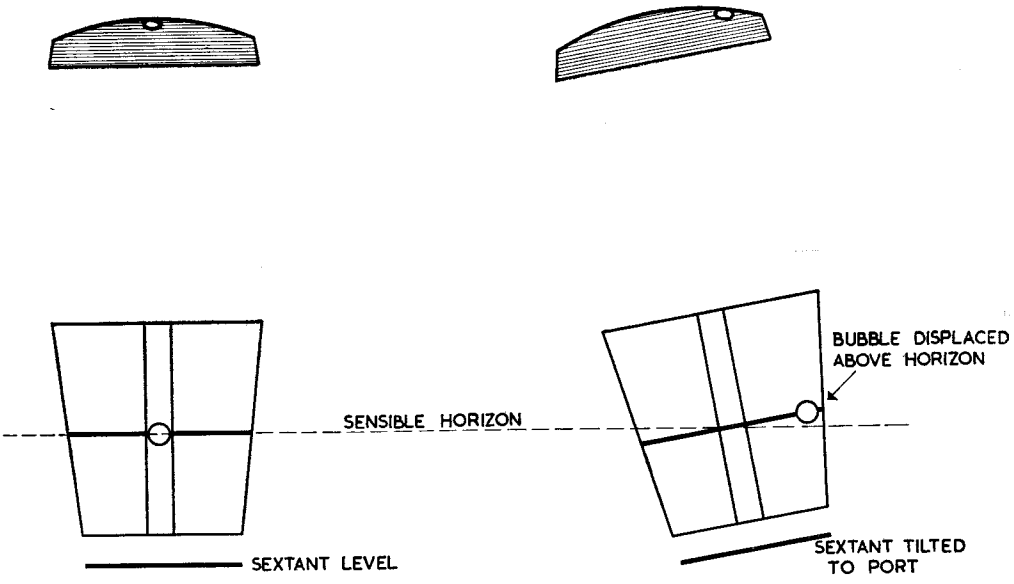


Fig. 16. Lateral Tilt—Effect on Bubble System and 45° Glass

will cause the bubble to define a false zenith and thus a false horizon.

40. The extent of the errors in measurement due to an acceleration will be observed from Fig. 17, where, owing to an acceleration acting on the bubble, the defined zenith is displaced from the true zenith Z to Z_1.

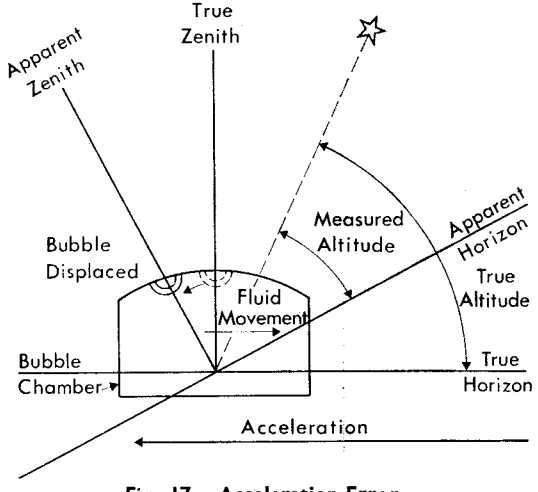


Fig. 17. Acceleration Error

41. The size and sign of the error in observed altitude varies according to the direction in which the sight is taken relative to the direction of the acceleration, and can be summarized as follows:—

(*a*) For a sight in the direction of the acceleration the error is equal to the angular displacement of the bubble due to the acceleration, and the observed altitude is too great.

(*b*) For a sight at 180° to the direction of acceleration the error is again equal to the angular displacement of the bubble but is opposite in sign, giving an observed altitude that is too small.

(*c*) Between these two extremes the size and sign of the error varies with the relative bearing of the sight, being approximately zero at 90° to the direction of acceleration.

42. The bubble will also be displaced by Coriolis acceleration. A correction is tabulated in the Air Almanac and A.P. 3270.

43. Random accelerations can be kept to a minimum by careful flying. The effect of those remaining is minimized by the averaging device.

Operation

44. **To Obtain a Bubble:—**

(*a*) Select a suitable shade (*e.g.* No. 3), check that the window cover plate is open and wind up the clockwork. At night, select shade No. 0, close the cover plate, wind up the clockwork and push the lighting trigger switch inwards.

(*b*) Tilt the sextant to the right and turn the bubble control with the left thumb in a clock-

wise direction until a bubble appears in the bubble chamber.

(*c*) Adjust the size of the bubble by varying the tension on the bubble control.

(*d*) Tilt the sextant to the left and release the tension on the bubble control.

(*e*) Check that the bubble remains in the chamber.

45. **To Remove a Bubble:—**

(*a*) Tilt the sextant to the right until the bubble appears at 10 o'clock and exert a slight clockwise tension on the bubble control.

(*b*) Rotate the bubble control in an anti-clockwise direction until the bubble disappears.

46. **Pre-flight Checks.**

(*a*) Ensure that the mirrors, matt perspex window and eyepiece are clean.

(*b*) Check the operation of the 5° increase mirror.

(*c*) Wind up fully the automatic attachment.

(*d*) Check that the counters have zeroed.

(*e*) Check the operation of the coarse setting control.

(*f*) Check the operation of the fine setting control.

(*g*) Check the operation of the sun shades.

(*h*) Obtain a bubble.

(*j*) Check that the sextant case contains spare batteries, bulbs and an adaptor.

(*k*) Select shade No. 7.

(*l*) Position the aperture disc to the largest clear opening. (This lies directly opposite the largest red filter opening.)

(*m*) Push the trigger switch outwards and check that the right handle lamp illuminates.

(*n*) Push the trigger switch inwards and, by holding the sextant away from any strong source of light, check through the eyepiece that the bubble lamp illuminates.

(*o*) Remove the bubble.

(*p*) With the 5° increase out adjust the fine setting knob so that 8° 00′ is set on the counters.

(*q*) If a Mk. 9BM sextant is being used select 2 minutes run.

(*r*) Depress the starting trigger and start timing.

(*s*) Look through the eyepiece and when the shutter comes down stop timing.

(*t*) The timing period should be 2 minutes ± 10 seconds.

(*u*) The averaging counters should read 8° 00′ ± 2′.

(*v*) If using a Mk. 9BM sextant, wind the clockwork and check the timing on a one minute run.

(*w*) The timing period should be 1 minute ± 5 seconds.

(*x*) Make a note of half the running time.

(*y*) Check the serviceability of the two eyepieces.

47. **Taking a Sight in the Air.**

(*a*) Wind up fully the automatic attachment and if using a Mk. 9BM sextant select one or two minute run.

(*b*) Set the fine and coarse setting knobs to the body's calculated altitude.

(*c*) Set the 5° increase as required for (*b*).

(*d*) Obtain a bubble.

(*e*) Adjust shades and aperture disc as necessary.

(*f*) Suspend the sextant in the dome.

(*g*) Give the pilot a preliminary warning.

(*h*) Direct the sextant towards the body maintaining the bubble in the centre of the chamber.

(*j*) At night, press the trigger switch inwards.

(*k*) Adjust the coarse setting knob until the body is coincident with the bubble.

(*l*) Engage the coarse setting knob so that the body is above the bubble.

Note.—Steps (*k*) and (*l*) apply to non-pre-calculated observations only.

(*m*) If the body goes out of the field of view, use the 5° increase.

(*n*) Inform pilot "Taking a sight—steady".

(*o*) Adjust the fine setting knob until the body is coincident with the centre of the bubble.

(*p*) Depress the starting trigger.

(*q*) By rotating the fine setting knob maintain the body in the centre of the bubble.

(*r*) When the shutter comes down, note the time.

(*s*) Inform the pilot "Sight taken".

(*t*) Note the average altitude.

(*u*) Note the dome refraction.

(*v*) Subtract half the running time from the time noted in (*r*) above.

(*w*) Log the time, altitude and dome refraction.

(*x*) If no more sights are to be taken remove the sextant from the astro-dome.

48. **Post-flight Checks.**

(*a*) Remove any bubble and release the tension on the bubble control knob.

(*b*) Make sure that the clockwork is not wound up.

(*c*) Stow the sextant in its carrying case.

49. **General Precautions.** The following precautions should be observed at all times:—

(*a*) Wind the automatic attachment fully before using the sextant.

(*b*) Always wind gently or the minute counter may be thrown past zero.

(*c*) Do not attempt to rewind the attachment until after the clockwork has run down.

(*d*) Never attempt a sun observation with zero showing on the sun shade selector indicator. Start with shade No. 3 and then adjust shades to desired intensity.

(*e*) Always release the tension on the bubble control knob after use.

(*f*) Keep the observed body concentric with the bubble during the entire run of the attachment.

(*g*) If there is any moisture on the cellophane eyepiece cover, allow it to dry in air. Do not wipe it with fingers or a cloth, or the eyepiece will become smeared, rendering it unserviceable.

(*h*) Apparent failure of the bubble lighting may result from:—

(i) Failure to wind the attachment.

(ii) The aperture disc being incorrectly positioned. A spring-loaded ball engages indentations in the disc when it is correctly positioned but it is possible to set the disc between the positions.

Depression Sights

50. The bubble sextant may be used for obtaining plan range position lines by measuring the depression of a natural feature from the observer. The coarse setting knob is set to D (a depression of 10°) and, with all shades removed, a one-minute sight is taken on the feature. The recorded angle of false elevation is subtracted from 10° to give the true depression angle, and the range is calculated on the circular slide rule as follows:—

$$\frac{\text{Relative Height}}{\text{Depression}} = \frac{\text{Range}}{9.4}$$

In this equation, relative height is in thousands of feet above the natural feature; depression is in degrees; 9.4 is a constant which takes account of the conversion of feet to nautical miles, and the equality of the tangent of a small angle to its value in radians; range is in nautical miles.

Note.—Depression sighting requires considerable practice, notably in the manipulation of the shades to render both bubble and object visible simultaneously.

Part 4 Sect. 1
Chap 2

CHAPTER 2.

PERISCOPIC SEXTANT, MARK 2

CONTENTS

	Paras.
Introduction	1–6
SEXTANT	
Principle	7
Bubble System	8–13
Altitude Measurement	14
Altitude Control	15–16
Automatic Averaging Attachment	17–19
Azimuth Measurement	20–21
Shade System	22
Scale Lighting	23
Eyepiece	24
SEXTANT MOUNTING	
General Construction	25–28
Sequence of Operation	29–32
Electrical Supplies	33–36
CASE AND TRAY	
Case	37–38
Tray	39
OPERATION	
To Insert Sextant in Mounting	40
Pre-Flight Checks	41
Periodic Checks	42–44
Taking a Sight in the Air	45–46
To Withdraw Sextant from Mounting	47
After Landing	48
ERRORS	
Instrumental Errors	49
Tilt and Acceleration Errors	50

Introduction

1. The Periscopic Sextant, Mk. 2, is intended for use in aircraft having pressure cabins or in which the drag caused by an astrodome is unacceptable. The instrument is dual purpose, having facilities for both determination of aircraft heading and obtaining astro-position lines.

2. The periscope tube of the sextant fits into a mounting which is attached to the inside of the aircraft skin. The sextant can then be raised or lowered at will through a small hole in the top of the fuselage. The sextant periscope moves in a sleeve in the mounting and forms an air-tight seal, which enables cabin pressure to be maintained.

3. When the sextant is required for use, it is put into the raised position with the periscope tube projecting only three inches from the skin of the aircraft. Otherwise the instrument is retracted into the lowered position. De-misting and de-

icing equipment enables the sextant to be operated at all temperatures likely to be met operationally.

4. The sextant uses a bubble level to provide a datum from which the altitude of a heavenly body can be measured, but no provision is made for observing the natural horizon as an alternative to the bubble horizon. The sextant has a field of view of 12° both in azimuth and elevation and a magnification factor of 2. Because of the narrow field of view, the object to be observed has to be identified by setting its precomputed altitude and azimuth on the sextant. The object should then appear approximately in the centre of the field of view. Great care must still be taken, however, to ensure that the object being observed is the required one, as the accuracy of the azimuth setting depends on the accuracy with which the true heading of the aircraft is known. If for any reason the heading set on the sextant is different from that which the aircraft is flying, the wrong object may be observed.

5. To overcome the inherent difficulties of obtaining accurate single sights using a bubble system, a clockwork averaging device is fitted.

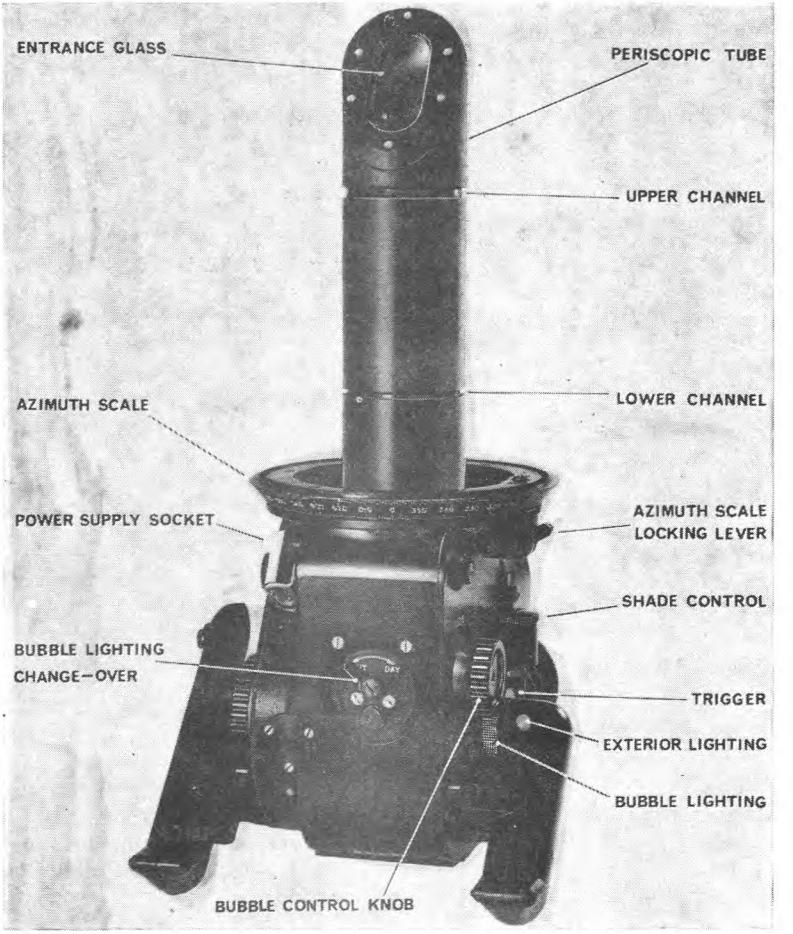


Fig. 1. Periscopic Sextant, Mk. 2—Left Front View

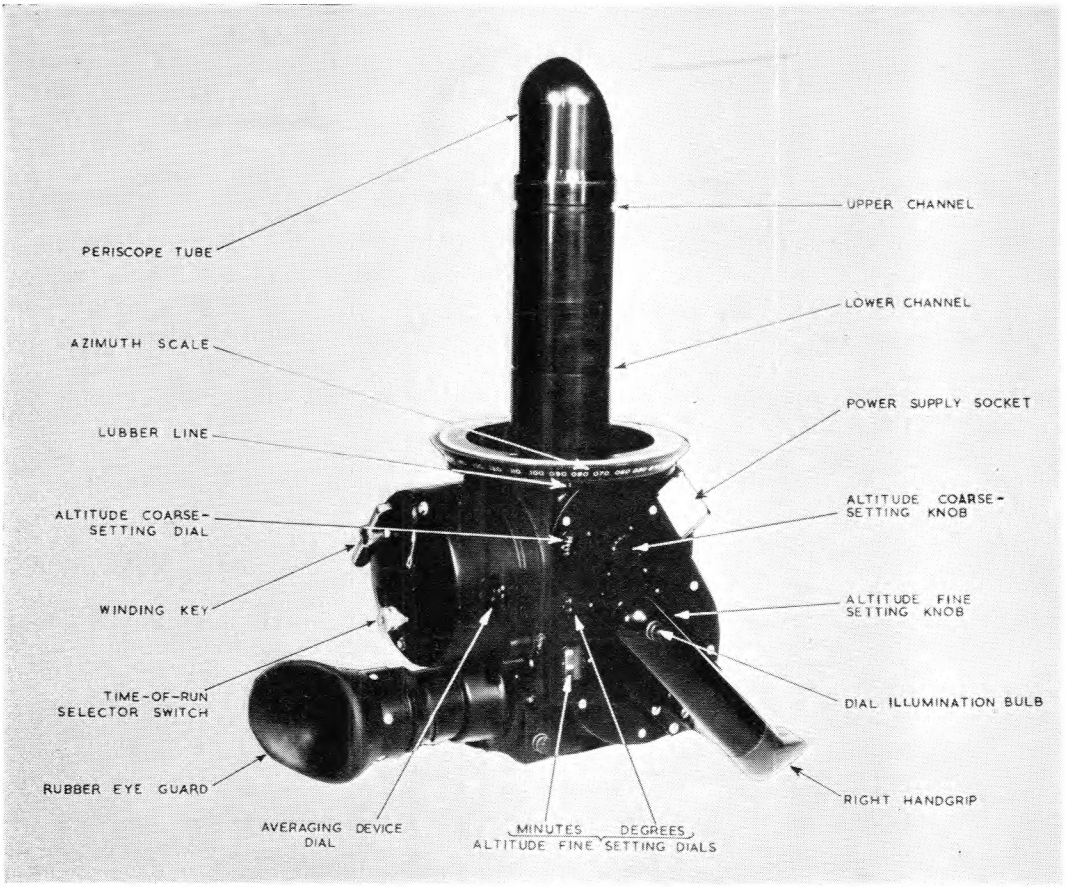


Fig. 2. Periscopic Sextant, Mk. 2—Right Rear View

It is similar to that used in the Mk. 9 series of sextants and averages 60 readings over a period of one or two minutes.

6. The complete sextant installation consists of:
(*a*) Sextant.
(*b*) Sextant mounting.
(*c*) Case and tray.

SEXTANT

Principle

7. The sextant makes use of the two basic principles of the bubble sextant:—

(*a*) The establishment of a datum from which the altitude may be measured; this is done by using a bubble levelling system.

(*b*) The angular measurement of the altitude of the body from this datum by means of the measuring prism.

The applications of these two principles are discussed separately in paras. 8 to 13 and 14 to 16.

Bubble System

8. The method of establishing the horizontal by the use of a bubble has been dealt with in Chapter 1 of this section.

9. The bubble system is shown diagrammatically in Fig. 3 and consists of:—

(*a*) An illuminating lamp.

(*b*) A bubble chamber.

(*c*) Projection lenses.

(*d*) A bubble prism.

The light from the illuminating lamp is reflected through the bubble chamber, the image so formed being erected, in the field of view of the eyepiece, by the bubble projection lenses and the bubble prism.

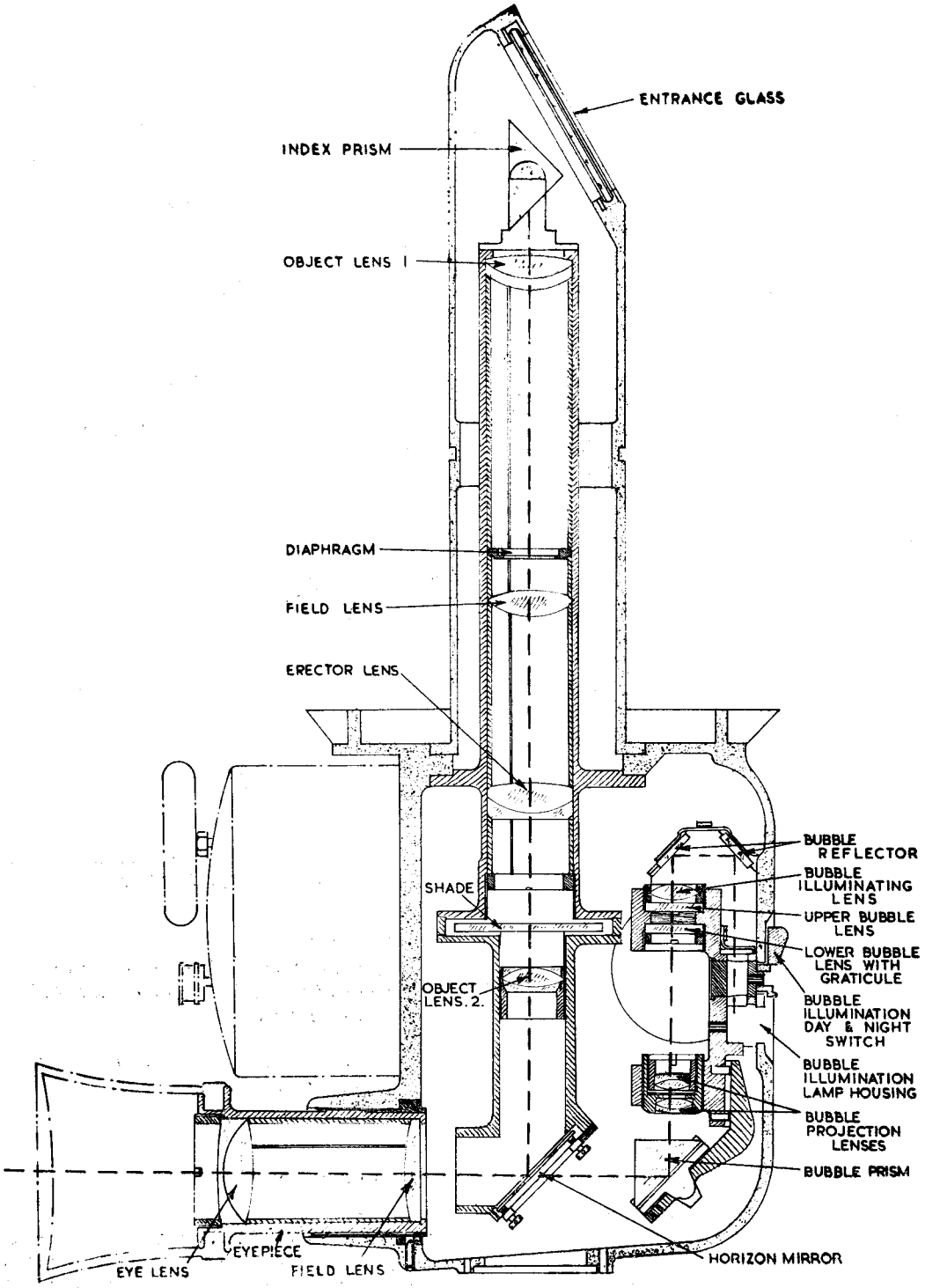


Fig. 3. Periscopic Sextant, Mk. 2—Optical System

10. The bubble chamber and its controls are similar to those used in the Mk. 9 series sextants, and are described in Chapter 1 of this section.

11. The bubble, formed by tilting the sextant towards the operator and increasing the diaphragm tension by means of the bubble control knob, enters the chamber at the top of the field of view. When a bubble of suitable size has been formed the sextant is tilted sharply in the opposite direction and the tension on the diaphragm decreased. The amount of diaphragm tension required to keep the bubble of constant size is a matter of experience. To remove the bubble, the diaphragm tension is increased, the sextant is tilted towards the operator, and all tension on the diaphragm removed.

12. Bubble illumination is by artificial light at all times, the type of illumination being controlled by a two-position switch on the front of the sextant (Fig. 1). With the switch on DAY, the bubble is seen as a dark ring against a bright background. When the switch is on NIGHT, two horizontally disposed red spots are seen against a dark background; the body should then be placed on an imaginary line joining these two spots. The degree of illumination of the bubble is varied by means of a rheostat, controlled by a knob at the top of the left handgrip (*see* Fig. 1).

13. Two parallel lines are engraved on the upper glass of the bubble chamber, and appear in the field of view as vertical *tram-lines* disposed equally about the centre. These lines are visible both with day and night illumination and, provided the bubble is kept between them when a sight is being taken, the sideways tilt of the sextant will not cause any appreciable error in altitude measurement.

Altitude Measurement

14. The optical system used to place the image of the celestial body in the field of view of the observer is shown in Fig. 3. An optically flat entrance glass is sealed in the outer casing at the top of the periscope tube, at an angle of 30° to the vertical. Light from the celestial body is reflected by the rotatable index prism down the periscope tube, through the lens system and a progressively variable shade, until it is reflected by the horizon mirror to form an erect image in the field of view of the eyepiece.

Altitude Control

15. To obtain coincidence between bubble and star, the index prism is rotated in the vertical plane. This rotation is performed in two stages:—

(*a*) In 10-degree steps over a range of —10° to +80°, by means of a pin and hole plate assembly, the rotation of which is transmitted to the index prism by a pair of push rods. The 10-degree steps are selected by the altitude coarse-setting knob on the right-hand side of the sextant and indicated by the coarse setting scale in the adjacent window.

(*b*) The altitude fine control wheel works a micrometer screw and lever, causing rotation of the pin and hole plate assembly over a range of 13° 30′. The fine altitude setting is indicated in degrees and minutes, in two windows below that of the coarse setting scale. The value of the fine setting is added to that of the coarse setting to obtain the altitude of the body.

16. When observing the sun, certain combinations of altitude setting and sextant tilt may cause a spurious image or *ghost* to appear in the field of view. The use of normal precomputation methods should ensure that the ghost is never observed in place of the real image. In addition, the ghost may be distinguished from the real image in the following ways:—

(*a*) The ghost is dimmer than the real image. If the shade is inserted progressively, the background to the ghost will remain bright as long as the ghost is visible, whereas with the real image, the background becomes completely dark long before the image disappears.

(*b*) The ghost is usually blue in colour.

(*c*) The ghost moves in the same direction in the field of view as the real image, but at twice the speed.

Automatic Averaging Attachment

17. The automatic averaging attachment is situated at the back of the sextant, above the eyepiece, with its starting trigger below and to the left (*see* Figs. 1 and 2). It is similar in operation to that used in the Mk. 9BM sextant (but its range is 0° 15′ to 13° 30′), is clockwork-driven, and is wound by a key at the back of the mechanism housing.

18. The averaging attachment automatically records 1/60th of the altitude, as indicated by the index prism setting, once every two seconds for a two-minute period, or once every second for a one-minute period, and gives the total of the 60 recordings on counters. The winding of the clockwork mechanism returns the counters to

zero, whilst the period of run is controlled by a selector switch on the averaging attachment case.

19. The clockwork mechanism is linked to the bubble lighting circuit so that the bubble is illuminated only when the averaging attachment is fully wound. At the end of each run the bubble lighting circuit is again broken and the bubble blacked out. A check on the functioning of the averaging attachment is also provided by the momentary cutting off of the bubble illumination after five seconds on the one-minute run, and after ten seconds on the two-minute run.

Azimuth Measurement

20. A graduated azimuth ring, fitted around the lower part of the periscope tube at its junction with the sextant body, carries a scale calibrated every degree, with values shown every whole 10°. The scale can be read to an accuracy of ½°.

21. By means of a small lever on the left side of the sextant (*see* Fig. 1), the azimuth ring may be locked either:—

(*a*) To the sextant, for heading checks, when the scale is read against the lubber line marked on the mounting.

(*b*) To the mounting, for setting and measuring azimuths, when the scale is read against the lubber line marked on the sextant.

Note.—For ease of reading of relative azimuths and bearings, the lubber lines on both the sextant and the mounting are offset from the fore-and-aft axis of the aircraft by approximately 135°. The azimuth scale is offset by the same amount so that correct readings are obtained.

Shade System

22. A shade system is fitted to the sextant to allow the operator to control the brightness of the celestial body, as seen through the eyepiece. The shade is a photographically darkened disc rotating horizontally immediately below the periscope lens system (*see* Fig. 3), the setting of which is controlled by a horizontal knob on the left side of the sextant (*see* Fig. 1). A click stop indicates when the shade is fully out, in which position there is no light loss as the light rays pass through a Vee cut-out in the disc.

Scale Lighting

23. Lighting for the various scales of the sextant is controlled by a white press-button switch on the left handgrip. When the switch is out, the bubble chamber is illuminated and the scales are unlit. When the switch is depressed, the bubble chamber illumination is cut out and a pea-lamp on the right handgrip is lit, so illuminating the scales through suitable slots cut in the handgrip. Also, by means of a perspex rod, a beam of light is projected downwards from the right handgrip to illuminate the operator's watch.

Eyepiece

24. The eyepiece is of fixed focus and has a rubber eyeguard which can be rotated so that the body may be viewed with either eye. A concertina-type eyeguard is also available.

SEXTANT MOUNTING

General Construction

25. The mounting (*see* Fig. 4) is designed to be fitted to the inside of the fuselage skin. Its principal functions are:—

(*a*) To locate the sextant in either the operating or retracted position.

(*b*) To prevent appreciable loss of cabin pressure.

The mounting has a sleeve or carrier tube into which the sextant periscope fits (*see* Figs. 5 and 6). The tube terminates at its upper end in a ball and socket joint which permits ±8° tilt in any direction from the vertical but does not allow rotation of the carrier tube in azimuth.

26. Directly above the upper end of the carrier tube is a sealing plate, which is withdrawn from its seating and swung back into the body of the mounting by operation of the lever on the side of the mounting. Interlocking safety catches are provided to prevent the sealing plate from being opened until the sextant has been inserted in the carrier tube. They also prevent the sextant from being withdrawn from the carrier tube until the sealing plate is closed.

27. At the lower end of the carrier tube is a flat index plate, carrying a lubber line. The index plate is rotatable and can be set in the fore-and-aft position and locked. Above this plate is a pair of finger catches, which hold the sextant in either the operating or retracted position; operation of these catches also allows the sextant to be removed from the carrier tube, provided that the sealing plate is closed.

28. Any condensation inside the mounting can be drained off by means of a drain plug on the underside of the mounting.

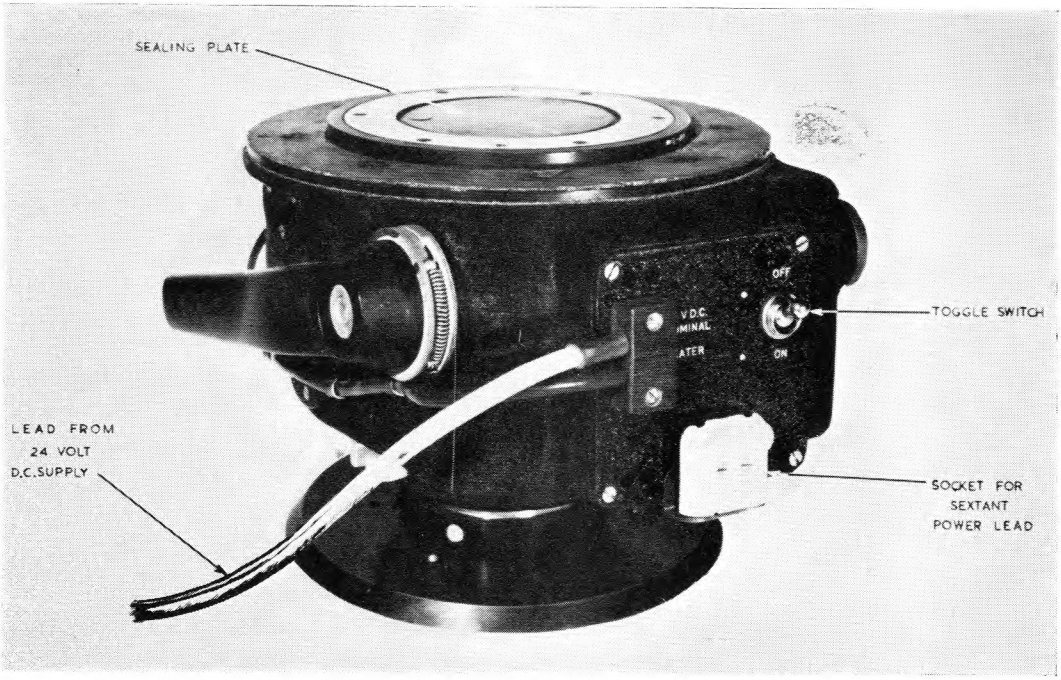


Fig. 4. Mounting, Showing Electrical Connector Plate

Sequence of Operation

29. Embodied in the lower end of the carrier tube are three types of catches which have the following functions:—

(*a*) To retain the sextant in the retracted position.

(*b*) To retain the sextant in the operating position.

(*c*) To ensure that the sealing plate cannot be opened until the sextant is inserted in the carrier tube; and that the sextant cannot be removed from the carrier tube until the sealing plate is closed.

30. As the periscope of the sextant is slid into the carrier tube of the mounting, the finger catches engage in the upper channel around the periscope (*see* Fig. 1) and retain the sextant in the retracted position. The action of inserting the periscope into the carrier tube automatically depresses one of the safety catches, thus permitting the operating lever on the side of the mounting to be pulled downwards. Operation of this lever withdraws the sealing plate from its seating and swings it back into the body of the mounting. At the same time, another safety catch is operated. It is shaped so as not to impede the upward movement of the periscope tube, but to engage with the upper channel in the tube and prevent complete withdrawal.

31. To raise the sextant to its operating position the finger catches are pressed together, the sextant raised, and the finger catches released so that they engage in the lower channel of the periscope tube.

32. To retract the sextant, the finger catches are again operated and the sextant is withdrawn until the safety catch and the finger catches engage in the upper channel of the periscope tube. In this position the sextant cannot be entirely withdrawn from the carrier tube until the operating lever is pushed upwards and the sealing plate closed. As the lever moves upwards, the safety catch is withdrawn and the sextant is then retained only by the finger catches, operation of which enables the sextant to be withdrawn completely.

Electrical Supplies

33. At the rear of the mounting is the electrical connector plate (*see* Fig. 4). To this is connected the aircraft 24-volt d.c. supply, which is fed through a toggle switch to:—

(*a*) The mounting heater,

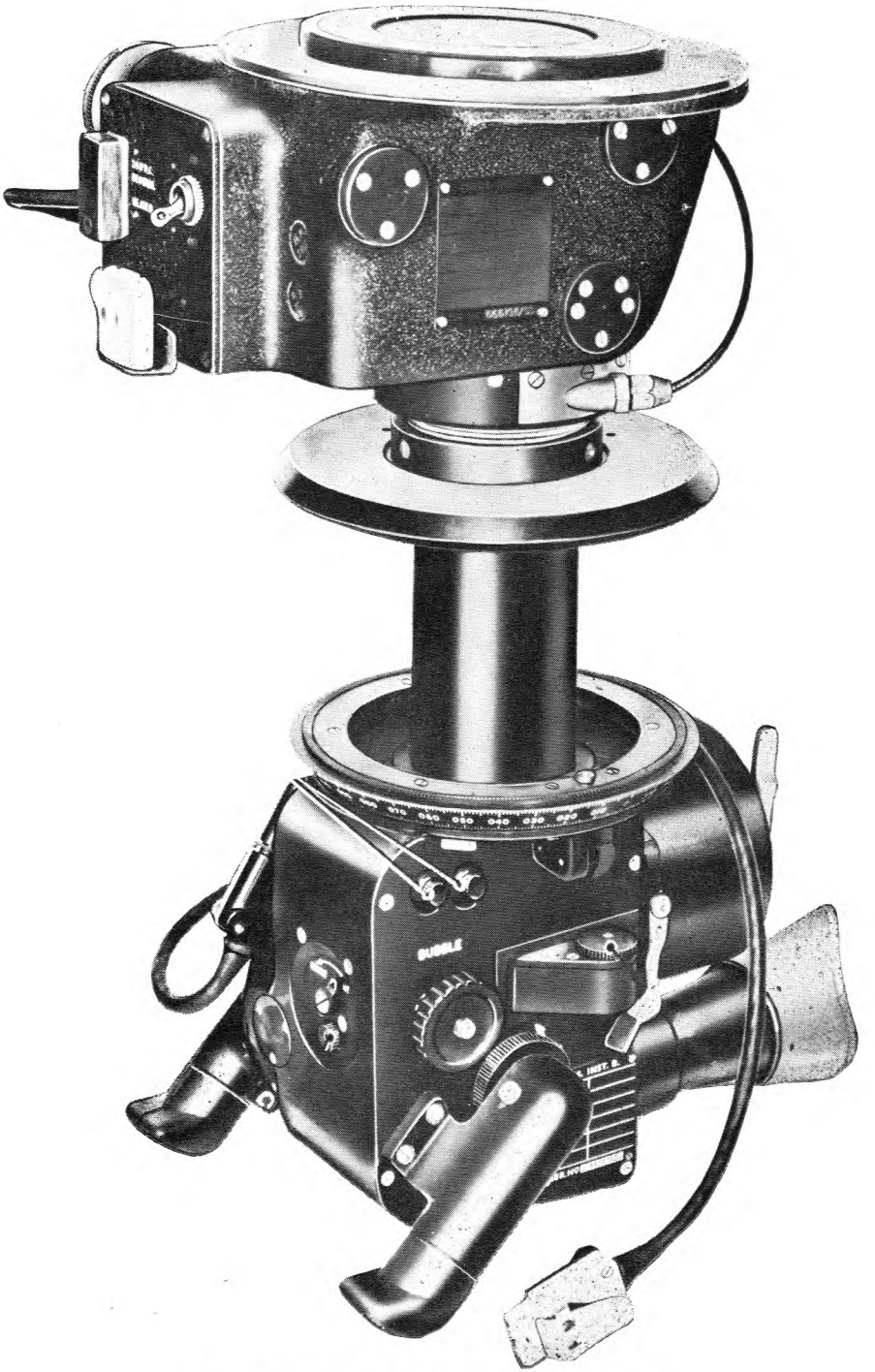


Fig. 5. Mounting, with Sextant in Retracted Position

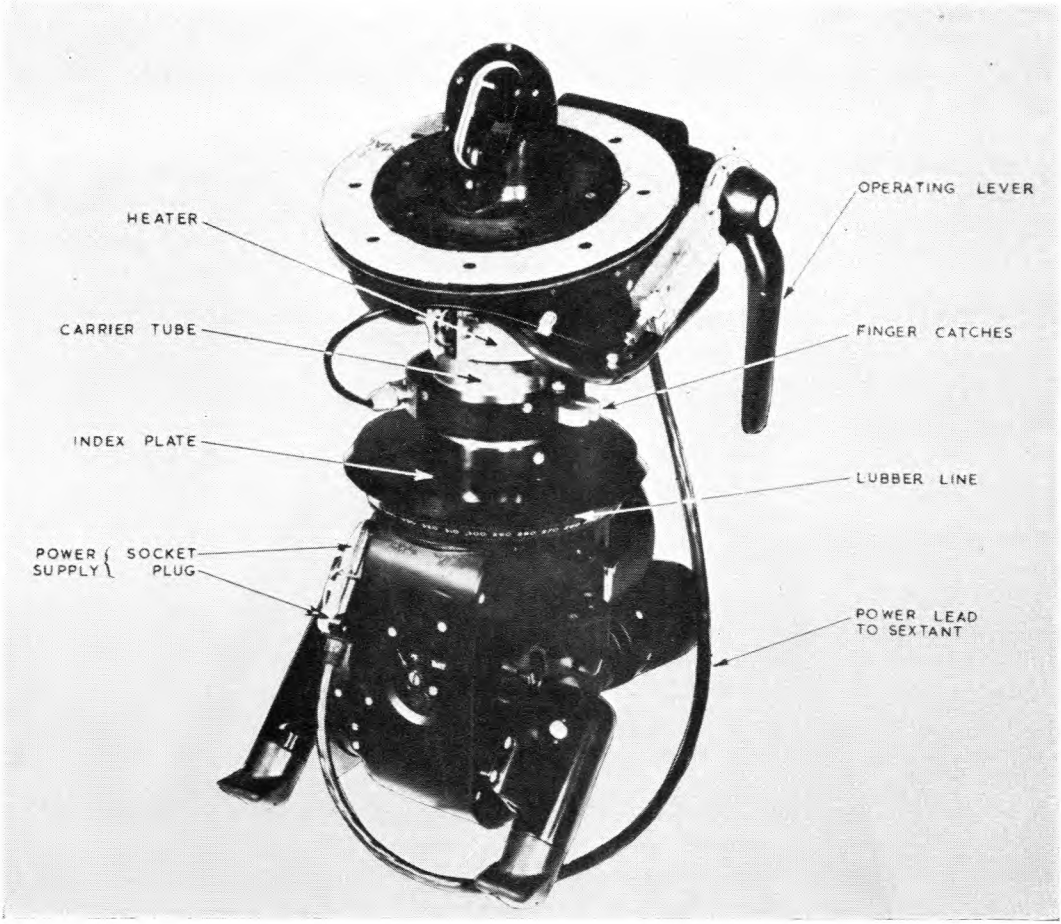


Fig. 6. Mounting, with Sextant in Operating Position

(*b*) The sextant, via a plug-in lead.

34. The mounting heater is fitted around the carrier tube (*see* Figs. 5 and 6), and the heat generated is sufficient to keep the sextant free from icing in the carrier tube, but is not intended to prevent the formation of rime ice on the body of the mounting. It is essential that the heater be switched on immediately before take-off, as the heat generated is insufficient to free the mounting once it has frozen up. The heater should be switched off on landing.

35. To prevent the formation of mist or frost, the window at the top of the periscope tube is electrically heated. Power for this heater and for the sextant lamps is obtained from the electrical connector plate through a connecting lead, the two ends of which plug into the sextant and the mounting respectively. To ensure that the correct connections are made, one side of each plug and each socket is coloured white and the other side black, so that each plug and socket is matched white to white. The lead is long enough to allow the sextant to be turned in the mounting by more than 360°.

36. As the averaging mechanism is prone to failure, and errors increase, at temperatures below −2°C, a heater is installed in the averager.

CASE AND TRAY

Case

37. The case (*see* Fig. 7) has been designed to carry the sextant when it is not in the aircraft; and to provide a storage for it on the flight deck. The front of the case forms a door which may be

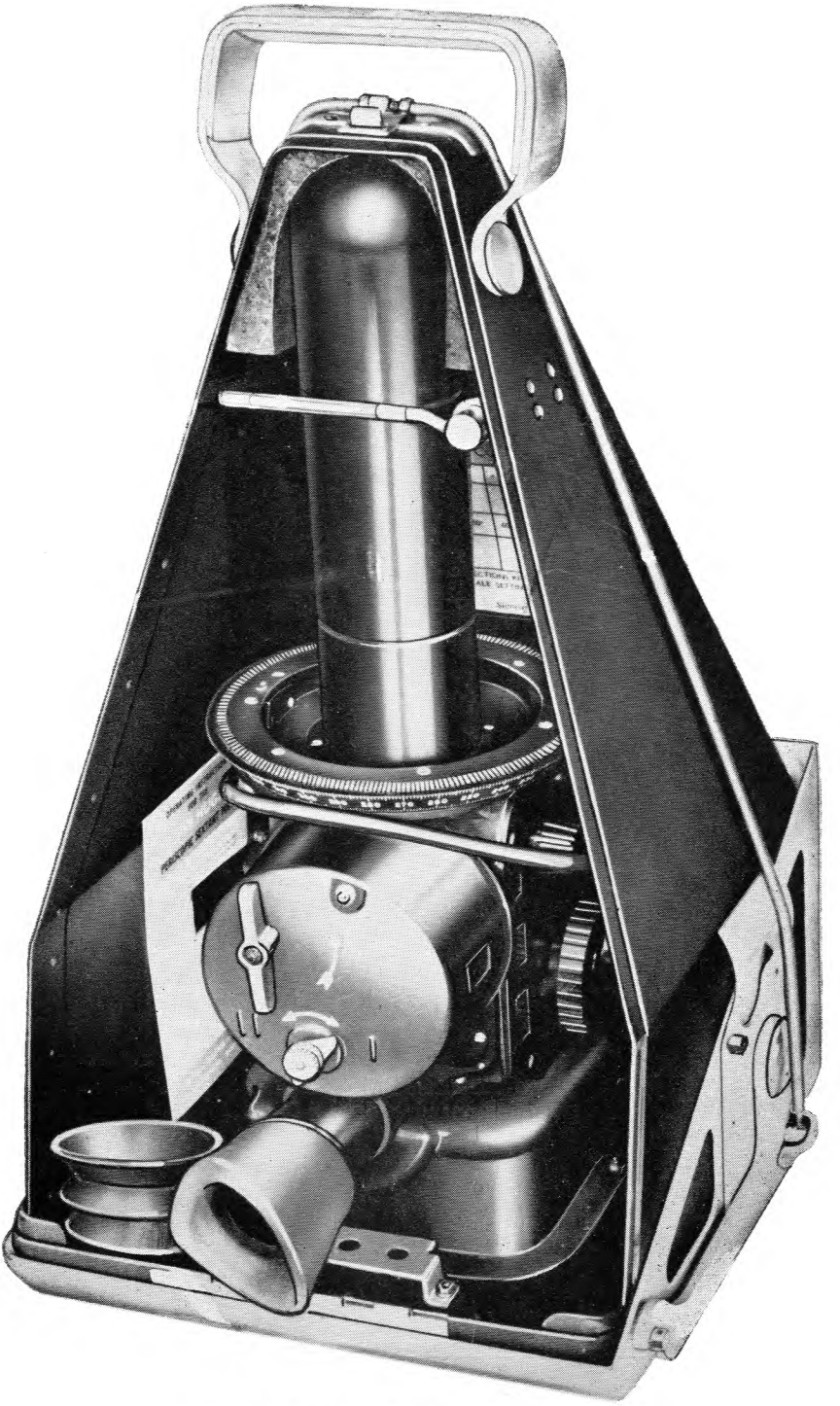


Fig. 7. Case, with Sextant in Stowed Position

lowered on a hinge, the hinge pin being removable to enable the door to be detached, if so required. A clip inside the door holds the hinge pin when it is not in use.

38. The sextant fits into a sponge rubber moulding on the floor of the case, and the periscope tube into another moulding at the top. The action of closing and securing the case door automatically secures the sextant inside the case but if the door is detached, the sextant can be secured by a clip which engages in the upper channel of the periscope tube.

Tray

39. The tray, which is a light-alloy pressing and is intended to hold the case when it is stowed in the aircraft, is permanently fixed on the flight deck in a position convenient to the navigator. Attached to a lever on the right-hand side of the tray is a cable whose other end terminates in a spring, housed on the left-hand side of the tray. The case is secured to the tray by this cable, a locating ring in the centre of which engages with the door catch on the case in such a manner that the locking of the case door also locks the case to the tray.

OPERATION

To Insert Sextant in Mounting

40. To insert the sextant into the mounting the following procedure should be followed:

a. Slide the sextant into the mounting carrier tube until the finger catches engage. The sextant will then be held in the retracted position.

b. Plug the sextant lead into the sextant and mounting sockets (white to white).

c. Pull the operating lever fully downwards to open the sealing plate.

d. Press the finger catches together and slide the sextant a little farther into the carrier tube. Release the finger catches.

e. Continue to slide the sextant gently into the carrier tube until the finger catches re-engage. The sextant is then held in the operating position.

Note: The sextant must be held firmly whilst it is being raised to the operating position. If this is not done, and there is a pressure difference between the cabin and the outside air, the sextant may be forced violently into the mounting, resulting in damage to the sextant and also, probably, injury to the operator.

Pre-Flight Checks

41. The pre-flight checks are as follows:

a. Check for superficial damage, cleanliness, and availability of spare bulbs and of electrical lead.

b. Insert sextant in mounting, wind averaging attachment, make electrical connection, Switch ON/OFF switch to ON and put DAY/NIGHT switch to appropriate position.

c. Form bubble, check illumination and lighting.

Note: The bubble adjusting knob must not be rotated to the extremes of its movement, but should be limited to some $1\frac{1}{4}$ turns at the centre of its range of travel.

d. Check shades "out" and carry out alignment check. (Details of the check for the aircraft type are given on a card in the sextant carrying case.) Note any correction.

e. Set automatic averaging attachment to 1 or 2 minutes, whichever is to be used during the flight. Set fine and coarse setting controls to mid-range positions.

f. Release averaging attachment control and check run-down time. It should agree with the calibration time given in the carrying case to within 5 seconds.

g. Compare final averaged reading with the setting of the fine setting control. They should agree to within 3 min of arc.

Periodic Checks

42. These periodic checks are the navigator's responsibility only when he holds the sextant on permanent personal loan; at other times they will be carried out by competent technical personnel.

43. **Checks to be Carried out Every Three Months.** At Intervals not exceeding three months the standard serviceability test given in AP 112B–0501–16, Chap 1–2 is to be carried out.

Taking a Sight in the Air

44. **Altitude Observations.**

a. Ensure that the power supply switch on the mounting is on and the sextant lead in position.

b. Select either the one-minute or two-minute run according to the required length of observation.

c. Fully wind the averaging attachment.

d. Set the shade control to a suitable position and the daylight lighting to maximum. Form bubble.

e. Using the coarse setting and fine setting controls, set the pre-computed altitude of the body on the sextant.

f. Rotate the azimuth ring until true heading is set against the mounting lubber line.

g. Clamp the azimuth ring to the mounting by placing the locking lever in the up position.

h. Rotate the sextant until its index mark is set to the pre-computed azimuth of the body.

j. Re-adjust the shade density and bubble lighting to suit viewing conditions.

k. Move sextant to position bubble between tram-lines in middle of field of view, and rotate fine altitude control to bring body and bubble into coincidence.

l. Press starting trigger and commence observation. Throughout observation maintain coincidence of bubble and body by operation of the fine altitude control.

m. When the bubble lighting is finally extinguished, note time. Calculate the mid-time of run.

n. Add the values shown by the averaging attachment counters and the coarse setting scale to obtain the average altitude.

45. **Azimuth Observations**

a. Fully wind the averaging attachment.

b. Set the shade control to a suitable position and the daylight lighting to a maximum. Form bubble.

c. Make heading checks:

(1) Calculate expected true azimuth and altitude of body for a given time ahead.

(2) Set pre-computed altitude of body by means of coarse and fine setting controls. Rotate the azimuth ring until pre-computed true azimuth of body is set against sextant lubber line.

(3) Clamp the azimuth ring to sextant by placing the locking lever in the down position.

(4) Rotate the sextant until the approximate true heading is shown against the mounting lubber line.

(5) Re-adjust the shade density and bubble lighting to suit viewing conditions.

(6) Move the sextant to position the bubble between tram-lines. Bring body and bubble into concidence.

(7) Read off true heading against mounting lubber line.

Note: Because of the narrow field of view of the sextant, great care must be taken to ensure that the body viewed is the correct one. This precaution also applies when obtaining the altitude of a body.

d. To measure range and bearing:

(1) Set coarse setting control to D, and rotate azimuth ring until true heading is set against the mounting lubber line. Clamp the azimuth ring to the mounting.

(2) Adjust shade density and bubble lighting to suit viewing conditions, and rotate sextant until object, bubble and tram-lines are coincident.

(3) Note the true bearing at the beginning and end of a one-minute shot and obtain the bearing for the mid-time.

(4) Note depression (10°— altitude) and calculate' range as explained in Chap 1 of this section.

To Withdraw Sextant from Mounting

46. To withdraw the sextant from the mounting, the following procedure should be followed:

a. Remove bubble and relax tension on bubble diaphragm. Ensure clockwork is run down.

b. Disconnect the electric lead from the sextant and the mounting.

c. Press the finger catches together and withdraw the sextant slightly from the operating position.

d. Release the finger catches and gently withdraw the sextant to the retracted position.

e. Push the operating lever fully upwards. This closes the sealing plate.

f. Holding the sextant firmly, press finger catches together and withdraw the sextant from the mounting. Stow in case.

After Landing.

47. Switch off the power supply to the mounting.

ERRORS

Instrumental Errors

48. The instrumental accuracy of the sextant is:

a. Altitude Measurement.

Optical system ± 2 min Automatic averaging device ± 3 min	Overall accuracy ± 4 mins

b. Azimuth measurement. ± ½°.

Tilt and Acceleration Errors

49. Tilt and acceleration errors are discussed in Chap 1.

3456D
Part 4 Sect 1
Chap 3.

CHAPTER 3.

PERISCOPIC SEXTANTS, MARKS 2A, 2B

CONTENTS

	Paras.
Introduction	1– 2
Principle	3– 4
SEXTANT	
Pendulous Reference Unit	5
Graticule System	6
Sextant Mechanism	7
Advantages	8
OPERATION	
General	9
Inserting Sextant in Mounting	10
Pre-Flight Checks	11
Periodic Checks	12–14
Altitude Observations	15
Azimuth Observations	16
Withdrawing Sextant from Mounting	17
After Landing	18

Introduction

1. The periscopic sextants, Mks. 2A and 2B, are developments of the periscopic sextant Mk. 2, which is described in Chapter 5 of this section. In both the Mks. 2A and 2B models the bubble reference system, which is standard in the Mk. 2 sextant, has been replaced by a pendulous reference system. In the Mk. 2A sextant the eyepiece is cranked downwards to provide for the observer the extra clearance required in some aircraft installations. In this cranked construction, the optical path is bent by right-angled prisms. The cranked eyepiece may also be swivelled through 180° of the lower arc; it is additionally provided with two breathers and internal air passages to overcome misting of the eyepiece optical surfaces. Apart from the cranked eyepiece, the Mks. 2A and 2B are identical, and will be described together as the pendulous sextant.

2. Since the pendulous sextant differs only in detail from the periscopic sextant Mk. 2, this chapter does not repeat the general description of the sextant and its operation given in Chapter 5. It describes the points of difference between the two types of sextant and gives full operating drills for the pendulous sextants.

Principle

3. The purpose of the pendulous reference system is to produce the artificial horizon from which the altitude of a celestial body is measured. The system (*see* Fig. 2) contains a lamp which illuminates a graticule plate. An image of the illuminated graticule is projected by prisms on to a pendulous mirror. This mirror is suspended from a flexible nylon filament and is balanced to hang with its reflecting surface in the horizontal plane so that the normal to the mirror defines the zenith. When the sextant is level the image of the graticule reflected by the mirror appears in the centre of the field of view of the eye lens. The image of the celestial body is combined in the beam-splitting cube with that of the graticule, so that the sextant is correctly adjusted when both images appear in the centre of the field of view. A *tram-line* graticule is included in the eye lens system to assist in location in the field of view.

4. When using a bubble sextant it is necessary to correct the sextant altitudes for the effects of any acceleration, *e.g.* Coriolis, change of heading, change of airspeed. Fig. 3 shows that the effect of an acceleration upon both the bubble system and the pendulous system is essentially the same,

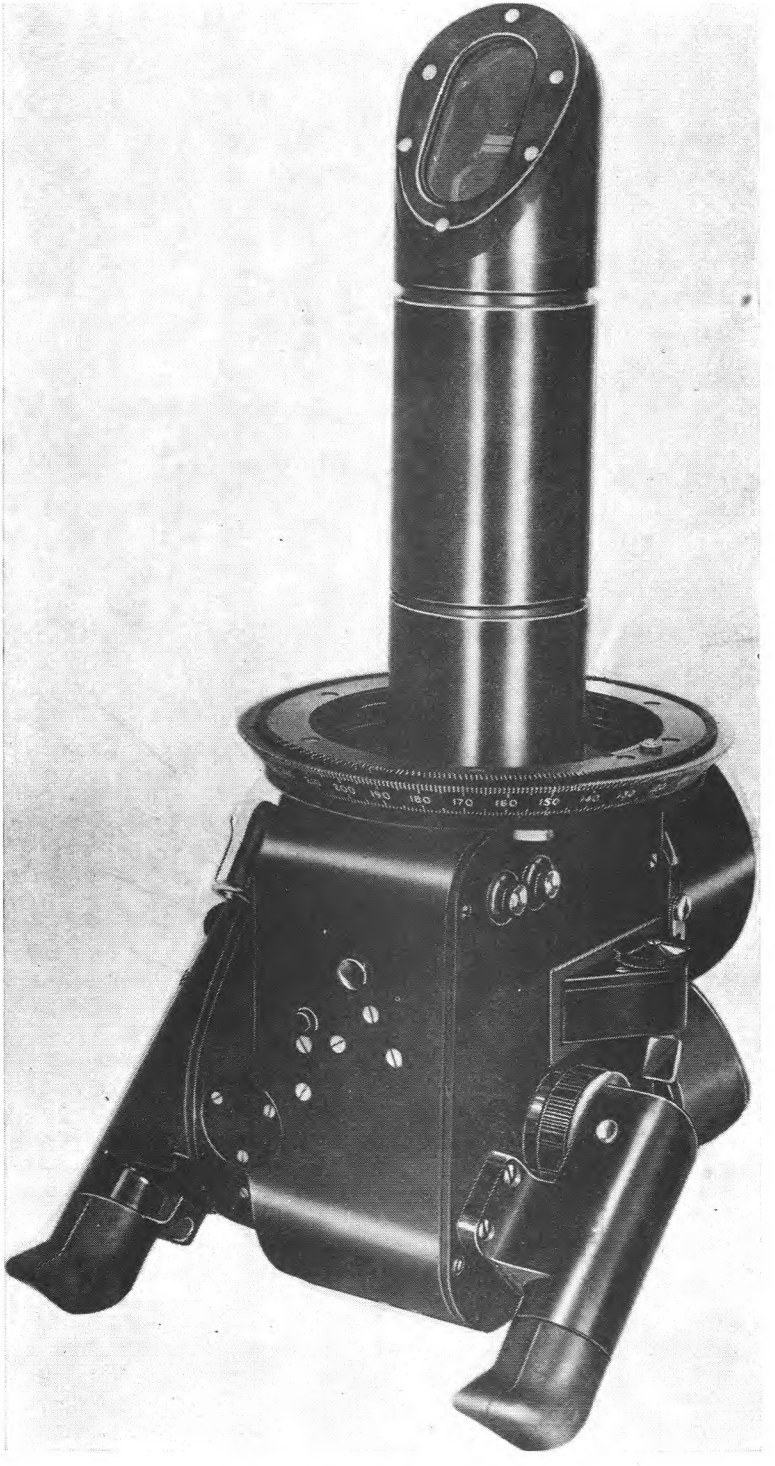


Fig. 1. Periscopic Sextant, Mk. 2B

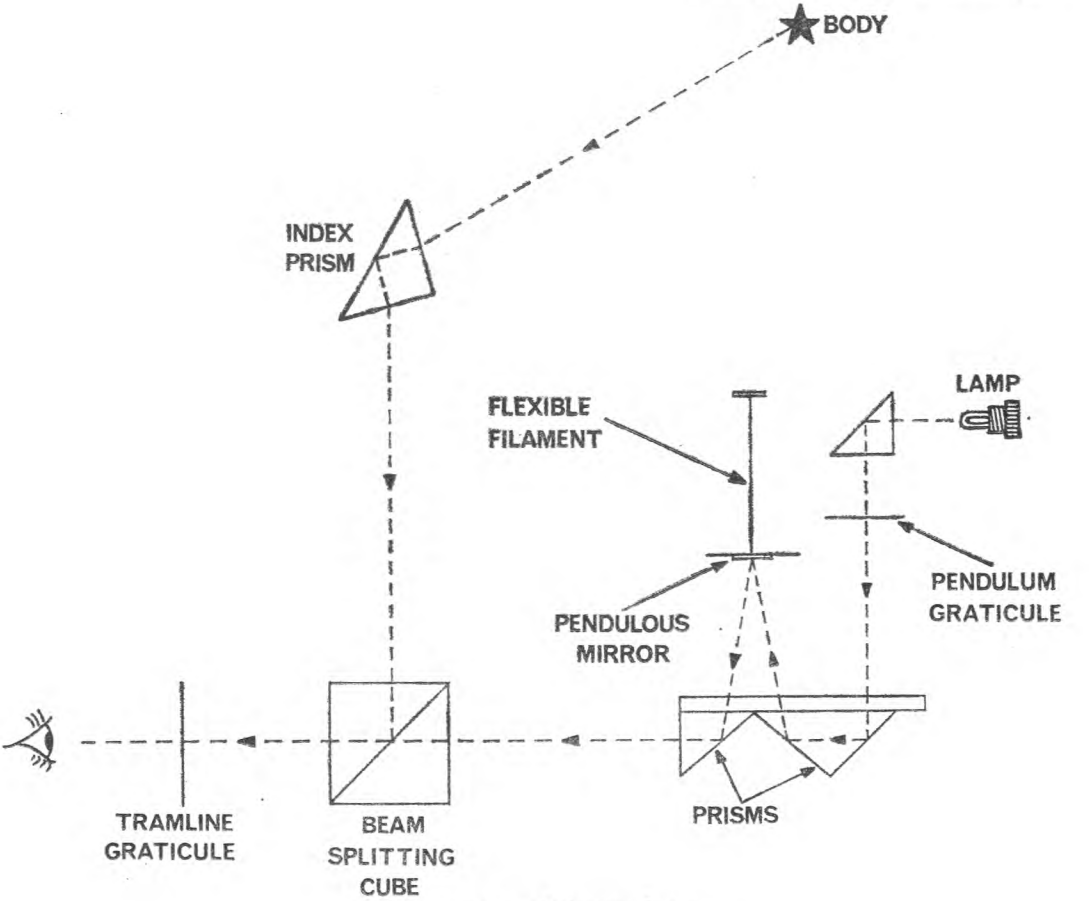


Fig. 2. Pendulous Reference System

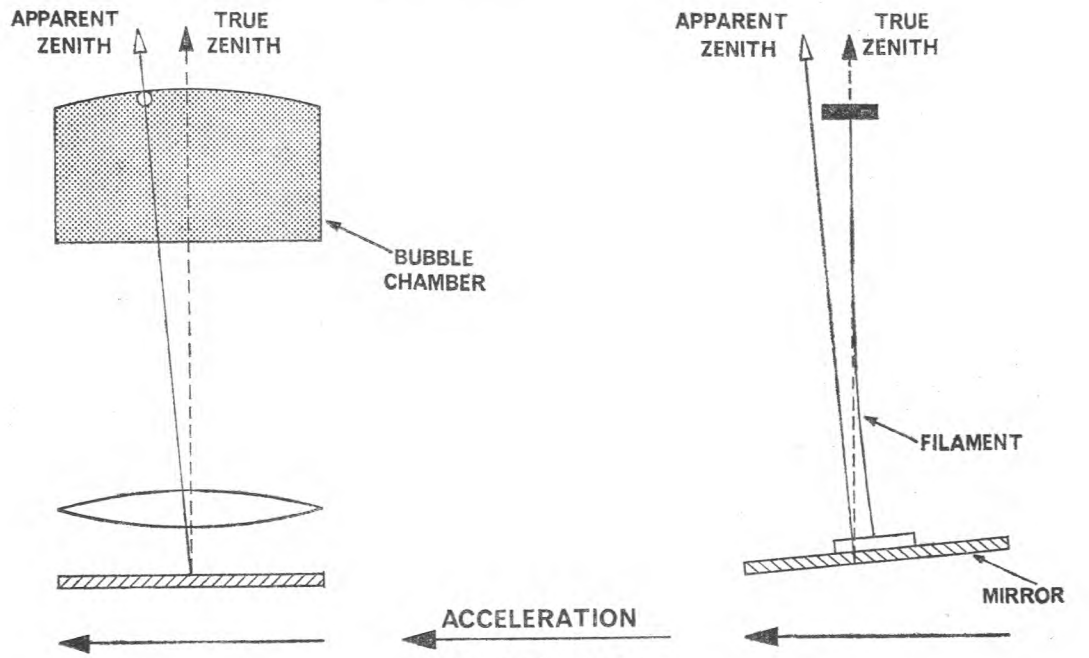


Fig. 3. Acceleration Effect

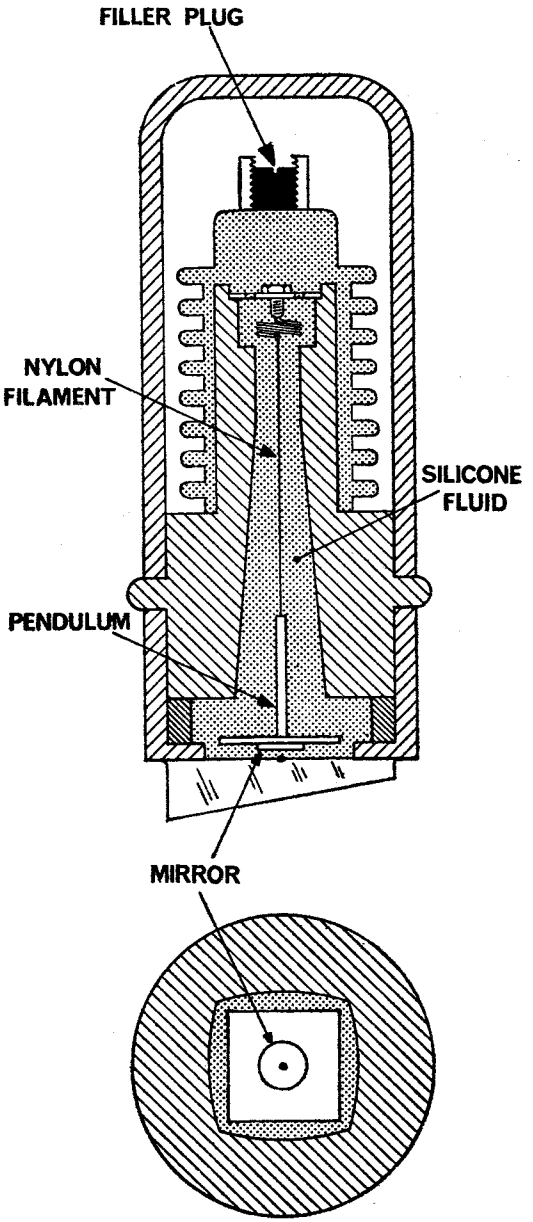


Fig. 4. Pendulous Reference Unit

and the acceleration corrections that have to be applied to a sight taken with a pendulous sextant are, therefore, identical in sign and quantity with those for the bubble sextant.

SEXTANT

Pendulous Reference Unit

5. The detail of the pendulous reference unit is shown in Fig. 4. The pendulum is suspended on a fine nylon filament and consists of a mirror, about 0.2 in. in diameter, mounted on a square plate. The movement of the plate is restricted by a guard ring, which is so shaped that the plate cannot rotate continuously. A small steel ball is let into the surface of the glass wedge beneath the mirror to prevent the mirror from becoming stuck to the glass by a vertical acceleration. The pendulum is surrounded by a silicone fluid which damps its oscillations, and the expansion bellows are incorporated to allow for changes in volume with variation in fluid temperature. Since changes of temperature alter the viscosity and refractive index of the fluid, a thermostatically-controlled heater is employed to keep the fluid temperature at a minimum of 20°C.

Graticule System

6. The brightness of the lamp, which illuminates both the pendulum graticule and the tram-lines, is regulated by the rheostat control on the left handgrip. The pendulum graticule appears as an illuminated circle 45 minutes of arc in diameter, with a split crossbar. Fig. 5 shows the graticule, tram-lines, and body, and the complete image obtained when the sextant is correctly adjusted.

Sextant Mechanism

7. The optical system of the pendulous sextant is, apart from that section associated directly with the pendulous reference unit, exactly the same as that of the Mk. 2 model. Externally there are two differences:—

(*a*) There is no bubble control knob.

(*b*) There is no day/night light switch.

Advantages

8. The periscopic sextants, Mks. 2A and 2B, possess the following advantages over the Mk. 2 model:—

(*a*) The size of the reference circle is constant and of convenient size.

(*b*) The graticule crossbars facilitate correct positioning of the celestial body.

(*c*) The graticule is always available for immediate use.

(*d*) The pendulous unit is more robust than the bubble unit.

(*e*) In severe conditions of vibration, when the oscillations of a bubble would render it useless as a datum, the graticule forms a diffuse but quite useable pattern.

(*f*) Although the instrumental accuracy of all three marks of sextant is the same, com-

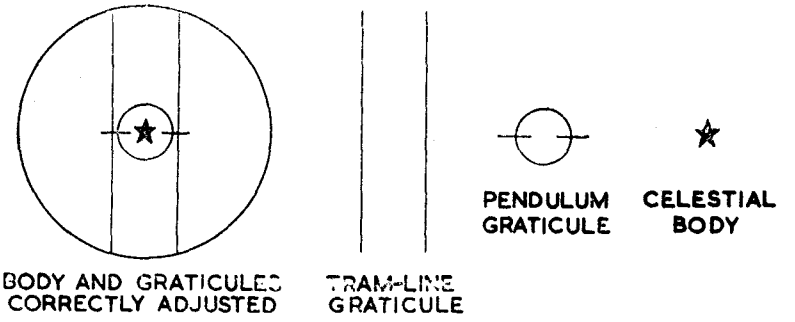


Fig. 5. Graticules

parative trials have shown that slightly more accurate sights can be taken with the pendulous models.

OPERATION

General

9. The operating drills for the pendulous sextants are given in full in the following paragraphs. These drills are similar in many respects to those for the Mk. 2 sextant and should be read in conjunction with Figs. 1, 2, 5, 6 and 7 of Chapter 5, of this section and Fig. 1 of this chapter. A single type of mounting serves all three periscopic sextants.

Inserting Sextant in Mounting

10. (*a*) Slide the sextant into the mounting carrier tube until the finger catches engage. The sextant will then be held in the retracted position.

(*b*) Plug the sextant lead into the sextant and mounting sockets (ensuring that the white lead engages with the white socket in the mounting.)

(*c*) Pull the operating lever fully downward to open the sealing plate.

(*d*) Press the finger catches together and slide the sextant a little further into the carrier tube. Release the finger catches.

(*e*) Continue to slide the sextant gently into the carrier tube until the finger catches re-engage. The sextant is then held in the operating position.

Note.—The sextant must be held firmly whilst it is being raised to the operating position. If this is not done, and there is a pressure difference between the cabin and the outside air, the sextant may be forced into the mounting so violently as to cause damage to the sextant and injury to the operator.

Pre-Flight Checks

11. (*a*) Check for superficial damage, cleanliness, and availability of spare bulbs and of electrical lead.

(*b*) Insert sextant in mounting, wind averaging attachment, make electrical connection. Switch ON/OFF switch to ON.

(*c*) Check graticule illumination and sextant lighting.

(*d*) Check shades "out" and carry out alignment check. (Details of the check for the aircraft type are given on a card in the sextant carrying case.) Note any correction.

(*e*) Set automatic averaging attachment to 1 or 2 minutes, whichever is to be used during the flight. Set the fine and coarse setting controls to any mid-altitude range setting.

(*f*) Release averaging attachment control and check run-down time. It should agree with the calibration time given in the carrying case to within 5 seconds.

(*g*) Compare the final averaged reading with the setting of the fine setting control. They should agree to within 3 mins. of arc.

Periodic Checks

12. These periodic checks are the navigator's responsibility only when he holds the sextant on permanent personal loan; at other times they will be carried out by competent technical personnel.

13. **Checks to be Carried Out Each Month.** At intervals not exceeding one month the serviceability checks given in paras. 5-7 and 14-16 of Appendix 1 (or 1A as appropriate) of A.P. 1275B, Vol. 1, Sect. 13, Chap. 6 are to be carried out. These checks comprise:—

(*a*) General function check.
(*b*) Lighting check.
(*c*) Automatic attachment check.

14. **Checks to be Carried Out Every Six Months.**

At intervals not exceeding 6 months the serviceability checks given in paras. 11-13 of Appendix 1 (or 1A as appropriate) of A.P. 1275B, Vol. 1, Sect. 13, Chap. 6 are to be carried out. These checks comprise:—

(*a*) Index error test.

(*b*) Backlash test.

Altitude Observations

15. (*a*) Ensure that the power switch on the mounting is on and the sextant lead in position.

(*b*) Select either the one-minute or two-minute run as required.

(*c*) Fully wind the averaging attachment.

(*d*) Using the coarse setting and fine setting controls, set the precomputed altitude of the body on the sextant.

(*e*) Rotate the azimuth ring until true heading is set against the mounting lubber line.

(*f*) Clamp the azimuth ring to the mounting by placing the locking lever in the up position.

(*g*) Rotate the sextant until its index mark is set to the precomputed azimuth of the body.

(*h*) Adjust the shade density and graticule lighting to suit viewing conditions.

(*j*) Move the sextant so that the graticule is positioned between the tram-lines in the middle of the field of view and rotate the fine altitude control to bring the body and graticule into coincidence.

(*k*) At the time for the start of the observation press the starting trigger. Throughout observation maintain coincidence of graticule and body by operation of the fine altitude control.

(*l*) When the graticule lighting is finally extinguished calculate the mid-time of the run.

(*m*) To obtain the average altitude, add the values shown by the averaging attachment counters to those on the coarse setting scale.

Azimuth Observations

16. (*a*) Fully wind the averaging attachment.

(*b*) To make a heading check:—

(i) Calculate the expected true azimuth and altitude of the body selected for a convenient time ahead.

(ii) Set the precomputed altitude of the body by means of coarse and fine setting controls. Rotate the azimuth ring until the precomputed true azimuth of the body is set against the sextant lubber line.

(iii) Clamp the azimuth ring to the sextant by placing the locking lever in the down position.

(iv) Rotate the sextant until the approximate true heading is shown against the lubber line of the mounting.

(v) Adjust the shade density and graticule lighting to suit viewing conditions.

(vi) Move the sextant to position the graticule between the tram-lines. Bring the body and graticule into coincidence.

(vii) Read off true heading against mounting lubber line.

Note.—Because of the narrow field of view of the sextant great care must be taken to ensure that the body viewed is the correct one. This precaution also applies when obtaining the altitude of a body.

(*c*) To measure range and bearing:—

(i) Set coarse setting control to D, and rotate azimuth ring until true heading is set against the mounting lubber line. Clamp the azimuth ring to the mounting.

(ii) Adjust shade density and graticule lighting and rotate sextant until object, graticule, and tram-lines are coincident.

(iii) Note the true bearing at the beginning and end of a one-minute shot and obtain the bearing for the mid-time.

(iv) Note depression (10°—altitude) and calculate range as explained in Chapter 1 of this section.

Withdrawing Sextant from Mounting

17. (*a*) Disconnect the electric lead from the sextant and the mounting.

(*b*) Press the finger catches together and withdraw the sextant slightly from the operating position.

(*c*) Release the finger catches and withdraw the sextant to the retracted position.

(*d*) Push the operating lever fully upwards. This closes the sealing plates.

(*e*) Holding the sextant firmly, press finger catches together and withdraw the sextant from the mounting. Stow in case.

After Landing

18. Switch off the power supply to the mounting.

Caution

19. Cases have occurred where the periscope sextant has been withdrawn with the sealing plate still open. In theory this should not be possible as the spring-loaded safety catch should enter the periscope's upper groove and so arrest further movement of the sextant. If the sextant is allowed to drop sufficiently quickly, the spring-loaded catch may not engage.

20. The built-in safety system can be beaten by one of several ways:

a. A mal-adjustment of the bowden cable between the safety catch and the operating lever.

b. Too fast a withdrawal of the sextant. The critical speed is a function of the adjustment of a. above.

c. If the finger catches are held in/stuck in continuously when withdrawing the sextant.

21. If the safety system is beaten in flight in a pressurized aircraft, the pressure differential should keep the sextant in the mounting.

CHAPTER 4

THE KOLLSMAN PERISCOPIC SEXTANT

CONTENTS

	Paras
Introduction	1– 2
THE MOUNT	
General	3– 6
The Sealing Plate	7
The Gimballed Suspension	8
The Azimuth Reference	9–10
THE SEXTANT	
General	11
The Optical System	12–17
The Averager	18–19
Lighting	20
OPERATION	
Checks	21–26
Inserting the Sextant	27–28
Altitude Observations	29
Azimuth Observations	30
Removing the Sextant	31
Accuracy	32

Introduction

1. The Kollsman periscopic sextant is designed for use in pressurized aircraft. When fitted into a special mount in the aircraft it can be used to measure the altitude of celestial bodies, or to determine the true heading of the aircraft. In this chapter it will be assumed that the sextant and mount are used in conjunction with a Synchronous Astro Compass (SAC) which is described in Part 4, Sect 2, Chap 3.

2. The sextant uses a pendulous mirror to define the horizontal. Altitude observations can be taken over any period between 30 secs and 2 minutes. An averaging device is incorporated to indicate the mean altitude whatever the selected period.

THE MOUNT

General

3. The sextant mount is fitted permanently to the inside of the fuselage skin. It has three functions:

a. To enable the sextant to be extended into or retracted from the observing position without loss of cabin pressure.

b. To provide a gimballed suspension for the sextant.

c. To provide an indication of the sextant's azimuth.

4. Aircraft power is provided at the mount for it and the sextant. Both supplies are controlled by a single on/off switch.

5. Any water which may have collected in the mount can be drained off by removing a plug at the lowest point of the mount.

6. Fig 1 is an illustration of the mount in isolation. Fig 2 shows the mount installed in an aircraft with the sextant and the servo drive unit (SDU) fitted.

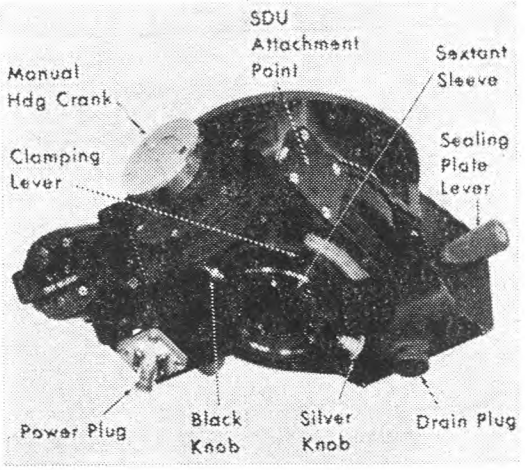


Fig 1 The Sextant Mount

The Sealing Plate

7. At the top of the mount is a cabin sealing plate, controlled by the large lever on the mount, which blocks off the aperture in the fuselage when the sextant is not extended for use. There is no automatic locking device to stop the plate being withdrawn when the sextant is not fitted. Therefore negligent operation of the lever could cause a loss of pressurization.

The Gimballed Suspension

8. Within the gimballed suspension system is a sleeve into which the sextant is positively locked when in use. The sextant and sleeve can be rotated through 360° in azimuth, and the gimbals allow 15° of tilt. The sextant is locked in the sleeve by two spring loaded pins, one black (marked "To insert/remove—Pull") which holds it in the retracted position, and one silver (marked "To retract sextant—Pull") which holds it in the

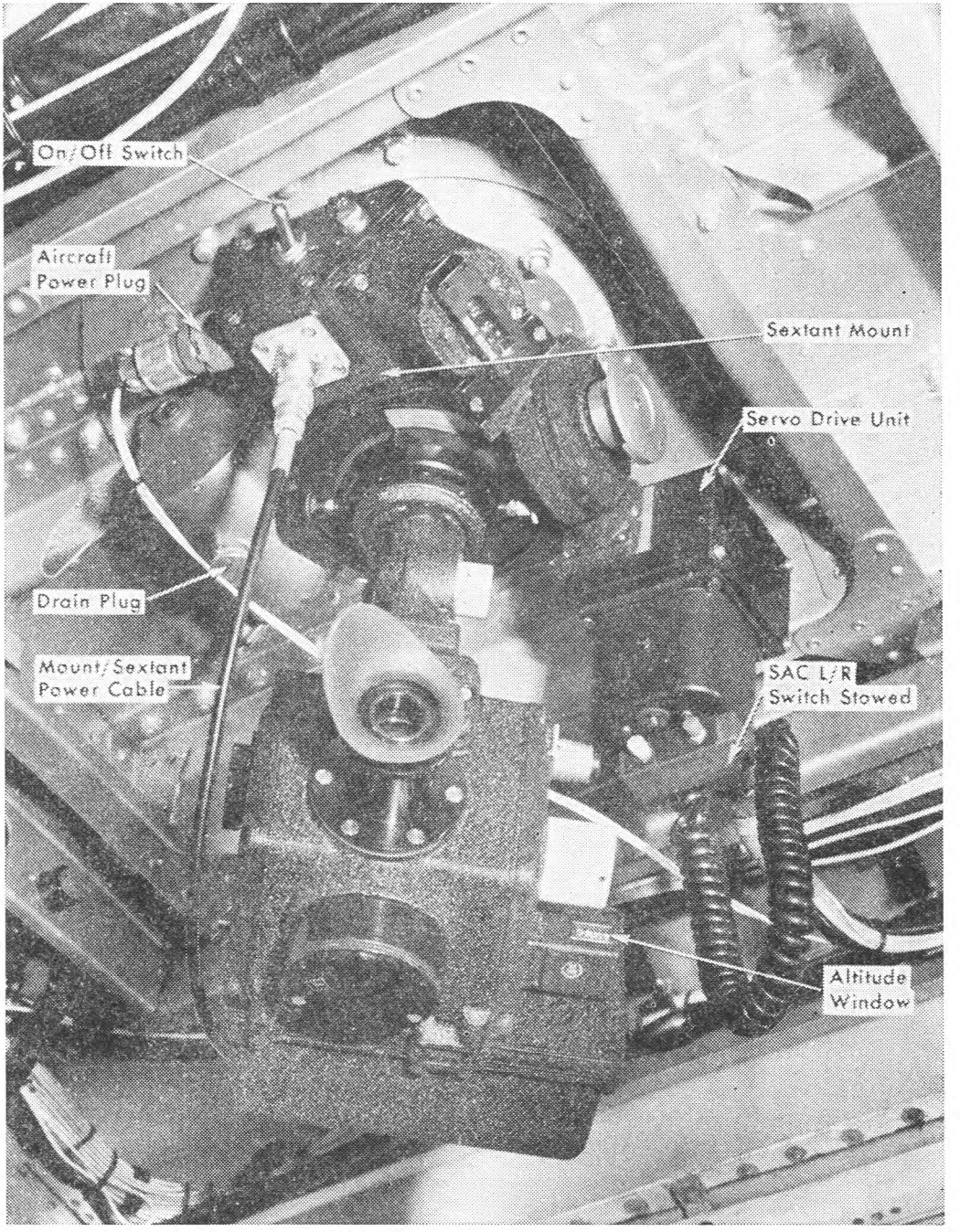


Fig 2 The Mount, the Sextant and the SDU Installed

extended or operating position. In the extended position the sextant protrudes through the aircraft skin by 1½ inches. The sextant and sleeve can be locked in azimuth by a lever ontrolled friction clamp.

The Azimuth Reference

9. The azimuth scale in the mount can itself be rotated. The sleeve is not mechanically connected to the azimuth scale. A plate attached to the sleeve covers all but a segment of the azimuth scale, rotation of the sleeve exposing different parts of the scale.

10. The azimuth ring is positioned by the servo drive unit which is controlled by the SAC controller (SACC). For normal operation the azimuth ring and counters will show the relative bearing. (*Note.* If the sextant is not used in conjunction with the SAC a manual crank is provided to position the azimuth scales.)

THE SEXTANT

General

11. Two views of the sextant are shown in Figs 3 and 4. Aircraft power for the sextant lighting is provided from the mount through a short detachable cable. The sextant, the cable, batteries, battery holder and spare bulbs are contained in the sextant carrying case which is strapped to the aircraft.

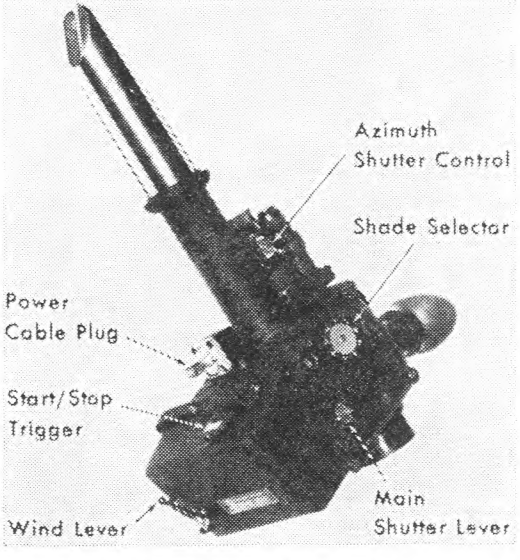


Fig 3 The Kollsman Periscopic Sextant

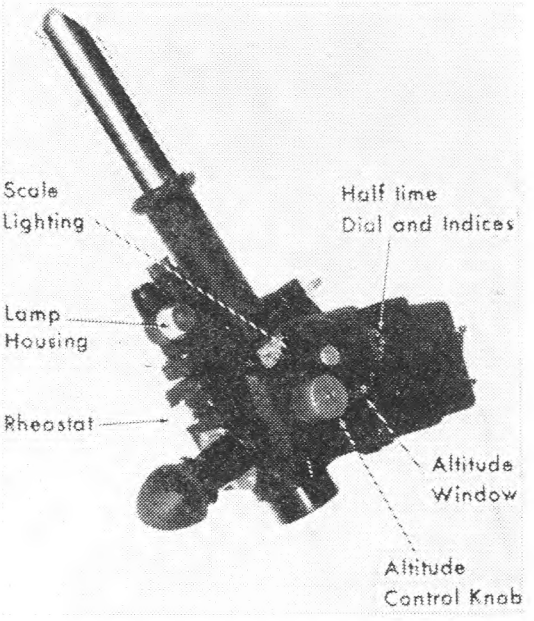


Fig 4 Another View of the Sextant

The Optical System

12. The optical system is designed so that three separate light paths can be viewed through the sextant eyepiece. The paths are those of the celestial body, the horizon reference and about 55° of the exposed part of the azimuth ring. A field of view of 15° in both the horizontal and vertical can be seen. The optical system is illustrated at Fig 5.

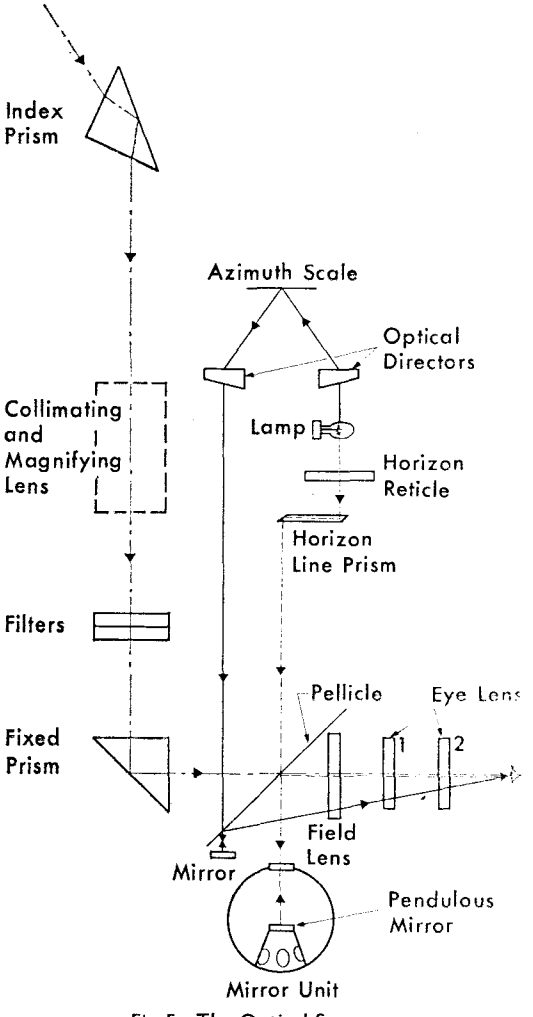


Fig 5 The Optical System

13. **The Horizon Reference.** The system's horizon reference is the transparent cross hair on the horizon reticle. As can be seen from Fig 5, light from the lamp, the brightness of which is controlled by the rheostat, passes through the horizon reticle. The vertical limbs of the cross hair are cut off by the horizon line prism. The horizontal line's image is then reflected by the pendulous mirror and the pellicle to appear as a real image at the field lens. The centre-pivoted mirror is made pendulous by its heavy, peripheral underskirt. The mirror unit is housed in an oil-filled chamber to provide damping, internal expansion bellows compensating for changes in volume of the oil.

14. **Altitude Measurement.** Operation of the altitude control changes the position of the rotatable index prism at the top of the sextant tube. The altitude control knob is geared to a dial counter. Light from the celestial body is directed by the index prism, through the periscope optics, on to the fixed prism. The real image of the body appears at the field lens, on which is engraved the cross hair which indicates the centre of the field of view.

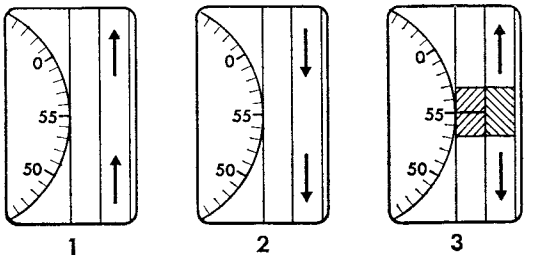
15. **Azimuth Display.** Light from the lamp referred to in para 13 is also directed upwards to illuminate the exposed part of the azimuth scale. The scale's image is then reflected by the rectangular mirror and the pellicle to appear at the bottom of the field as viewed through the eyepiece. The vertical line of the field lens cross hair (reticle) is the azimuth datum. The scale can be blacked out by use of the azimuth scale shutter control.

16. **The Eyepiece Lens System.** The focal plane of the eyepiece lens system is coincident with the focal plane of the erecting system at the field lens. Thus the images of the horizon reference and the body appear together without parallax. The eyepiece is adjustable for correct focus.

17. **The Shade System.** The filter knob is used to insert one of eight degrees of shade into the path of the light from the celestial body. Usually a combination of shade and sextant lighting can be obtained so that the body, the horizon reference and the azimuth scale can be viewed together. Light from the body can be eliminated altogether by operating the spring-loaded main shutter lever.

The Averager

18. **Principle.** The averager is basically a time-controlled ball and disc integrator. The disc is driven by the altitude control; the ball is moved radially from the centre, its speed being time controlled to arrive at its maximum radius after two minutes. The ball in turn drives the cylinder and its indices. It is arranged that the rotation of the cylinder represents a linear function of altitude against time. At the end of the observation the cylinder is rotated back through its angular displacement while the ball retains its radial position; when the indices are aligned the average altitude is indicated on the altitude counters.



1. To obtain the average, rotate so that the arrows move upward.
2. To obtain the average, rotate so that the arrows move downward.
3. Index set to obtain the average altitude angle.

Fig. 6 The Half-Time Dial and Indices

19. **The Controls and Indicators.** There are two controls, one to wind the timer which automatically resets the averager and opens the main shutter, and the start/stop trigger. The averager is started by depressing the start/stop trigger. The averager will stop and the main shutter fall after two minutes, or when the start/stop trigger is operated again at any time between 30 seconds and two minutes. The half-time dial shows the period in seconds to apply to obtain the time of the observation. It appears in the same window as the averager indices, as shown at Fig 6.

Lighting

20. In addition to the lamp which illuminates the horizon reference and the azimuth scale, a second lamp above the altitude control knob illuminates the altitude window, the half-time dial and indices. Both these lamps can be operated from the battery which is stored in the sextant case. The battery case is connected by a cable to the sextant.

OPERATION

Checks

21. The checks and operating procedures described below apply when the sextant is used with the SAC.

22 **Checking the Mount.** The checks for the mount are as follows:

a. Remove the drain plug and allow any water to drain out. Replace the plug.

b. Operate the sealing plate lever to ensure the operation of the plate. This check should not be done in flight unless the sextant is installed in the retracted position, and held firmly.

c. Switch on the on/off switch and check that the azimuth counters are illuminated, switch off.

d. Alter the HRS read-out and check that the azimuth counters rotate.

23. **Checking the Sextant.** The checks for the sextant are:

a. Insert the sextant in the mount (*see* para 28).

b. Connect the power cable and switch on.

c. Operate the clamping lever to ensure correct functioning.

d. Rotate the altitude dial from —10° to +92°.

e. Check the functioning of the eight filters

f. Operate the azimuth shutter control.

g. With the timer wound up, operate the main shutter manual control.

24. **Checking the Averager.** The averager, which should be left unwound when not in use, should be checked as follows:

a. Depress the winder and check that the indices align, that the half-time dial goes past zero and then back to zero, and that the main shutter is removed from the field of view.

b. Depress the start/stop trigger.

c. Depress the start/stop trigger again after exactly one minute and check that:

(1) The main shutter is across the field of view.

(2) The averager has stopped.

(3) The half-time dial indicates 30 secs $\pm\frac{1}{2}$ sec.

d. Repeat a and b and let the averager run for its full time. Check that:

(1) The main shutter falls after 2 mins ± 2 secs.

(2) The half-time dial indicates 60 secs ± 1 sec.

25. **Checking the Sextant/Mount Alignment.** The method of checking the mount and sextant alignment is described in Part 4, Sect 2, Chap 3, paras 16 and 17.

26. **Periodic Checks.** As the sextant is part of the aircraft equipment, periodic checks are the responsibility of the ground engineers.

Inserting the Sextant

27. The sextant is normally to be put in or withdrawn from the mount before the aircraft is pressurized. If it is necessary to insert the sextant during pressurized flight, it must be secured in the mount in the retracted position before the sealing plate is opened. Similarly, to withdraw the sextant during pressurized flight, the sextant must be secured in the retracted position and the sealing plate closed before it is withdrawn from the mount.

28. The insertion procedure is as follows:

a. Check that the clamping lever is to the left (the unlocked position).

b. Rotate the locking pin ring counter-clockwise (looking upwards) to the limit of its travel.

c. Insert the sextant with the arrows on the tube and mount aligned, pulling out the black knob.

d. Rotate the retaining ring clockwise until the knob seats itself; the sextant is now in the retracted position. It should not be left in this position when not in use, but should it prove necessary to do so the clamping lever should be locked.

e. Open the sealing plate, holding the sextant firmly to prevent it slamming home should there be a pressure differential.

f. Raise the sextant into the mount until the silver knob snaps into place.

g. Connect the power cable and switch on.

Altitude Observations

29. To measure the altitude of a body:

a. Insert the sextant as in para 28.

b. Wind up the averager.

c. Set the pre-computed altitude of the body.

d. Check that the correct relative bearing from the SAC is shown at the azimuth counters and against the lubber line.

e. Rotate the sextant until the vertical cross-hair passes through 000° on the scale.

f. Adjust the rheostat, the filters and the focus for comfortable viewing.

g. Identify the body and make the final adjustments to altitude and azimuth to align the body and the datums.

h. Check that the averager is fully wound.

j. Note the time, depress the start/stop trigger, and maintain the coincidence of the body and datums.

k. At any time after 30 seconds the observation may be stopped by depressing the trigger again. Greater accuracy is achieved by taking the observation over the full two minutes.

l. Add the half-time to the start time to give the time of the observation.

m. Turn the indices in the direction shown, and read off the average altitude in the altitude window.

Azimuth Observations

30. The full procedure for obtaining a heading check when the sextant is used with the SAC is described in Part 4, Sect 2, Chap 3, para 25.

Removing the Sextant

31. The procedure to remove the sextant is:

a. Switch off and disconnect the power cable.

b. Holding the sextant securely, pull out the silver knob which may have quite a stiff action.

c. Lower the sextant firmly to the retracted position.

d. Close the sealing plate.

e. Still holding the sextant securely, pull out the black knob.

f. Rotate the retaining ring counter-clockwise until the arrows on the tube and the mount are aligned.

g. Remove the sextant.

h. Ensure that the averager is unwound, and store the sextant and cable in the case.

Accuracy

32. Altitude can be measured from —10° to +92° with an accuracy of ±2′ of arc. Azimuth accuracy is discussed in Part 4, Sect 2, Chap 3, para 14.

PART 4

SECTION 2

ASTRO COMPASSES

Chapter

1 **Astro Compass, Mark 2A**

2 ~~**Sky Compasses**~~ SPARE AL 29

3 ~~**The Synchronous Astro Compass**~~ SPARE AL 29

3456D
Part 4 Sect 2
Chap 1.

CHAPTER 1

ASTRO COMPASS, MARK 2A

CONTENTS

	Paras.
Introduction	1
Principle	2– 7
Description	8–14
Sighting	15–17
Aligning the Standards	18–21
Operation in the Air	22–26
Accuracy	27–28

Introduction

1. The astro compass is a navigational instrument designed to provide:—

(*a*) A means of identifying the stars used in air navigation.

(*b*) The true heading of an aircraft.

(*c*) True and relative bearings of distant objects.

Principle

2. The astro compass is a mechanical device for reconstructing the astro triangle from known data. It can also be used to act as a bearing plate.

3. The azimuth of a celestial body is the angle measured clockwise in the plane of the celestial horizon between the observer's true meridian and the vertical circle passing through the body. The astro compass incorporates an azimuth scale which can be levelled by means of two bubble levels at right angles, with appropriate levelling screws. Thus the azimuth plate can be set parallel to the celestial horizon.

4. The local hour angle is the angle measured westwards along the celestial equator between the planes of the observer's and body's celestial meridians. The astro compass incorporates an hour circle which can be set parallel to the plane of the celestial equator thus enabling the setting and measurement of the L.H.A. of a body.

5. The declination is the angle at the centre of the celestial sphere between the body and the celestial equator measured in the plane of the celestial meridian of the body. This may also be set on the astro compass.

6. If all the data mentioned above (*i.e.* azimuth L.H.A. and declination) are known for any given body, together with the latitude of the observer, and their values are set on the astro compass, then the sighting bar will be lined up on the body in question. This situation arises when it is required to locate a celestial body.

7. By setting on known values and lining up the sights on a celestial body, the triangle can be solved for unknown values as in paras. 22-26.

Description

8. The Astro Compass, Mk. 2A may be considered to consist of two parts, an upper and a lower. The lower part comprises an azimuth scale, which is free to rotate against a lubber line. It is mounted on a levelling plate, the lower part of which comprises a plug fitting a Type 05A Standard. The instrument can be levelled against the two cross levels by adjustable screws. The compass is about $8\frac{3}{4}$ in. high when in the standard and measures about $6\frac{1}{4}$ in. above the levelling plate.

9. The upper part, or L.H.A. drum assembly, is mounted on two side plates attached to the azimuth platform. The drum is secured so that the axis of its supports is at right angles to the N-S line of the azimuth scale. One of the side plates is provided with a worm wheel driven by a worm to which the latitude scale is attached. This scale is marked in tens of degrees for approximate setting, while a scale of single degrees for fine setting is engraved on the latitude knob on the worm shaft spindle. North latitudes are marked in white, south latitudes in red.

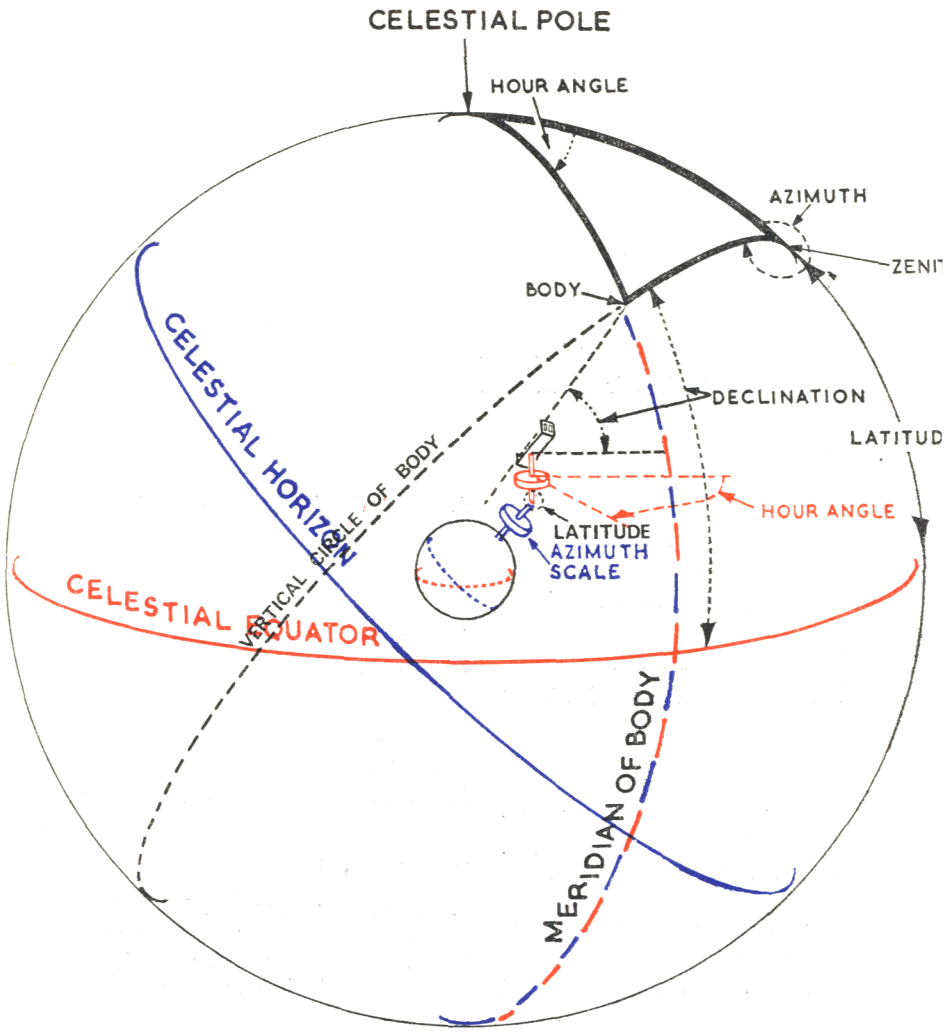


Fig. 1. Principle of Astro Compass

10. Two hour circles, graduated in opposite directions from 0 to 360°, one for north and the other for south latitudes, are mounted on either side of the horizontal axis of the L.H.A. drum. These are rigidly connected and are turned by a knob at the opposite end of the axis to the latitude knob.

11. To set hour angle, the knob should be rotated until the appropriate hour circle reads the required value. The hour circle for north latitudes is in white and is read against an index marked L.H.A., N. LAT. That for south latitudes is marked in red and is read against an index marked L.H.A., S. LAT. These two index marks are 180° removed and lie in a vertical plane passing through the 0 to 180° marks of the azimuth scale. Another white index TRUE BEARING is engraved above S. LAT., and is for use when taking bea both north and south latitudes.

12. Attached to the upper hour circ aligned parallel to the 0 to 180° line is a bar and screen for use on the Sun and combined with a star sight for use o celestial and terrestrial bodies. The sig mounted in a frame and can be tilted rel the plane of the hour circle and set to any of declination from 64°N. to 64°S.

13. The declination is shown against an e arc by a white pointer. Beneath the enc arc at the shadow bar end are the lette

white and N in red. At the other end of the arc are the letters N in white and S in red. The white letters are used in north latitudes and the red in south latitudes. The shadow bar is situated over the white 0° mark of the hour circle, and the screen and star sight over the 180° mark. The star sight consists of a lens mounted above the screen, and a foresight consisting of two luminous tubes above the shadow bar.

14. The following items are available:—

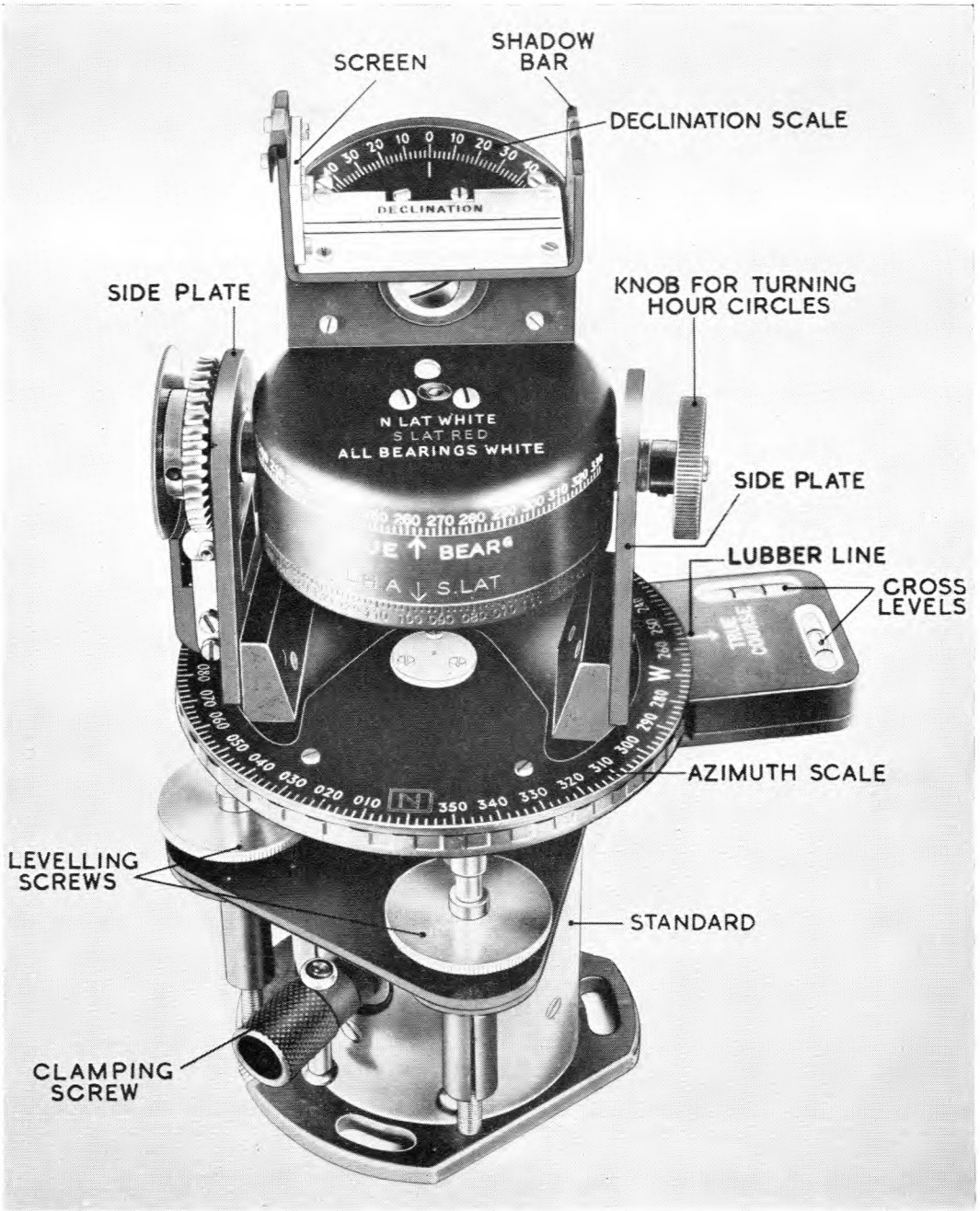


Fig. 2. Astro Compass, Mk. 2A, in Standard

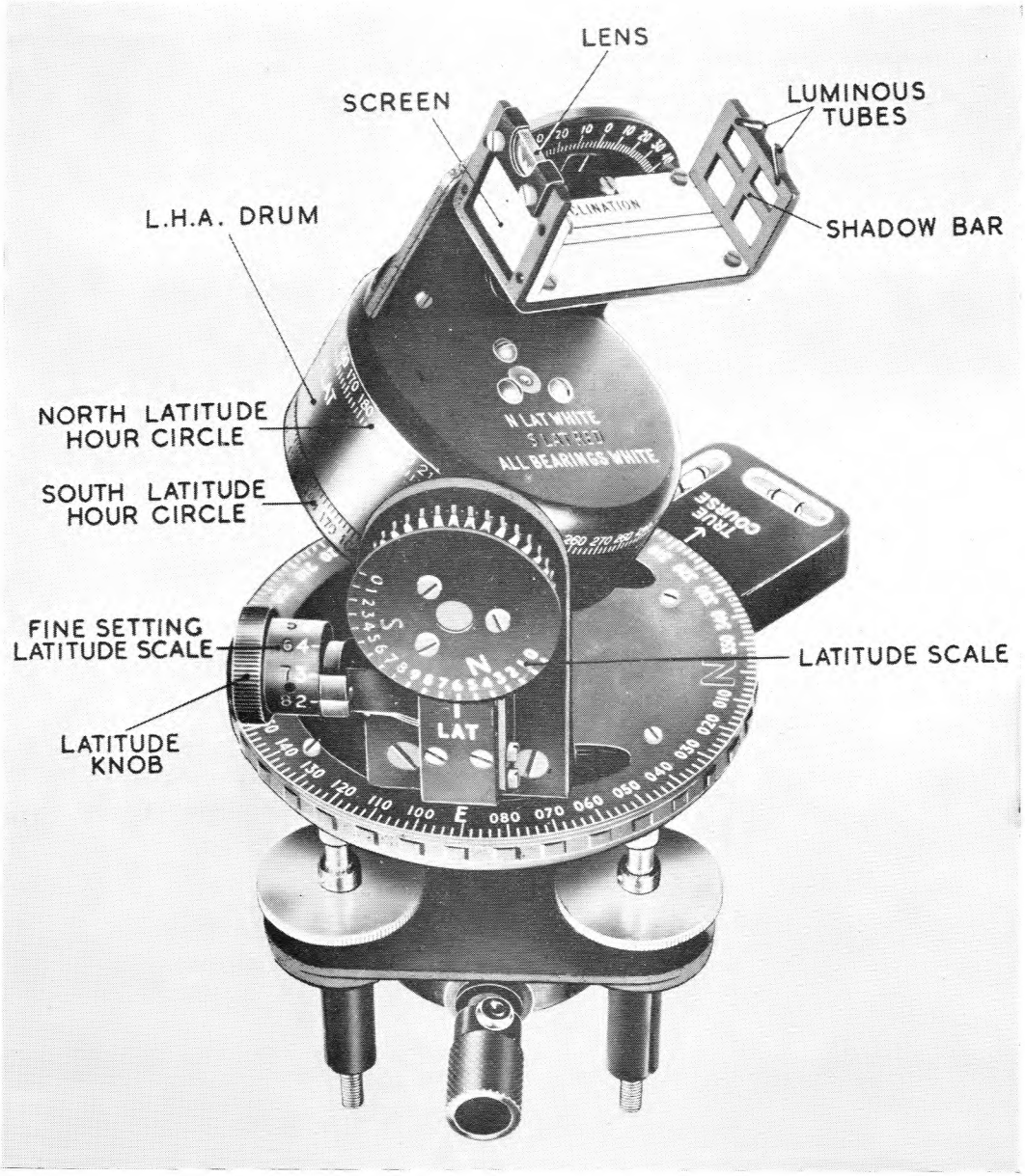


Fig. 3. Astro Compass, Mk. 2A

(*a*) Astro Compass Mk. 2A (Stores Ref. 6B/399), weight 2lb 3oz.

(*b*) Transit Case (Stores Ref. 6A/1196).

(*c*) Type 05A Standard (Stores Ref. 6B/400).

Sighting

15. A star, planet or terrestrial object can be sighted by looking both through and over the lens simultaneously, when the image of the foresight and the object will be seen clearly. The object is correctly sighted when it appears at the point at which the axes of the two luminous tubes would intersect, if produced.

16. If all the settings made are correct and the compass properly levelled, the star or planet will pass through the line of sight when the compass is

rotated in azimuth. However, this ideal is rarely attained in practice. Thus, if the body does not pass through the line of sight, the correct azimuth setting can still be obtained when the star is vertically above or below the point of intersection of the luminous tubes, ignoring the positioning of the tubes themselves with regard to the vertical (*see* Fig. 4).

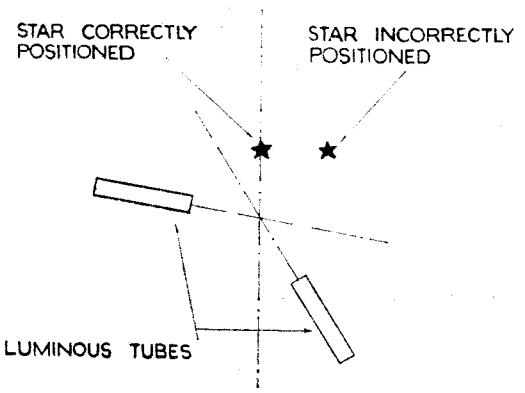


Fig. 4. Correct Sighting

17. When using the astro compass on the Sun or bright Moon, in latitudes above 30°, the cross may be ignored. The declination should be set to zero, and the shadow bar vertical to the L.H.A. drum used. In latitudes below 30° it is necessary to use the shadow cross, which provides greater sensitivity. Sighting should be carried out so that the shadow cross is vertically above the backsight cross. It is also advisable to use the azimuth method of checking heading in these low latitudes (*see* para. 23).

Aligning the Standard

18. The Type 05A Standard should be fixed in the position set out in the appropriate aircraft schedule and should then be aligned as described in the following paragraphs. When not in use the astro compass should be placed in the stowage standard, or kept in its transit case. The compass should be clamped securely by means of the clamping screw when placed in the standard.

19. The standard can be aligned with the fore-and-aft line of the aircraft on the ground with the tail down. The astro compass should be placed in position in its standard, rotated fully clockwise, and the clamping screw tightened securely. It should then be levelled as accurately as possible. Levelling plays an important part in the operation of the astro compass, since levelling error can give an azimuth error equal to the error in levelling in degrees × the tangent of the altitude of the body. For example, a levelling error of 1° with an altitude of 64° may give an azimuth error of 2°.

20. The true heading of the aircraft should then be determined by the same method as that used in air operation (para. 22). Then, by some suitable external means, such as the medium landing compass, the true heading of the aircraft should again be determined and the two headings compared. If there is a discrepancy, the standard should be rotated until the astro compass heading agrees with the correct true heading.

21. If the sky is not visible the true heading of the aircraft should be set against the lubber line and the latitude scale set to 90°. A medium landing compass should then be set up at a distance and the true bearing of the astro compass observed. The reciprocal of this true bearing should be set against the bearing index on the astro compass, which should then be rotated with its standard until the sights are lined up on the landing compass. The standard should then be screwed down.

Operation in the Air

22. **To Check True Heading:—**

(*a*) Determine the L.H.A. and declination of a suitable celestial body for some convenient time ahead.

(*b*) Set declination.

(*c*) Set D.R. latitude.

(*d*) Set L.H.A.

(*e*) Approximately one minute before the predetermined time warn the pilot.

(*f*) When on a straight and level heading, level the instrument.

(*g*) At the predetermined time rotate the azimuth scale until either:—

(i) The body is correctly aligned in the sights (stars and planets).

(ii) The shadow of the shadow bar lies between the lines on the screen (Sun or bright Moon).

(*h*) Read off true heading against the lubber line.

(*j*) Inform the pilot when the operation is completed.

23. **To Check True Heading—Azimuth Method:-** For heading checking in low latitudes it is advisable to use the following method:—

(*a*) Calculate the azimuth and the approximate altitude of a suitable celestial body for some convenient time ahead.

(*b*) Set latitude to 90°.

(*c*) Set the precalculated azimuth against the true bearing lubber line.

(*d*) Set the approximate altitude on the declination scale.

(*e*) Approximately one minute before the predetermined time warn the pilot.

(*f*) When on a straight and level heading, level the instrument.

(*g*) At the predetermined time rotate the azimuth scale until either:—

(i) The body is correctly aligned in the sights (stars and planets).

(ii) The shadow bar lies between the lines on the screen (Sun and bright Moon).

(*h*) Read off the true heading against the true heading lubber line.

(*j*) Inform the pilot when the observation has been completed.

24. **Steering a Heading:**—

(*a*) Check the heading.

(*b*) Instruct the pilot to set this heading on the directional gyro.

(*c*) Instruct the pilot to turn on to the required heading using the directional gyro.

(*d*) Carry out a heading check regularly altering heading on the directional gyro as necessary.

Note.—For complete drill see A.P. 1234 A—Gyro Steering.

25. **To Identify an Unknown Star:**—

(*a*) Set D.R. latitude.

(*b*) Set true heading on the true heading lubber line.

(*c*) Warn the pilot.

(*d*) When on a straight and level heading, level the instrument.

(*e*) Rotate the L.H.A. drum until the body is correctly aligned in the sights.

(*f*) Note the time.

(*g*) Inform the pilot that the observation has been taken.

(*h*) Read off L.H.A. star.

(*j*) Read off declination.

(*k*) Enter the Air Almanac with G.M.T. and obtain G.H.A. Aries.

(*l*) Calculate L.H.A. Aries.

(*m*) Calculate S.H.A. star.

(*n*) Enter Air Almanac with S.H.A. and declination to identify star.

26. **To obtain a Terrestrial Bearing:**—

(*a*) Set latitude to 90°.

(*b*) Set true heading on the true heading lubber line.

(*c*) Warn the pilot.

(*d*) When on a straight and level heading, level the instrument.

(*e*) Rotate the L.H.A. drum until the object lies in the sights.

(*f*) Note the time.

(*g*) Inform the pilot that the observation has been made.

(*h*) Read off the true bearing against the true bearing lubber line.

Accuracy

27. When using the astro compass it is possible to introduce appreciable errors in the headings and bearings obtained, through small errors in settings and adjustments. The instrument itself is accurate to within 2°, if the settings and adjustments are correctly made. More often than not, the result will be within 1° or less as individual errors tend to cancel out rather than to accumulate.

28. To ensure the most accurate results the following precautions should be taken when using the instrument:—

(*a*) Between 30°N. and 30°S. use the azimuth method of checking heading.

(*b*) At latitudes greater than 30°N. or 30°S. use a body with a fairly low altitude.

PART 5

MISCELLANEOUS AIRCRAFT INSTRUMENTS

Section

1 Engine and Ancillary Instruments

2 Optical Instruments

PART 5

SECTION 1

ENGINE AND ANCILLARY INSTRUMENTS

Chapter

1 Tachometers and Position Indicators
2 Pressure Gauges
3 Fuel Contents Gauges and Flowmeters
4 Warning Systems and Devices
5 Accelerometers and Fatigue Meters
6 Torquemeters

CHAPTER 1

TACHOMETERS AND POSITION INDICATORS

CONTENTS

TACHOMETERS

	Paras
Introduction	1 – 3
The Mechanical Tachometer	4 – 5
Electrically Operated Tachometers	6 – 7
Helicopter Tachometers	8

POSITION INDICATORS

Undercarriage Indicators	9–12
Calibrated Indicators	13

TACHOMETERS

Introduction

1. Tachometers are velocity measuring instruments which are used to indicate engine rpm, helicopter rotor speed, and propellor speed in turbo-prop aircraft.

2. Early types of tachometer employed a centrifugal mechanism driven by a flexible shaft from the engine gearbox. Because of the problems of shaft wear and weight penalty inherent in a purely mechanical system electrical transmission systems were used in later versions.

3. The type of tachometer in current use consists of an electrical indicator powered by the output of a generator coupled to the engine or shaft.

The Mechanical Tachometer

4. The mechanical tachometer shown in Fig 1 relies on the action of centrifugal forces which are produced by, and are proportional to, the rotational speed of the drive shaft.

5. A flexible drive is taken from a convenient part of the engine such as the end of the camshaft. The instrument shaft is connected to the flexible drive and rotates at a speed proportional to crankshaft speed. Rotation of the instrument shaft sets up centrifugal forces in the weighted frame pivoted about the shaft, tending to turn the frame into a horizontal position. The action is resisted by an involute spring, so that for a given speed the frame will turn until the tension in the spring balances centrifugal force. The higher the speed the more the frame approaches the horizontal. One end of the frame is linked to a sliding muff which is free to move up and down the instrument shaft, so that the higher the speed the higher the muff rises. A steel arm, interposed between the flanges of the sleeve, communicates this movement to a toothed sector which causes the pointer pinion to rotate. The pointer moves about a calibrated scale (*see* Fig 2). A hairspring attached to the pointer spindle absorbs mechanical backlash.

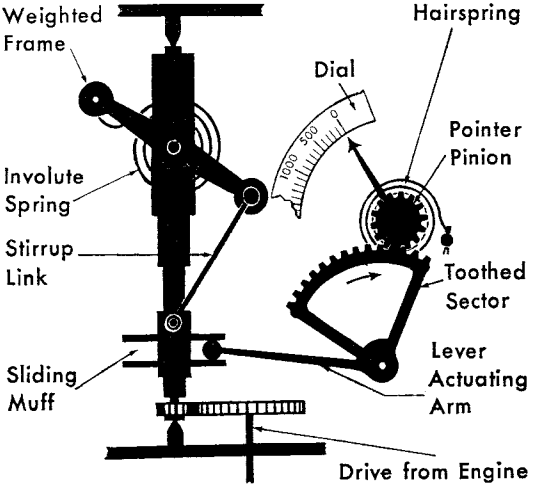


Fig 1 Mechanical Tachometer

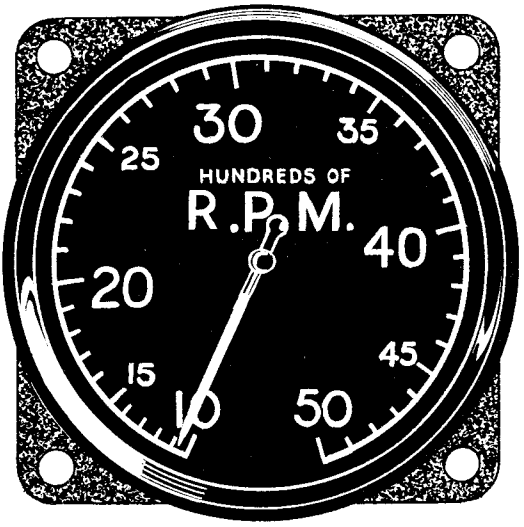


Fig 2 Tachometer Indicator

Electrically Operated Tachometers

6. An electrically operated tachometer consists of two units, a transmitter and an indicator which are connected by cable. The transmitter is a small three phase ac generator which is driven by the engine, rotor or propellor shaft. The generator output is fed to a small squirrel cage synchronous motor housed in the indicator.

7. Since the speed of a synchronous motor is governed by frequency, and the output frequency of an ac generator depends on its speed, line resistance does not affect the accuracy of the indicator. The speed of the motor is measured by the mechanism shown in Fig 3. An extension of the motor shaft carries a four pole permanent magnet which revolves inside a copper alloy drag cup. The rotation of the permanent magnet produces eddy currents in the cup which in turn set up magnetic fields. These fields interact with the field of the permanent magnet causing a torque which turns the cup and its attached handstaff. This torque is balanced by a hairspring. The movement of the handstaff actuates a gear train which rotates a pointer over the dial face. The dial is often calibrated in percentage of maximum rpm with a second needle geared at 10 : 1 ratio (*see* Fig 4). The generator is usually driven through a reduction gearing as it is undersirable for it to operate at the high rpm associated with gas turbines.

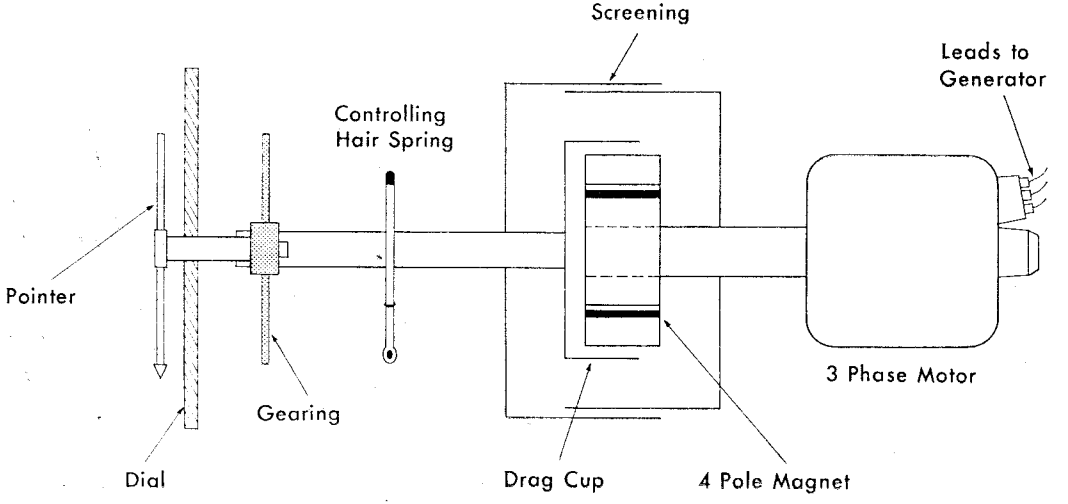


Fig 3 Tachometer Indicator Mechanism

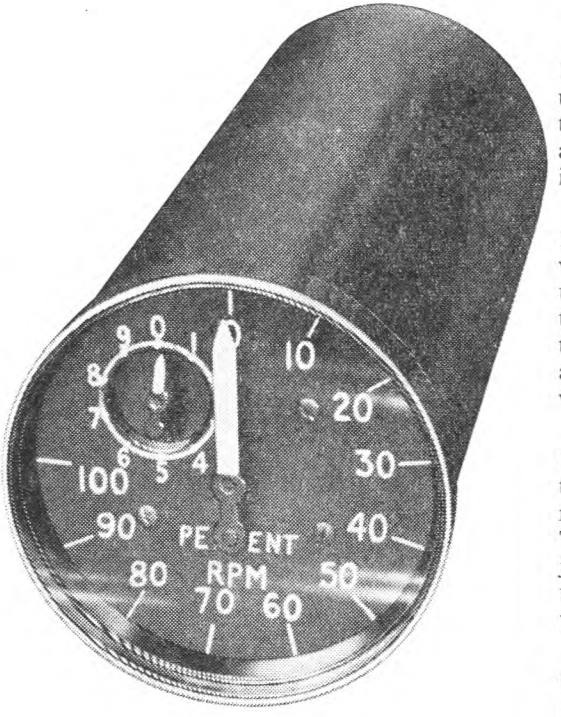


Fig 4 Percentage rpm Indicator

Helicopter Tachometers

8. Helicopter tachometers usually consist of a single dial with two pointers, one indicating rotor speed and the other engine speed. They are driven by separate generators fitted to the appropriate shafts. The dial is calibrated with two concentric scales and has co-axially mounted pointers.

POSITION INDICATORS

Undercarriage Indicators

9. Although varying in minor detail, undercarriage indicators conform to a standard pattern in which each retractable unit of the undercarriage is represented by a red and a green light. The lights are operated by microswitches fitted to the undercarriage locks, and indicate position as follows:

a. Green light on—unit is locked down.

b. Red light on—unit is between the up and the down lock.

c. Red and Green lights off—the unit is locked up (provided that power is available to the circuits).

10. The green lights corresponding to each undercarriage unit are normally duplicated, and the duplicate lights can be selected by operating a changeover switch. A dimmer screen can be interposed for night operation.

11. **Warning Systems.** Many aircraft have a warning system operating in conjunction with the undercarriage indicator. It is actuated by closing the throttle beyond a predetermined point with the undercarriage not locked down: in some aircraft it is actuated by selecting approach flap with the undercarriage retracted.

12. **Mechanical Indicators.** In some aircraft the electrical indicator in the cockpit is supplemented by additional mechanical indicators. These usually consist of small rods which project from the upper surface of the wing when the undercarriage is locked down, and retract flush when the undercarriage is locked up.

Calibrated Indicators

13. Pointer and scale type indicators are used to show the position of flaps, trimming surfaces, radiator shutters *etc.* The majority of these indicators are actuated by desynn transmission systems which are described in Part 3, Sect 1, Chap 1. A flap and gill position indicator is shown in Fig 5.

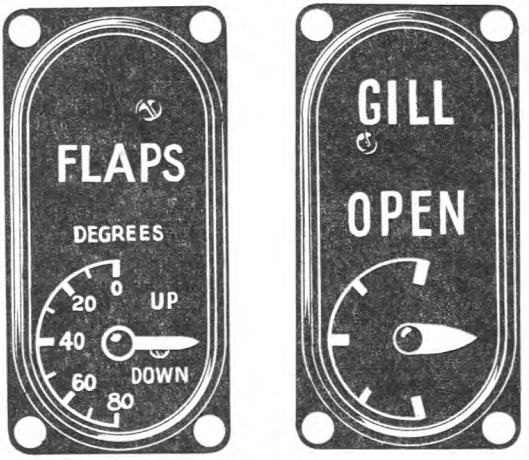


Fig 5 Desynn Position Indicator

CHAPTER 2

PRESSURE GAUGES

CONTENTS

	Paras
Introduction	1– 2
DIRECT READING INSTRUMENTS	
Hydraulic Transmitters	3
Pressure Indicators	4– 7
REMOTE INDICATING INSTRUMENTS	
General	8
Resistor Type	9
Inductor Type	10
Desynn Type	11

Introduction

1. A wide variety of gauges are used to display information about the pressure of the gases and liquids in the various aircraft systems. This chapter briefly describes the general principles applicable to pressure transmitters and indicators. Details of specific instruments are to be found in the AP 112 series.

2. Early types of pressure gauge were fed with pressure either directly or indirectly from the appropriate aircraft system. The disadvantages of feeding high pressure liquids and gases to panel mounted instruments in the cockpit led to the development of pressure transmitters which fed remote indicators electrically.

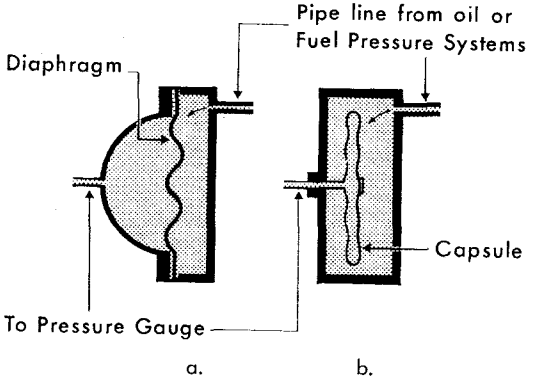


Fig 1 Hydraulic Pressure Transmitters

DIRECT READING INSTRUMENTS

Hydraulic Transmitters

3. The device shown at Fig 1a is a diaphragm pressure transmitter. The liquid in the aircraft system is fed into one side of the chamber where it acts on a diaphragm. The pressure on the diaphragm is communicated by the transmission fluid through a capillary tube to the indicator.The capsule pressure transmitter at Fig 1b operates in a similar fashion.

Pressure Indicators

4. The pressure indicator or gauge usually consists of a needle or needles moving over a calibrated dial as shown in Fig 2. The needle movement is driven by a pressure sensitive device: either a Bourdon tube or a capsule:

a. *Bourdon Tube.* The Bourdon tube (*see* Fig 3) is a coiled metal tube which tends to uncoil when fed with gas or liquid under pressure. It is suitable for measuring high pressures.

b. *Capsule.* An expanding capsule can be used to measure pressures in the same fashion as a simple altimeter. A development of this principle is the diaphragm shown at Fig 4, in which pressure is fed to one side of a diaphragm which actuates a lever movement.

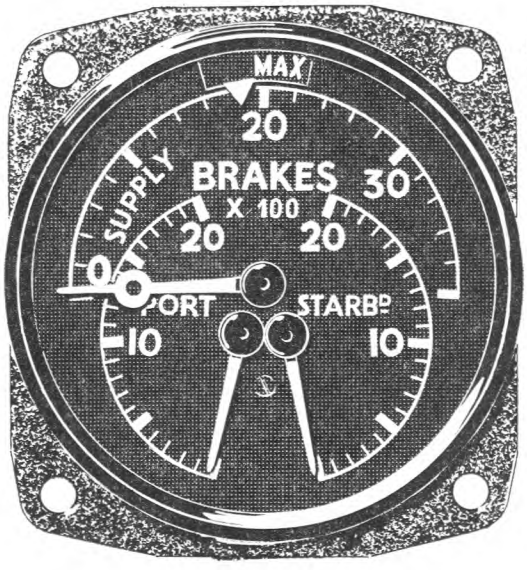


Fig 2 Pressure Indicators

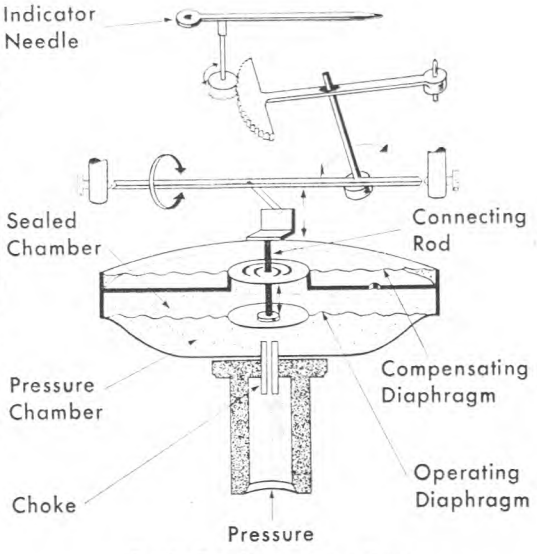
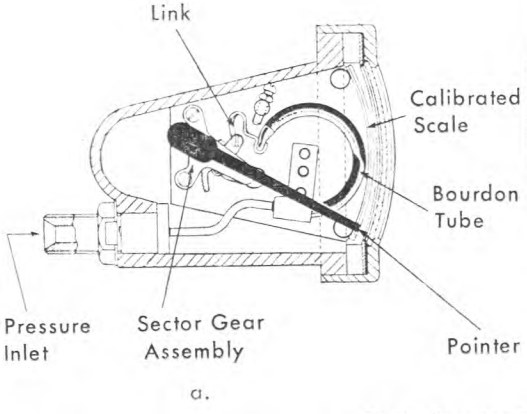
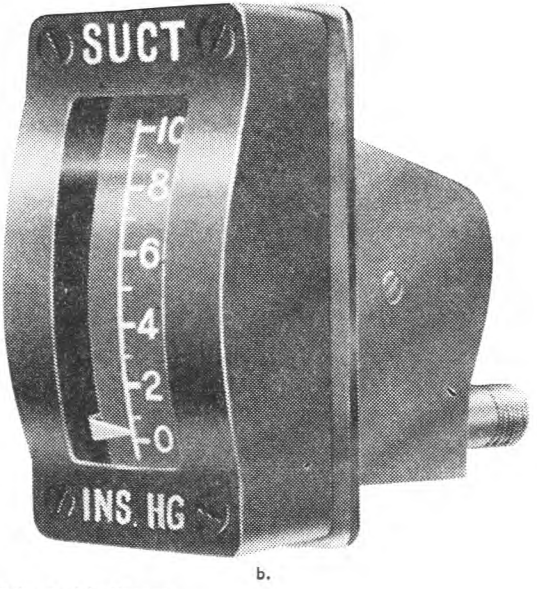


Fig 4 Diaphragm Type Gauge



a.



b.

Fig 3 Bourdon Tube and Indicator

5. **Boost Gauges.** Boost gauges indicate the gas pressure in the induction manifold of a piston engine. The gauge may have either a capsule or a diaphragm movement.Calibration of the dial is in inches of mercury.

6. **Oil Pressure Gauges.** Oil pressure gauges are fed with pressure from a hydraulic transmitter. The gauge mechanism is usually of the Bourdon tube type.

7. **Brake Pressure Gauges.** Brake pressure gauges are Bourdon tube instruments fed directly with air pressure from a pneumatic braking system or by a hydraulic transmitter in the case of a hydraulic braking system.

REMOTE INDICATING INSTRUMENTS

General

8. Remote indicating pressure gauges consist of a pressure sensor which converts pressure measurements into electrical anologue form. The electrical signals are passed to an indicator and displayed by a suitable meter movement against a calibrated dial.

Resistor Type

9. In the resistor type of pressure gauge changes in pressure vary the value of a resistor. The indicator is connected to the transmitter and consists of two windings A and B (*see* Fig 5). The values of the currents flowing through each winding depend on the value of the resistance R of the transmitter. A ratiometer mechanism measures the ratio of the currents in windings A and B and operates a needle moving against a calibrated dial.

Inductor Type

10. The inductor type consists of a transmitter, an indicator and an auto-transformer (*see* Fig 6). Pressure applied to the sensitive element of the transmitter causes linear movement of a laminated iron armature which moves in the bore of a stator. The stator has two windings, and the

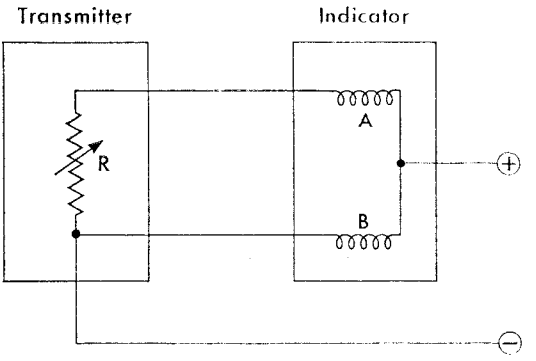


Fig 5 Resistor Type Pressure Gauge Circuit

relative values of the two currents alter as the armature moves. The ratio of these currents is measured by an ac type ratiometer in the indicator.

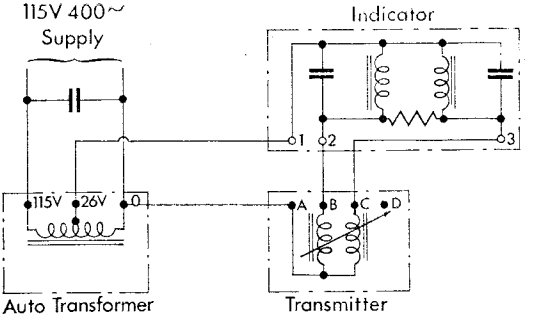


Fig 6 Inductor Type Pressure Gauge Circuit

Desynn Type

11. The Desynn type gauge consists of a pressure sensitive element, usually a diaphragm, which operates a resistor controlling the current to the indicator stator (*see* Part 3 Sect 1 Chap 1).

CHAPTER 3

FUEL CONTENTS GAUGES AND FLOWMETERS

CONTENTS

FUEL CONTENTS GAUGES

	Paras
Introduction	1– 2
Pacitor Type	3– 4
Errors in Gauges	5– 7
Volumetric Gauges	8– 9
Mass Unit Gauges	10–12
Summary of Errors	13–14

FLOWMETERS

Introduction	15
Description	16–18

FUEL CONTENTS GAUGES

Introduction

1. Fuel contents gauges indicate the amount of fuel contained in the aircraft fuel tanks. In this chapter only the pacitor type is described; information about other installations is provided in the AP 112G series. The Aircrew Manual for a particular type of aircraft gives the characteristics of the fuel contents gauge system installed, and particular attention should be paid to details of calibration accuracy, since not all the fuel indicated may be usable because of the design of the tanks and other engineering considerations.

2. Early fuel gauges were actuated by a mechanical linkage from a float in the fuel tank; the gauges were usually mounted externally on the upper surface of the wing where they could be seen from the cockpit in flight. A development of the float actuated gauge used a Desynn electrical transmission system in place of the mechanical linkage, and the fuel gauges were placed in the cockpit. Later systems use the electrical impedance of fuel to measure the quantity in the tank.

Pacitor Type

3. The principle used in the pacitor gauge is that the capacitance and therefore the impedance of a condenser varies according to the substance between the plates. Each condenser or tank unit consists of two concentric tubes arranged vertically in the tank, the tubes being separated by a gap which is filled with fuel, air, or both. As the fuel level in the tank falls, the ratio of fuel to air in the gap decreases, thereby altering the impedance of the tank unit. A diagram of a typical system is shown at Fig 1.

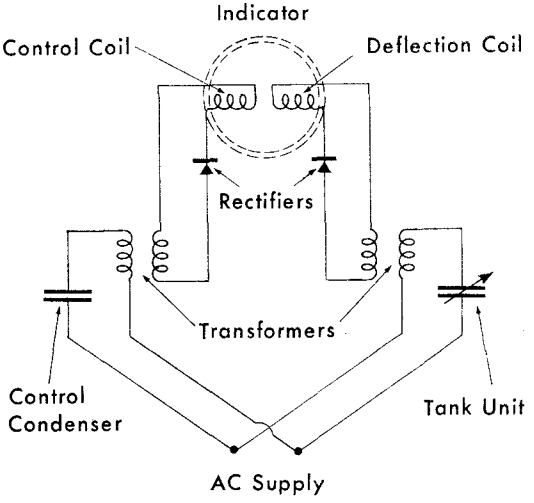
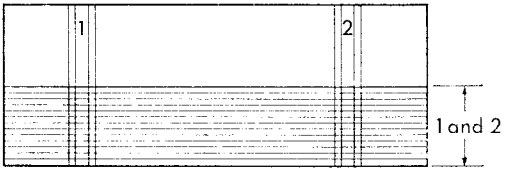


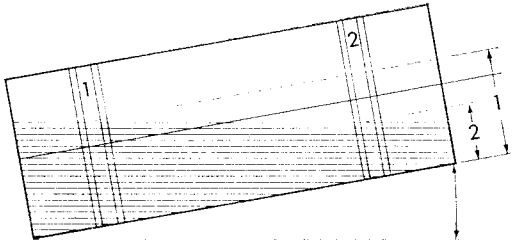
Fig 1 Pacitor Fuel Gauge Diagram

4. Taking first the right hand side of Fig 1, it is seen that the tank unit varies the current flowing in the transformer primary winding to which it is connected. The corresponding alternating

current induced in the secondary winding is converted to direct current by a rectifier and then fed to the deflection coil in the indicator. Meanwhile the same process is taking place in the control circuit on the left, except that, as the control condenser is of fixed value, the current in the control coil remains constant. Variations in the supply voltage affect both circuits so that the ratio of control coil current to deflection coil current remains constant for a given tank unit impedance. Unlike float operated gauges the pacitor type maintains an accuracy of $\pm 5\%$ for aircraft attitude changes of up to 150°. This is achieved by fitting two units in each tank as shown in Fig 2 and connecting them in parallel. The fall in fuel level at one end of the tank is compensated for by the rise at the other end.



Tank Level (capacitance of 1 and 2 equal)



Tank Tilted (capacitance of 1 increased and 2 decreased

Fig 2 Compensation for Tilt in the Pacitor System

Errors in Gauges

5. Fuel gauges are subject to instrument errors, installation errors and calibration errors. Instrument and installation errors are virtually constant for any one gauge. However, calibration error may vary widely, since it is caused by inconsistencies in the electrical conducting property of the fuel.

6. The pacitor type fuel gauge uses the following two properties of fuel to measure the quantity:

a. *Relative Permittivity.* The relative permittivity of fuel is the ratio of the condenser capacitance using fuel as the dielectric to its capacitance using air as the dielectric.

b. *Density.* This is the mass of fuel per unit volume at the ambient temperature.

7. Although no fixed relationship exists between the permittivity and the density of fuel it is assumed that an increase in density results in an increase in permittivity. The rate of increase varies according to the type of fuel used.

Volumetric Gauges

8. Volumetric gauges are usually calibrated in gallons assuming a constant value of permittivity. No allowance is made for variations in density. Gauges calibrated for either AVTUR or AVGAS are not subject to appreciable error since the values of permittivity and density for these fuels are virtually constant.

9. AVTAG fuel has a wide range of associated density values giving rise to large calibration errors. The higher the density and hence permittivity compared to the assumed values the greater will be the tendency of the gauge to overread.

Mass Unit Gauges

10. Mass unit gauges indicate the fuel remaining in terms of pounds and are calibrated to the expression: (Permittivity—1)/Density. Since permittivity increases with density, calibration errors are reduced as compared with the errors in volumetric gauges.

11. Since there are several fuels available for use with gas turbine and turbo-prop engines, each with its own value of permittivity and density, fuel contents gauges are calibrated to a formula using the mean of the highest and lowest values of permittivity in the range of fuels, and an approximate density value.

12. **Inferred Density Compensation.** Calibration error is reduced by incorporating a reference condenser into the electrical circuit. This unit consists of a condenser placed in the base of the tank so that it is immersed in fuel, and its capacitance is determined by the permittivity of the fuel. Since density is assumed to be a function of permittivity, the unit corrects for density error.

Summary of Errors

13. The percentage errors in fuel contents indication due to variations in the permittivity and/or density of fuel at +20°C are summarized in Table 1.

British fuel	American equivalent	Permittivity	Density gm/ml	% Error		
				Volume: uncompensated	Mass: uncompensated	Mass: inferred density (compensated)
AVGAS —DERD 2485	100/130 and 115/145	1·935 2·095	0·695 0·740	–8·3 +7·4	–4·7 +4·8	–3·0 +3·0
AVTAG —DERD 2486	JP4 and JP4B	1·995 2·185	0·740 0·800	–4·1 +11·3	–4·8 +4·9	–2·6 +2·9
AVTUR —DERD 2482	JP1 and JP1B	2·055 2·225	0·775 0·825	–4·1 +11·3	–3·7 +5·0	–2·8 +2·4
AVCAT —DERD 2488	JP5 and JP5B	2·070 2·260	0·785 0·850	–2·7 +14·4	–3·5 +4·9	–2·9 +1·5

14. It is evident from Table 1 that the errors in the mass unit gauge are about half those in the volumetric gauge. Therefore most modern aircraft gauges are of the mass unit type.

FLOWMETERS

Introduction

15. Flowmeters measure the amount of fuel being delivered to the aircraft engine, and consist of a transmitter and an indicator. The indicator usually shows the amount of fuel passing through the flowmeter as well as the total amount that has already passed. Most flowmeters measure volume, and a manual setting of fuel density is used to convert the volume to a mass reading. Information about specific types of flowmeter is to be found in the AP 112G series.

Description

16. **The Transmitter.** The transmitter usually consists of a chamber inserted in the fuel line which contains a rotor turning at a rate determined by the fuel flow through the line. A pick-off attached to the transmitter detects the rotation rate and passes an electrical signal to the indicator unit.

17. **The Indicator.** The indicator incorporates electrical circuits which convert the signal from the pick-off into either analogue or digital form. A manual setting of fuel density determines the conversion of volume into mass for display on a dial. Integrating circuits convert the flow rate into total fuel gone, which can then be shown on veeder counters. A reset control is usually provided for the counters.

18. **The Gravimetric Transmitter.** The gravimetric transmitter measures the mass of fuel passing through a chamber and eliminates the requirement for a manual setting of fuel density. The measuring device consists of a vane restrained by a calibrated spring. The fuel flows through the chamber impinging on the vane and deflecting it through an angle which is proportional to the rate of mass flow. A bleed vent provides compensation for changes in viscosity at low temperatures. The angle of the measuring vane is detected by a pick-off and passed as an electrical signal to the indicator unit.

CHAPTER 4

WARNING SYSTEMS AND DEVICES

CONTENTS

	Paras
Introduction	1
The Centralized Warning System	2– 6
The Auxiliary Warning Panel	7
Pressure Warning Lights	8– 9
Electromagnetic Indicators	10–13

Introduction

1. The failure of one of the vital systems in a modern aircraft can prejudice the success of its task and may lead to the loss of or damage to an aircraft. Warning devices are included in the various aircraft systems to indicate to the crew if there is a malfunction and enable them to take swift remedial action.

The Centralized Warning System

2. In the centralized warning system provision is made in the cockpit for immediate indication on a central panel of failure in a vital system. The type of warning installation varies according to the aircraft type, and reference must be made to the appropriate Aircrew Manual and the AP 112 series for details of particular installations.

3. Warnings of failure of essential services affecting the safety of the aircraft are notified to the pilot by the central attention getting system (CAGS) which flashes a prominent light and gives an audible warning signal in the headset. The appropriate indicator light on the warning panel is illuminated to specify the particular system that has failed. Warnings of failure in secondary systems may or may not be notified by the action of the CAGS lights and an audible warning. Primary and secondary warnings may be presented on separate panels or in separate sections of a common indicator unit, in which case the primary emergency warnings will be indicated by red lights and the secondary or auxiliary warnings by amber lights.

4. When an essential service fails an indicating caption on the panel is illuminated and will remain displayed until action has been taken to remedy the fault. Each panel is fitted with an adjustable screen to reduce the intensity of illumination at night. When triggered by a warning the CAGS will flash continuously. These lamps are operated by a flasher warning and excitation unit and are dimmed by a resistor. The audio unit when energized produces the sound of a fire bell in the crew's headphones.

5. After any part of the warning system has been caused to function by a service failure, the operation of the CAGS lamps and the audio warning unit can be cancelled but the warning caption will remain displayed. Further service failures will cause the system to respond as for the first failure.

6. **Warning Indicator Panel.** The panel captions are illuminated by miniature lamp filaments; each panel usually has four switches which function as follows:

a. *Test Switch.* Operation of this switch tests the warning system but not the systems from which the warnings are derived. All warning captions will be tested together with the lamps inset in the panel switch buttons. The CAGS lamps and audio warning unit will also function. When the test switch is released the panel lamps are extinguished but the CAGS and the audio unit remain energized until the cancel switch is depressed.

b. *Cancel Switch.* The cancel switch is a push-switch sprung to the off position. When the button is depressed the CAGS lamps and the audio warning are cancelled. If the system was activated by a genuine warning, the caption will remain illuminated until action is taken to remedy the fault.

c. *Mute Switch.* When the mute button is pulled out all warnings are rendered inoperative except for fire warnings. The M button must always be in during flight. When the M button is pulled out the switch becomes illuminated.

d. *Fire Extinguisher Switch.* The button of the fire extinguisher push-switch contains a lamp which is illuminated at the same time as the fire warning caption. Whenever the button is depressed it will cause the fire extinguisher to be operated.

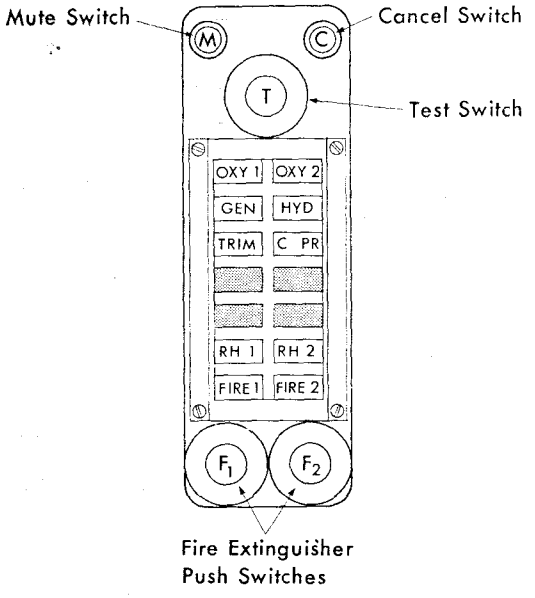


Fig 1 A Typical Standard Warning Panel

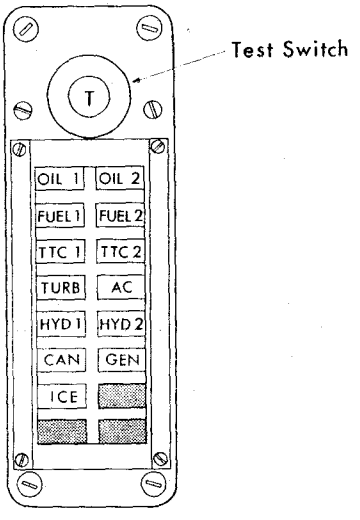


Fig 2 A Typical Auxiliary Warning Panel

The Auxiliary Warning Panel

7. Some aircraft in addition to being fitted with centralized warning panels are also fitted with auxiliary or secondary warning panels. These panels group the less important warnings together in one convenient display unit.

Pressure Warning Lights

8. **Fuel Pressure Warning Lights.** Where it is necessary to draw attention to abnormal pressures a light is fitted which, under normal circumstances, is off. Such lights are operated by a pressure sensitive switch of the diaphragm type. Should the fuel pressure drop below a pre-detmined value, movement of the diaphragm closes the switch and completes a circuit which lights the warning lamp.

9. **Cabin Pressure Warning Lights.** A cabin pressure warning light switch similar in principle to that described above differs only in that both of the pressures affecting the opposite sides of the diaphragm are variable. The switch closes when the difference in pressure between the air outside and that within the cabin exceeds a certain value.

Electromagnetic Indicators

10. Commonly known as "dolls eyes" these are used for a variety of warning and other indications. Their advantage over the electric bulb type indicators is that they are immune from the risk of filament failure.

11. The indicator shows either a black or a white face. By convention the white surface shows whenever the indicator is drawing attention to any particular occurence or operating condition.

12. **Construction.** The indicator consists of a rotating ball under a hemispherical window, one half of the ball coloured black and the other half coloured white. A small electromagnet is contained in the case behind the window and an iron armature is embedded in the eyeball.

13. **Operation.** When the indicator is energized the electromagnet attracts the iron armature, causing the eyeball to rotate and present a new surface at the window. When the indicator is de-energized the eyeball rolls back to the original position. Some indicators show white when energized and others black; the choice depends on whether the indicator is required to attract attention when energized or when not energized.

CHAPTER 5

ACCELEROMETERS AND FATIGUE METERS

CONTENTS

ACCELEROMETERS

	Paras
Introduction	1– 4
Basic Principles	5– 6
Accelerometer Type KAE 0201	7–12
Accelerometers Mk 2 and Mk 2A	13–16

FATIGUE METERS

Description	17–20
Six Counter Type	21–22
Eight Counter Type	23–24

ACCELEROMETERS

Introduction

1. An indicating accelerometer is an instrument used in aircraft to provide visual indication of acceleration components in the direction of the Z axis (Fig 1). Operational aircraft accelerometers embody in addition auxiliary pointers which preserve a reading of the maximum and minimum accelerations sustained during any period; these can be re-set as required.

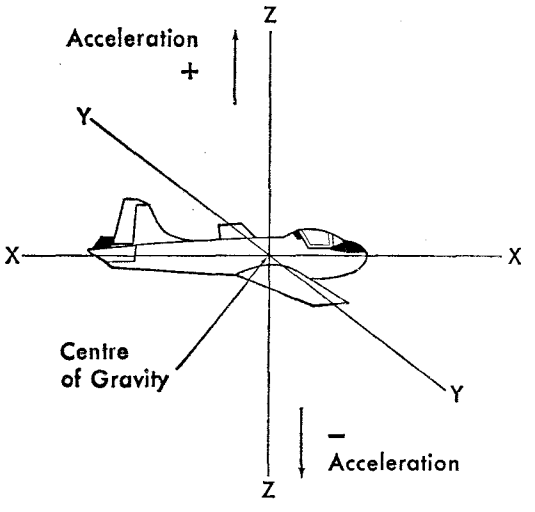


Fig 1 Aircraft Axes

2. Indicating accelerometers are reference instruments used to indicate the Z accelerations acting on the aircraft, so that manoeuvres which impose excessive loads on the aircraft will not be executed. The indications of the accelerometer also serve to show the accelerations imposed on an aircraft by gusts and turbulent air in otherwise level flight. This serves as an index to some of the stresses encountered under normal operating conditions.

3. The indicating accelerometer gives a reasonably accurate indication of the accelerations encountered under flight conditions. However, the indications of the instrument with respect to accelerations of extremely short duration, such as landing shocks, should be treated with reserve, since the accuracy under these conditions is dependent on the damping characteristics and no generalization is possible.

4. The accelerometer should be mounted on a rigid part of the aircraft structure in the cockpit. Accurate results cannot be obtained from accelerometers mounted on anti-vibration mountings, which would tend to reduce the effect of accelerations on the instrument.

Basic Principles

5. **General Description.** In general, an accelerometer consists of a mass suspended from calibrated springs. The weight of the mass is statically balanced by the tension in the springs.

When the aircraft accelerates the mass is displaced and the spring tension alters until it once more balances the force imposed (equal to the mass times the acceleration). In the indicating accelerometer, the suspension is arranged so that the mass responds to accelerations along the Z axis of the aircraft (subject to the instrument being installed correctly).

6. **Maximum Indications.** The accelerometer pointer registers the acceleration along the Z axis at any instant. However, it is even more important to know the maximum acceleration experienced during a manoeuvre or when flying in turbulent conditions. For this purpose accelerometers in current use are provided with two auxiliary pointers, mounted concentrically with the main pointer, and which indicate respectively maximum positive and maximum negative accelerations. The auxiliary pointers are driven by a small plate attached to the staff of the main or instantaneous pointer. This plate has projections which move two ratchet wheels, engaging in opposite directions, one of which carries along the maximum positive pointer, and the other the maximum negative pointer, to the position of the main pointer. A light pawl and spring hold the auxiliary pointers at their respective maximum indications until that acceleration is exceeded or until the instrument is re-set by the operation of a re-setting knob. Both auxiliary pointers are then returned to the position of the main pointer by the action of hairsprings.

Accelerometer Type KAE 0201

7. Basically the mechanism consists of two masses elastically suspended about parallel axes and so interconnected that acceleration in the vertical axis only is indicated. Fig 2 shows a schematic arrangement of the mechanism.

8. The masses are carried on the ends of two cantilever arms which are attached to two parallel shafts. The rotation of these shafts is controlled by two springs. Each spring is anchored to the body of the mechanism at one end while the other end is attached to the appropriate shaft through a connecting link. The shafts each carry a sector gear, the sector gears being in mesh with each other. An additional sector gear meshes with a pinion to the staff of which the indicating pointer is attached.

9. As the accelerometer scale is linear, there must be a linear relationship between acceleration and angular movement of the weight arms. This is achieved by using linear springs attached to connecting links, the links being at right angles to the cantilever arms, so that the rotation of the masses is directly proportional to the extension of the springs, and hence to the imposed acceleration. Thus as the effect of acceleration on the masses decreases owing to a decrease in the effective radius of the weight arms, a proportional decrease in the effective length of the spring arms takes place.

10. Acceleration in any direction in the plane of the dial may be resolved into two components, vertical and horizontal, which may be considered separately. Vertical accelerations will cause the rocking shafts to rotate in opposite directions and thus operate the indicating pointer. Horizontal acceleration components exert equal

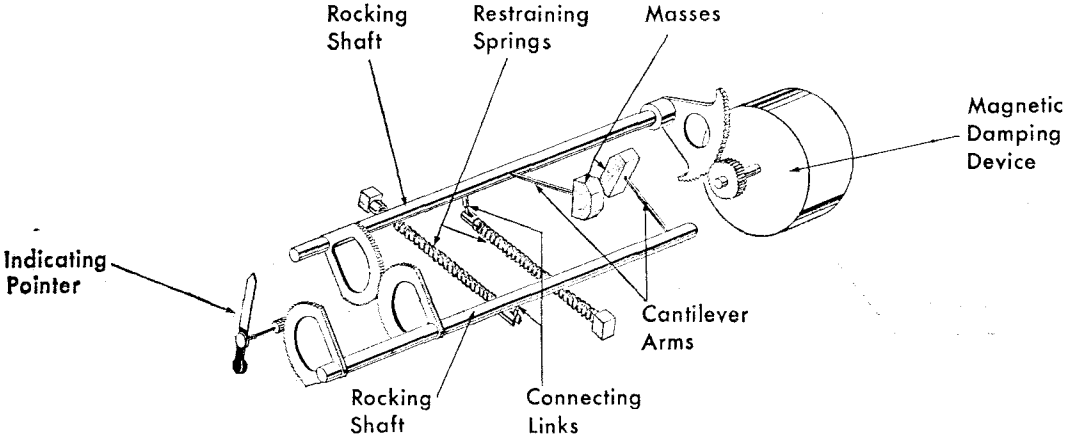


Fig 2 Schematic Arrangement of Accelerometer Mechanism Type KAE 0201

forces on each of the masses but will tend to rotate the rocking shafts in the same direction and therefore, since the forces are identical, will cancel each other as the shafts are geared together. Thus only vertical acceleration components will be indicated provided that the two spring mass systems are identical. It is therefore essential that these should be properly balanced and adjusted.

11. In the Type KAE 0201 (KB 482/01) and Mk 3 accelerometers the rocking shaft is extended to carry a sector gear which drives a magnetic drag damping device to prevent violent pointer fluctuation under short period accelerations.

12. **Dial.** The accelerometer KAE 0201 is designed to operate over the range -5*g* to +12*g* and the dial (Fig 3) is marked in equal graduations of 0.2*g* throughout this range. The figures and graduations are fluorescent.

Fig 3 Accelerometer KAE 0201—Dial

Accelerometers Mk 2 and Mk 2A

13. The principle used in the Mk 2 and Mk 2A accelerometers is that a mass-weight is arranged so that it is free to move on a vertical axis and is coupled to a main shaft, so that when acceleration forces are imposed on the mass, the main shaft is caused to rotate. The linear movement of the mass is thus converted to rotary movement of a set of three pointers, one to indicate instantaneous accelerations and the other two to remain at maximum indications, plus and minus, until reset manually.

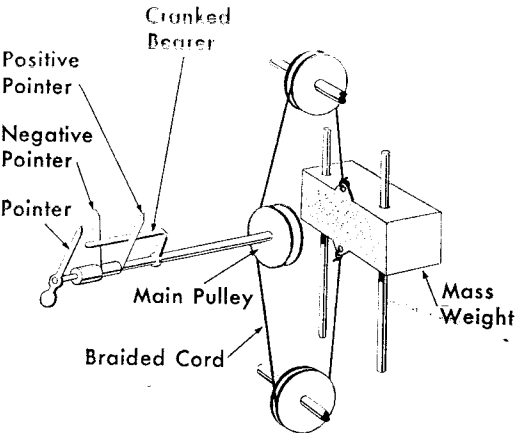


Fig 4 Accelerometers Mk 2 and 2A Mechanism

14. The basic mechanism (Fig 4) consists of a mass-weight free to move up and down on two vertical slides, carrying two braided silk cords which pass over two free pulleys and are attached to a main pulley, after making a partial turn around it. The main pulley is secured at the rear of a main shaft, to which a pointer is attached at the front end. Movement of the weight is therefore communicated to the shaft, and this in turn will cause the pointer to indicate the angle through which the shaft has been moved.

15. A cranked lever is attached to the shaft, and the horizontal arm of this lever is interposed between the positive and the negative pointers so that they will be moved when the shaft rotates, and will remain in their new positions on the return of the shaft to the neutral position. A device is fitted to the mechanism to enable the recording pointers to be reset to the neutral position when so desired. A further device is fitted to damp out vibrations and prevent violent pointer fluctuations under short period accelerations.

16. **Dials.** The accelerometer Mk 2 is designed to operate over the range -5*g* to +10*g*, and the dial (Fig 5) is graduated in equal divisions of 0.5*g* throughout this range. The figures and graduation marks are fluorescent. The indicating accelerometer Mk 2A (Fig 6) is designed to operate over a range of -2*g* to +5*g*, and the scale is graduated in equal divisions of 0.2*g*.

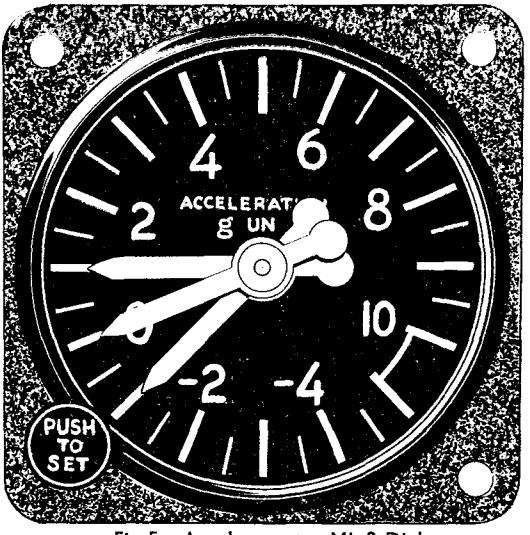


Fig 5 Accelerometer Mk 2 Dial

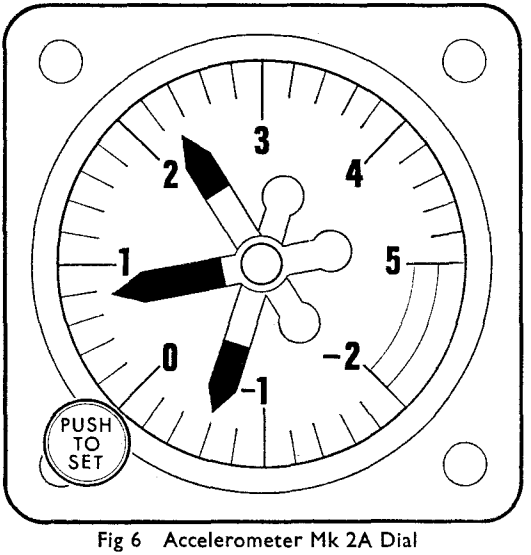


Fig 6 Accelerometer Mk 2A Dial

FATIGUE METERS

Description

17. The fatigue meter is a counting accelerometer, the primary function of which is to measure and record, at the centre of gravity of an aircraft, the accelerations normal to the flight-path to which an aircraft is subjected whilst in flight. It will record the number of times pre-determined acceleration threshold values have been exceeded. The acceleration thresholds are arranged so that a count of acceleration is not completed until the acceleration returns very nearly to 1*g*. This arrangement ensures that the counters record the large accelerations, but not small superimposed accelerations which cause negligible fatigue damage to normal aircraft, and at the same time it also ensures that excessive fatigue counts are not obtained on aircraft with a large dynamic response.

18. The instrument is required to function only while the aircraft is in flight. This condition is attained by operating an air speed switch on the pitot-static system to switch on the fatigue meter soon after unsticking, and to switch off over the runway threshold. For fighter-type aircraft that spend an insignificant part of a sortie in the turbulent air conditions associated with having flaps and wheels down, an air speed switch is unnecessary and fatigue meter switching is accomplished by an undercarriage up-lock microswitch.

19. An aircraft fitted with a fatigue meter may be allowed to fly until its predetermined safe fatigue life, in terms of acceleration counts recorded, has been reached.

20. There are two basic types, having either six or eight counters. Various forms of each type are available for different applications.

Six Counter Type

21. The fatigue meter comprises an accelerometer incorporating a commutator system to give the acceleration threshold pick-offs, six electrical relays, and electro-magnetic counters for recording the counts of acceleration. A nominal 28V DC electrical supply at 0.7A maximum is required to operate the relays and counters.

22. **Accelerometer.** The accelerometer used has a double mass and spring system with eddy current damping. It will record low frequency accelerations and attenuate all accelerations of higher frequencies.

Eight Counter Type

23. The fatigue meter comprises an accelerometer incorporating a commutator system to give eight acceleration thresholds. The counts of acceleration are recorded through eight electrical relays and electro-magnetic counters.

24. **Accelerometer.** The accelerometer used has a double mass and spring system with eddy current velocity damping. It will record low frequency accelerations and attenuate all accelerations of higher frequency.

Printed for Her Majesty's Stationery Office by Wilfred Edmunds Ltd., Station Road, Chesterfield.
1608/646190/3,750/4-70/468/2.

CHAPTER 6

TORQUEMETERS

CONTENTS

	Paras
Introduction	1–2
THE ELECTRONIC TORQUEMETER	
Components	3–6
Principle of Operation...	7–11
THE PHASE COMPARISON TORQUEMETER	
General	12
Principle of Operation	13
THE HYDRAULIC TORQUEMETER	
General	14
System No 1	15–19
System No 2	20–23

Introduction

1. The piston engine has been superseded by the jet engine in all medium and large helicopters and in the majority of small helicopters and fixed wing aircraft. This has meant an increase in available power. The pilot must be aware of the amount of power being absorbed by the transmission system. To exceed laid down limits would inevitably result in the structural failure of a transmission component. The amount of torque being applied to the transmission system at any given time can be related to fuel consumption, air temperature and density, rpm and pitch setting. By incorporating a torquemeter system the pilot has an immediate indication of the power being applied to the helicopter transmission system at all times and under all conditions of flight.

2. There are several types of torque registering systems in use on helicopters. This chapter describes in broad outline the operation of the three main types of torque registering systems in current use, *ie* hydraulic, phase comparison and electronic.

THE ELECTRONIC TORQUEMETER

Components

3. A typical electronic torquemeter consists of three basic units.

4. **Electronic Unit.** The electronic unit contains an oscillator and an amplifier.

5. **Transducer Unit.** The transducer unit contains a shaft, attached strain gauges and a rotary transformer which transfers signals to and from the strain gauges.

6. **Indicator(s).** The indicator consists of an electronic meter with the presentation calibrated

in foot-pounds or as a percentage. The unit contains a press-to-test facility.

Principle of Operation

7. A block schematic diagram of an electronic torquemeter is shown at Fig 1. An 8kHz signal generated by the oscillator is fed via one of the transformer windings to the strain gauges. These gauges are firmly attached to the shaft and form an electrical bridge circuit with the transformer windings. The outer windings and structure of the transducer are prevented from rotating by a strut secured to the aircraft frame or gear-box casing.

8. The resistance of the strain gauges varies in proportion to the amount of torque in the transmission shaft. Under zero torque conditions each bridge circuit will be in a state of electrical balance and there will be no output. As torque is applied and increased the bridge will become increasingly unbalanced and the bridge circuit's output will increase in proportion.

9. This bridge circuit output is fed via the transducer windings to the electronic unit amplifier. After amplification the signal is fed to the indicator unit thereby showing the torque applied to the shaft.

10. The press-to-test facility on the indicator disconnects the output from the transducer and feeds a pre-set "unbalanced" signal to the amplifier. The signal will cause the indicator to register a specific torque thus proving the electronic unit and indicator but not the transducer. Some helicopters have only one or two pilot indicators but others make use of the following three types of output from the electronic unit:

a. Output to an indicator or indicators.

b. Output to drive a recorder unit.

c. Output to an overtorque relay unit to give an audio warning.

11. The electronic torquemeter is normally positioned between the power unit and the transmission system thus indicating the total torque due to the power being absorbed by the main and tail rotors. Some aircraft types have two torquemeters, one supplying information about the main rotor, the other information about the tail rotor. The two signals are added by the electronic amplifier and shown on one indicator as total torque.

THE PHASE COMPARISON TORQUEMETER

General

12. A phase comparison torquemeter is part electrical and part mechanical. A typical unit compares the angular displacement of a length of shaft under torque, the output shaft, with that of a second shaft, the datum shaft, not under load. As shown in Fig 2, the torquemeter consists of the front section of the power turbine shaft and the datum shaft. Toothed wheels fixed to both shafts constitute the rotors of the phase generator. The phase generator is also known as a torque transducer.

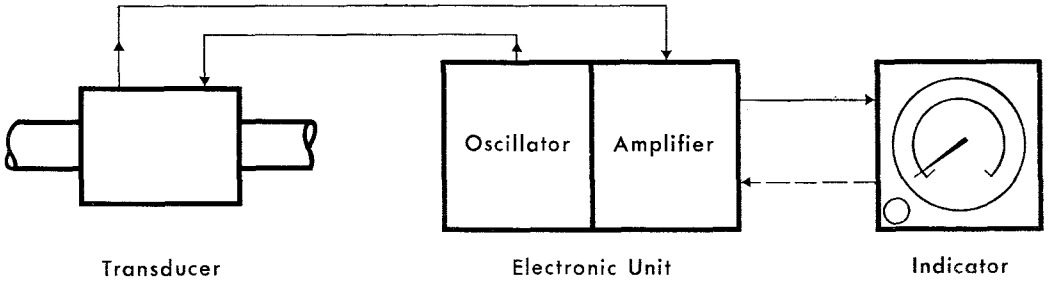


Fig 1 An Electronic Torquemeter

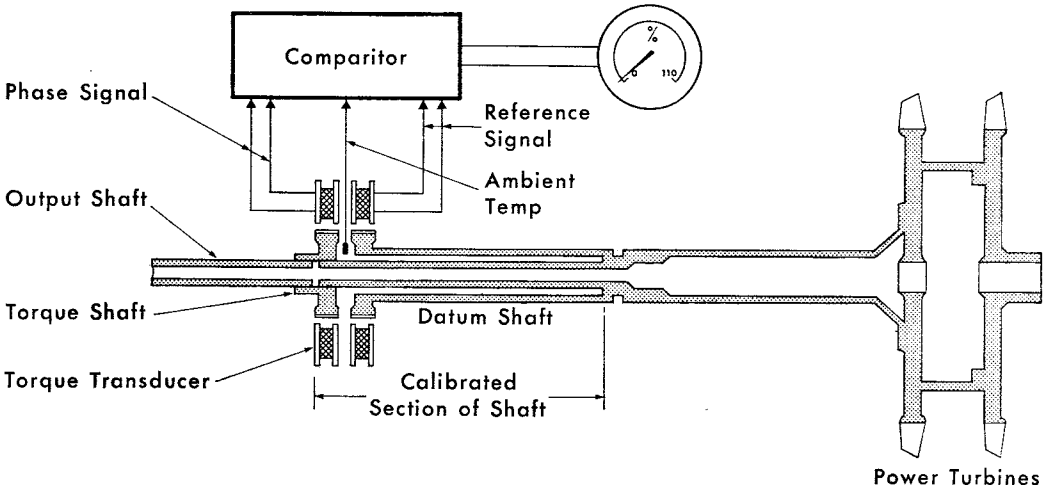


Fig 2 A Phase Comparison Torquemeter

Principle of Operation

13. Under no-load conditions the toothed rotors will turn without any relative movement. As torque is applied and increased the output shaft will twist along its length. This movement will have no effect on the datum shaft. The toothed rotor on the output shaft will have an angular displacement when compared with the rotor on the datum shaft. This displacement is conveyed, by the transducer coil windings, as a phase difference to the comparator. The comparator generates a signal, dependent on the phase difference, to drive the pilot's indicator thereby registering torque.

THE HYDRAULIC TORQUEMETER

General

14. There are two types of hydraulic torquemeter systems in general use. In this chapter they are called System No 1 and System No 2.

System No 1

15. System No 1 provides a direct and accurate indication of engine power output and is housed in the engine main reduction gear casing. The principal components of the torquemeter are the annulus gear, a steel torquemeter ring which is fixed within the reduction gear casing, and the torquemeter oil pump. System No 1 is shown at Fig 3.

16. The engine output torque is proportional to the torque experienced by the annulus gear of the epicyclic gear train and can be determined by the effort required to prevent the gear rotating. The annulus gear is allowed limited rotational freedom inside a fixed torque ring. Rotation is prevented by high pressure oil acting against stubs projecting from the periphery of the annulus gear and torque ring. By measuring the oil pressure required to prevent the gear rotating, engine output torque may be determined.

17. The torque ring and annulus gear each have fifteen intermeshing stubs. Each stub carries a spring-loaded seal. Steel cover plates on both sides enclose and locate the annulus gear axially without interfering with rotational freedom. A circumferential gallery round the outer surface of the torque wing interconnects the high pressure chambers.

18. The high pressure pump is a gear type and is supplied with engine oil under pressure. The pump boosts the pressure up to that required by the torquemeter assembly. A relief valve reduces the pressure should a pre-determined torque delivery pressure be exceeded.

19. The torquemeter pump delivers oil to the high pressure gallery where it is fed through drillings to the fifteen high pressure chambers. The flow of oil from the chambers is controlled

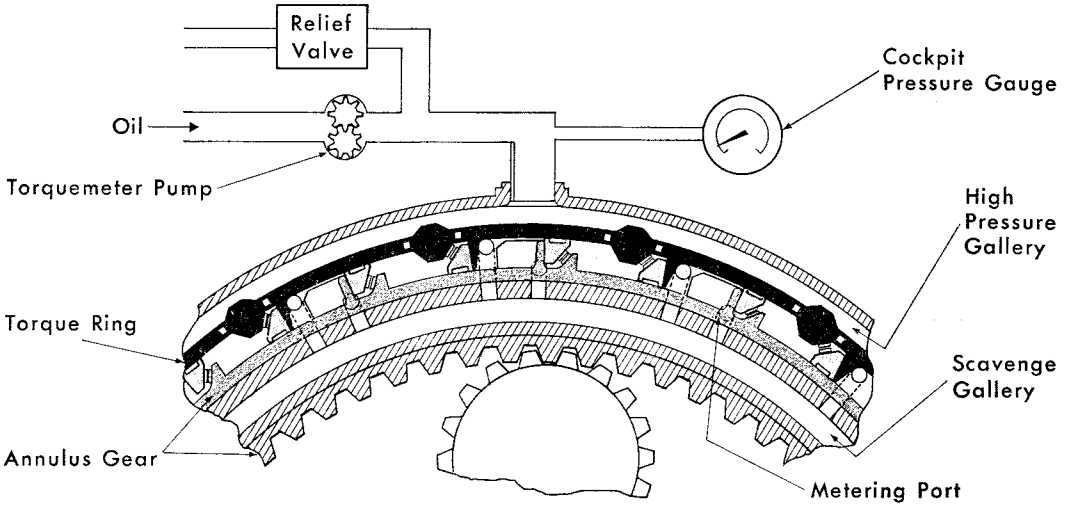


Fig 3 Hydraulic Torquemeter System No 1

by movement of the annulus gear which varies the relative position of the stubs and metering ports in the outlet side plate. As torque increases so the metering ports progressively close, thereby raising the pressure in the oil supply line which also has a tapping to the pilot's torque gauge. Any increase in oil pressure will be indicated on the pilot's torque gauge as an increase in torque.

System No 2

20. System No 2 has two torquemeter mechanisms built into the main gear-box input section. This system is used in some twin-engine helicopters. The mechanisms measure the torque applied by each engine to the transmission system and indicate it on two dual needle instruments in the cockpit. These readings enable a pilot to accurately match the torque inputs so that each engine is carrying an equal load. They also indicate the total torque input to the transmission system.

21. The power turbine of each engine is connected to the main gear-box by a drive shaft. The drive shaft connects to the high speed input gear in the input section of the main gear-box (*See* Fig 4). The high speed input gear drives a spur gear on the free wheel unit which in turn drives the helical gear. This gear meshes with the input bevel gear helical drive. The meshing of these two gears can be compared to pushing two ramps or inclined planes together; the harder one pushes against the other, the further up the surface it slides. Thus the two gears tend to move apart in opposite linear motions. Tapered roller bearings prevent the input bevel gear from moving axially but movement is allowed on the free-wheel unit helical gear.

22. The free-wheel unit assembly is mounted in straight roller bearings which allow the entire gear assembly to move linearly. Therefore all the gear reaction is taken up by the free-wheel unit assembly. As the unit moves forward (in the direction of the arrow) it carries with it a piston that is mounted on the outer race of a ball-bearing. This bearing allows the piston to remain rotationally fixed but allows the free-wheel unit to rotate. Spring loaded against the torquemeter piston is a torquemeter valve which will be compressed by the forward motion of the piston.

23. A geared oil pump supplies oil under pressure to both torquemeter valves. If no torque is being applied the torquemeter valves will be closed. As torque is applied the valve will be forced to crack open allowing some of the high pressure oil to enter the piston chamber. When

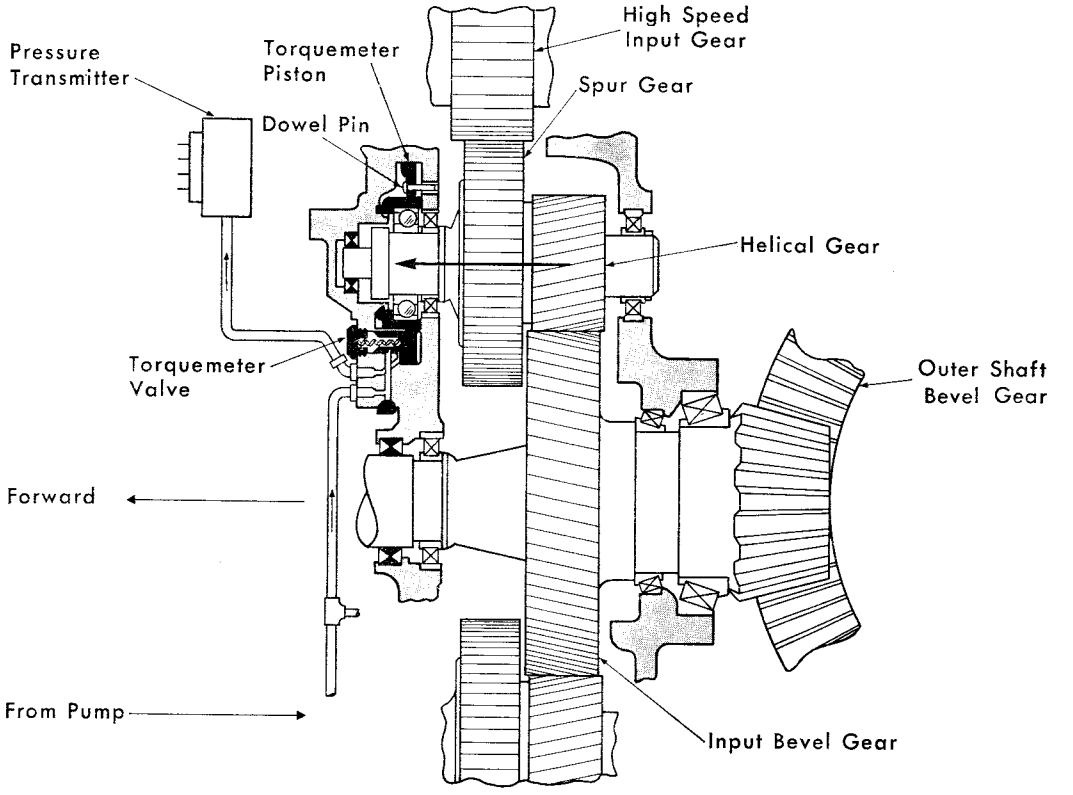


Fig 4 Hydraulic Torquemeter System No 2

the oil pressure acting upon the piston in the chamber is sufficient to overcome the movement of the free-wheel unit it will tend to close off the valve, thus retaining a specific oil pressure in the chamber. Increasing the torque will cause the valve to crack open again thereby increasing the oil pressure in the chamber. This fine balance of shaft movement to oil pressure is continuously maintained. The oil pressure in the chamber will be proportional to the valve movement which in turn will be proportional to the torque applied to the free-wheel unit. The piston chamber is connected to an external pressure transmitter which in turn operates a cockpit torquemeter gauge. The gauge reads percentage of torque and has two needles indicating the torque input from both engines.

PART 5

SECTION 2

OPTICAL INSTRUMENTS

Chapter

1 **Drift Recorder, Mark 2**

3456 D
Part 5. Sect 2.
Chap 1.

CHAPTER 1

DRIFT RECORDER, MARK 2*

CONTENTS

	Paras.
Introduction	1
Description	2–12
Cleaning	13–14
Operating Drills	15–18
Alignment	19

Introduction

1. The Drift Recorder Mk. 2* is designed to provide a simple and convenient means of measuring drift and checking groundspeed. The instrument can be fitted at about the height of, and close to, the navigator's table. It can be used at low altitudes, in bumpy flying conditions, over broken cloud, over choppy sea, and over lighted territory at night.

Description

2. An image of the ground almost vertically below the aircraft is formed by an optical system, the object end of which protrudes a short distance through the side of the aircraft. A pointer in the focal plane is made to follow the apparent track of a ground object as it is seen to move across the field of view. This pointer is connected, through a pantograph mechanism, to a pencil which is moved by hand over a fixed, slightly frosted, glass plate. A record on a larger scale of every movement imparted to the pointer is by this means produced on the plate.

3. Underneath the glass plate is a circular grid card bearing a number of parallel lines, which can be clearly seen through the glass plate. This grid card is rotatable to make the direction of the lines on it as nearly as possible parallel to the average direction of the recorded tracks on the glass plate. The angular movement of the grid card from its zero position necessary to align the grid lines and recorded tracks is equivalent to the angle of drift, and can be read off on a fixed scale against a pointer on the grid card.

4. Rotation of the grid card is transmitted by means of a linkage to one of the field lenses on which is superimposed a graticule, so that grid card and graticule rotate together. The graticule consists of four parallel grid lines, which always remain parallel to the lines on the grid card, and two transverse groundspeed timing lines. The latter converge slightly to allow for the perspective effect due to the oblique view of the ground given by the instrument.

5. The parallel lines on the graticule can be lined up with the general direction of the ground movement to obtain drift readings on objects moving past the graticule too fast to be followed by the pointer, *e.g.* at low altitudes.

6. When the drift records have been taken, and the grid card is aligned with them to obtain the drift angle, the timing lines are thereby brought into position for groundspeed measurement. By observing the height and by timing the passage of an object between the two timing lines, the groundspeed can be calculated on the simple circular computer provided.

7. The recorder is retractable and can be withdrawn to protect the projecting end from sea spray or dirt when landing and taking-off, and to avoid wind resistance when it is not in use for a long time. It is supported on slide rails attached to the aircraft, and can be moved along them although it is normally held by springs in a channel at each side.

8. A light metal casting houses the optical system. Soft rubber pads are provided, one to act as a buffer against the inside of the fuselage, and to exclude rain and draught where the viewing prism projects, and the other to act as an eyeguard.

9. The left half of the frame has a hinged box lid which houses the pantograph and graticule mechanisms (*see* Fig. 2). The lid contains the pantograph mechanism and also carries the eyepiece, its lens and computer. The base carries, in addition to the graticule lens, a link mechanism connecting the graticule lens to the card carrier under the ground glass screen.

10. To protect the optical system against moisture, a window is provided inside the aperture at the outer end of the tube; the inner end is sealed by the lens beneath the graticule.

11. **Red Polarizing Filter.** A red polarizing filter (Stores Ref. 106B/69) can be supplied. It is designed to reduce glare and reflections when flying over the sea. It can be inserted, glass end upwards, in the rubber eyepiece, the eyepiece then being loosened and rotated on its screw threads until reflections reach their mimimum intensity. When not in use it should be stowed in the canvas bag provided, and the latter tied to the neck of the recorder.

12. **Field of View.** The field of view of the instrument is approximately 38 degrees and this allows drift measurements to be made down to about 400 ft., at 150 knots. The central line of view is inclined 15 degrees to the vertical in a plane at right angles to the fore-and-aft axis of the aircraft.

Cleaning

13. Pencil tracks on the glass plate are best removed with a damp rag. India rubber should not be used as the resulting small fragments tend to clog the instrument.

14. If the lid is raised to clean the field lens and graticule, care must be taken not to damage the pointer whilst doing so.

Operating Drills

15. **To Obtain Drift:—**

(*a*) Inform pilot so that he can maintain steady heading and airspeed.

(*b*) Follow the movement of an object on the

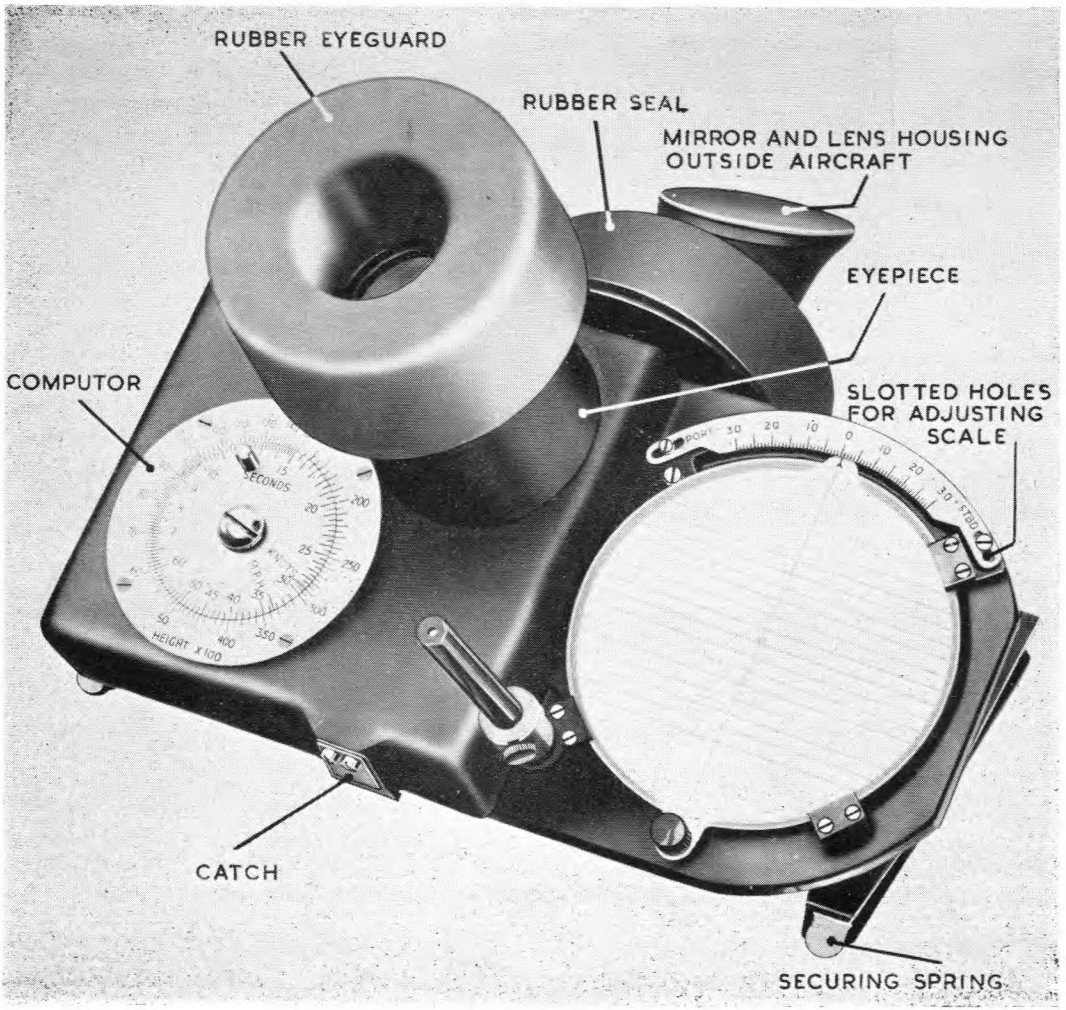


Fig. 1. Drift Recorder, Mk. 2*

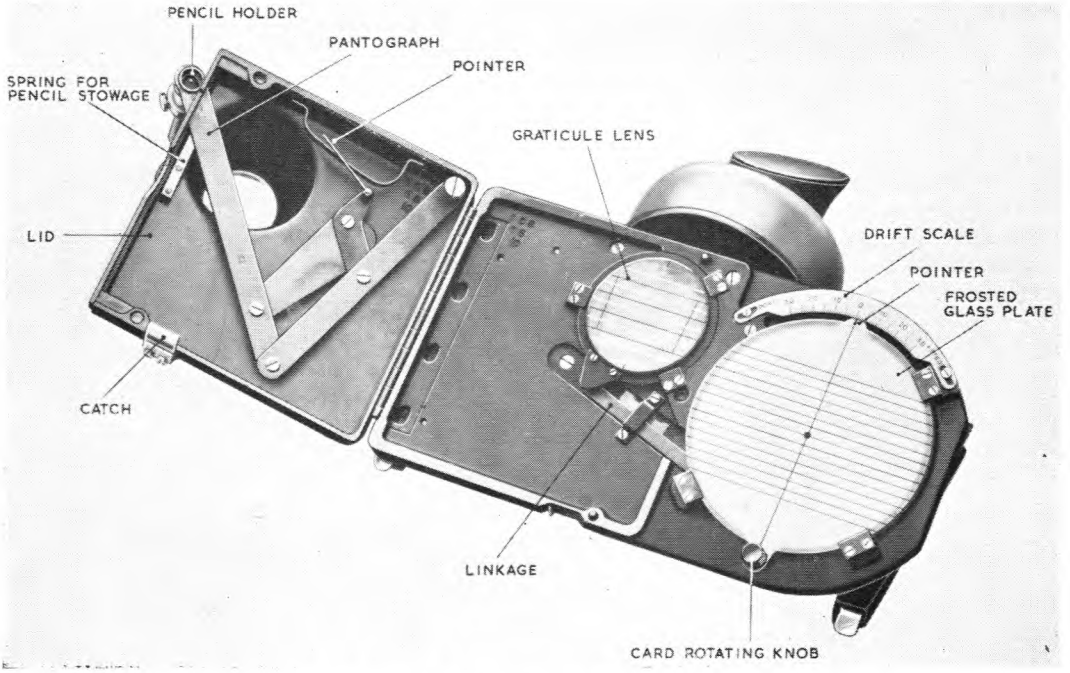


Fig. 2. Drift Recorder, Mk. 2*—Mechanisms

ground across the field of view with the pointer. This will cause a pencil track to be drawn on the glass plate.

(*c*) Repeat (*b*) with other objects.

(*d*) Rotate the grid plate in azimuth until the lines thereon are parallel to the average direction of the pencil tracks.

(*e*) Read off the drift angle on the scale.

(*f*) Inform the pilot that the observation has been completed.

16. **To Obtain Drift at Night:—**

(*a*) Use the drill as in para. 15 but follow illuminated objects on the ground with the luminous end of the pointer.

17. **To Obtain Drift at Low Levels:—**

(*a*) Inform pilot.

(*b*) Rotate the graticule until objects on the ground appear to travel along the parallel lines.

(*c*) Read drift on scale.

(*d*) Repeat three or four times and average the results.

(*e*) Inform the pilot when finished.

18. **To Measure Groundspeed:—**

(*a*) Inform pilot.

(*b*) Obtain drift and set on the instrument.

(*c*) Time the passage of a ground object between the timing wires.

(*d*) Repeat (*c*) three or four times and take average time.

(*e*) Compute the true height of the aircraft or obtain it from a radar altimeter.

(*f*) Set height in hundreds of feet against time in seconds obtained at (*d*) above on the circular slide rule.

(*g*) Read off groundspeed against the appropriate pointer.

(*h*) Inform pilot when finished.

Alignment of the Drift Recorder

19. To align the Drift Recorder correctly in relation to the fore-and-aft axis of the aircraft proceed as follows:—

(*a*) Select a level piece of ground.

(*b*) Drop plumb lines from the nose and tail of the aircraft and mark the fore-and-aft axis on the ground.

(*c*) Set drift reading at zero.

(*d*) Set a peg in the ground so that it can be observed centrally in the field of view on the side nearest the observer. Remember that the optical system is inverting.

(*e*) From this peg, lay off a piece of string to cut the fore-and-aft axis at right angles. To do this, take a piece of string about half as long again as the perpendicular from the peg to the fore-and-aft axis. With the peg as centre, mark off the two points on the fore-and-aft line equidistant from the peg. Insert a second peg halfway between these points and draw the string taut between this peg and the one mentioned in sub-para. (*d*) above.

(*f*) Draw a track on the Drift Recorder with the pointer following the line of the string now stretched between the two pegs.

(*g*) Set the cross line of the grid card so that it is parallel to the track drawn.

(h) Slacken the screws holding down the drift scale, adjust it so that the pointer reads zero (being careful not to move the grid card) and screw down the scale.

PART 6

GLOSSARY OF TERMS

GLOSSARY OF TERMS

The following key is used to indicate the source of the definition:

ASCC = Air Standardization Co-Ordinating Committee Terms and Definitions

NATO = NATO Glossary of Military Terms and Definitions

ICAO = International Civil Aviation Organisation

(For a Glossary of Computer Terms *see* Part 3, Sect 2, Chap 2)

Accelerometer Performance Parameters

a. *Threshold.* The minimum acceleration input to an accelerometer required to produce an electrical output.

b. *Sensitivity.* The minimum change in acceleration input required to produce a decisive change in electrical output and which, when reversed, also causes a decisive change in output. It is usually assumed that the incremental change is made from an arbitrary, but relatively large input compared to the incremental change.

c. *Zero Uncertainty (Null Uncertainty).* Zero uncertainty is a measure of the maximum accelerometer output when there is zero acceleration input, measured under test conditions. It is caused by misalignment of the accelerometer mounting pad with the required datum with zero feed back current in the circuit. It is usually expressed as the equivalent angle of misalignment which would cause that maximum output with zero acceleration input.

d. *Zero Stability (Null Stability).* Zero stability is a measure of the random changes in accelerometer output whilst there is zero acceleration input. The changes are caused mainly by fluctuations in bias compensation, but can also be caused by mechanical instabilities. Zero stability is usually expressed as the equivalent angle of misalignment which would cause that degree of change.

Air Mileage Indicator

An instrument which gives a continuous display of air distance flown.

Air Speed Indicator

An instrument which displays the speed of an aircraft relative to the surrounding air.

Altimeter, Pressure

An instrument which displays vertical distance above a selected pressure datum.

Altimeter Setting

The pressure datum in millibars or inches of mercury set on the altimeter's sub-scale.

Artificial Horizon

A device that indicates attitude with respect to the true horizon. A substitute for a natural horizon, determined by a liquid level, bubble, pendulum or gyroscope, incorporated in a navigating instrument. (ASCC)

Astrodome

A transparent bubble, calibrated for refraction, mounted in the top of an aircraft fuselage through which celestial observations are taken. (ASCC)

Astro Compass

An instrument used primarily to obtain true headings by reference to celestial bodies. (ASCC)

Astro Tracker

A navigation equipment which automatically acquires and continually tracks a celestial body in azimuth and altitude. (ASCC)

Attitude

The position of a body as determined by the inclination of the axes to some frame of reference. If not otherwise specified, this frame of reference is fixed to the Earth. (ASCC)

Bubble Horizon

An artificial horizon parallel to the celestial horizon, established by means of a bubble level. (ASCC)

Calibration Card

A card mounted near an instrument indicating the corrections for instrument and installation errors. (ASCC)

Central Air Data Computer

A device which computes height, vertical speed, air speed and Mach number from inputs of pitot and static pressure and temperature.

Compass Acceleration Error

The error induced in a magnetic compass by vertical magnetic components when acceleration deflects the detecting element from its normal position. (ASCC)

Compass Calibration

The process of swinging and compensating an aircraft compass by detecting and reducing the deviation coefficients and recording the residual deviations. (ASCC)

Compass, Direct Indicating

A compass in which the dial, scale, or index, is carried on the detecting element.

Compass Direction

The horizontal direction expressed as an angular distance measured clockwise from Compass North. (ASCC)

Compass, Gyro-Magnetic

A direction gyroscope whose azimuth scale datum is maintained in alignment with the magnetic meridian by applying precession torques derived from a magnetic detector unit.

Compass North

The uncorrected direction indicated by the North-seeking end of a compass needle. (NATO)

Compass, Remote Indicating

A magnetic compass in which the dial, scale, or index is not carried on the detecting element.

Compass Swing

See Compass Calibration.

Compass, Sky

An instrument for determining the azimuth of the sun by using the polarization of sunlight in the sky. (ASCC)

Declination

The angular distance of a body on the celestial sphere measured North or South through 90° from the celestrial equator along the hour circle of the body. It is comparable to latitude on the terrestial sphere. (NATO)