AP3456
The Central Flying School (CFS) Manual of Flying

Volume 5 – Flight Instruments

CHAPTER 1 - INTRODUCTION TO BAROMETRIC HEIGHT MEASUREMENT

Introduction

1. The atmospheric pressure at a point on the Earth's surface is equivalent to the weight of the whole column of air standing on the area of that point. As distance increases from the Earth, the weight of the air above will be less, therefore atmospheric pressure decreases (Fig 1). Pressure altimeters operate on this principle, and indicate aircraft height relative to a selected pressure datum. Pressure altimeters are, in fact, aneroid barometers graduated to indicate height rather than pressure. In order for such an instrument to be calibrated, certain assumptions must be made concerning the manner in which air pressure decreases with height and this has given rise to a number of model atmospheres.

2. **Units of Measurement**. The units used in pressure measurement are:

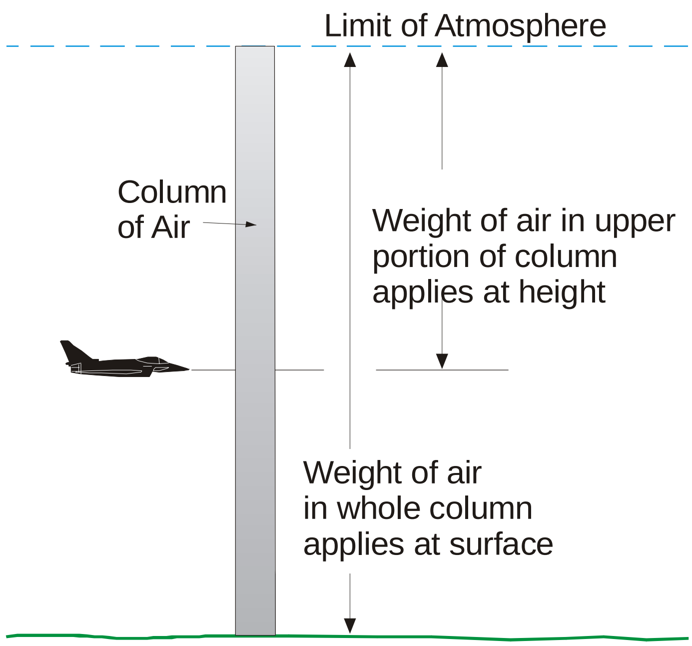
a. **Hectopascal**. The hectopascal (hPa) is the unit of measurement of pressure in common use. At mean sea level (MSL), the atmospheric pressure is of the order of 1,000 hPa; at 50,000 ft it is of the order of 100 hPa.

b. **Inches of Mercury**. Some countries (notably the USA), measure pressure in inches of mercury (Hg). At MSL, atmospheric pressure is about 30 inches Hg.

c. **Millibar**. Although the hPa is now in common usage, the millibar (mb) is still used by many aircrew. The hPa and the mb have equivalent values and so can be considered to be identical for all practical purposes.

Both the hPa and the mb will be found in AP3456, but references to the mb will be gradually replaced by the hPa as chapters are routinely checked and amended.

5-1 Fig 1 Decrease in Atmospheric Pressure with Height



The Atmosphere

3. The atmosphere is described in detail in Volume 1, Chapter 1. It is a relatively thin layer of gases surrounding the Earth, becoming more diffuse with increasing height. Water vapour is present in variable amounts, particularly near the surface.

4. The atmosphere can be divided into a number of layers, each with a tendency to a particular temperature distribution. The names, heights and characteristics of these layers may vary according to which standard atmosphere is being defined. However, in all cases the lower layer, the troposphere, is characterized by a fairly regular decrease of temperature with height. The upper limit of the troposphere is named the 'tropopause'. The height of the tropopause varies with latitude, season and weather. In general, it is lowest at the Earth's poles (around 25,000 ft) and highest over equatorial regions (up to 54,000 ft). The layer above the tropopause is known as the stratosphere. Within this layer, the temperature is assumed to remain more or less constant, although, in reality, there is a noticeable increase near the top of the layer. The upper boundary of the stratosphere is called the 'stratopause', the height of which varies depending on which definition is being employed, but can be taken to be about 30 miles (166,000 ft).

5. **Pressure Lapse Rate**. As height increases, pressure decreases. However, this decrease is not proportional to the increase in height because the density of air varies with height. 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, close to sea level, is that a decrease in pressure of one hPa equates to an increase in height of 30 feet.

6. **Temperature Lapse Rate**. Temperature 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.

7. **Height Assumptions**. The correlation between indicated barometric altitude and actual altitude is poor because of:

a. Variations in conditions of temperature and pressure.

b. The assumptions used in altimeter calibration.

c. Real errors induced by the instrument itself.

A barometrically derived height must therefore be used with extreme caution as a basis for terrain clearance. However, provided that all aircraft use the same datum (and the same assumptions in the calibration of their altimeters), safe vertical separation between aircraft can be achieved.

Standard Atmospheres

8. 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.

9. The first widely accepted standard was proposed by the International Commission on Air Navigation (ICAN) in 1924. Between 1950 and 1952 the International Civil Aviation Organization (ICAO) proposed and adopted another standard which varied only slightly from the ICAN model. Equations were formulated for determining height from barometric pressure which were valid up to 65,617 ft. The ICAO standard atmosphere is taken as the International Standard Atmosphere (ISA) and the assumed characteristics are:

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

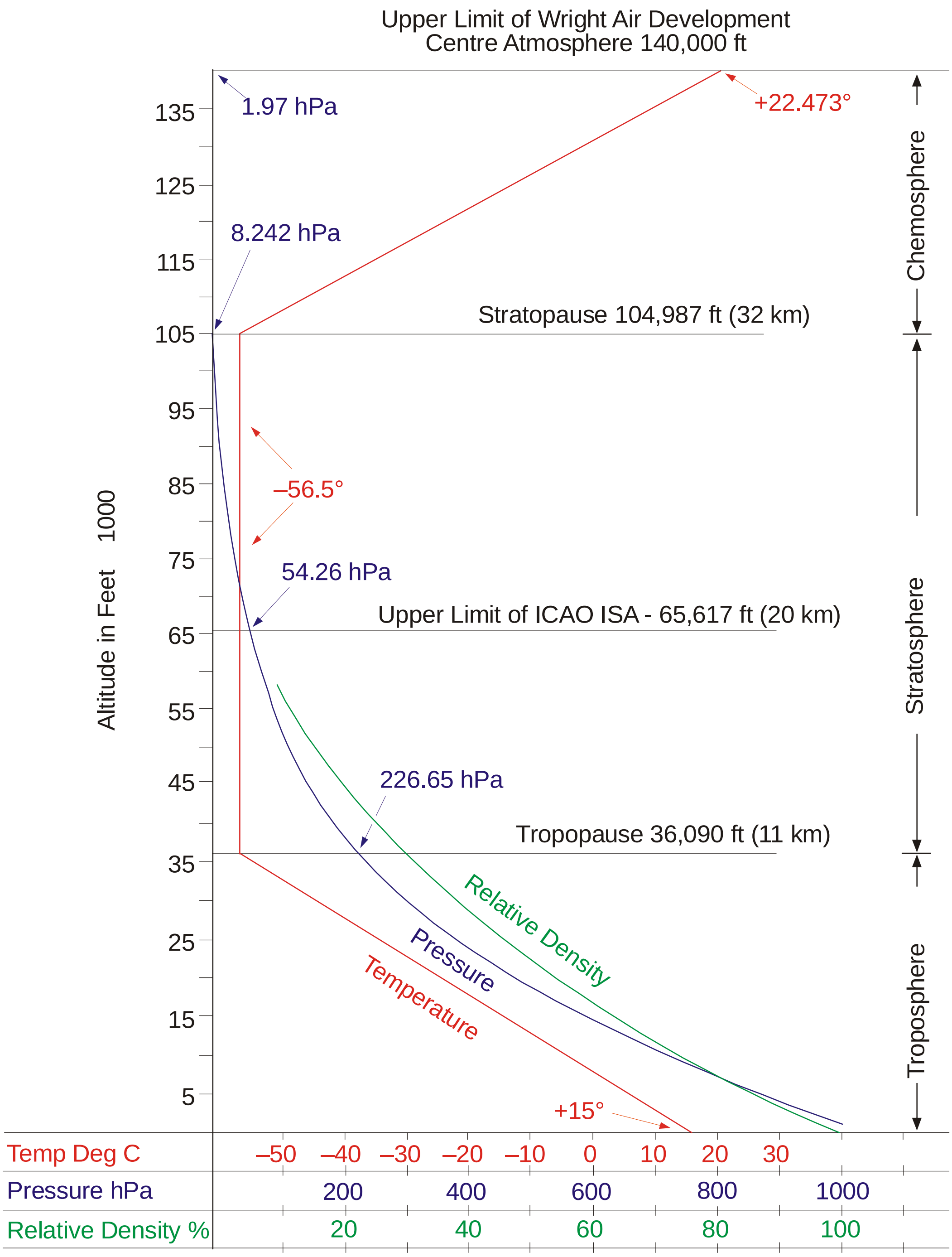
b. The value of g is constant at 980.665 cm/sec^2.

c. The temperature and pressure at mean sea level are 15 ºC and 1013.25 hPa, respectively.

d. The temperature lapse rate is 1.98 ºC per 1,000 ft up to a height of 36,090 ft above which the temperature is assumed to remain constant at –56.5 ºC.

10. A number of other standard atmospheres have been formulated, mainly in response to the need to extend the height limit of the model beyond 65,617 ft to accommodate the requirements of missiles and certain high performance aircraft. The assumptions of these models are very similar to the ICAO standard and the differences in the relation of height to pressure are minimal in the lower altitudes. However, in the stratosphere and beyond, heights, lapse rates and layer names differ markedly. A comparison of Fig 2, which depicts the Wright Air Development Centre (WADC) Standard Atmosphere with Fig 1 in Volume 1, Chapter 1 will reveal some of the differences.

5-1 Fig 2 WADC Standard Atmosphere



CHAPTER 2 - ALTIMETERS

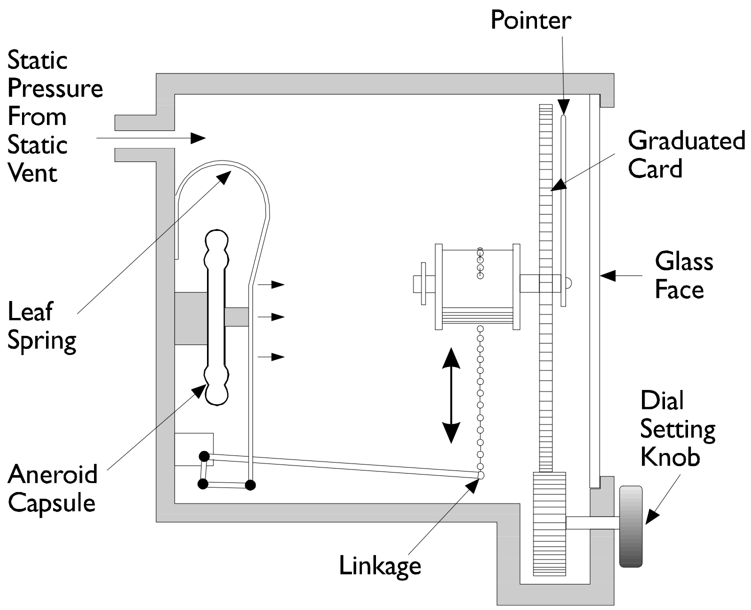
PRINCIPLE OF OPERATION

The Simple Altimeter

1. Air pressure is linked to height and this relationship is exploited by altimeters, which measure pressure but display height.

2. Fig 1 is a schematic diagram showing the main components of a basic, or 'simple' altimeter. Changes in air pressure are detected by an aneroid capsule (constructed from thin, corrugated metal) which is sealed and partially evacuated, and mounted inside a case. The case is fed with static pressure from the aircraft's static tube or vents. As the aircraft climbs, the static pressure in the case will reduce, and the aneroid capsule will expand, assisted by a leaf spring. The linear movement of the capsule face is magnified and transmitted, via a system of gears and linkages, to a pointer moving over a scale (graduated in feet according to one of the standard atmospheres). Conversely, as the aircraft descends, the static pressure increases, and the capsule is compressed. An equilibrium is maintained between the pressure of the atmosphere on the face of the capsule and the tension of the spring.

5-2 Fig 1 Simple Altimeter – Schematic



3. A simple altimeter will normally be calibrated according to the International Standard Atmosphere (ISA), and will, therefore, normally be set to indicate height above the 1013.25 mb pressure level. The dial setting knob allows the indicator needle to be moved away from the normal datum. Thus, for example, before take-off, the altimeter could be set to read airfield elevation, so that it will thereafter indicate height above mean sea-level (providing that the prevailing sea-level pressure does not change). Alternatively, by setting zero feet before take-off, the altimeter will indicate height above the airfield (again, assuming constant surface pressure).

Sensitive Altimeters

4. The 'sensitive' altimeter is designed for more accurate height measurement than the simple altimeter, although the principle of operation is the same. The single capsule is replaced by two or more capsules to

give greater sensitivity for small changes in pressure. Multiple pointers are provided, typically one rotating every 1,000 ft, another every 10,000 ft and possibly a third every 100,000 ft.

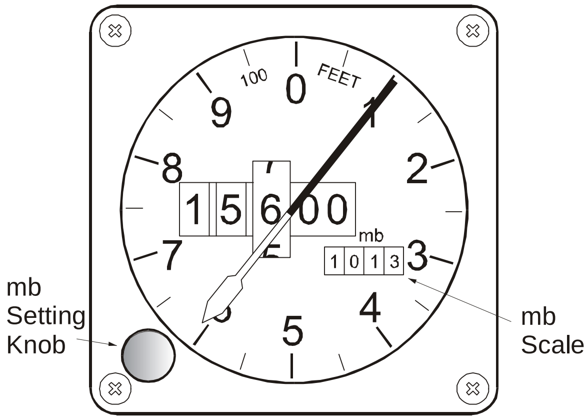
5. A sensitive altimeter has a millibar scale, adjusted by means of the setting knob, allowing the user to set whatever datum pressure is desired. Thus, if airfield level pressure (QFE) is set, the altimeter will read zero on the runway, and height above the airfield once airborne. If sea-level pressure (QNH) is set, the altimeter will indicate height above sea-level. The millibar setting can be altered in the air to reflect changes of pressure with time, location or required datum level.

6. **Limitation**. The chief limitation of the directly operated capsule altimeter is its increasing inaccuracy and lack of sensitivity with increasing height above approximately 60,000 ft. At these altitudes, the change in height for a given pressure change is very much greater than at ground level. For example, a change of pressure of 1 mb at sea-level equates to only 30 ft, whereas at 60,000 ft a similar pressure change relates to a height change of 325 ft. Thus small changes in pressure, which can represent significant changes in height, have to overcome inertia in the mechanical linkages and therefore tend to cause the altimeter to lag significantly behind the aircraft's true change of height.

Servo-assisted Altimeters

7. The servo-assisted altimeter is designed to relieve the capsule of the work required to drive the mechanical linkage. Changes of barometric pressure are still sensed by the contraction or expansion of evacuated capsules, but the mechanical transmission is replaced by a position control servo system. The movement of the capsule is now transferred to the pointers by means of amplified electrical signals. This results in increased accuracy and sensitivity.

5-2 Fig 2 Servo-assisted Altimeter – Display



8. The servo drive can also be used to transmit altitude information to remote displays and to other systems, e.g. IFF/SSR. In current servo altimeters, the multiple needle display is replaced by a digital display, with an auxiliary pointer moving over a scale graduated in 50 ft increments from 0 to 1,000 ft (Fig 2).

Cabin Altimeters

9. Cabin altimeters indicate cabin pressure in terms of altitude and are normally of the simple type, having one pointer moving over a scale graduated in tens of thousands of feet. The static pressure is

a sample from cabin pressure and any change causes the capsules to expand or contract in the normal way. They do not usually have error compensating devices, although they may be compensated to allow for fluctuations in cabin temperature. As with all pressure altimeters, cabin altimeters suffer from errors (see para 10), but at cabin altitudes below 30,000 ft the instrument should be accurate to better than ±500 ft.

ERRORS OF THE PRESSURE ALTIMETER

10. Pressure altimeters are subject to errors which may be considered under two categories; instrument and installation errors, and errors caused by non-standard atmospheric conditions. In addition, they can incur errors through blockages and leaks.

Instrument and Installation Errors

11. **Instrument Error**. Instrument error is caused by small irregularities in the mechanism during manufacture. Certain instrument tolerances have to be accepted. Any residual error will be noted on a correction card, but is usually insignificant.

12. **Pressure Error**. Pressure error only applies when the aircraft is moving or airborne and occurs when the true external static pressure is not accurately supplied to the altimeter. A false static pressure can be created by the effect of the airflow passing over the static vent. Although the error is generally negligible at low speeds and altitudes, it can become significant at high speeds, during flight manoeuvres, or when services such as flaps, airbrakes, or gear are operated. Avoidance or reduction of the effect is accomplished by careful design and location of the static probe or vent. Residual error is calibrated for each aircraft type and detailed in the Aircrew Manual or ODM, or automatically removed in an air data computer (ADC) or pressure error corrector unit (PECU). At high speeds, near Mach 1, if a shock wave passes over the static source, a rapid change in static pressure will occur. This gives an error in the altimeter indication (known as 'Transonic Jump') for the duration of the disturbance.

13. **Lag Error**. Since the response of the capsule and linkage is not instantaneous, the altimeter pointer lags whenever height is changed rapidly causing an under-read on climbs and an over-read on descents. The latter situation could be dangerous and should be allowed for in rapid descents. The amount of lag varies with the rate of change of height. Lag error is virtually eliminated in servo-assisted altimeters and may be reduced in others by the fitting of a vibration mechanism.

14. **Hysteresis Loss**. 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 is difficult to predict, and is most noticeable after sharp climbs or descents.

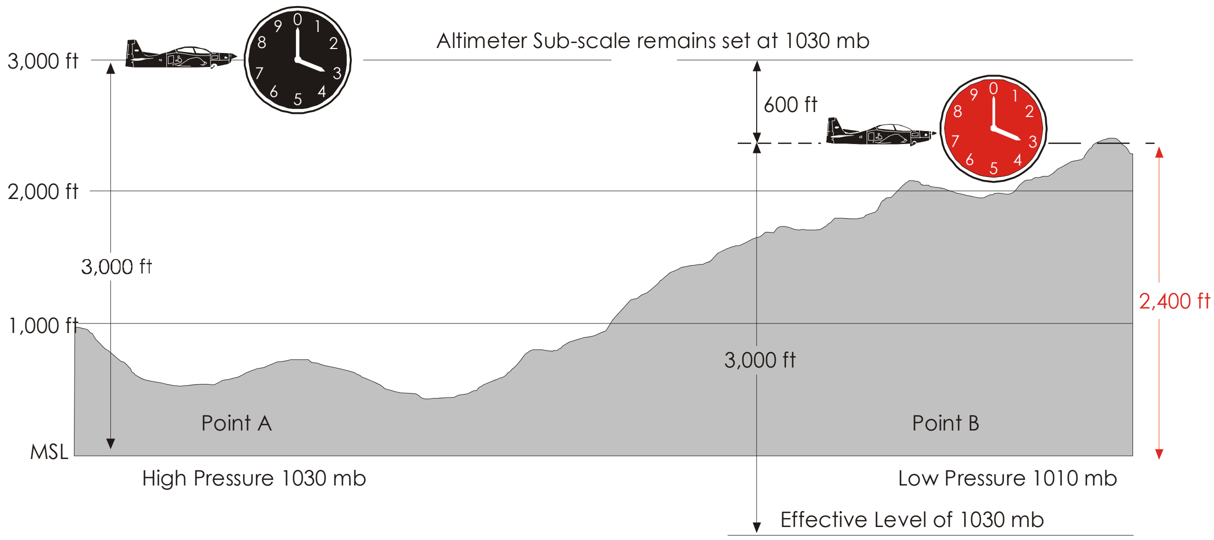
Errors due to Non-standard Atmospheric Conditions

15. Variations from ISA conditions may be brought about by the development of weather systems, and local geographic effects. Errors can also result if an incorrect millibar datum is set by the operator.

16. **Barometric Error**. Barometric error occurs when the actual datum pressure differs from that to which the altimeter has been set. Fig 3 illustrates the effect of this error on an aircraft flying at a constant indicated altitude, from an area of high pressure to one of low pressure. In this example, the aircraft flies from Point A where the MSL pressure is 1030 mb, to Point B where the MSL pressure is 1010 mb. The 1030 mb setting is retained on the altimeter, and the aircraft is flown at 3,000 ft indicated. As the MSL

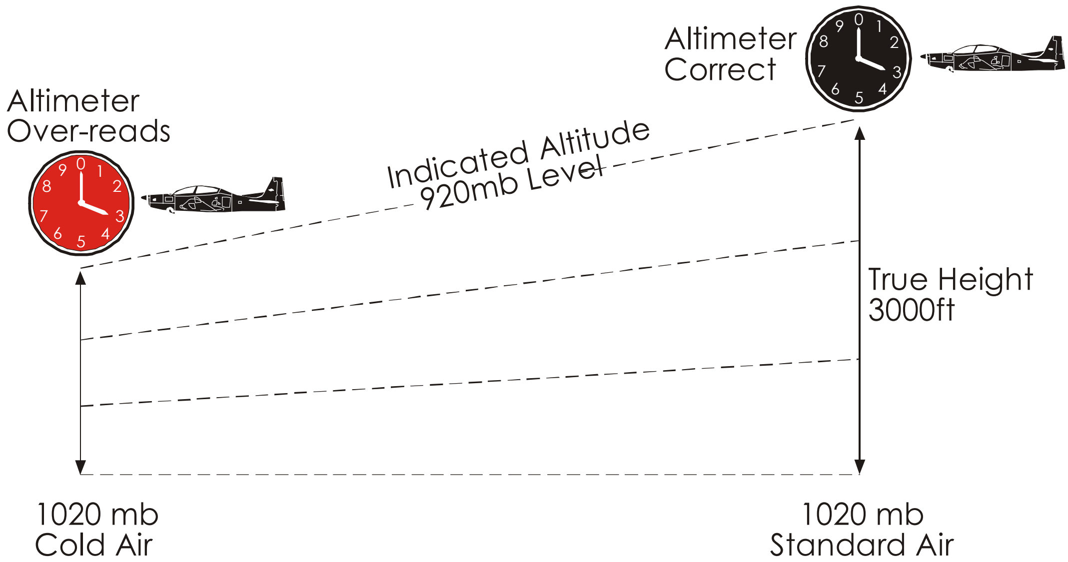
pressure decreases, the effective level of the pressure datum (1030 mb in Fig 3) becomes progressively lower. As a result, after a period of flight, the altimeter reads high. Conversely, if the flight was from an area of low pressure to one of high pressure the altimeter would read low if not corrected. In summary, from HIGH to LOW the altimeter reads HIGH, and from LOW to HIGH the altimeter reads LOW. Barometric error is overcome by changing the millibar scale, as appropriate, for the region of operation.

5-2 Fig 3 Effect of Barometric Error



17. **Temperature Error**. Temperature error arises when the atmospheric conditions differ from those assumed by the standard atmosphere used to calibrate the altimeter. ISA assumes a temperature lapse rate of 1.98 ºC per 1,000 ft up to 36,090 ft, with a constant temperature of –56.5 ºC above that. If the actual temperatures differ from the assumed ones, as they very often do, then the indicated height will be incorrect. In a cold air mass, the air density is greater than in a warm air mass, the pressure levels are more closely spaced and the altimeter will over-read (Fig 4) - the error being zero at sea-level and increasing with altitude. The error is not easy to compensate for, since in order to do so it would be necessary to have a knowledge of the temperature structure from the surface to the aircraft. The magnitude of the error is approximately 4 ft/1,000 ft for each 1 ºC of difference from ISA. Corrections can be made for low altitudes by use of the table in the Flight Information Handbook and this may be necessary, for example, when calculating decision heights in arctic conditions. The table is reproduced in Fig 5 to give an indication of the magnitude of the error. The Terminal Approach Procedure (TAP) used for the example is shown at Fig 6.

5-2 Fig 4 Effect of Temperature Error



18. **Orographic Effect**. When a current of air meets a barrier of hills or mountains there is a tendency, often marked, for much of the air to sweep round the ends of the barrier, so avoiding the ascent. This gives rise to areas of low pressure to the lee of the barrier. The altimeter readings will therefore be affected due to barometric error as described in para 16. Additionally, if standing waves are present above the barrier, the rising or descending air in the wave will change temperature at very nearly the normal adiabatic lapse rate. The temperature profile in the affected area may then be significantly different from the unaffected airmass, thereby inducing temperature errors (as described in para 17) in altimeter readings.

Blockages and Leaks

19. Blockages and leaks are not common occurrences. Blockages may occur if water in the pipework freezes, or there are obstructions such as insects. A slight obstruction may increase altimeter lag. A complete blockage will cause the pressure in the instrument case to remain constant, and the altimeter will then continue to register the height indicated when the blockage occurred. The effect of leaks varies with the size and location of the leak; leaks in pressurized compartments cause under-reading.

5-2 Fig 5 Altimeter Temperature Error Correction

Aerodrome Temperature (Sea level A/Ds) °C	Aerodrome ISA Deviation °C	HEIGHT ABOVE TOUCHDOWN OR HEIGHT ABOVE AERODROME IN FEET														
		200	300	400	500	600	700	800	900	1000	1500	2000	3000	4000	5000	6000
0	–15	20	20	20	40	40	40	40	60	60	100	120	180	240	300	360
–10	–25	20	40	40	60	60	80	80	100	100	160	200	300	400	500	600
–20	–35	20	40	60	80	80	100	120	120	140	220	280	420	560	700	840
–30	–45	40	60	80	100	100	120	140	160	180	280	360	540	720	900	1080
–40	–55	40	60	80	100	140	160	180	200	220	340	440	660	880	1100	1320
–50	–65	60	80	100	140	160	180	200	240	260	400	520	780	1040	1300	1560

20. Pressure altimeters are calibrated to ISA conditions. Any deviation from ISA will result in error proportional to ISA deviation and the height of the aircraft above the aerodrome pressure datum. The error is approximately 4ft/1000ft per °C of difference. When temperature is **LESS** than ISA an aircraft will be **LOWER** than the altimeter reading. Table values should be **ADDED** to published/calculated altitudes or heights.

21. The error corrections in the table are properly a function of deviation from ISA, but for simplicity the aerodrome temperature may safely be used for aerodromes up to 1000 ft above sea level. (This will include virtually all UK aerodromes). At higher aerodromes the ISA deviation should be used. The temperature at ISA is +15 °C minus 2 °C per 1000 ft above sea level. The ISA deviation is the ambient temperature minus the temperature at ISA. (e.g. an airfield 2,500 ft above sea level at –30 ºC has ISA deviation of –30 – (+10) = – 40).

WHEN TO APPLY CORRECTIONS

22. When the aerodrome temperature is **0 °C** or colder, temperature error correction **must** be added to:

a. DH/DA or MDH/MDA and step down fixes inside the FAF.

b. ALL low altitude approach procedure altitudes in mountainous regions (defined as terrain of 3000 ft amsl or higher).

23. When pilots intend to apply corrections to the FAF crossing altitude, procedure turn or missed approach altitude, they **must** advise ATC of their intention and the correction to be applied.

24. Pilots may refuse IFR assigned altitudes if altimeter temperature error will reduce obstacle clearance below acceptable minima. However, once an assigned altitude has been accepted, it must not subsequently be adjusted to compensate for temperature error.

MINIMUM SECTOR ALTITUDE (MSA)

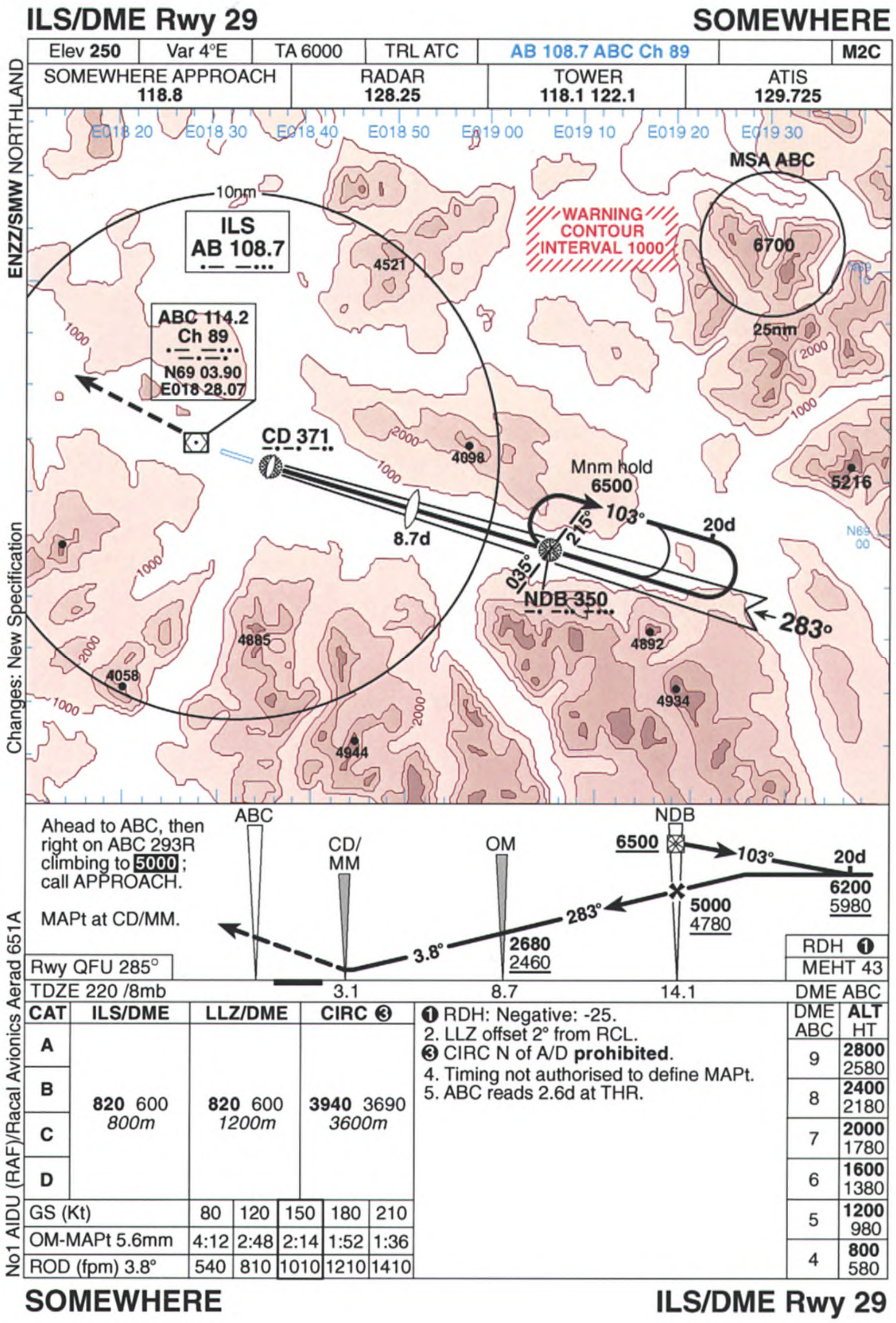
25. When the aerodrome temperature is **–30 °C** or colder, add 1000ft to the MSA to ensure obstacle clearance.

EXAMPLE

26. SOMEWHERE; ILS/DME Rwy 29; TDZE/THR Elev 220; Surface Temp –30 °C.

	Published Altitude	HAT	Add	Altitude to Fly
MSA	6700ft	N/A	1000ft	7700ft
NDB Mnm Hold	6500ft	6280ft	1140ft	7640ft
Turn Inbound	6200ft	5980ft	1080ft	7280ft
NDB Inbound	5000ft	4780ft	860ft	5860ft
OM Inbound	2680ft	2460ft	450ft	3130ft
DA/DH	820ft	600ft	100ft	920ft

5-2 Fig 6 TAP for Temperature Error Correction Example in Fig 5



CHAPTER 3 - RADAR ALTIMETERS

Introduction

1. Barometric altimeters provide a standard datum for the safe vertical navigation and separation of aircraft; their limitations and errors are considered in Volume 5, Chapter 1. However, the nature of certain air operations (such as low level flying over the desert or sea) determines the need for an indication of actual surface clearance and proximity warning, and additionally, many on-board computing system require an accurate input of the instantaneous vertical distance of an aircraft from the Earth's surface. These include systems for:

a. Terrain following.

b. Weapon aiming.

c. Navigation.

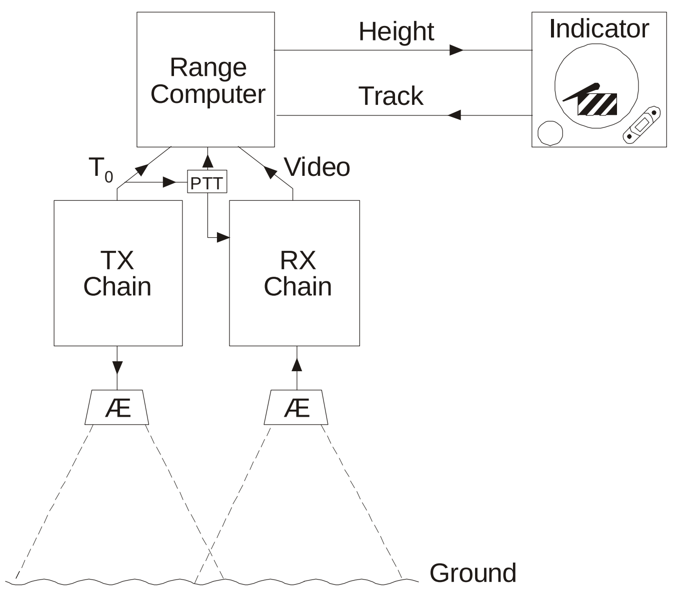
d. Helicopter automatic transitions to and from the hover.

2. Pulsed radar altimeters are range finding radar devices mounted to point downwards and measure the distance to the ground directly below the aircraft. Early systems had limited value because minimum range, determined by pulse width, was too great for very low level use. However, developed, modern systems can now be used down to zero feet with an accuracy of ±1.5 ft.

Pulse Radar Altimeter

3. Radar altimeters use conventional pulse radar techniques. The time taken for a short pulse to travel to the ground and back is measured and displayed on an indicator. The display time base is synchronized with the transmitter pulse. From the block diagram, at Fig 1 it can be seen that the main difference from a 'conventional' range finding system is that two aerials are used, one for transmission and one for reception. Due to the very small ranges that have to be measured, very narrow short duration pulses are transmitted at a typical frequency of 4.3 GHz.

5-3 Fig 1 Block Diagram of a Typical Radar Altimeter Installation



4. **Indicator**. A typical height indicator is a remote position control (RPC) servo, fed with the height voltage from the range computer. A manual index bug sets contacts which are operated by a cam on the pointer shaft. When the pointer indicates a height less than that set by the index, a low-level warning lamp lights, and the contacts can also operate an audio warning tone. A yellow striped electromagnetic flag indicates power off or failure, or signal unlocked condition but is removed or shows black when the instrument is functioning correctly. Some instruments have a NO TRACK flag to show when they are unlocked or heights are unreliable. Two types of indicator are available, 0 to 5,000 feet or 0 to 1,000 feet depending on role.

5. **Aerials**. The aerials are usually identical, suppressed, and mounted flush with the aircraft skin. They are so positioned that the receiving aerial cannot acquire signals directly from the transmitting aerial, which would result in a permanent zero indication.

6. **Power Supply**. Depending upon system type, power is switched either from a remote RAD ALT ON/OFF switch or by an OFF/SET/PUSH-TO-TEST switch on the face of the instrument. In all cases, a short period of warming up is required before the system will record heights.

Principle of Operation

7. Radar Altimeters have three principal components:

a. **Transmitter Chain**. The transmitter (TX) chain produces pulses of RF energy. A low power time-zero (T_0) pulse is also produced to trigger the range computer at the exact instant the transmitter fires.

b. **Receiver Chain**. The receiver chain (RX) amplifies the returned echo and passes the video pulse to the range computer.

c. **Range Computer**. The range computer measures the time interval between the T_0 pulse and the video pulse and produces a DC voltage proportional to the time interval. The track line is energized when the circuits 'lock-on' to and track a received signal, removing the POWER-OFF/NO TRACK flag from view.

8. The range computer has three states of operation:

a. **Search**. This is the state of operation if signals are not present or are too weak. The computer searches over the range of voltages from a minimum, equivalent to 0 feet, to a maximum, equivalent to 1,000 or 5,000 feet (depending upon type), 4 times per second. The indicator shows the NO TRACK or yellow striped POWER OFF flag.

b. **Lock/Follow**.

(1) When a returning signal is detected and tracked for a pre-determined period the system 'locks-on', producing an output voltage proportional to the height.

(2) This internal voltage is fed to the indicator which causes the pointer to move to the correct height. The track line is energized and pulls the POWER-OFF/NO TRACK flag out of view.

(3) The tracking circuits can maintain lock with vertical range changes of up to 2,000 ft/sec.

c. **Memory**.

(1) If the signal is lost, the circuitry is prevented from reverting to search for approximately 0.2 seconds. If signals are detected during this time, the indicator is unaffected.

(2) If signals are not detected, the range computer goes into its search condition, but the indicator display is held steady for a further 1.0 seconds.

(3) If lock is not obtained within this period, the track line will be de-energized and the indicator will show the NO TRACK or POWER OFF flag.

Self Test

9. The aircrew, or the maintenance crew, can check the operation of the equipment, with the exception of the aerials and their cables, by means of the PUSH-TO-TEST (PTT) button. Detailed instructions for testing the equipment are contained in the relevant Aircrew Manuals.

CHAPTER 4 - VERTICAL SPEED INDICATORS

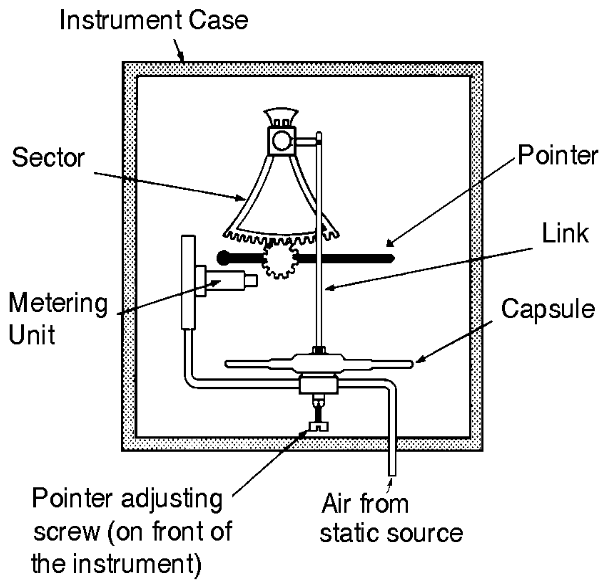
Introduction

1. A vertical speed indicator (VSI), also known as a rate of climb and descent indicator (RCDI), is a sensitive differential pressure gauge, which displays a rate of change of atmospheric pressure in terms of a rate of climb or descent.

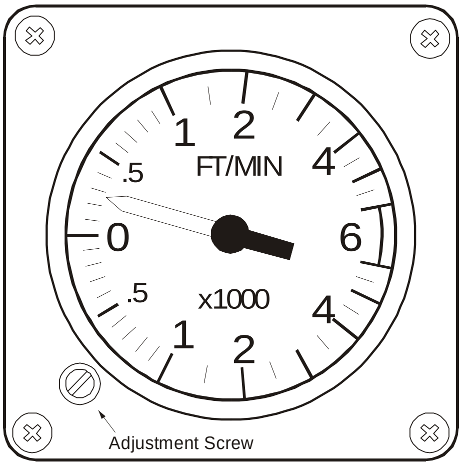
Principle

2. The principle employed is that of measuring the difference of pressure between two chambers, one within the other. Static atmospheric pressure is fed directly to the inner chamber (an aneroid capsule) and through a metering unit to the outer chamber, which forms the instrument case. The metering unit restricts the flow of air into and out of the case, whereas the flow to the inside of the capsule is unrestricted. Therefore, if the static pressure varies due to changing altitude, the pressure change in the case lags behind that in the capsule. The resultant differential pressure distorts the capsule and this movement is magnified and transmitted to the pointer by means of a mechanical linkage. The construction of a VSI is shown schematically in Fig 1 and a typical display is illustrated in Fig 2.

5-4 Fig 1 VSI - Schematic Construction



5-4 Fig 2 VSI - Typical Display



3. It is important that any given pressure difference between the inside and outside of the capsule should represent the same rate of climb or descent, regardless of the ambient atmospheric pressure and temperature variations with altitude. The function of the metering unit, in the manner in which it restricts the flow into the case, is to compensate for these changes in ambient conditions.

4. In level flight, the pressure inside the capsule and the case are the same, and the pointer remains at the zero (horizontal) position. When the aircraft climbs, the static pressure decreases and the capsule collapses slightly, causing the pointer to move upwards 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, causing the pointer to move downwards.

Errors

5. The VSI can suffer from the following errors:

a. **Instrument Error**. Instrument error is the result of manufacturing tolerances and is usually insignificant. Before flight, pilots should ensure that the pointer reads zero, or is within permissible limits. With some VSIs, the zero setting can be adjusted by means of a screw on the face of the instrument.

b. **Pressure Error**. If the static head or vent is subject to disturbed airflow, the static pressure may be in error, and the VSI will briefly indicate a wrong rate of climb or descent. These disturbances may be due to:

(1) Accelerations (such as take-off or missed approach) or decelerations.

(2) Change of aircraft configuration.

(3) Movement of a shock wave over the static vents (resulting erroneous indications are referred to as Transonic Jump).

c. **Lag Error**. When an aircraft begins a climb or descent, the instrument will indicate the change in pitch, but there is few seconds delay before the pointer settles at the appropriate rate of climb or descent. This delay is known as 'lag' and is caused by the time required for the pressure difference to develop. A similar delay occurs before the pointer indicates zero when the aircraft is levelled.

6. **Static Line Blockage**. If the static line or vent becomes blocked by ice (or any other obstruction), the VSI will be rendered unserviceable and the pointer will remain at zero regardless of the vertical speed.

Instantaneous Vertical Speed Indicator

7. The Instantaneous Vertical Speed Indicator (IVSI), also sometimes referred to as the Inertia Lead VSI, was developed to overcome the initial lag error (described in sub-para 5c) when a climb or descent is started. The IVSI is similar in operation to the VSI, except that two accelerometer units, working in opposite directions, are added to the linkage between the capsule and the pointer. The accelerometer units rely upon inertial mass, which is moved with a change in vertical speed in either direction. At the initiation of a climb or descent, inertia now causes the appropriate accelerometer to produce an immediate response, which is transmitted to the pointer on the instrument face, well before any static pressure differential has been established. After a few seconds, the effect of the accelerometer response dies away, but, by this time, the static pressure change will have become effective in the normal way. Thus, within the IVSI, initial lag error is virtually eliminated. However, because the accelerometers are not vertically stabilized, some error is produced during turns, and at large angles of bank (in excess of 40º), the IVSI is unreliable.

CHAPTER 5 - AIR SPEED INDICATORS

Introduction

1. A knowledge of the speed at which an aircraft is travelling through the air, ie the air speed, 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:

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

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

p = the static pressure

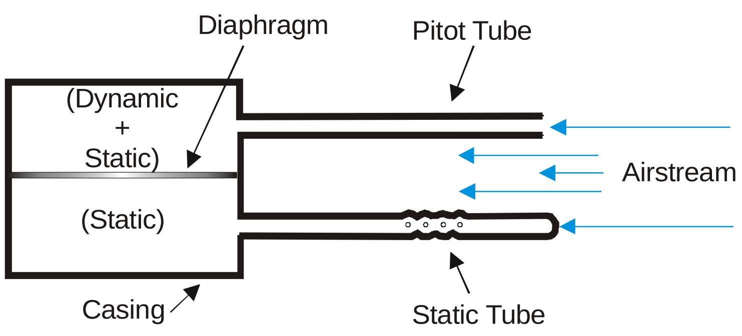
ρ = the air density

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.

5-5 Fig 1 Principle of Air Speed Indicator

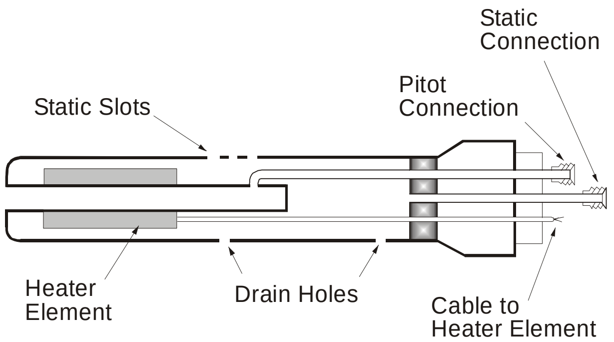


4. When moved through the air, the pitot tube will pick up pitot pressure made up of static pressure and dynamic pressure. The pitot pressure is led through a pipeline to one side of a sealed chamber, divided by a thin flexible diaphragm. The static tube is unaffected by dynamic pressure as its end is 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.

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 some installations, the pitot tube and the static tube are 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 may be used to prevent dirt and water from affecting the instrument. A combined pressure head is shown in Fig 2.

5-5 Fig 2 A Combined Pressure Head



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 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 characteristics 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.

5-5 Fig 3 A Typical Simple ASI

9. **Sensitive and Servo Air Speed Indicators**. Sensitive and servo ASIs are identical in principle to the simple ASI and operate from the normal pitot/static system. Extra sensitivity 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 capsules. This capsule assembly has a linear pressure/deflection characteristic which is more closely controlled than the single capsule used in the simple ASI. In a servo, ASI the mechanical linkage is replaced by an electrical linkage utilizing error actuation and power amplification. A typical sensitive ASI display is shown in Fig 4.

5-5 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.

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. Depending upon aircraft type, the error may be influenced by:

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.

d. The configuration of the aircraft (i.e. 'clean'/flaps/gear/airbrakes/etc).

e. The presence of sideslip.

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 is 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 pressure error correction (PEC) is tabulated in the Aircrew Manual for the aircraft type and is also combined with that for instrument error correction (IEC) and recorded on a correction card mounted adjacent to each ASI. The card correction (IEC + PEC) should be applied to the indicated air speed (IAS) to obtain calibrated air speed (CAS).

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 Mk4A or 5A. Application of the compressibility error correction (CEC) to CAS 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:

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

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/5A.

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

CAS = IAS $\pm$ PEC $\pm$ IEC

EAS = CAS – CEC

TAS = EAS $\pm$ DEC

Blocked or Leaking Pressure Systems

17. **Blockages**.

a. **Pitot**. If the pitot tube is blocked, e.g. by ice, the ASI will not react to changes of airspeed in level flight. However, the capsule may act as a barometer producing an indication of increase in speed if the aircraft climbs or a decrease in speed if the aircraft dives. If the pitot tube contains a small bleed hole for drainage, partial blockage of the 'nose' of the tube (the most common effect of icing) will result in an under-reading. More extensive icing will cause the reading to reduce towards zero, as the dynamic pressure leaks away through the bleed hole.

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 pressure outside the pipe is lower than static (ie most unpressurized aircraft), will cause the ASI to over-read. Where the outside air is higher than static (i.e. in a pressurized cabin) the ASI will under-read.

19. **Effects**. The under- or over-reading of an ASI is potentially dangerous. The former may cause problems in adverse landing conditions (e.g. in a strong cross-wind), and the latter condition may result in an aircraft stall at a higher indicated airspeed than that specified for the aircraft.

CHAPTER 6 - MACHMETERS

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 buffeting. The onset of these shock waves and their subsequent effects occur, for a given aircraft type, when the true airspeed is a certain proportion of the local speed of sound. For convenience, the ratio of true airspeed to the local speed of sound is considered as a single entity. It is called Mach number and is usually expressed thus:

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

where: V = True airspeed

a = Local speed of sound

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 may have an adjustable index which is usually set to the Limiting Indicated Mach Number of the aircraft in which it is installed.

Basic Principle

3. As explained in para 1, the local Mach number varies with the true airspeed and the local speed of sound. True airspeed is a function of dynamic 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:

$$M = \frac{V}{a} \propto \left[\frac{p_t - p}{p}\right]$$

where: V = True airspeed

a = Local speed of sound

p_t = Pitot pressure

p = Static pressure

The machmeter uses an airspeed capsule to measure ($p_t - p$), an altitude capsule to measure p, and is calibrated to show the quotient as the corresponding Mach number.

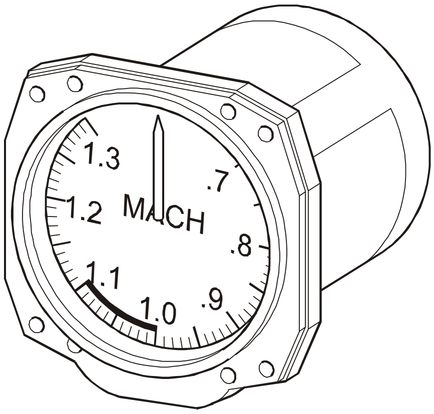
4. The actual calibration of the instrument is more complex than the basic principle suggests, since the behaviour of air changes as speed is increased, especially once shockwaves form. As Mach number increases therefore, the actual formula used to derive an indicated Mach reading requires and receives considerable modification.

Construction

5. 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 airspeed capsule, is connected to the

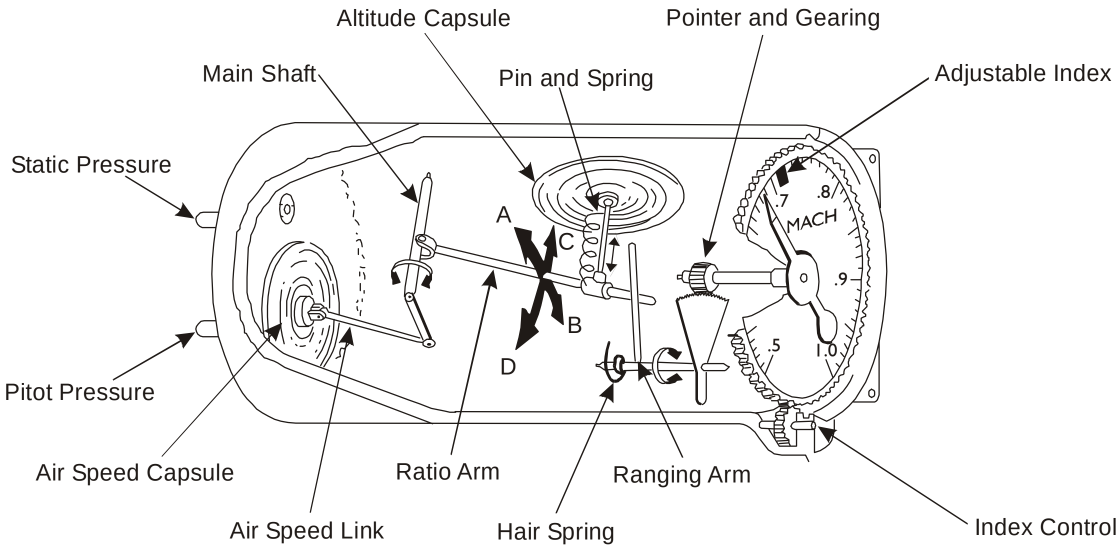
pitot pressure pipeline. The second capsule unit, the altitude capsule, is sealed and evacuated to respond to static pressure changes.

5-6 Fig 1 A Typical Machmeter



6. The airspeed capsule measures the pressure difference between pitot and static pressure and therefore expands or contracts as airspeed increases or decreases. The movement of the capsule is transferred by the airspeed 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).

5-6 Fig 2 Principle of Operation of a Machmeter



7. 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).

8. 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 the corresponding Mach number. An increase of altitude and/or airspeed results in a display of higher Mach number.

9. Critical or Limiting Mach Number is indicated by a specially shaped lubber mark located over the dial of the machmeter. It is adjustable so that the relevant Mach number for the particular type of

aircraft in which the machmeter is installed may be preset. Presetting can be carried out by an adjusting screw on the front of the instrument.

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 ± 0.01M over a range of 0.5 to 1.0M.

12. **Pressure Error**. The machmeter operates from the same pressure source as the airspeed indicator and is therefore subject to the same pressure errors. 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.

CHAPTER 7 - COMBINED SPEED INDICATORS

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 two forms; a simple capsule operated dial presentation or a capsule operated IAS dial with a synchro operated digital Mach number presentation.

Principle

2. The construction of the dial-type combined speed indicator is very similar to the machmeter and the same principles are employed.

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 rotatable 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.

d. Undercarriage warning.

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 kt, at 10,000 ft to 509 kt, at 20,000 ft to 425 kt, at 30,000 ft to 347 kt, etc. However, at low level it may be restricted to 490 kt. 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 speed pointer is set on the ground to the particular relevant limit speed, in this case 490 kt. 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 kt 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 reached 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.

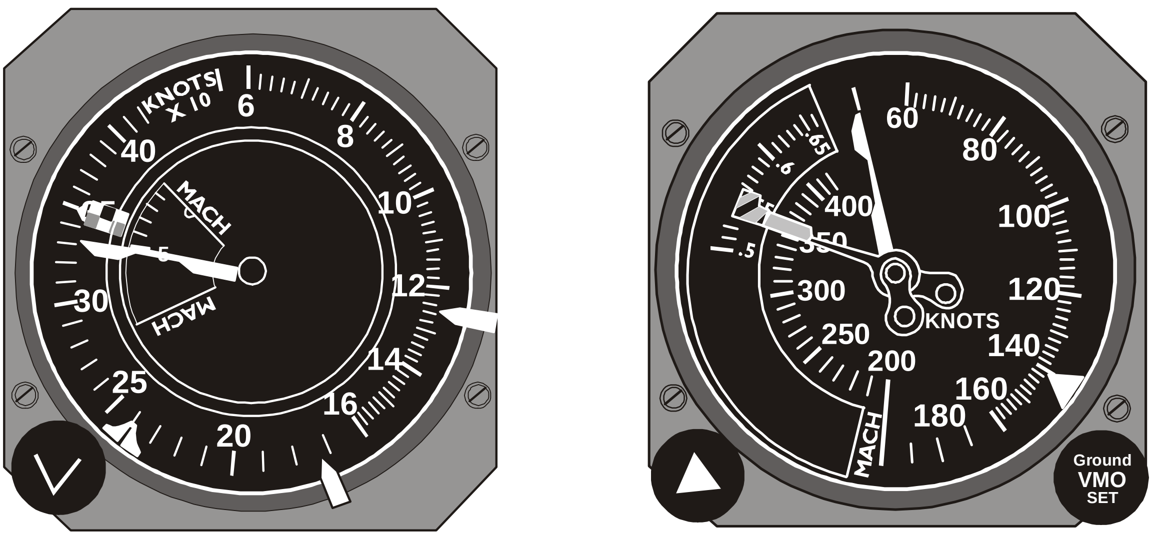
8. **Undercarriage Warning**. An internal switch is fitted in some CSIs which will close at a pre-set figure in the aircraft approach speed range to provide a signal for a visual or audio warning if the undercarriage is not selected down.

Presentation

9. A single pointer is read against a fixed IAS dial calibrated in knots and a rotatable 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 (VMO) 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.

5-7 Fig 1 Typical CSI Dial Presentations

Fig 1a Mach Aperture Inside IAS Scale **Fig 1b Mach Aperture Outside IAS Scale**



Digital Mach/Air Speed Indicators

10. 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 6 and 7.

11. **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 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 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

12. The operating range of the CSI varies with the particular model but, typically, air speeds up to 800 kt and Mach number up to 2.5 can be covered. Typical instrument accuracies are ± 3 kt and ± 0.010M.

5-7 Fig 2 Typical Digital Mach/Air Speed Indicator

CHAPTER 8 – OUTSIDE AIR TEMPERATURE GAUGES

Introduction

1. Temperature measurement and the basic principles of temperature sensing elements can be researched in other publications.

2. Thermometers installed in aircraft provide information on:

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

b. The temperature of various compartments within the aircraft (eg bomb bays).

c. The operating temperatures of various engine components, lubricants, exhaust gases, etc. These sensors are covered fully in Volume 5, Chapter 26.

3. This chapter describes two categories of OAT thermometer - those that are direct reading, and those utilizing a 'total air temperature' probe.

DIRECT READING THERMOMETERS

4. Direct reading thermometers, used mainly in slow-speed aircraft, can be considered as single units, even though some of them include a remote indication facility. They are of fairly simple construction, and do not require a power supply. They employ sensing elements based on one of the following physical characteristics:

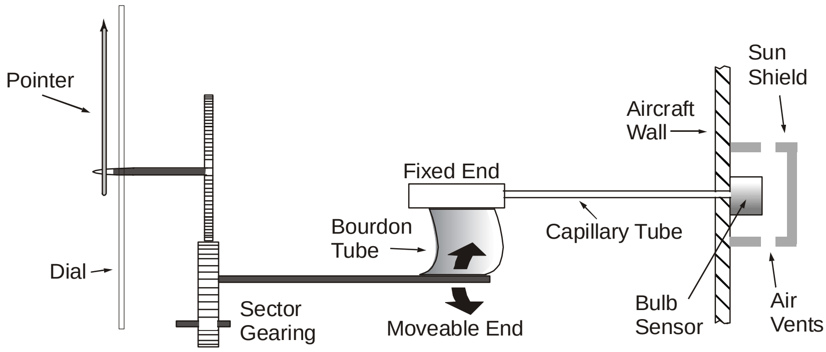
a. The expansion of mercury when heated.

b. The differential expansions of dissimilar metals, exploited in the use of bi-metallic sensing elements.

Mercury Type

5. The mercury thermometer consists of a steel bulb connected by capillary tubing to a Bourdon pressure gauge. This pressure gauge is linked to the pointer on the indicator gauge by a suitable linkage (Fig 1). (A Bourdon gauge utilizes a flattened tube, bent to a curve or spiral, which tends to straighten under internal pressure.) The mercury-filled bulb is situated in the airstream. An increase in temperature will cause the mercury to expand and flow via the capillary, to the Bourdon tube. The resulting movement of the Bourdon tube is transmitted to the pointer. A decrease in temperature will have the reverse effect.

5-8 Fig 1 Mercury Thermometer – Schematic

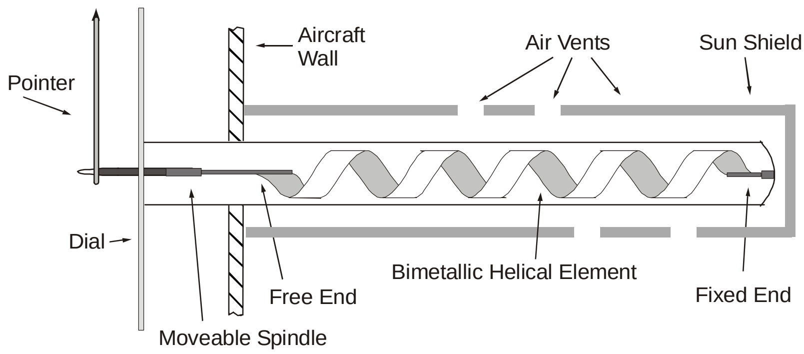


6. To minimize errors due to variations in the temperature of components other than the bulb, the volumes of the capillary tube and the Bourdon tube are kept small relative to the volume of the bulb. To compensate for changes of temperature at the indicator, a bi-metallic strip, which reacts in opposition to the motion of the Bourdon tube, is incorporated between the free end of the Bourdon tube and the pointer spindle.

Bi-metallic Element Type

7. Bi-metallic devices take advantage of the different rates of thermal expansion in different metals. The sensing element consists of two strips of dissimilar metals, welded together and formed into a helix (see Fig 2). The element is housed in a metal tube and coils and uncoils as a result of variations in air temperature. One end of the helix is anchored to the tube, while the other end is free to rotate. The free end is attached, via a spindle, to a pointer that moves over the graduated temperature scale.

5-8 Fig 2 Bi-metallic Thermometer - Schematic



OAT Indicator

8. The OAT Indicator may be integral to the thermometer or be in a remote location. Where a remote indicator is employed, it may be fed from the sensor by:

a. Electrical transmission signals.

b. A long capillary tube, carefully designed to ensure that variations in temperature of the capillary tubing do not affect the reading of the indicator.

5-8 Fig 3 OAT Indicator

Errors of Direct Reading Thermometers

9. Thermometer errors are described in detail in a later section within this chapter. However, direct reading thermometers are particularly prone to errors caused by kinetic heating and skin friction (see paras 30 and 31). These phenomena result in the temperature sensed, and shown at the indicator, being warmer than that of the ambient free air.

10. The effect of heating errors is to render direct reading thermometers of limited practical use at true airspeeds above 150 kt. Corrections to be applied to the indicated OAT are normally given in the aircraft Operating Data Manual.

TOTAL AIR TEMPERATURE PROBES

Total Air Temperature

11. **Total Air Temperature**. Where an element is mounted in a probe projecting into the airflow, some of the air coming into contact with the probe will be brought wholly or partially to rest. When brought to rest, air will release kinetic energy in the form of heat, and, if in contact with the sensor, the temperature registered will be higher than the temperature of the ambient air. Therefore, the temperature sensed by a thermometer probe in the airstream will be the free airstream temperature plus any temperature rise due to the kinetic release. Where the air is brought totally to rest, the temperature (i.e. ambient plus kinetic) is known as the 'total air temperature'.

12. **Total Air Temperature Probes**. A total air temperature probe is designed to bring part of the airstream as close to total stagnation as possible and measure its total air temperature. This value can then be used as an input to an air data computer (see Volume 5, Chapter 9).

13. **Recovery Factor**. The proportion of the kinetic energy of the airstream that the sensing element recovers in reducing the velocity of the airstream is known as the 'recovery factor' (k). 'k' is typically around 0.80, i.e. the sensor will measure the ambient air temperature plus 80% of the kinetic rise.

General Principles

14. The temperature-sensing element will be located within a probe that is situated in the free airstream, away from the boundary layer, and free from airframe skin-friction. The element is either a temperature-sensing resistance bulb, or a thermocouple.

a. **Temperature Sensitive Resistance Bulbs**. Temperature-sensitive resistance bulbs work on the principle that, in certain metals, the electrical resistance will vary with a change in temperature. Resistance bulbs consist of a resistance coil, contained in a sealed steel tube, and connected to the electrical circuit of the indicator unit. The resistance coil may be made of nickel or platinum wire.

b. **Thermocouples**. A thermocouple consists of two strips of dissimilar metals, joined at one end. Changes in temperature at their junction induce an electromotive force (emf) between the other ends. This emf, which increases as temperature rises, provides an electrical signal proportional to temperature.

15. The total air temperature sensed by the probe can be used as an input to an air data computer (ADC), where it will be processed, and an accurate OAT extracted. The accurate OAT can then be combined with Mach input to determine true airspeed (TAS).

16. The Total Head Thermometer and the Rosemount OAT Probe are examples of electrical thermometers used at higher Mach numbers to detect total air temperature.

Total Head Thermometer

17. The total head thermometer, illustrated in Fig 4, utilizes a platinum resistance coil as the temperature-sensitive element. The element is enclosed in a protective housing, positioned in the free airstream.

5-8 Fig 4 Total Head Thermometer

18. The housing is designed such that air entering the probe head proceeds into a venturi. Inside, most of the air that comes into contact with the venturi walls is expelled through ports in the housing. However, a small central flow of air is passed over the tube containing the element at a very reduced speed, and then discharged through holes in the outlet ring.

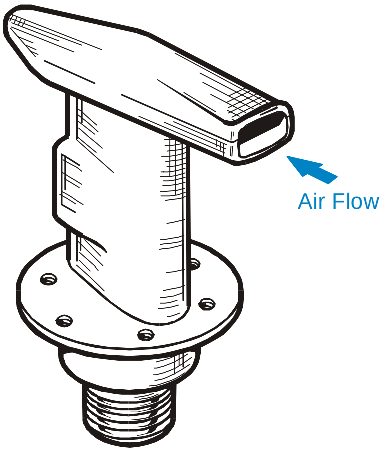
19. The total head thermometer has an accuracy of ± 1º C.

Rosemount Outside Air Temperature Probe

20. In the Rosemount probe, 'k' approaches the theoretical maximum value of 1. It therefore gives an accurate total air temperature during flight, even in icing conditions.

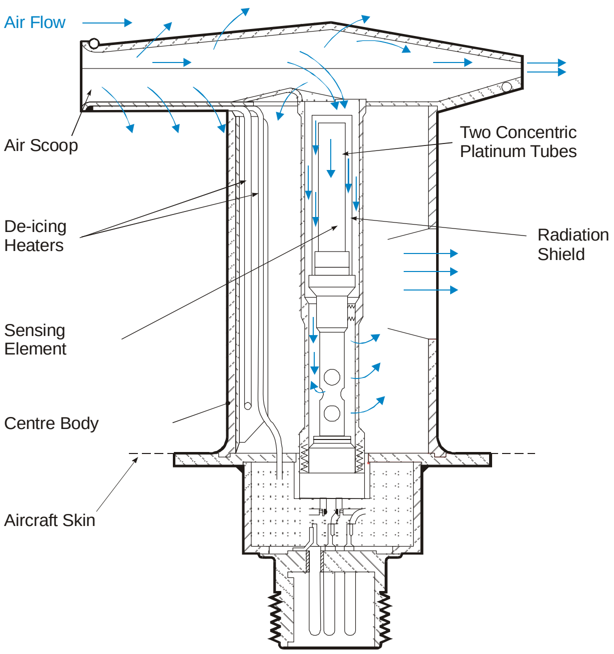
21. The Rosemount probe (Figs 5 and 6) consists of a centre body, mounted on the aircraft skin, containing a hermetically sealed platinum resistance element, and incorporating an air scoop and de-icing element.

5-8 Fig 5 Rosemount Probe - Outside View



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 6. In addition, the flow within the probe separates, part of the flow turning through a right angle before passing around the sensor. This change of direction produces particle separation, which prevents droplets of water from 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 and sensitive resistance element with fast response.

5-8 Fig 6 Rosemount Probe - Sectional View



23. **De-icing**. The de-icing heater, embedded in the material of the probe, is a tube containing an axial wire element, which 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 de-icing element heats the exterior of the probe. A cylindrical radiation shield protects the sensor from the heating effects of the element.

24. **Limitations**. The Rosemount probe will operate only with a satisfactory airflow over the probe. This condition is met under all conditions of flight. However, 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. In cases where the heater is on for extended periods in zero airflow, the pointer will move past full-scale. On the ground, with no airflow over the probe, the probe body temperature could reach a maximum of approximately 300 ºC. However, the indicator movement will not be damaged by these conditions.

Note: On some systems, the probe heater may be automatically modulated to eliminate excessive heating when on the ground, or at very low airspeeds.

THERMOMETER ERRORS

General Errors

25. 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.

OAT Thermometer Errors

26. Aircraft thermometers used for measuring OAT are subject to three types of error:

a. Instrument error.

b. Environmental error.

c. Heating error.

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

28. **Environmental Errors**. Environmental errors such as solar heating or ice accretion can cause errors.

a. **Solar Heating**. Solar heating effects can be reduced by mounting a flat plate sensitive element beneath the wing or fuselage, or for a probe-mounted element, fitting it in a sun shield through which air is allowed to pass freely (see Fig 2).

b. **Ice Accretion**. Protection of the sensing element from ice accretion effects can be achieved by incorporating a heater. The sensing element is protected from the heater by a shield, as in the Rosemount probe (see Fig 6).

Residual errors due to environmental effects can not be calculated, therefore corrections can not be made for them.

29. **Heating Errors**. Heating errors in OAT thermometers are dependent upon the type of temperature-sensing element mounting. The types of mounting are:

a. Probes, protruding from the skin of the aircraft into the airflow.

b. Flat plates, let into the skin of the aircraft.

30. **Probe Sensors**. As described in para 11, where an element is mounted in a probe projecting into the airflow, some of the air coming into contact with that probe will be brought wholly or partially to rest and will release kinetic energy in the form of heat. As a result, the temperature registered by the sensor will be in excess of the temperature of the ambient air. Assuming that pressure changes are adiabatic, the rise in temperature at the probe may be calculated from Bernoulli's equation for compressible flow. Although the equation would assume a full value of adiabatic temperature rise, this is not realized in practice, since no energy exchange is perfect. A useful approximation of the formula for dealing with kinetic heating is:

$$T_1 = T_2 - \left(\frac{V_T}{100}\right)^2$$

where:

T_1	=	Correct outside air temperature (ºC)
T_2	=	Indicated outside air temperature (ºC)
V_T	=	TAS (kt)

31. **Flat Plate Sensors**. The flat plate sensor is unaffected by adiabatic heating, as it does not protrude into the airflow. However, the passage of air across 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 flat plate and stagnation point sensors.

CHAPTER 9 - AIR DATA COMPUTER

Introduction

1. Although conventional pressure instruments can provide satisfactory information for the crew, they have a number of limitations, especially in the context of modern aircraft systems. In particular, the information that an instrument measures can only be presented in one form and cannot easily be transmitted for use by other equipment, or to other crew positions, resulting in a need to duplicate the instrument. An Air Data System (ADS) overcomes these limitations.

2. An ADS can take a number of forms which will vary between aircraft types, however all systems are similar in principle and this chapter will describe a typical, rather than any specific, system.

3. The core of an ADS is an Air Data Computer (ADC) which forms an essential part of a modern flight/navigation/weapon aiming system. The ADS measures the basic air inputs of pitot pressure, static pressure, outside air temperature, angle of attack (α angle), sideslip (β angle), and outputs flight parameters for the various systems and displays. A comprehensive ADS thus consists of:

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

b. Local incidence vanes for α angle and β angle computation.

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

d. Air Data Computer to process the data and provide the required outputs to the aircraft systems and displays.

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

Probes

4. **Pitot/Static**. Pitot and static pressures are taken from the aircraft's pressure head or the pitot head and static vents.

5. **Temperature**. Temperature is determined from outside air temperature (OAT) probes, as described in Volume 5, Chapter 8.

6. **Angle of Attack (α Angle)**. Angle of attack is the angle, in the 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 (see Volume 5, Chapter 23). The vane is usually mounted on the side of the fuselage near the nose or on a probe forward of the wing or nose.

7. **Angle of Sideslip (β Angle)**. The sideslip angle is the angle in the horizontal plane at which the free stream airflow meets an arbitrary longitudinal datum line on the aircraft. The β sensor is normally identical to the α sensor and mounted on the underside of the airframe along the aircraft centre line. In simpler ADS the β sensor is often omitted.

Transducers

8. Transducers, which convert pressures, temperatures and angles to voltages or digital pulses, are the most vital elements of the air data systems, and are the limiting factors in the system accuracy.

Transducers vary in type depending on the parameter which is to be measured, ie pressure transducers, temperature transducers and angular transducers. Various techniques are employed to convert the measured data into usable, repeatable and accurate signals which can be transmitted to the ADC, e.g. using the expansion of a diaphragm or capsule to actuate an electrical pick-off, or to vary the electrical resistance of a wire by changing the wire's tension.

Air Data Computer

9. The air data computer processes the data input from the sensors, applies any necessary corrections, and supplies output data in the form required by other equipment, either directly or via a central computer. Particularly in older systems, where there is no central computer, the same output parameter may be in several forms, e.g. pressure altitude may be processed as a voltage, a synchro output, and a digital code. Fig 1 shows a typical ADS arrangement.

10. Compared with conventional pressure instruments, the ADS has the following advantages:

a. The bulk and complexity of pipe work is avoided.

b. Duplication of units is avoided.

c. Errors can be automatically corrected before display.

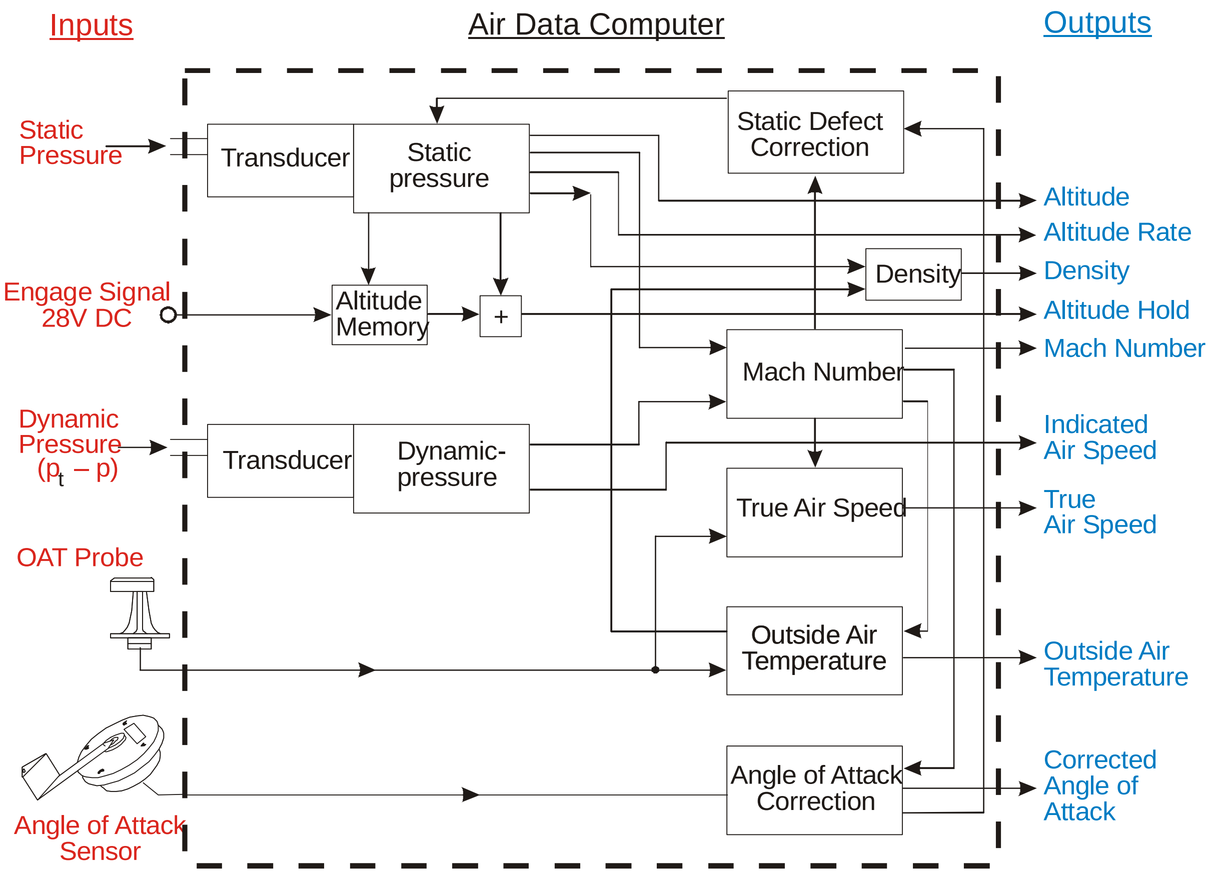
d. There are accuracy and sensitivity gains.

e. There is a reduced time lag.

f. There is the potential for flexibility in presentation.

The disadvantage of the ADS is that it needs power to work whereas conventional pressure instruments do not. It is therefore usually necessary to provide back-up systems, either in the form of alternative power supplies or simple pressure instruments.

5-9 Fig 1 Air Data System

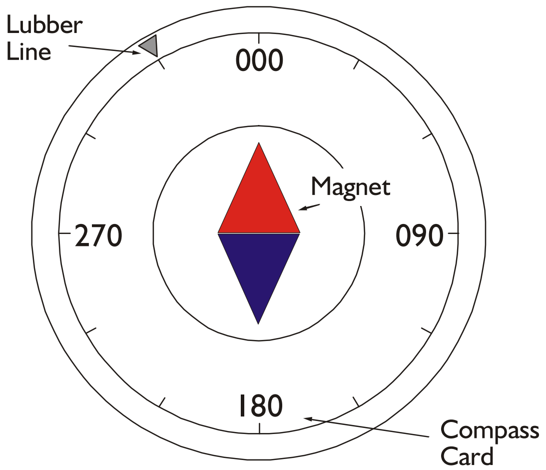


CHAPTER 10
DIRECT INDICATING COMPASSES AND DIRECTION INDICATORS

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.

5-10 Fig 1 A Basic DICS



2. **Properties**. For a DICS to operate satisfactorily in conditions encountered in flight, it must exhibit the following properties:

   a. Horizontality.

   b. Sensitivity.

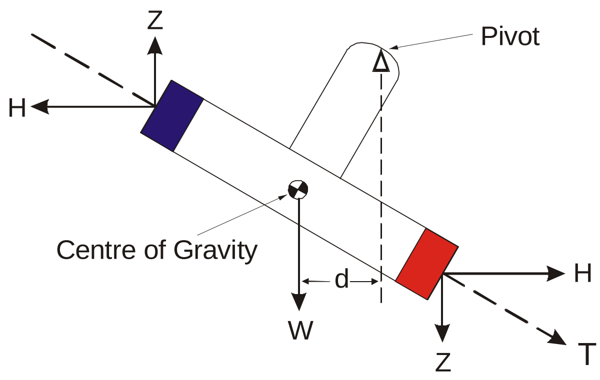
   c. Aperiodicity.

These properties are discussed in detail in the following paragraphs.

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 (H) and vertical (Z) fields (see Fig 2).

5-10 Fig 2 Pendulous Suspension



4. If the magnet was allowed to align itself with the T field, it would be difficult to align the compass card accurately. Moreover, the 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's tendency to tilt. When the pendulously suspended magnet 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 Wd, which acts to restore the system to the horizontal. In UK latitudes, the residual tilt in a well-designed compass is approximately 2º.

Sensitivity

5. The 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 (the magnetic moment of a compass needle is dependent upon the length of the needle and its magnetic strength).

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 compass system is prone to vibrations and accelerations in flight, and these can cause undesirable periodic oscillations. To make the system aperiodic (i.e. without a natural period) the design may incorporate:

a. A magnet system with a low moment of inertia and high magnetic moment (the same measures as applied for sensitivity).

b. Some 'damping out' of the oscillations by immersing the moving parts of the compass system in fluid.

DICS - ERRORS AND LIMITATIONS

8. In addition to the errors caused by external magnetic fields, DICS are subject to the errors and limitations covered in the following paragraphs.

Turning and Acceleration Errors - Cause

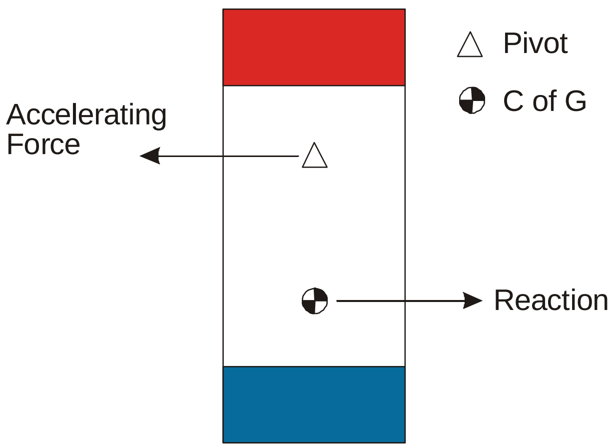
9. If an aircraft fitted with a DICS is subjected to horizontal accelerations, the accelerating forces may cause errors in the indicated heading. The accelerations may be the result of speed changes or from the central acceleration experienced in a turn; both have similar effects on the compass system, the resultant errors being greatest when the accelerating force acts at right angles to the magnetic meridian with which the compass is aligned, i.e. 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

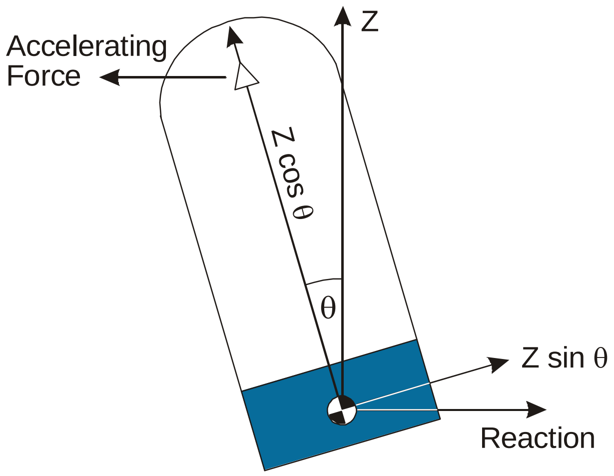
10. Consider an aircraft in the Northern Hemisphere increasing speed whilst heading West, or turning from North or South on to West. In both cases, the accelerating force acts through the pivot which is the magnet system's point of attachment to the aircraft. The reaction force acts, not through the 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 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 90º 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).

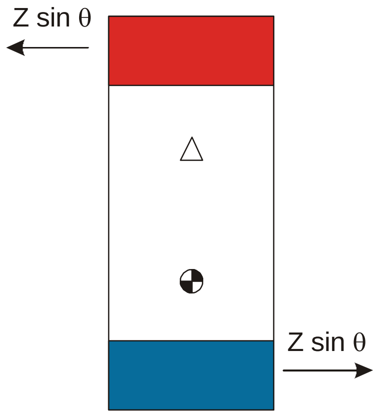
5-10 Fig 3 Accelerating Force Producing Couple



5-10 Fig 4 Acceleration Causing Tilt



5-10 Fig 5 Couple Causing Turn



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, i.e. 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 20º for an actual turn through 45º. 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, i.e. the compass indicates a turn of perhaps 40º for an actual turn of 20º.

Summary

13. The effects of turning and acceleration errors are 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.

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 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.

Operational Limitations

15. A DICS has the following limitations which make it unsuitable for use as the primary heading system of a modern aircraft:

a. It depends upon the size of the horizontal component of the Earth's magnetic field for its drive and thus it becomes insensitive and unreliable at high magnetic latitudes.

b. It must be installed in the aircraft cockpit, which is normally an area of high magnetic deviation.

c. It can only provide magnetic heading, whereas true or grid heading may be required on occasions.

d. Turning and acceleration errors make it only suitable for use in straight, unaccelerated flight.

e. There is insufficient torque to enable it to drive transmission systems to feed other aircraft equipment.

Advantages

16. Despite the limitations of a DICS it is likely to be fitted to most aircraft for the foreseeable future as a standby compass. In this application it has the advantages of being cheap to purchase and install, small and light, simple and easy to maintain and operate, and requiring no power, except for lighting.

A PRACTICAL DICS

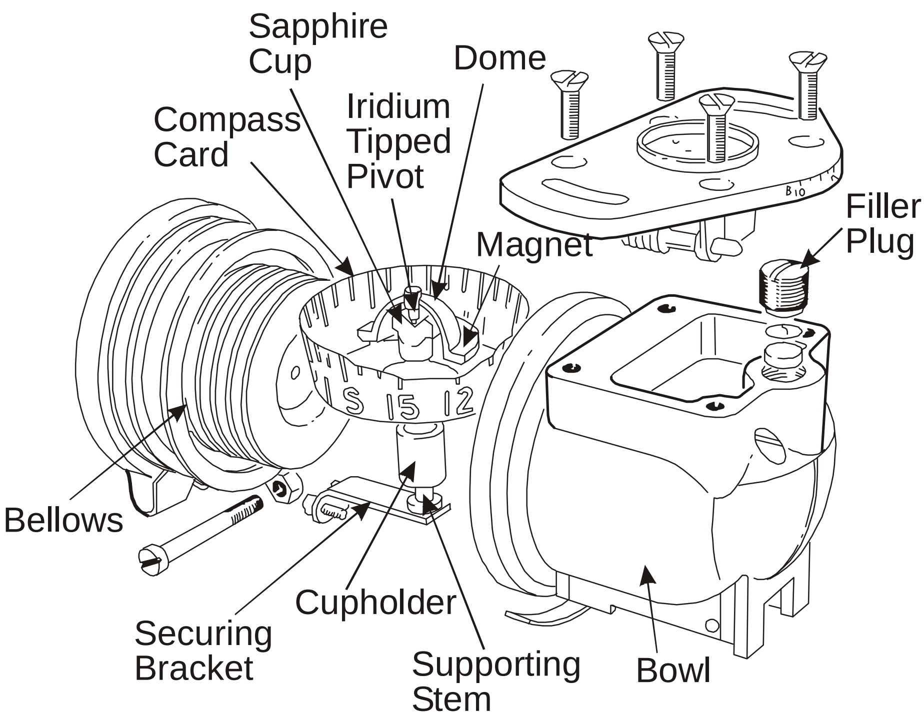
The E2 Series

17. The principles of the DICS are exemplified in the E2 series of standby compasses which are widely used (Fig 6). The differences between the E2A, E2B and E2C are minor and mostly concern the lighting arrangements. The compasses have a vertical card fastened to the magnet system, graduated every 10 degrees, with figures every 30 degrees. The cardinal points are marked with the appropriate letter. The compasses are designed to give an operational accuracy of $\pm 10^{\circ}$, in good, stable flight conditions the accuracy may approach the bench accuracy of 2.5°.

5-10 Fig 6 E2 Compass

18. **Design**. Fig 7 shows an exploded view of an E2 compass. The bowl is plastic with a lubber line marked on the front inside. The magnet is a steel ring to which a dome is attached. 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 compass bowl is filled with a silicone fluid and a bellows at the rear of the bowl allows for a change of the volume of the liquid due to variations in temperature. Provision is made for correction of coefficients A, B, and C (see Volume 5, Chapter 16).

5-10 Fig 7 E2 Compass - Exploded View



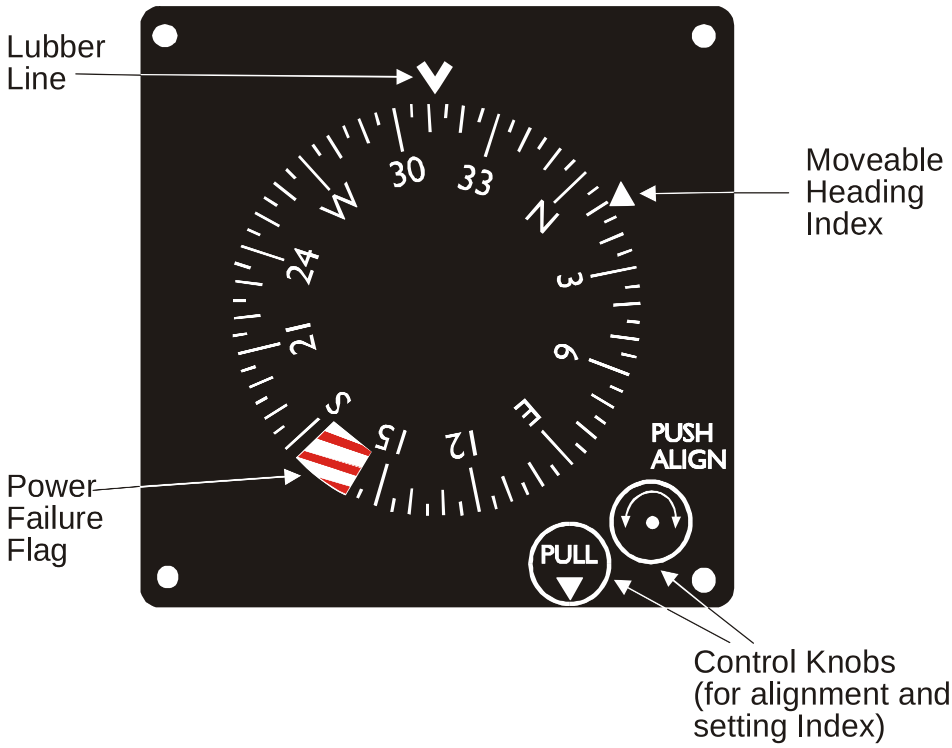
19. **Serviceability Checks**. Before use the compass should be checked to ensure that the bowl is not cracked or damaged and is completely filled with fluid that is free from excessive discolouration, bubbles and sediment.

DIRECTION INDICATORS

Operation

20. The direction indicator (DI) is used, mostly in light aircraft, as a simple heading reference. It consists of an air or electrically driven, two degree of freedom, displacement gyro with its spin axis mounted horizontally (see Volume 5, Chapter 11). The DI must initially be set to a known heading such as that obtained from a direct indicating compass. Thereafter it may be used as a heading reference during level flight provided that it is checked and reset if necessary to the correct heading periodically. The display is usually in the form of a conventional plan form compass rose and the only controls provided are to reset the indicated heading, and to position a moveable heading index (see Fig 8).

5-10 Fig 8 Direction Indicator Display



21. The spin axis is maintained in the horizontal plane either by the action of a gravity actuated torque motor or by air jets initiated by a liquid level switch.

Errors

22. The direction indicator is subject to the normal wander errors associated with gyros. Topple is controlled within acceptable limits by the action of the levelling system.

23. The combination of real and apparent drift could make the total error rate accrued by a direction indicator to be in the order of 10 to 20º/hr, hence the need to reset the instrument at regular intervals. Resetting should be done in straight, unaccelerated flight. Clearly the direction indicator cannot be relied upon as a primary heading reference.

CHAPTER 11 - INTRODUCTION TO GYROSCOPES

Introduction

1. Modern technology has brought about many changes to the gyroscope. The conventional spinning gyroscope is still in current use for flight instruments in smaller and simpler aircraft. More sophisticated aircraft however, make use of devices which are termed 'gyros', but this is because of the tasks they perform rather than their manner of operation. Gyroscopes can therefore be categorised as:

a. Spinning Gyroscopes.

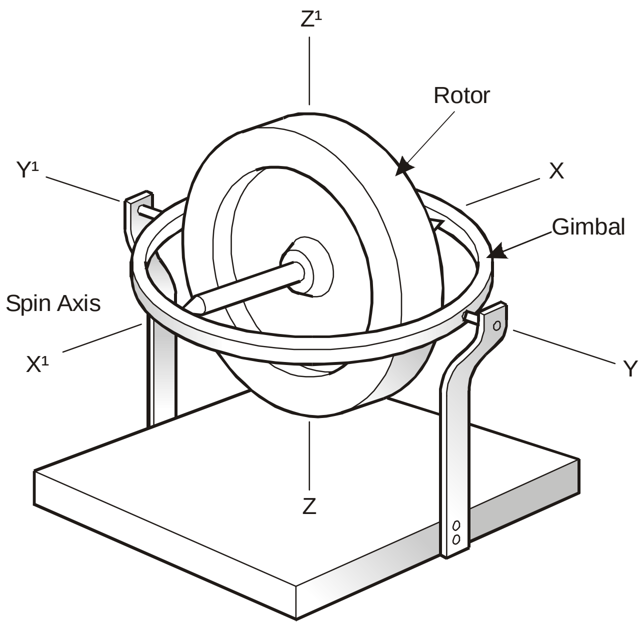
b. Optical Gyroscopes.

c. Vibrating Gyroscopes.

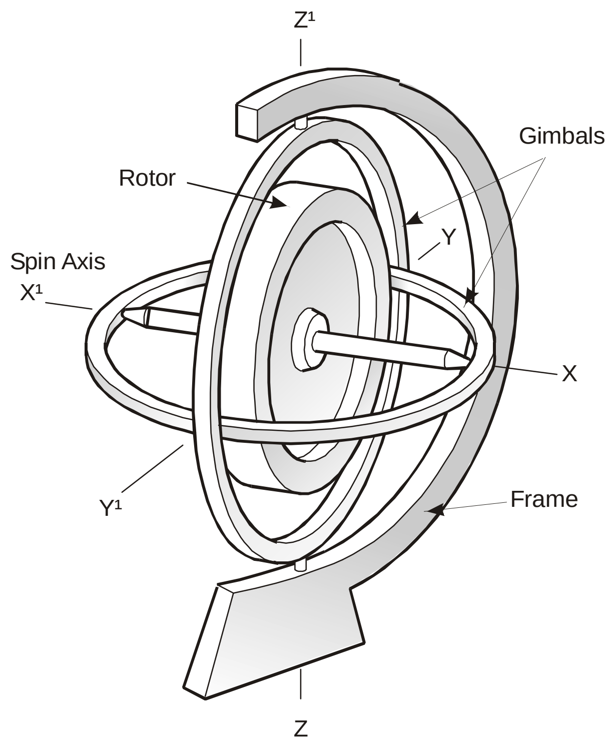
This chapter will concentrate for the most part on the spinning gyroscope.

2. A conventional gyroscope consists of a symmetrical rotor spinning rapidly about its axis and free to rotate about one or more perpendicular axis. Freedom of movement about one axis is usually achieved by mounting the rotor in a gimbal, as in Fig 1 where the gyro is free to rotate about the YY^1 axis. Complete freedom can be approached by using two gimbals, as illustrated in Fig 2.

5-11 Fig 1 One-degree-of-freedom Gyroscope



5-11 Fig 2 Two-degrees-of-freedom Gyroscope



3. The physical laws which govern the behaviour of a conventional 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 and precession. These properties, which are explained later, are exploited in some heading reference and inertial navigation systems (INS) and other aircraft instruments.

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 (ω) is the tangential velocity (v) at the periphery of a circle, divided by the radius of the circle (r), so $\omega = \frac{v}{r}$. Angular velocity is normally measured in radians per second.

c. **Moment of Inertia**. Since a rotating rigid body consists of mass in motion, it possesses kinetic energy. This kinetic energy can be expressed in terms of the body's angular velocity and a quantity called 'Moment of Inertia'. Imagine the body as being made up of an infinite number of particles, with masses m_1, m_2, etc, at distances r_1, r_2, etc from the axis of rotation. In general, the mass of a typical particle is m_x and its distance from the axis of rotation is r_x. Since the particles do not necessarily lie in the same plane, r_x is specified as the perpendicular distance from the particle to the axis. The total kinetic energy of the body is the sum of the kinetic energy of all its particles:

$$K = \tfrac{1}{2} m_1 r_1^2 \omega^2 + \tfrac{1}{2} m_2 r_2^2 \omega^2 + \ldots$$

$$= \Sigma_x \tfrac{1}{2} m_x r_x^2 \omega^2$$

Taking the common factor $\tfrac{1}{2}\omega^2$ out of the expression gives:

$$K = \tfrac{1}{2} \omega^2 (m_1 r_1^2 + m_2 r_2^2 + \ldots)$$

$$= \tfrac{1}{2} \omega^2 (\Sigma_x m_x r_x^2)$$

The quantity in parenthesis, obtained by multiplying the mass of each particle by the square of the distance from the axis of rotation and adding these products, is called the Moment of Inertia of the body, denoted by I:

$$I = m_1 r_1^2 + m_2 r_2^2 + \ldots = \Sigma_x m_x r_x^2$$

In terms of the moment of inertia (I), the rotational kinetic energy (K) of a rigid body is

$$K = \tfrac{1}{2} I \omega^2$$

d. **Angular Momentum**. Angular Momentum (L) is defined as the product of Moment of Inertia and Angular Velocity, ie $L = I\omega$.

e. **Gyro Axes**. In gyro dynamics 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. The XX^1, YY^1 and ZZ^1 axes shown in the diagrams are not intended to represent the x, y and z axes of an aircraft in manoeuvre. However, if the XX^1 (rotational) axis of the gyro is aligned with the direction of flight, the effects of flight manoeuvre on the gyro may be readily demonstrated.

Classification of Gyroscopes

5. Conventional gyroscopes are classified in Table 1 in terms of the quantity they measure, namely:

a. **Rate Gyroscopes**. Rate gyroscopes measure the rate of angular displacement of a vehicle.

b. **Rate-integrating Gyroscopes**. Rate-integrating gyroscopes measure the integral of an input with respect to time.

c. **Displacement Gyroscopes**. Displacement gyroscopes measure the angular displacement from a known datum.

Table 1 Classification of Gyros

Type of Gyro	Uses in Guidance and Control	Gyro Characteristics
Rate Gyroscope	Aircraft Instruments	Modified single-degree-of-freedom gyro.
Rate-integrating Gyroscope	Older IN Systems	Modified single-degree-of-freedom gyro. Can also be a two-degree-of-freedom gyro.
Displacement Gyroscope	Heading Reference Older IN Systems Aircraft Instruments	Two degrees of freedom. Defines direction with respect to space, thus it is also called a space gyro, or free gyro.

6. 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-degrees-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 rotation about a chosen axis.

LAWS OF GYRODYNAMICS

Rigidity in Space

7. If the rotor of a perfect displacement gyroscope is spinning at constant angular velocity, and therefore constant angular momentum, no matter how the frame is turned, no torque is transmitted to the spin 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 spin axis continues to point in the same direction in inertial space. This property of a gyro is defined in the First Law of Gyro dynamics.

The First Law of Gyro dynamics

8. The first law of gyro dynamics states that:

> "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."

9. 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.

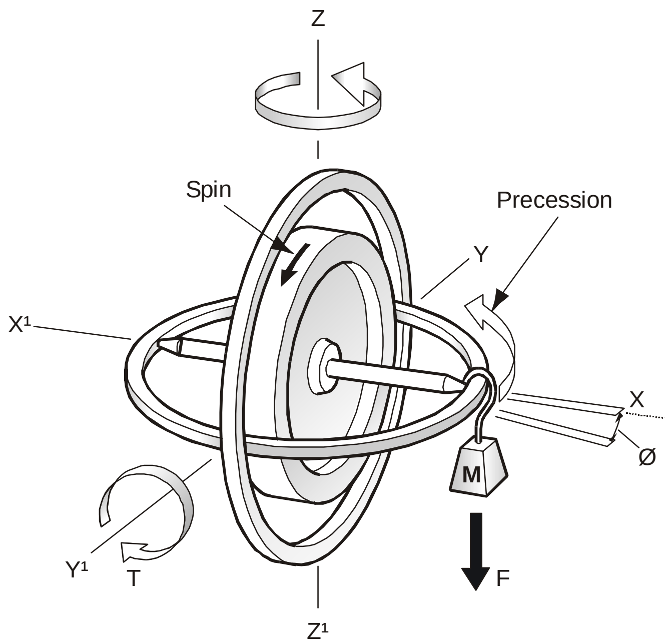
Precession

10. 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:

a. Initially, the gyro spin axis will tilt through a small angle (∅ in Fig 3), after which no further movement takes place about the YY^1 axis. The angle ∅ is proportional to T and is a measure of the work done. Its value is almost negligible and will not be discussed further.

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 Gyro dynamics.

5-11 Fig 3 Precession



The Second Law of Gyro dynamics

11. The second law of gyro dynamics states that:

"If a constant torque (T) is applied about an axis perpendicular to the spin axis of an unconstrained, symmetrical 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 $\Omega = \frac{T}{I\omega}$."

12. 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.

Direction of Precession

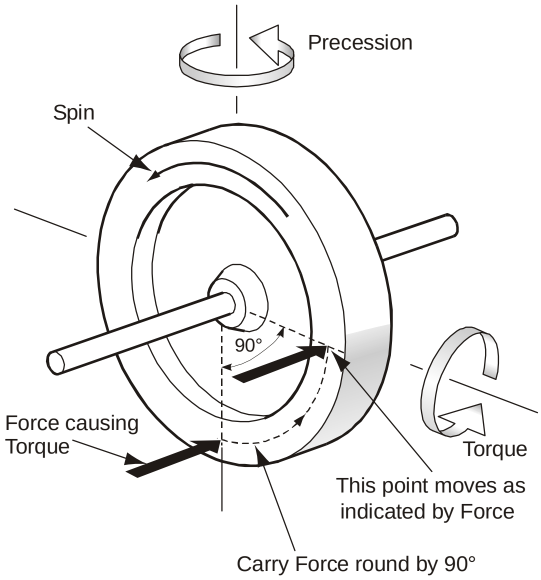
13. Fig 4 shows a simple rule of thumb to determine the direction of precession:

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 90º in the direction of rotor spin.

c. The torque will apparently act through this point and the rotor will precess in the direction shown.

5-11 Fig 4 Determining Precession



CONSERVATION OF ANGULAR MOMENTUM

Explanation

14. 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 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.

15. 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.

Cause of Precession

16. 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 in quadrants 1, 2, 3 and 4 tend to conserve angular momentum about YY^1, therefore:

a. Particles in quadrants 1 and 3 exert forces opposing their movement about YY^1.

b. Particles in quadrants 2 and 4 exert forces assisting their movement about YY^1.

17. 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.

Gyroscopic Resistance

18. 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 a resultant torque L, acting about this axis 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.

19. 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 16, 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.

5-11 Fig 5 Instant of Spin and Precession

Fig 5a

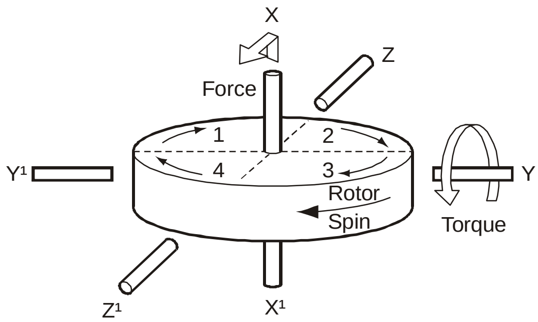


Fig 5b

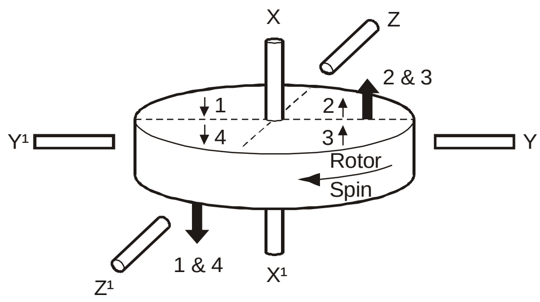
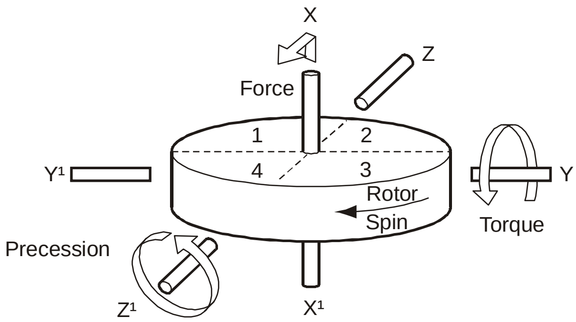
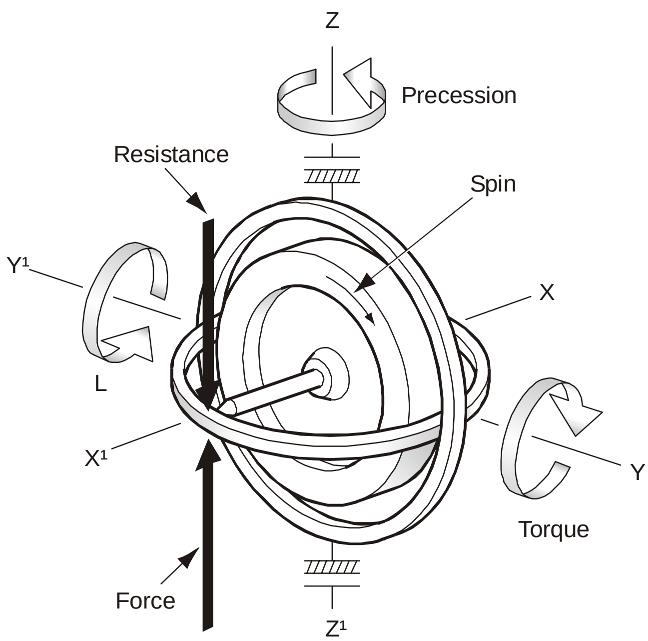


Fig 5c



5-11 Fig 6 Gyroscopic Resistance



20. 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 is as easy to move the spin axis when it is 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.

Secondary Precession

21. 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.

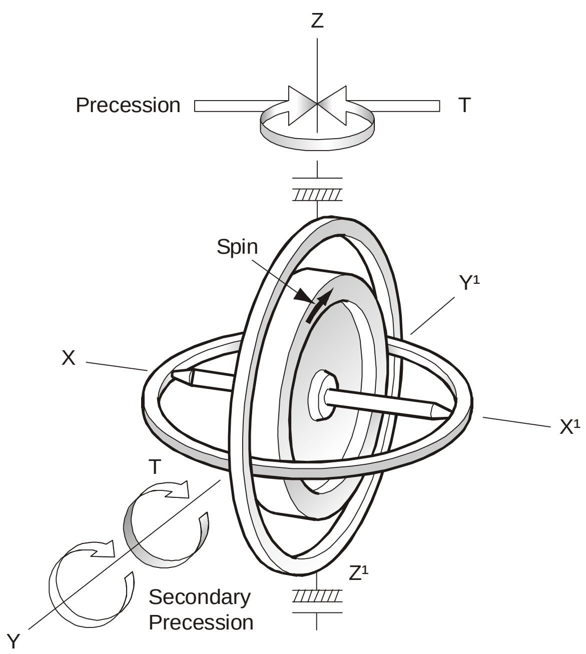
b. A deflection takes place about the torque axis, (dip), which remains constant provided that the gyro is perfect, and the applied torque is also constant.

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

22. 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 21b.

23. 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 13, it will be seen that the torque T causes the spin axis to dip through a larger angle. This precession is known as secondary precession.

5-11 Fig 7 Precession Opposed by Secondary Precession



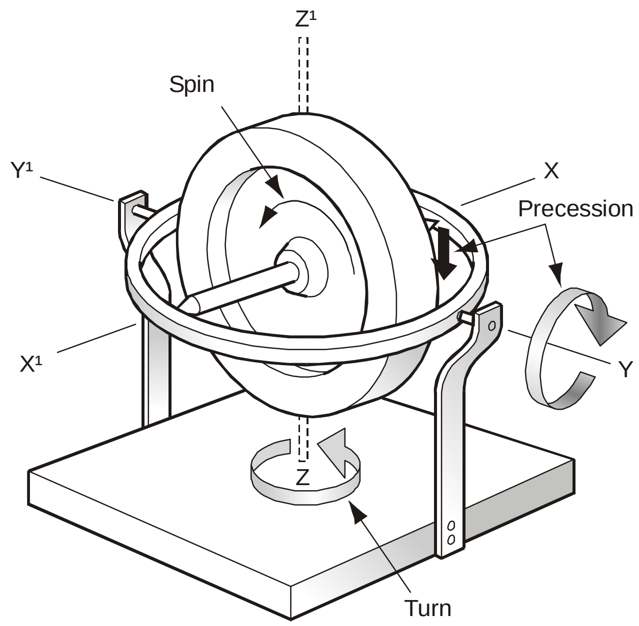
24. Secondary precession can only take place when the gyro is already precessing, thus its name. Note also that secondary precession acts in the same direction as the originally applied torque.

THE RATE GYROSCOPE

Principle of Operation

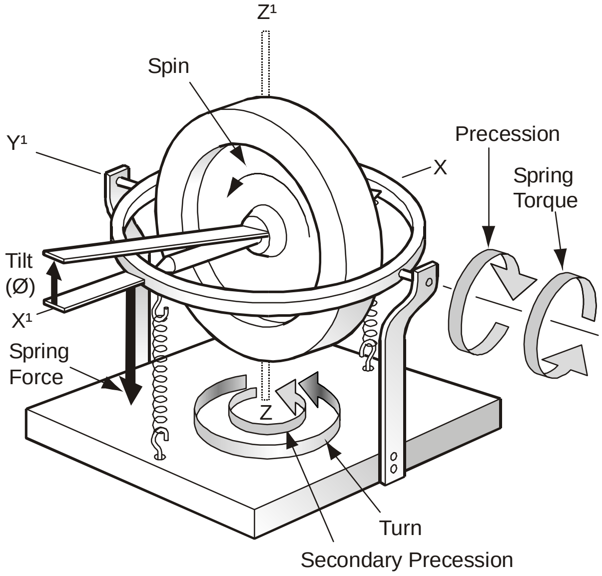
25. Fig 8 shows a gyroscope with freedom about one axis YY^1. If the frame of the gyro is turned about an axis ZZ^1 at right angles to both YY^1 and XX^1, then the spin axis will precess about the YY^1 axis. The precession will continue until the direction of rotor spin is coincident with the direction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

5-11 Fig 8 Gyro with One degree of Freedom – Precession



26. 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 24). 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.

5-11 Fig 9 Rate Gyroscope



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

Spring Torque is proportional to ∅ or

Spring Torque = K∅ (where K is a constant)

At equilibrium:

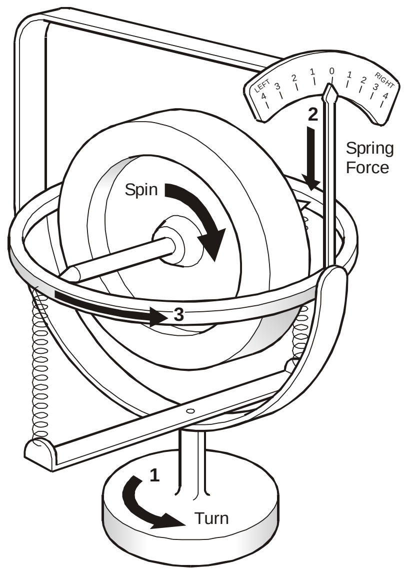
Rate of Secondary Precession = Rate of Turn

ie $\frac{K\varnothing}{I\omega}$ = Rate of Turn

$\therefore$ ∅ is proportional to 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 arrangement shown at Fig 10 and the scale calibrated accordingly.

5-11 Fig 10 Rate of Turn Indicator

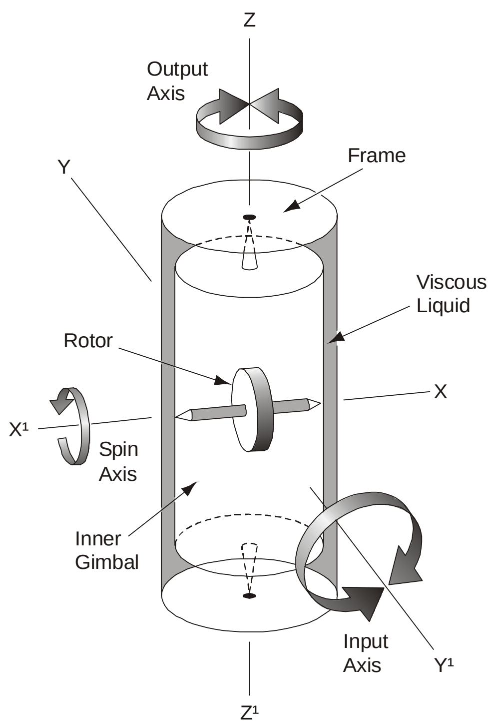


THE RATE-INTEGRATING GYROSCOPE

Principle of Operation

28. A rate-integrating gyroscope is a single degree of freedom gyro using viscous restraint to damp the precessional rotation about the output axis. The rate-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).

5-11 Fig 11 Simple Rate-integrating Gyroscope



29. The rate-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 rate-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 rate-integrating gyro has certain advantages over the displacement type. These are:

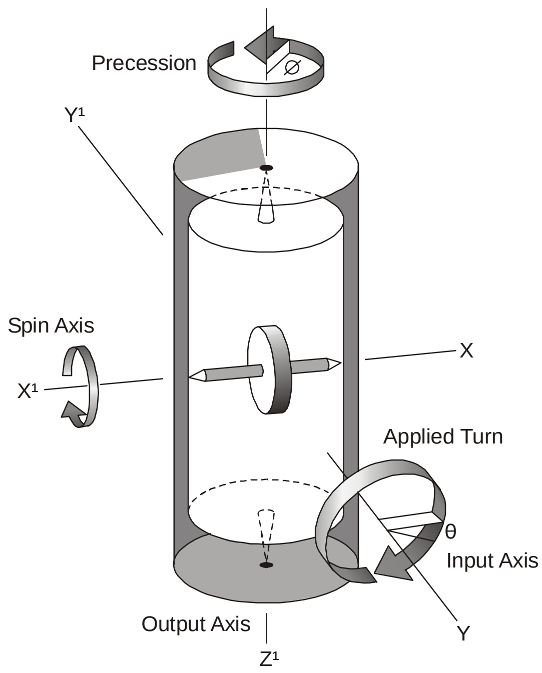
a. A small input rate causes a large gimbal deflection (gimbal gain).

b. The gyro does not suffer from nutation.

30. Fig 11 shows a simple rate-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, ba1l bearings support the entire gimbal weight and define the output axis. In the floated rate-integrating gyro the entire weight of the rotor and inner gimbal assembly is supported by the viscous liquid, thereby minimizing 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 utilizing this system for gimbal axis alignment, with fluid to provide support, the bearing friction is reduced to a very low figure.

31. 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 25 to 27 puts out a rate of turn only). The movement about the output axis may be made equal to, less than, or greater than 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.

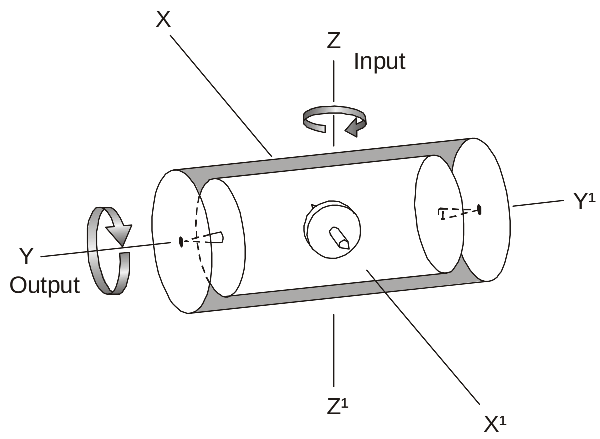
5-11 Fig 12 Function of Rate-integrating Gyroscope



32. A gyro mounted so that it senses 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 using conventional gyros align the input axis of their levelling gyros with True North and East.

33. Motion around the third axis, the vertical axis, is measured by an azimuth gyro, ie one in which the input axis is aligned with the vertical, as in Fig 13.

5-11 Fig 13 Rate-integrating Azimuth Gyroscope



THE DISPLACEMENT GYROSCOPE

Definition

34. 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

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

36. **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 move physically. 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 minimized by better engineering techniques.

b. A gyro defines direction with respect to inertial space, whilst the navigator requires Earth directions. In order to use a gyro to determine directions on Earth, it must be corrected for apparent wander due to the fact that the Earth rotates or that the gyro may be moving from one point on Earth to another (transport wander).

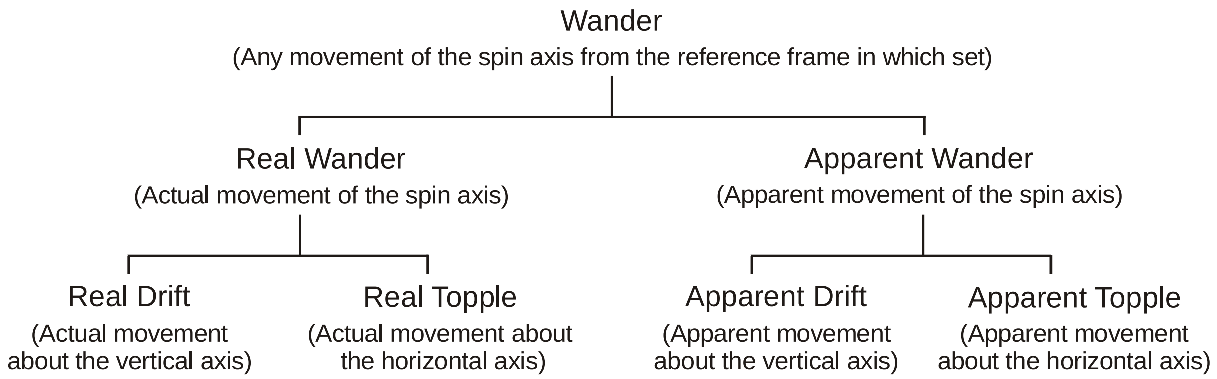
37. **Drift and Topple**. It is more convenient to study wander by resolving it into two components:

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

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

38. **Summary**. Table 2 summarizes the types of wander. From para 36 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.

Table 2 Types of Wander



Earth Rotation

39. 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.

40. Consider a gyro positioned at a point A in Fig 14. It would be affected by Earth rotation according to how its input axis was aligned, namely:

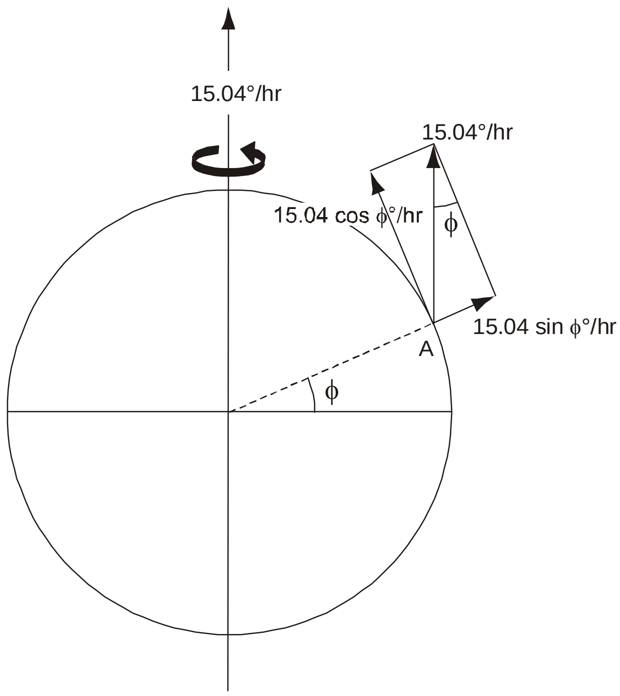
a. If its input axis was aligned with the Earth's spin axis, it would detect Earth rate (Ωe) of 15.04 º/hr.

b. **Azimuth Gyro**. If its input axis was aligned with the local vertical it would detect $15.04 \times \sin \phi$ º/hr, where ϕ = latitude. Note that, by definition, this is drift.

c. **North Sensitive Levelling Gyro**. If its input axis were aligned with local North, it would detect $15.04 \times \cos \phi$ º/hr. Note that, by definition, this is topple.

d. **East Sensitive 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.

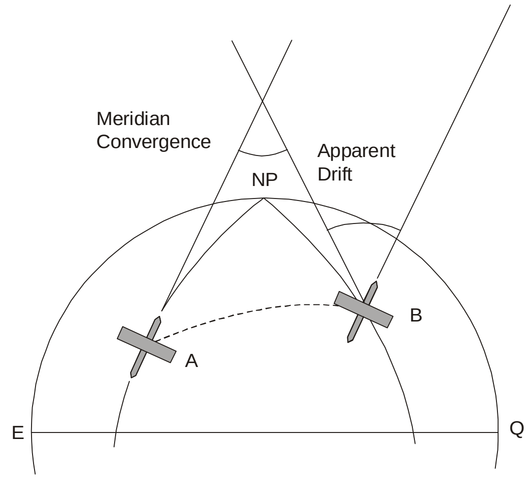
5-11 Fig 14 Components of Earth Rate



Transport Wander

41. 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. Transport rate drift thus depends on the convergence of the meridians and the rate of crossing them; i.e. the East-West component of ground speed (U). The amount of convergence between two meridians (C) is ch long × sin lat. Any given value of U thus produces an increase in apparent gyro drift as latitude increases.

5-11 Fig 15 Apparent Drift



The amount of drift due to transport rate may be found as follows:

$$C\ (^\circ/\text{hr}) = [\text{ch long/hr}] \times \sin\phi.$$

$$\text{Now, ch long/hr} = \frac{\text{ch Eastings (nm / hr)}}{60} \times \sec\phi$$

and, since 1º = 60 nm and ch Eastings (nm/hr) = U

$$C = \frac{U}{60} \times \sec\phi \times \sin\phi\ (^\circ/\text{hr})$$

$$\text{but, } \sec\phi \times \sin\phi = \frac{1}{\cos\phi} \times \sin\phi$$

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

$$\therefore\ C = \frac{U}{60} \times \tan\phi\ \ (^\circ/\text{hr})$$

This can be converted to radians/hour by multiplying by $\frac{\pi}{180}$

$$\therefore\ C = \frac{U}{60} \times \tan\phi \times \frac{\pi}{180} = U \times \tan\phi \times \frac{\pi}{60 \times 180}$$

Now an arc of length 60 nm on the Earth's surface subtends an angle of 1º (π/180º) at the centre of the Earth

$$\therefore\ R \times \frac{\pi}{180} = 60 \text{ where R= Earth's radius}$$

$$\text{or, } \frac{1}{R} = \frac{\pi}{60 \times 180}$$

Substituting into the above equation for Meridian Convergence (radians/hour)

$$C = U \times \tan\phi \times \frac{1}{R}$$

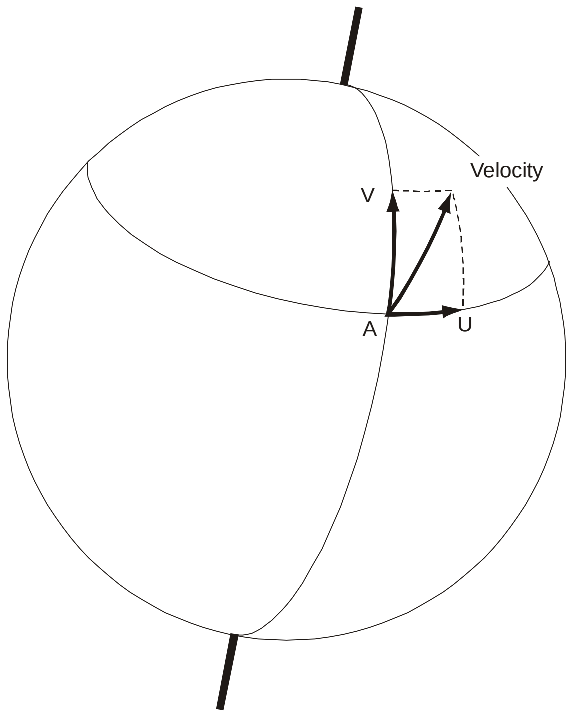
$$\text{or,} \quad C = \frac{U}{R} \times \tan\phi \text{ (radians/hour)}$$

42. 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 $\frac{V}{R}$.

5-11 Fig 16 Transport Wander



Apparent Wander Table

43. All of 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.

44. **Correction Signs**. The correction signs of Table 3 apply only to the drift equations, and they should be applied to the gyro readings to obtain true directions. These correction signs will be reversed for the Southern Hemisphere.

Table 3 Components of Drift and Topple – Earth Rate and Transport Wander Rate

	Input Axis Alignment			
	Local North	**Local East**	**Local Vertical**	**Correction Sign**
Earth Rate degrees (or radians) per hour	$\Omega e \cos \phi$	Nil	$\Omega e \sin \phi$	+
Transport Wander radians per hour	$\frac{U}{R}$	$\frac{-V}{R}$	$\frac{U}{R} \tan \phi$	+E –W
	Topple		Drift	

Ωe = Angular Velocity of the Earth   R = Earth's Radius ϕ = Latitude

U = East/West component of groundspeed   V = North/South component of groundspeed

Practical Corrections for Topple and Drift

45. 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 wander of a gyro and consequently an error growth proportional to the magnitude of the real drift and topple will exist. As a rough rule of thumb, an inertial platform employing gyros with real drift rates in the order of 0.01º/hr will have a system error growth of 1 to 2 nm/hr CEP.

46. Flight instruments, on the other hand, employ cheaper, lower quality gyros whose drift rates may be in the order of 0.1º/hr. If these real drift rates were not compensated for, system inaccuracies would be unacceptably large. For this reason, some flight instruments make use of the local gravity vector to define the level plane, thus compensating for both real and apparent drifts.

47. Specifically, gyro wander 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 17 and 18), or by case levelling devices (see Fig 19). 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 approximately 1º.

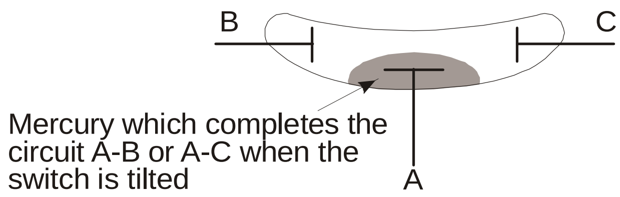
b. **Drift**. Drift corrections can be achieved by:

(1) Calculating corrections using Table 3 and applying them to the gyro reading.

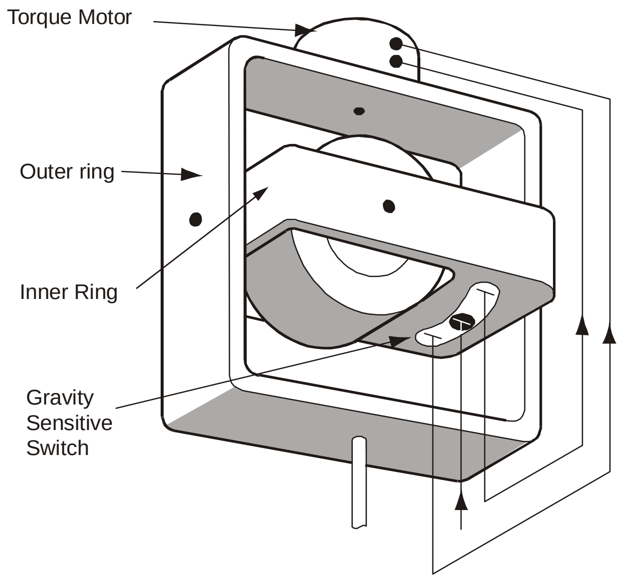
(2) Applying a fixed torque to the gyro so that it precesses at a rate equal to the Earth rate for a selected latitude. Although this method is relatively simple, it has the disadvantage that the compensation produced will only be correct at the selected latitude.

(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 coupling the azimuth gyro to a flux valve.

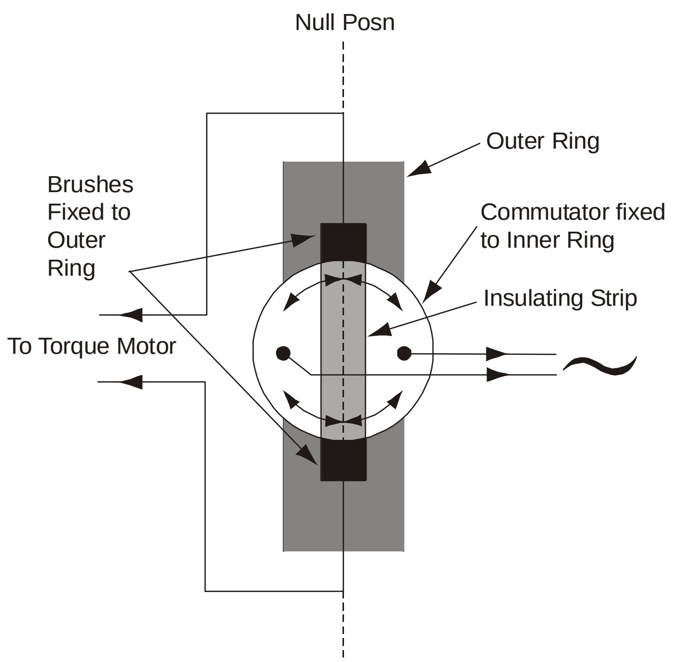
5-11 Fig 17 Gravity Sensitive Switch



5-11 Fig 18 Gravity Levelling



5-11 Fig 19 Synchro Case Levelling Device



48. To complete this study of the displacement gyro, it remains to mention a limitation and an error peculiar to this type of gyro, namely gimbal lock and gimbal error.

Gimbal Lock

49. 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

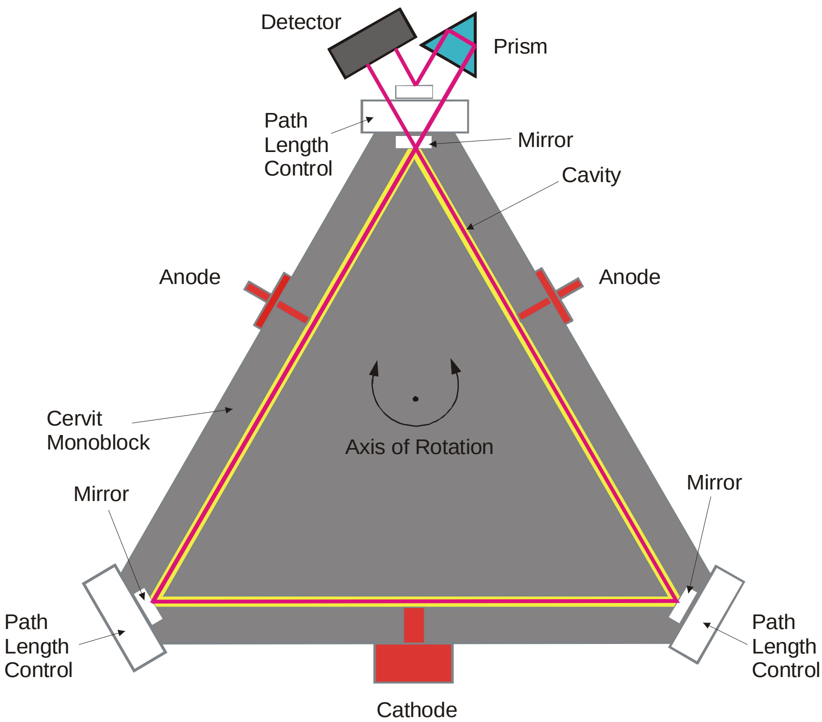
50. When a 2-degree-of-freedom gyroscope with a horizontal spin axis is both banked and rolled, the outer gimbal must rotate to maintain orientation of the rotor axis, thereby inducing a heading error at the outer gimbal pick-off. The incidence of this error depends upon the angle of bank and the angular difference between the spin axis and the longitudinal axis and, as in most systems, the spin axis direction is arbitrary relative to North, the error is not easily predicted. Although the error disappears when the aircraft is levelled, it will have accumulated in any GPI equipment, producing a small error in computed position.

OTHER TYPES OF GYRO

The Ring Laser Gyro

51. The ring laser gyro is one of the modern alternatives to conventional gyros for a number of applications including aircraft inertial navigation systems and attitude/heading reference systems. The ring laser gyro has no moving parts and is not a gyroscope in the normal meaning of the word. The ring laser gyro is, however, a very accurate device for measuring rotation and became the system of choice for use with strapdown inertial navigation systems. A schematic diagram of a ring laser gyro is at Fig 20.

5-11 Fig 20 Schematic Diagram of a Ring Laser Gyro



52. **Principle of Operation**. In ring laser gyros, the rotating mass of the conventional gyro is replaced by two contra-rotating beams of light. The main body of the gyro consists of a single piece (or 'monoblock') of a vitreous ceramic of low temperature coefficient (typically 'Cervit' or 'Zerodour'). A gas-tight cavity is accurately machined into the monoblock and this cavity is then filled with an inert gas, typically a mixture of Helium and Neon. A DC electrical discharge ionizes the gas and causes the lasing action. Two beams of light are produced, flowing in opposite directions in the cavity. Mirrors are used to reflect the beams around the enclosed area, producing a 'laser-in-a-ring' configuration. The frequency of oscillation of each beam corresponds to the cavity resonance condition. This condition requires that the optical path length of the cavity be an integral number of wavelengths. The frequency of each beam is therefore dependant on the optical path length.

53. **Effect of Movement**. At rest, the optical path length for each beam is identical; therefore, the frequencies of the two laser beams are the same. However, when the sensor is rotated about the axis perpendicular to the lasing plane, one beam travels an increased path length, whilst the other travels a reduced path length. The two resonant frequencies change to adjust to the longer or shorter optical path, and the frequency difference is directly proportional to the rotation rate. This phenomenon is known as the Sagnac effect. The frequency difference is measured by the beaming of an output signal for each wave on to photo detectors spaced one quarter of a wavelength apart, causing an optical effect known as an interference fringe. The fringe pattern moves at a rate that is directly proportional to the frequency difference between the two beams. It is converted to a digital output, where the output pulse rate is proportional to the input turn rate, and the cumulative pulse count is proportional to the angular change. This effect can be quantified using simplified maths, where it can be shown that the frequency difference Δf of the two waves is:

$$\Delta f = \frac{4A\Omega}{\lambda L}$$

Where:
A is the area enclosed by the path.
λ is the oscillating wavelength.
L is the length of the closed path.
Ω is the rate of rotation.

54. **Gyro Control**. The reason why the two beams have to occupy the same physical cavity is the sensitivity of laser light to cavity length. If they did not, a temperature induced difference in path length could result in a large frequency mismatch between the two beams. The path length control mechanism is used to alter the intensity of the laser and thus control expansion due to excess heat. To help avoid perceived differences in path length due to flow of the Helium/Neon gas mix, two anodes are used to balance any flow caused by ionization.

55. **Error Sources**. Ring laser gyros are subject to a number of errors, the most notable are:

a. **Null Shift**. Null shift arises due to a difference in path length as perceived by the two opposite beams, thus producing an output when no rotation exists. The major causes of perceived path length difference are:

(1) Differential movement of the gas in the cavity.

(2) Small changes in the refractive index of the monoblock material as the direction of travel of the laser light changes.

b. **Lock-in**. Lock-in occurs when the input rotation rate of the gyro is reduced below a critical value causing the frequency difference between the clockwise and anti-clockwise beams to drop to zero. One of the main causes of this phenomenon is backscatter of light at the mirrors. Some of the clockwise beam is reflected backwards, thereby contaminating the anti-clockwise beam with the clockwise frequency. Similarly, backscattering of the anti-clockwise beam contaminates the clockwise beam. With low input rotational rates, the two beams soon reach a common frequency which renders detection of rotation impossible. Several methods are used to ensure that lock-in is minimized. One method is to physically dither the gyro by inputting a known rotation rate in one direction, immediately followed by a rotation rate in the opposite direction. As the dither rate is known, it can be removed at the output stage. The dither ensures that the two frequencies are kept far enough apart to avoid lock-in.

56. **Advantages of the Ring Laser Gyro**. The main advantages of the ring laser gyro are:

a. Its performance is unaffected by high 'g'.

b. It has no moving parts and therefore has high reliability and low maintenance requirements.

c. It has a rapid turn-on time.

57. **Disadvantage of the Ring Laser Gyro**. The technical problems associated with ring laser gyros can all be overcome. However, solution of these problems inevitably increases costs which are already very high due to the complex, 'clean-room' manufacturing facilities needed to provide:

a. Precision machining and polishing.

b. High quality mirrors.

c. Very good optical seals.

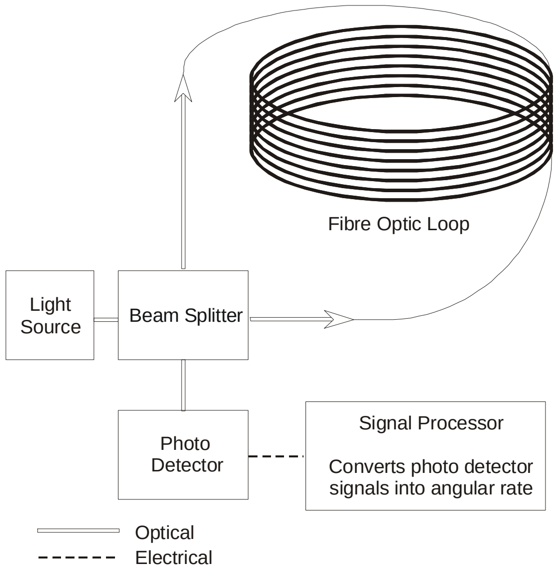
d. A carefully balanced mix of Helium and Neon, free of contaminants.

58. **Summary**. While the ring laser gyro represents a major advance over the traditional spinning gyro, it is only one of a number of possible alternatives. The search for new gyroscopic devices continues, driven by considerations of both cost and accuracy.

Fibre Optic Gyros

59. As previously outlined, the major disadvantage of ring laser gyros is their high cost due to the precise engineering facilities required to manufacture them. The fibre optic gyro (see Fig 21), first tested in 1975, works on the same principal as the ring laser gyro (the Sagnac effect) but no longer relies on a complex and costly block and mirror system since it uses a coil of fibre optic cable.

5-11 Fig 21 Fibre Optic Gyro

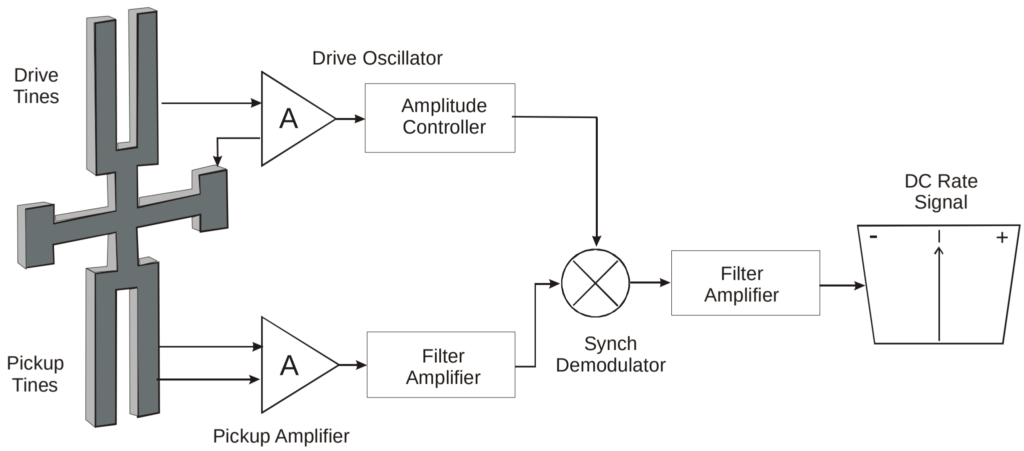


Vibrating Gyros

60. Vibrating gyros work by exploiting the Coriolis effect and, while not yet as accurate as optical gyros, are much smaller and cheaper to produce.

61. The 'GyroChip' (Fig 22) uses a vibrating quartz tuning fork as a Coriolis sensor, coupled to a similar fork as a pickup to produce the rate output signal. The piezoelectric drive tines are driven by an oscillator to vibrate at a precise amplitude, causing the tines to move toward and away from one another at a high frequency. This vibration causes the drive fork to become sensitive to angular rate about an axis parallel to its tines, defining the true input axis of the sensor.

5-11 Fig 22 The 'GyroChip' Vibrating Gyro



62. Vibration of the drive tines causes them to act like the arms of a spinning ice skater, where moving them in causes the skater’s spin rate to increase and moving them out causes a decrease in rate. An applied rotation rate causes a sine wave of torque to be produced, resulting from 'Coriolis Acceleration', in turn causing the tines of the Pickup Fork to move up and down (not toward and away from one another) out of the plane of the fork assembly.

63. The pickup tines thus respond to the oscillating torque by moving in and out of plane, causing electrical output signals to be produced by the Pickup Amplifier. These signals are amplified and converted into a DC signal proportional to rate by use of a synchronous switch (demodulator) which responds only to the desired rate signals. The DC output signal of the 'GyroChip' is directly proportional to input rate, reversing sign as the input rate reverses, since the oscillating torque produced by Coriolis reverses phase when the input rate reverses.

CHAPTER 12 - GYRO-MAGNETIC COMPASSES

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 the 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, i.e. 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

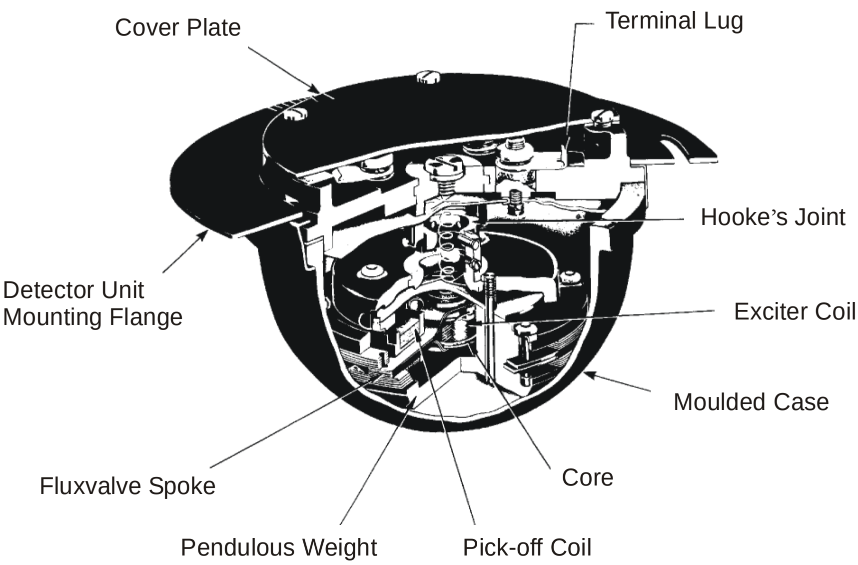
5. The gyro-magnetic compass consists essentially of a magnetic compass whose indications are stabilized gyroscopically so that the effects of turning and acceleration errors are reduced. A gyroscope is unaffected by changing magnetic fields or by normal aircraft accelerations but its heading indications may be inaccurate due to the effect of precessional forces caused by friction, incorrect balance etc. Since the commonly used detecting element, the fluxvalve, 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.

5-12 Fig 1 Detector Unit with Fluxvalve Element



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 can be attached to a digital counter, a moveable pointer against a fixed card or a moveable card against a fixed lubber line.

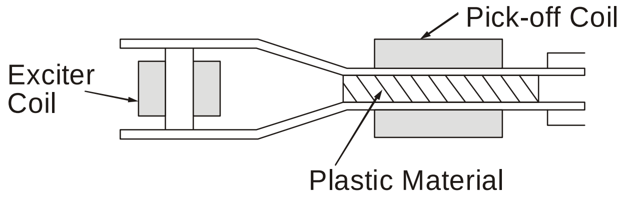
9. **The Gyroscope**. Short-term azimuth stability is typically provided by a two degree-of-freedom gyro with the input axis vertical, i.e. the spin axis in the local horizontal plane.

Fluxvalve Theory

10. The fluxvalve consists of a sensitive pendulous element mounted within the detector unit (see Fig 1). The fluxvalve is free to swing within limits (usually ± 25º) but is fixed to the aircraft in azimuth. The element is suspended by a Hooke's Joint (a common form of universal 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 120º 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 core. This core 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).

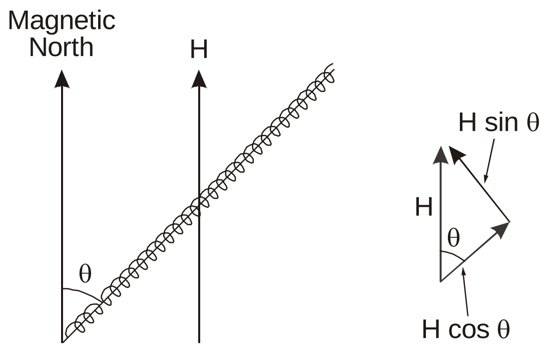
5-12 Fig 2 Vertical Cross-section of a Spoke



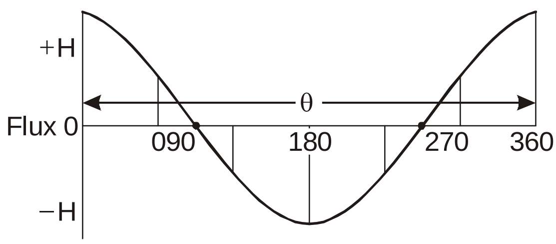
12. In order to appreciate the operation of the fluxvalve it is necessary to consider an individual spoke. The function of a spoke will be developed in a series of diagrams (Figs 3 to 10).

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 180º 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 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.

5-12 Fig 3 Magnetic Flux Components



5-12 Fig 4 Variation of Flux with θ

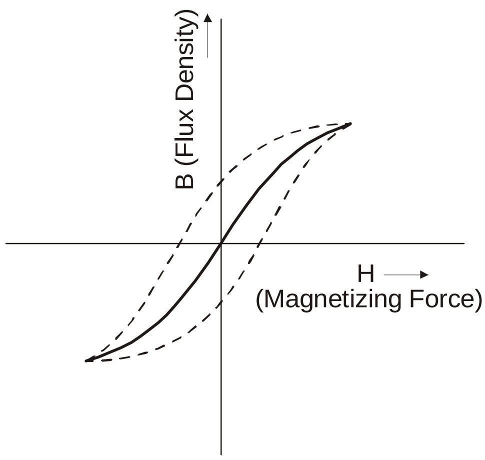


14. Unfortunately, the simple concept just described cannot be used without modification as a heading reference system for two important reasons. Firstly, the voltage induced into a coil depends on the rate of change of flux. Therefore, once established on a heading, there would be no change of flux and, consequently, no induced voltage. Secondly, the output of the simple detection device would be subject to heading ambiguity, i.e. 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.

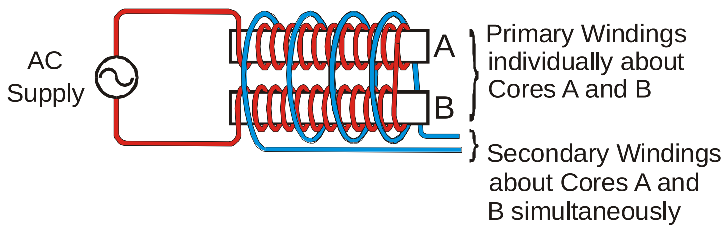
15. Fig 5 shows the relationship between flux density (B) and magnetizing force (H) known as the hysteresis loop for the permalloy commonly used in the legs of the flux valve spokes. Permalloy has a very high magnetic permeability (μ = B/H) and a corresponding low hysteresis loss. In the following discussion, the hysteresis loop is represented by a single line curve.

5-12 Fig 5 Hysteresis Curve for Permalloy



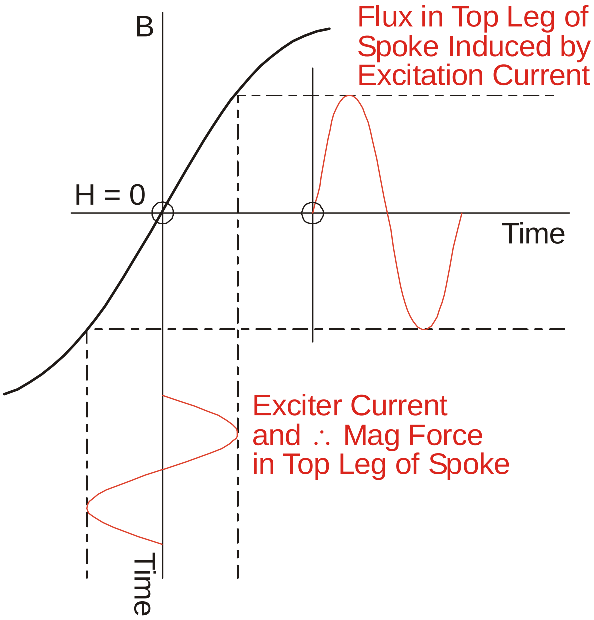
16. One spoke of the three-spoke fluxvalve is shown diagrammatically in Fig 6. It consists of a pair of soft iron (usually permalloy) cores each wound with a primary coil. The winding on one core is the reverse of that on the other. The AC supply is just sufficient, at peak power, to saturate magnetically 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.

5-12 Fig 6 Simple Fluxvalve



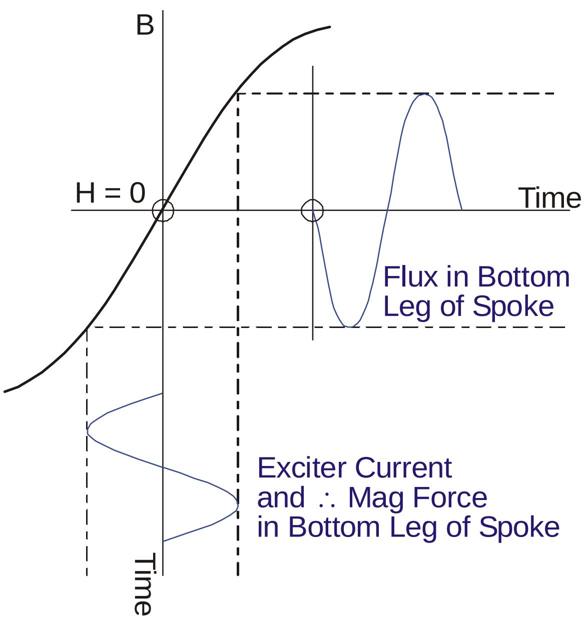
17. Fig 7 shows the 400 Hz alternating flux induced in the top leg by the excitation current considering only the top leg of the spoke and the effect of the excitation.

5-12 Fig 7 The Effect of Excitation Current in the Top Leg Only



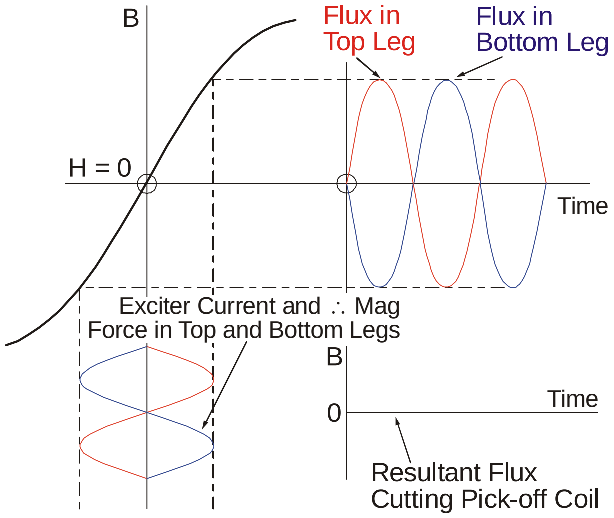
18. Now considering the bottom leg only; the flux induced in this leg by the excitation current will at any instant be in the opposite direction to that induced in the top leg, i.e. the flux in the bottom leg is 180º out of phase with the flux in the top leg as shown in Fig 8.

5-12 Fig 8 The Effect of the Excitation Current in the Bottom Leg Only



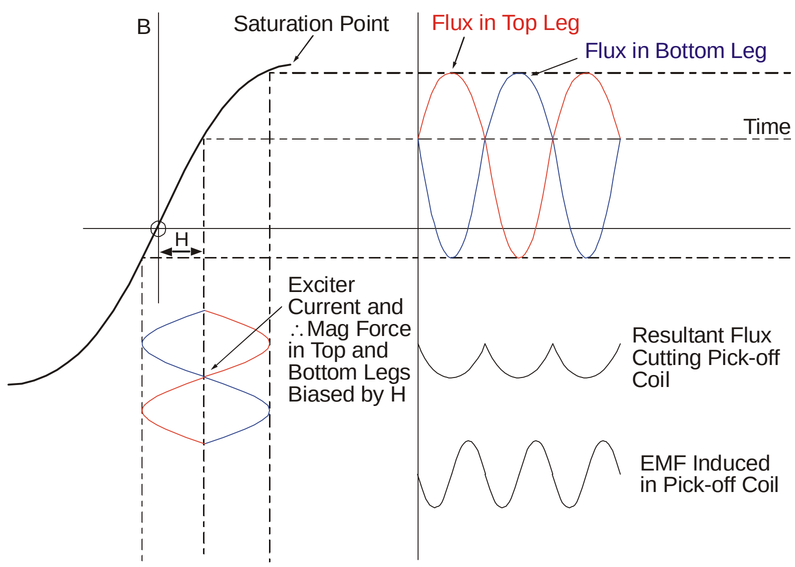
19. Since the top and bottom legs are identical, the amplitudes of the flux of the two legs are equal but 180º out of phase with each other relative to the pick-off coil, which is wound round both legs. Therefore, the resultant flux cutting the pick-off coil, which is the algebraic sum of the flux in the top and bottom legs is zero as shown in Fig 9.

5-12 Fig 9 The Effect of the Excitation Current in Both Legs



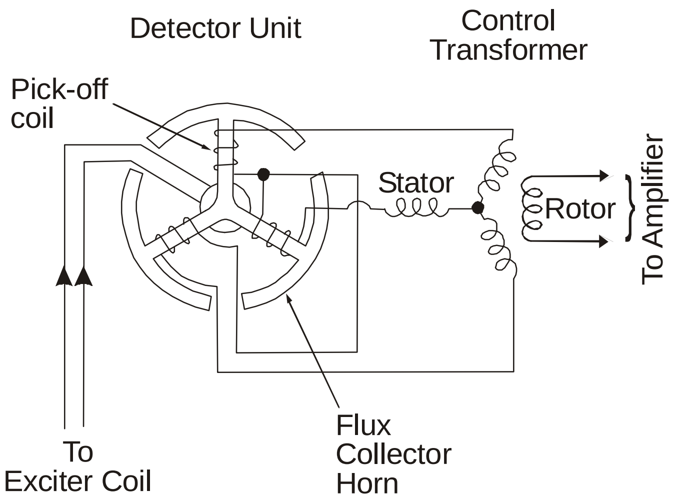
20. If the horizontal component of the Earth's magnetic field (H) is now added in line with the spoke, it will induce a steady flux in both legs of the spoke which will be added to the flux due to the excitation current. The effect, as shown in Fig 10, will be to bias the datum for the magnetizing force, 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, i.e. 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.

5-12 Fig 10 The Combined Effects of the Excitation Current and the Component of the Earth's Field

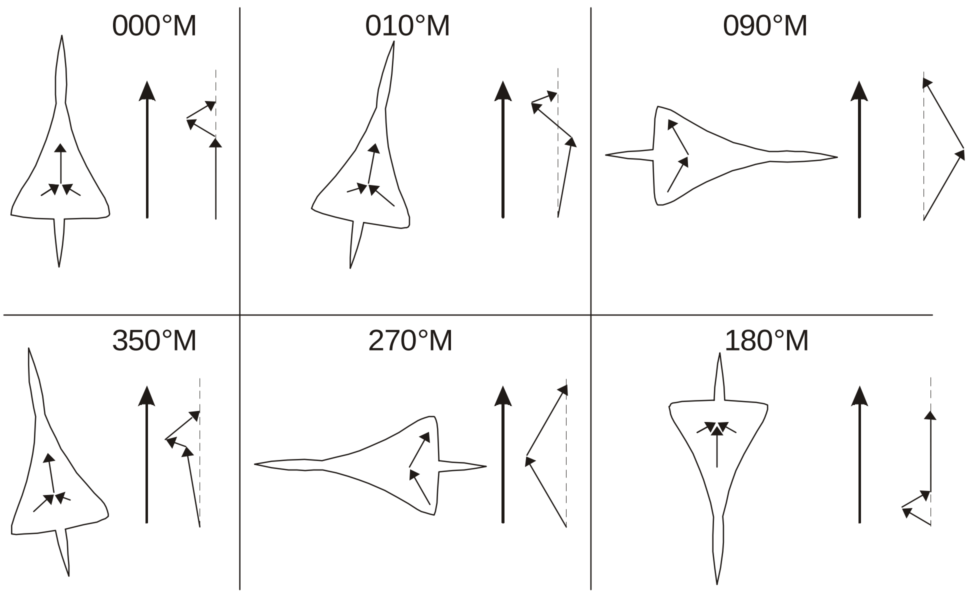


21. It should be apparent that there are two magnetic headings corresponding to zero flux (90º and 270º) and two headings corresponding to a maximum flux. The two maximum values give the same reading on an AC voltmeter since the instrument cannot take into account the direction of the voltage. For any 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 similar to the single spoked device previously discussed) with 120º 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 proportion to the change in H. The resultant field across the receiver stator is still aligned with H (see Fig 13).

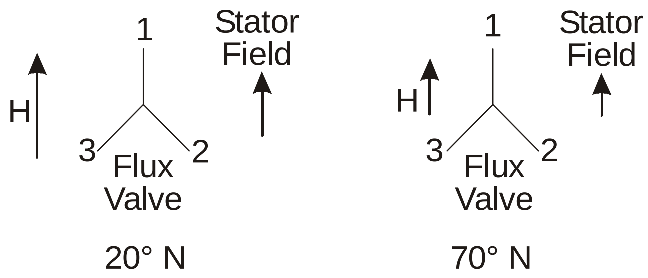
5-12 Fig 11 Detector Unit and Transmission System – Schematic



5-12 Fig 12 Operation of the Three-spoke Fluxvalve



5-12 Fig 13 Eliminating Latitude Ambiguity



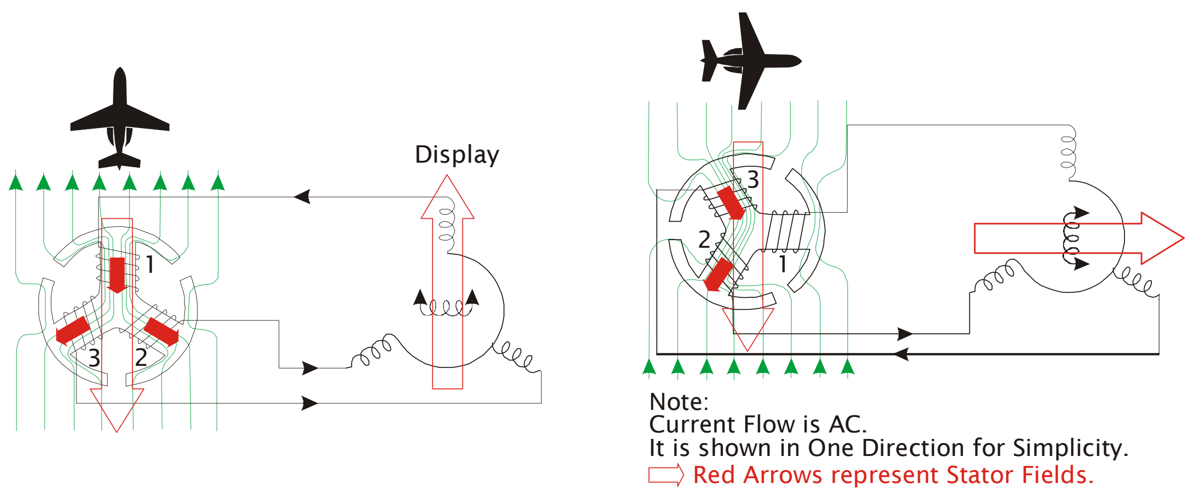
22. In the three-spoke fluxvalve, a single primary coil excites all six cores. If a single arm of the fluxvalve is considered, it will be apparent that the top and the bottom of the exciter coil have opposite polarity. The flux induced in the upper core of the spoke is equal and opposite to that induced 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 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 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.

5-12 Fig 14 Action of the Fluxvalve and Transmission System



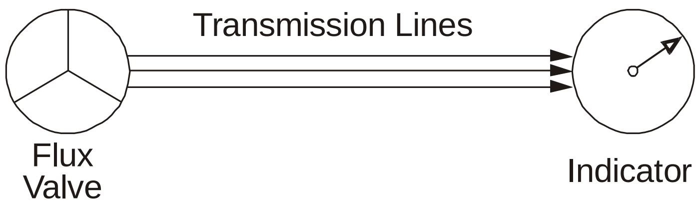
25. The outputs from the second and third fluxvalve spokes may be wired to the second and third receiver stator coils respectively or vice versa. The wiring will depend on whether it is necessary to drive a compass needle or a compass card. If the aircraft alters heading to starboard, the field across 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 anti-clockwise 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, i.e. 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.

5-12 Fig 15 Simple Remote Indicating Compass



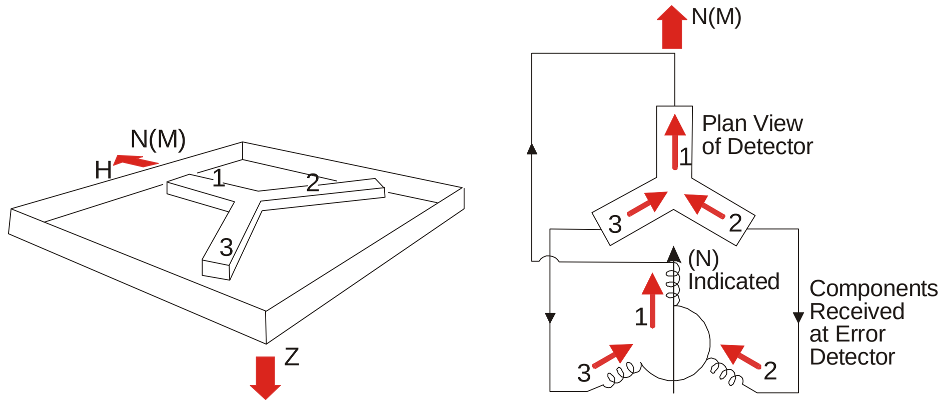
Detector Tilt Error

27. The fluxvalve will provide a correct output of magnetic heading only if the detecting element is maintained in the local horizontal plane, i.e. only detecting the horizontal component of the 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.

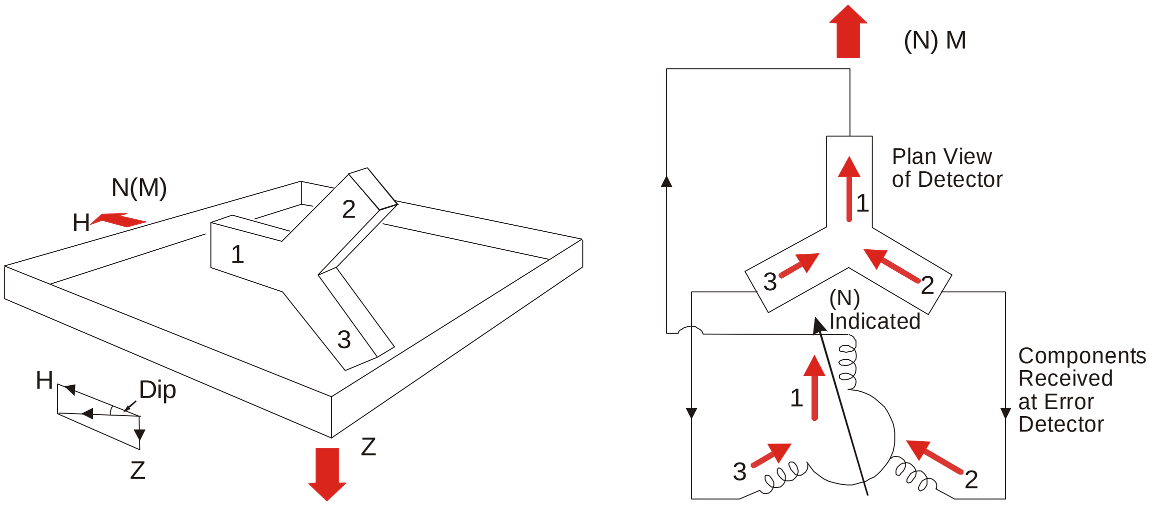
28. Fig 16 illustrates a fluxvalve fitted in an aircraft on a heading of magnetic North. The currents induced in spokes 1, 2 and 3 are such that they produce component magnetic fields in the error detector which compound to produce a resultant magnetic field in a direction indicating magnetic North. Only the horizontal component (H) threads the fluxvalve spokes to produce this result.

29. In Fig 17 the fluxvalve is tilted through 90º 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.

5-12 Fig 16 Indication of Magnetic North

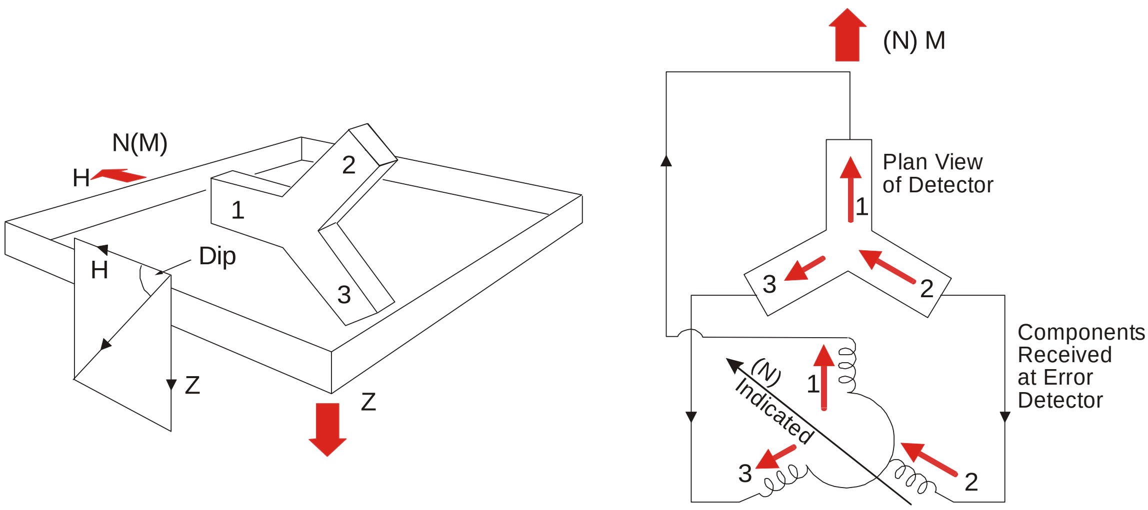


5-12 Fig 17 Effect of a Gross Tilt to Port



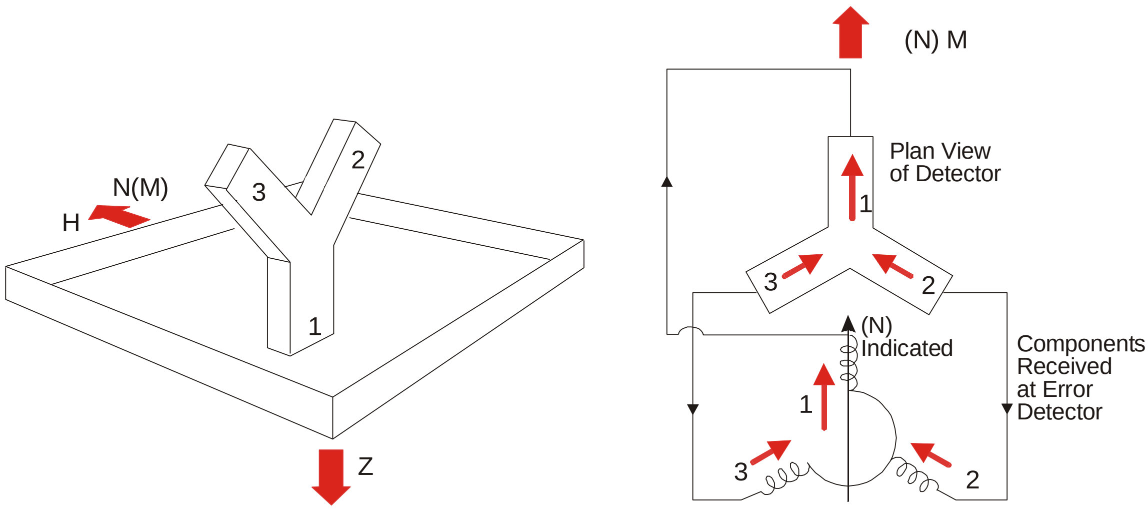
30. The error also depends on magnetic dip for, if the case at Fig 17 is repeated with a different dip, the components threading the spokes will alter. In Fig 18, the dip is increased, thereby increasing the error and reversing one component in this particular case.

5-12 Fig 18 Effect of Change of Dip

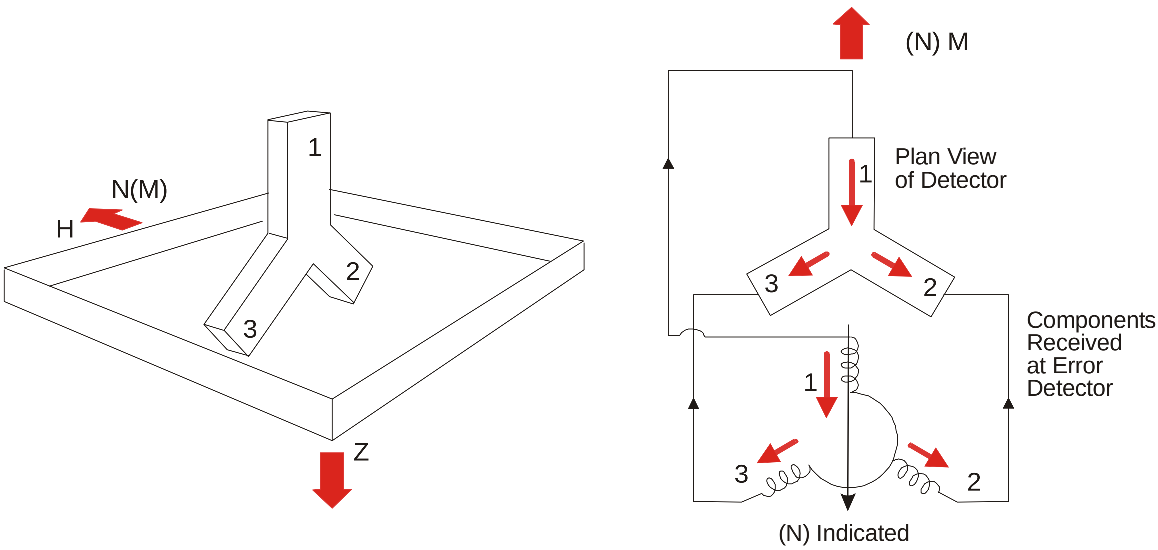


31. The direction of tilt relative to the total field is also important. Fig 19 shows how a tilt in the direction of the total field may produce no error. In this case, the flux flow through each spoke changes but the proportion of one to the other remains unchanged. The intensity of the resultant field 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 (90º – dip), the flux flow in each spoke is reversed and the error is 180º.

5-12 Fig 19 Effect of Direction of Tilt



5-12 Fig 20 Tilt Exceeds (90º – Dip)



32. Therefore, the error produced by tilting depends on the following factors:

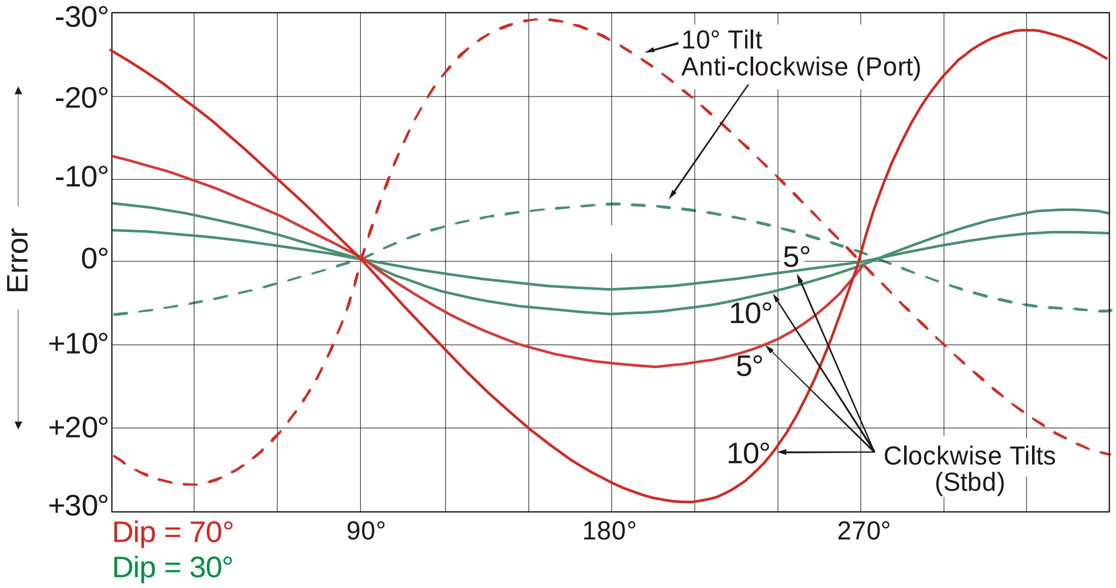
a. Angle of tilt

b. Direction of tilt.

c. Magnetic dip (δ)

Typical values of the error in the 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 (90º – δ) due to field reversal (see para 31).

5-12 Fig 21 Typical Errors in Magnetic Heading Due to Tilt



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.

These are discussed in paras 44-49.

THEORY OF THE GYRO-MAGNETIC COMPASS

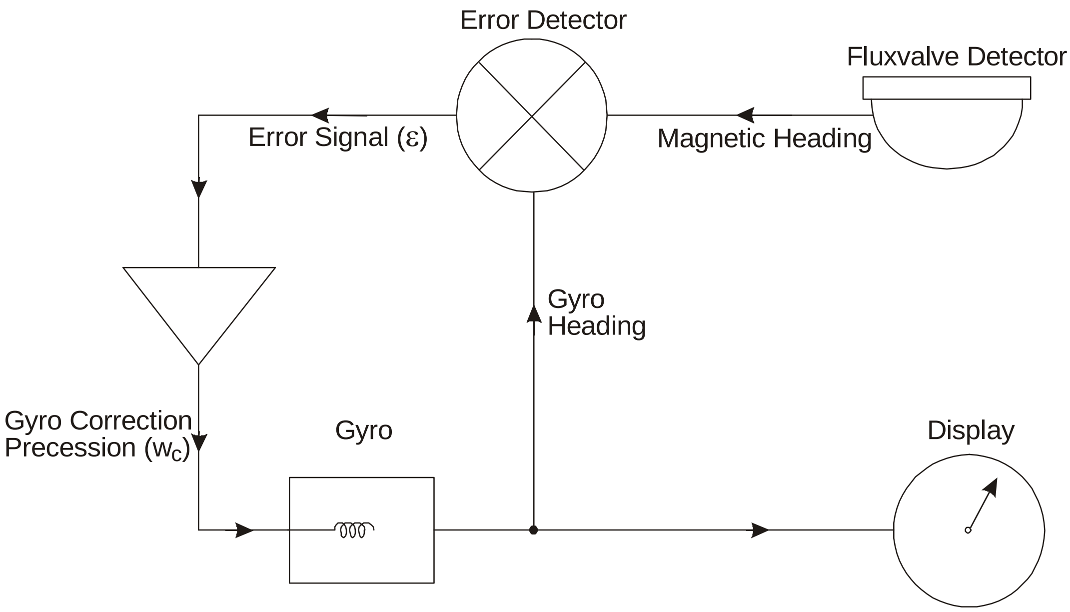
General

34. 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

35. 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.

5-12 Fig 22 Basic Gyro-magnetic Compass



36. 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.

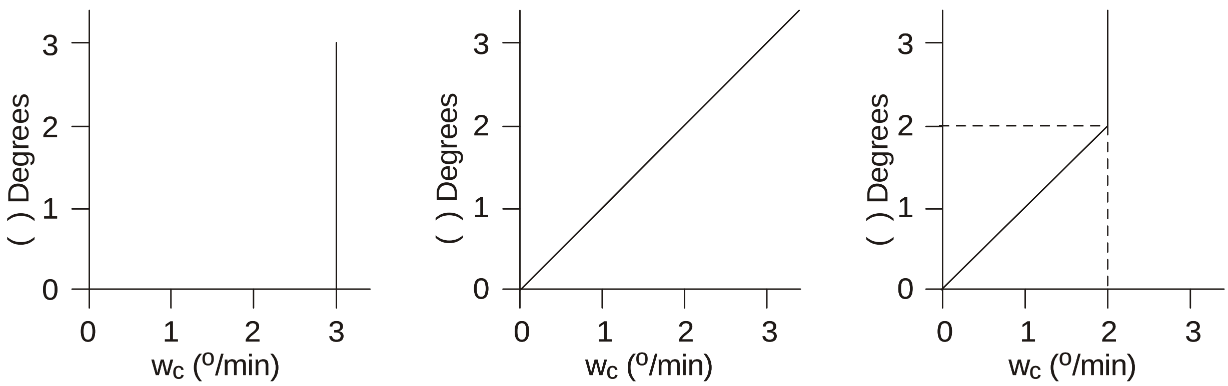
b. Linear function correction.

c. Limited linear function correction.

37. 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 also gyro behaviour suffers severely from nodding or nutation and secondary precession.

5-12 Fig 23 Gyro Correction Techniques

Fig 23a Step Function | **Fig 23b Linear Function** | **Fig 23c Limited Linear Function**



38. The linear correction technique (Fig 23b) appears to be ideal since the correction rate (W_c) is proportional to the error signal (ε), i.e. 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, 180º can exist between gyro and magnetic heading. If the system was mechanized to provide an adequate rate of precession for small errors, 180º would demand an excessive precession rate. Therefore, the purely linear system also has its limitations.

39. 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. For example, in Fig 23c, the gyro precession rate (W_C) is proportional to ε, where ε is $\leq 2^\circ$, however, W_C cannot exceed 2º per min regardless of the size of ε.

40. **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 of $W_C K = \varepsilon$ (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.

$$\varepsilon = W_C\,\tau$$

Therefore if $\varepsilon = 2^\circ$ and $\tau = 0.5$ minutes, the rate of precession (W_C) is given by:

$$W_C = \frac{\varepsilon}{\tau} = \frac{2^\circ}{0.5\,\text{min}} = 4^\circ\ \text{per min}$$

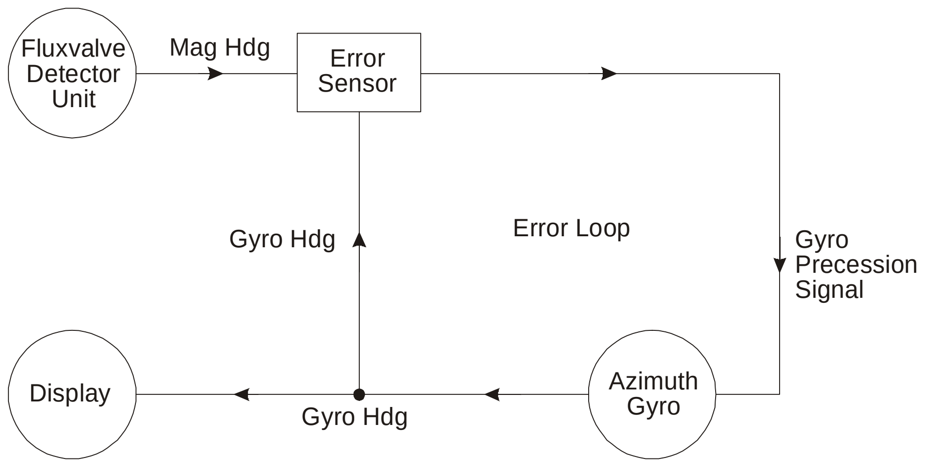
Obviously the larger the time constant, the slower is the rate of precession. Notice that τ does not express explicitly the time to correct a given error since the rate of correction reduces as the error reduces so it takes much longer than τ minutes to correct the error. Since the error reduces exponentially, τ directly gives the time it takes to remove 63% of the error. It would require approximately 5τ to remove all the error in a step error function. Therefore, for an initial error of 2º and a τ of 2 minutes, the error will reduce exponentially until at the end of 5τ (10 mins) the error is effectively reduced to zero.

41. **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.

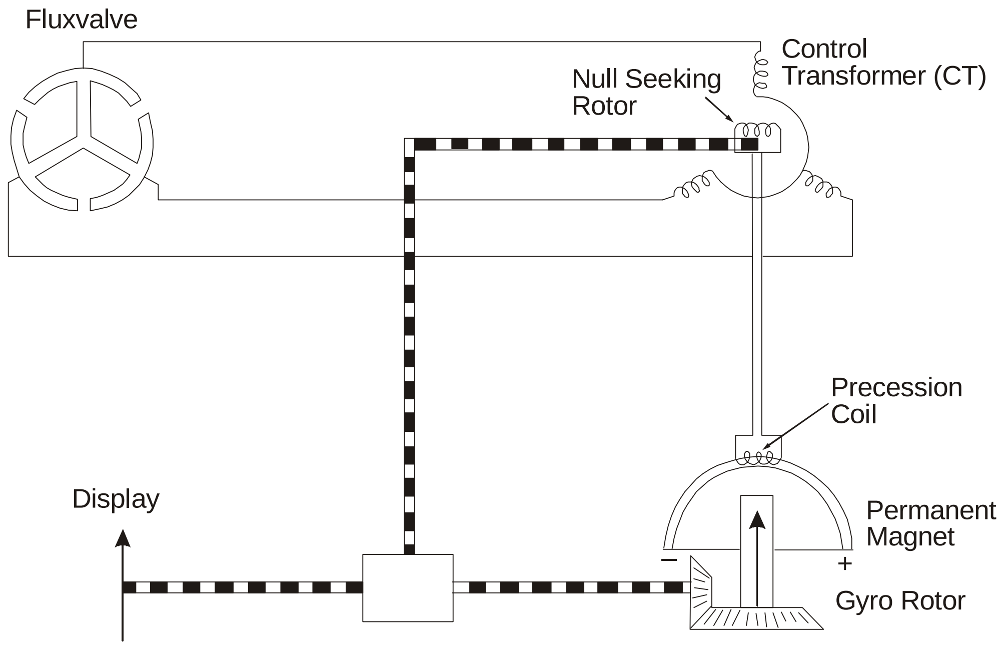
42. **Typical Gyro Slaving Mechanization**. The implementation of a typical limited linear control is illustrated in the block diagram at Fig 24 and the schematic at Fig 25, the currents induced in the spokes of the fluxvalve are passed to a receiver synchro (CT) and produce a field across the rotor 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 90º 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.

5-12 Fig 24 Gyro-magnetic Compass Block Diagram



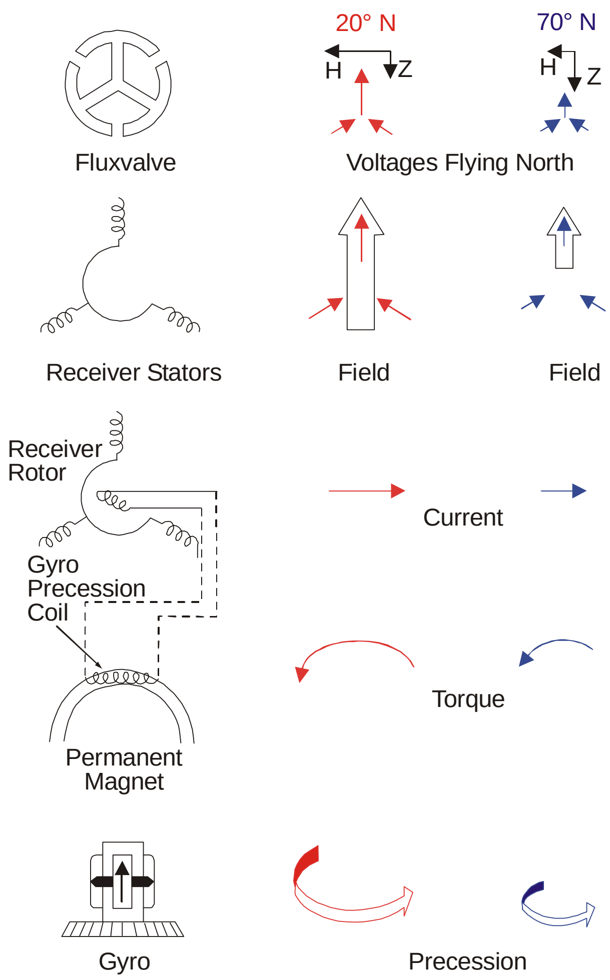
5-12 Fig 25 Typical Gyro Slaving Mechanization (Simplified Schematic)



43. **The Change in τ with H**. Fig 26 illustrates the relationship between H field strength and gyro precession rate in a typical 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 is 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 50). 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 is the common datum quoted by British gyro-magnetic compass system manufacturers.

5-12 Fig 26 Effect of a Change in H on the Time Constant



GYRO-MAGNETIC COMPASS SYSTEM ERRORS

Fluxvalve Tilt Errors

44. All of the horizontal accelerations which cause fluxvalve tilt can cause heading errors in a simple uncompensated gyro-magnetic compass system. Accelerations are caused by coriolis, vehicle movement (rhumb line), aircraft turns, linear changes of velocity and fluxvalve vibrations. 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.

45. **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, little of the error is displayed since the time spent in the turn is minimal. Slow prolonged turns at high speeds generate the greatest errors. The errors decay after level flight is resumed. Fluxvalve induced errors due to tilt can be limited by switching the system to an unslaved directional gyro mode whenever turns are sensed by suitable detection devices.

46. **Coriolis Error**. An aircraft flying relative to a spherical rotating Earth flies a curved path in space and, in consequence, there will be a central force acting to displace the pendulously suspended fluxvalve. When established on a given heading for approximately 5τ the entire error would be included in the gyro-magnetic compass heading display. The error is calculable, depending on groundspeed, latitude, dip and track, and can be compensated automatically.

47. **Vehicle Movement Error**. Whenever flying a true or magnetic rhumb line the aircraft must turn to maintain a constant track with reference to converging meridians. As with coriolis error, the acceleration displaces the detector from the local horizontal plane and the entire resultant heading error would appear in the displayed heading after about 5τ. A correction can be applied in a similar manner to the coriolis error.

48. **Fluxvalve Vibration**. Fluxvalve vibration results in a heading oscillation, the mean of which is not the actual mean heading. 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. The effect can be limited to small values by careful design of the pendulous detector damping mechanism and through consideration of the location of the detector in the aircraft.

Northerly Instability

49. 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 looking at this is to imagine that the magnetic meridian rotates clockwise. Thus, if an aircraft on North banks to starboard to correct a 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 regain a northerly-indicated heading. This tilts the fluxvalve which rotates the meridian to port. The new false meridian is chased until, upon resuming level flight, the sensor detects the true meridian again and precesses the gyro to starboard. The cyclic pattern is repeated and the amplitude can be as great as 6º. The amplitude of the weave tends to increase with an increase in dip and aircraft velocity, and decreases with an increase in τ. Weaving can thus be reduced to a certain extent by increasing the time constant of the compass system. However, this leads to a sluggish response and a large hang-off error (para 50).

Hang-off Error

50. Gyroscopic drift is a constant source of error signal in a gyro-magnetic compass system, and although it will be compensated for by the precession loop at a rate dependent on τ, at any given time there must be an increment of error present. This is known variously as hang-off error, stand-off error, or simply as velocity lag. Gyro drift may be due to:

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 sin lat º/hr. The correction can be supplied through a manually set latitude correction mechanism, automatically from a computer-generated latitude, or through a constantly biased gyro. The latter technique employs a mass imbalance in the gyro which constantly precesses the gyro at a predetermined rate, usually to compensate for an appropriate latitude for the aircraft's area of operation, say 51 ºN.

c. **Transport Wander**. To compensate for transport wander due to the convergence of geographic meridians the gyro must be corrected at a rate equal to:

$$\frac{U}{60} \tan \text{lat} \ ^{\circ}/\text{hr where U = East-West groundspeed}$$

The correction can be applied manually or through a computer using inputs of groundspeed, heading, and latitude. However, although the gyro can be compensated in this way for the apparent change in the direction of geographic North, the output from the fluxvalve is in terms of magnetic North. Therefore, as the aircraft moves over the Earth, there will be a difference between fluxvalve and gyro since the variation is changing (unless the aircraft is flying along an isogonal). To remove this error variation must be applied to the output of the detector unit before the gyro error loop so that both the gyro and fluxvalve give directional information relative to true North. The value of variation can be inserted manually or by means of an automatic variation setting control unit. Failure to update the variation value will result in small hang-off errors.

Gimbal Error

51. When a 2 degree of freedom gyroscope with a horizontal spin axis is both banked and rolled, the outer gimbal must rotate to maintain orientation of the rotor axis, thereby inducing a heading error at the outer gimbal pick-off. The incidence of this error depends upon the angle of bank and the angular difference between the spin axis and the longitudinal axis and as in most systems the spin axis direction is arbitrary relative to North the error is not easily predicted.

Transmission Errors

52. Overall system accuracy is lowered by the errors in the synchro systems. Typically, each synchro might be expected to a have an error in the order of 0.1º with an overall system error of perhaps 0.5º. This shows in a compass swing as a D or E error.

Compass Swinging Errors

53. It is not possible to obtain absolute accuracy in compass swinging, and even refined methods are considered to be only accurate to 0.2º.

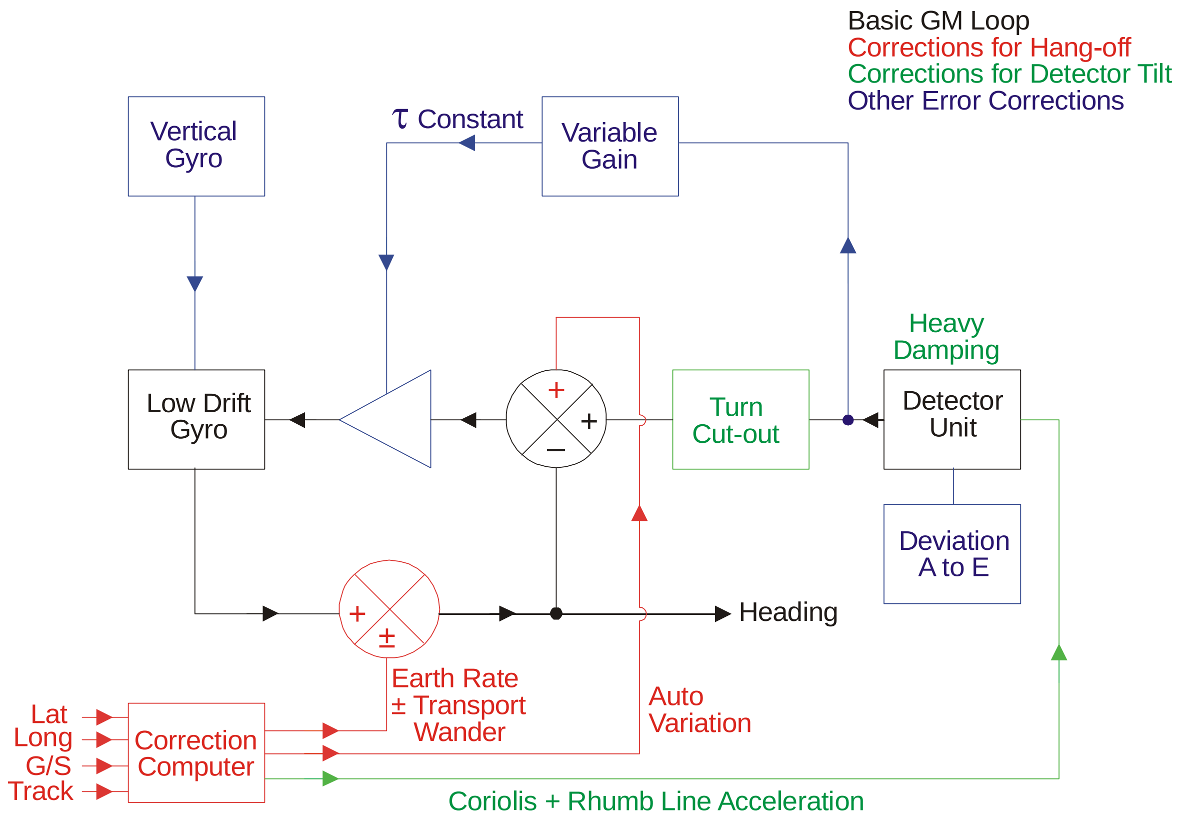
Variation and Deviation Errors

54. There are no reliable statistical data on the errors in charted values of variation, but they might be considered to vary between 0.1º and 2º. Over the UK, the uncertainty at height is considered to be within 1º but the value varies both with height and locality. Setting of variation and deviation is likely to be accurate to 0.25º.

A Refined Compass System

55. Fig 27 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 disadvantages to all approaches.

5-12 Fig 27 Ideal Gyro-magnetic Compass



56. The following description applies to Fig 27:

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 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. **Northerly 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.

CHAPTER 13 - HORIZONTAL SITUATION INDICATORS

Introduction

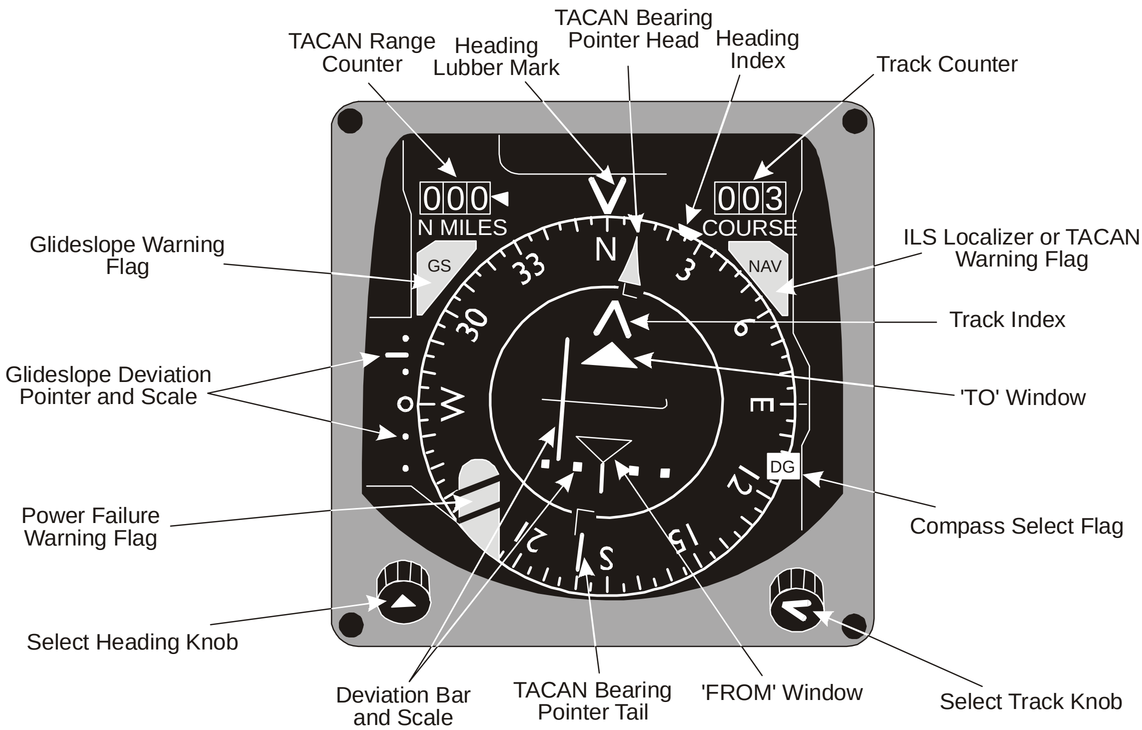
1. The Horizontal Situation Indicator (HSI) is an instrument for displaying both the compass system and the radio navigation aids in an aircraft (usually TACAN and VOR/ILS). An electronic version, employing a coloured liquid crystal display, functions in a similar manner and is able to handle more services.

CONVENTIONAL DISPLAYS

Display and Features

2. Although installations will vary slightly between aircraft types, a typical conventional display is illustrated in Fig 1 and the features are described below:

5-13 Fig 1 Horizontal Situation Indicator



a. **Heading**. Heading is indicated at the top of the display by a rotating compass card moving against a fixed 'V' lubber mark. The card is graduated at 5º intervals and is marked alphanumerically at 30º intervals with the numerical annotations being in tens of degrees.

b. **Heading Index**. A heading index registers against the outside edge of, and rotates with, the compass card. The index can be manually set relative to the compass card by a select heading knob which is marked with a symbol representing the heading index and is located at the lower left-hand corner of the instrument.

c. **Compass Select Flag**. When the compass system is set to the directional gyro mode the compass select flag appears with the letters DG displayed.

d. **Track Index and Counter**. A track index, which is on the centre display assembly, registers against the inside edge of, and rotates with, the compass card. The index can be manually set relative to the compass card by a selector knob at the lower right-hand corner of the instrument. The reciprocal of the track set is indicated by a track index tail on the centre display assembly. A

3-digit display of the selection is given on a track (COURSE) counter at the top right of the display. The selector knob is marked with a symbol representing the track index.

e. **Deviation Bar**. A deviation bar and a fixed scale of two dots either side of a centre index are on the centre display assembly. The bar moves left or right of the centre index to indicate deviation from the selected track when TACAN or ILS information is selected. The bar indicates the relative position of the chosen track as selected by the track index.

f. **TACAN Bearing**. The magnetic bearing to a TACAN ground beacon is indicated by a green pointer head when read against the compass card. The tail of the pointer indicates the TACAN radial. The TACAN bearing and radial are also displayed when ILS is selected.

g. **To/From Indication**. Two triangular indicator windows, 'to' and 'from' are on the centre display assembly. The 'to' window is adjacent to the track index and the 'from' window is adjacent to the tail of the track index. With TACAN selected, a TACAN radial set on the track index and the bearing pointer locked on to a TACAN beacon, a white flag is displayed in the 'to' or 'from' window. The 'to' flag is displayed whenever the bearing from the TACAN is less than 90º from the selected TACAN radial. Conversely the 'from' flag shows white whenever the bearing from the TACAN beacon is 90º or more from the selected TACAN radial.

h. **TACAN Range**. Range to a TACAN or DME beacon in nautical miles is shown on a 3-digit counter at the upper left corner of the instrument. A yellow bar obscures the counter when range information is invalid.

i. **Glidepath Deviation Pointer**. A pointer to the left of the compass card moves over a fixed vertical scale consisting of two dots above and two dots below a circle (representing the aircraft). The pointer is driven by the ILS equipment and indicates the vertical position of the ILS glidepath relative to the aircraft, eg if the pointer is above the circle the aircraft is below the glidepath.

j. **Glidepath Warning**. A red flag appears above the glidepath deviation scale when the glidepath information is invalid.

k. **ILS Localizer or TACAN Bearing Warning**. A red flag appears below the COURSE counter when the ILS localizer or the TACAN bearing information is invalid.

l. **Power Failure Warning**. An orange flag with black diagonal stripes appears when the power to the HSI has failed or when an invalid signal is transmitted from the compass system.

ELECTRONIC HSI (EHSI)

Description

3. The EHSI can be configured to provide more information than the HSI and employs a colour active matrix liquid crystal display. It receives inputs from the aircraft compass, TACAN, VOR/ILS and, depending on aircraft fit, UHF and VHF homers and specialist navigational aids. In helicopters, the EHSI is also linked to the hovermeter.

Mode Select Panel

4. A mode select panel will be available to each pilot position with buttons for selection of each available feature. A Transfer Mode (TM) switch enables the course selector display of one instrument to be transferred to the other. Therefore, for example, the TACAN radial can be set on one instrument and the ILS QDM on the other. Pushing the TM switch associated with the instrument configured for ILS will transfer the ILS information to the other instrument.

Displays

5. In different installations, display colours may vary depending upon which mode is selected. Fig 2 shows a typical instrument in VOR mode. When first switched on or after a power break the EHSI will have no mode indicated in the bottom right hand corner. In most cases, the numbered items described in the key are displayed only when the appropriate inputs are valid.

6. Displays and Controls. Operation of the controls annotated on Fig 2 may depend upon the mode selected. A brief description of each is given below. The relevant Aircrew Manual will set out the precise operation of the system.

a. **Heading Select Knob**. When enabled by the Heading Select Pushbutton, rotating the Heading Select Knob sets the heading bug. There is a slight ratchet effect to give positive feel. The knob is normally disabled 5 seconds after its last rotation.

b. **Heading Select Pushbutton***.* A positive press of the Heading Select Pushbutton of at least 0.1 sec is required to enable the Heading Select Knob.

c. **Track Select Knob***.* When enabled by the Track Select Pushbutton, rotation of the Track Select Knob allows the track pointer to be set to the required track. Positive feel is given by a slight ratchet effect, one click of the ratchet equating to 1-degree change in selected track. The associated track digital readout (10) follows the pointer setting. The knob is automatically disabled 5 sec after its last rotation.

d. **Track Select Pushbutton***.* A positive press of the Track Select Pushbutton of at least 0.1 sec is required to enable the Heading Select Knob.

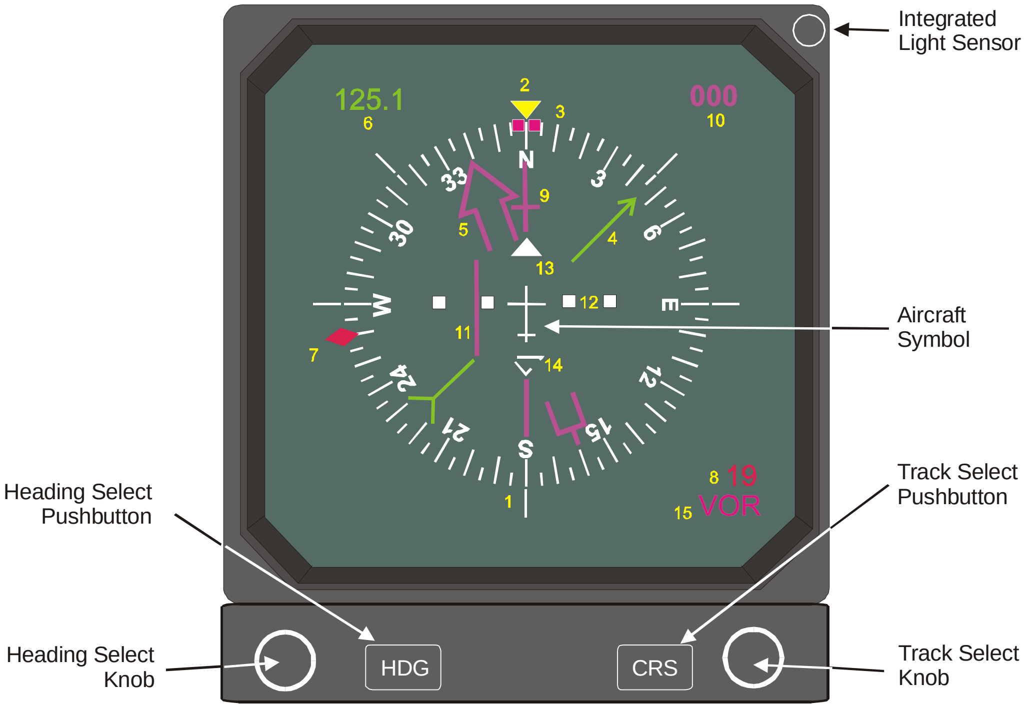
e. **Aircraft Symbol***.* The aircraft symbol is always aligned pointing towards the heading lubber line at the top of the instrument. When a navigation mode is active, the symbol represents aircraft orientation against the steering pointer or deviation bar.

f. **Integrated Light Sensor**. The sensor automatically adjusts the display brightness in daylight. A separate manually operated dimmer sets the brightness for night operations.

g. **Mode Displays**. The key to Fig 2 describes the numbered indicators shown on the diagram for VOR/ILS mode. In other modes the names, functions and colours of the displayed features may change from those depicted.

h. **Functionality**. Most coloured symbols are cleared when the service is not activated or the compass input fails. The appropriate Aircrew Manual should be consulted for precise details.

5-13 Fig 2 EHSI - VOR Mode Selected



1. **Compass Card**. The compass card indicates gyrocompass heading in conjunction with the lubber line (2). The card rotates clockwise as the aircraft turns left. If the compass fails, the card freezes and a red HDG FAIL caption is superimposed.

2. **Lubber Line**. The lubber line is the index against which heading is shown on the compass card.

3. **Heading Bug**. The red heading bug is set by the Heading Select Knob to indicate the required heading. The bug clears if the compass input fails.

4. **Single Bar Pointer (TACAN Bearing)**. The arrowhead on the green single bar pointer indicates the bearing of the TACAN station locked on. The pointer clears if a TACAN station is not locked on or if the compass input fails.

5. **Double Bar Pointer (VOR Bearing)**. The arrowhead on the purple double bar pointer indicates the bearing of the VOR station locked on. The pointer clears if the VOR receiver is not locked to a station or if the compass input fails.

6. **TACAN/DME Range Readout**. The TACAN/DME readout is a digital display which shows the slant range to a locked on TACAN or DME station. When not locked on, the display shows 4 dashes.

7. **Wind Direction Indicator**. The direction from which the wind is blowing is shown by a red diamond. The diamond clears when there is no source data or if the compass input fails.

8. **Wind Speed Readout**. Wind speed to the nearest knot is shown by a red digital display. The display clears when there is no source data or if the compass input fails.

9. **Track (Course) Pointer**. The cross end of the purple Track Pointer indicates the track selected.

10. **Selected Track (Course) Readout**. The readout shows the track selected on the Track Pointer (9).

11. **Track (Course) Deviation Bar**. The purple deviation bar shows track deviation left or right of that selected on the Track Pointer (9). To return to track, the aircraft should be turned towards the bar until the bar centralizes and then on to a new heading to keep the bar in the centre.

12. **Track (Course) Deviation Scale**. The scale comprises two white dots to the left and right of the centre of the Track Pointer (9) creating a scale over which the Track Deviation Bar (11) moves. The outside dots represent full-scale deflection (±10°), the intermediate dots indication ±5°.

13. **'To' Flag**. The 'To' Flag is a white arrowhead which is displayed until the aircraft passes over or abeam the locked station, after which it is replaced by the 'From' Flag.

14. **'From' Flag**. The 'From' Flag is a white dotted arrowhead which is displayed once the aircraft has passed over or abeam the locked station.

15. **Mode Annunciator**. The selected mode (VOR in the example) is displayed provided the service is on and functioning.

CHAPTER 14 - DATUM COMPASSES

Introduction

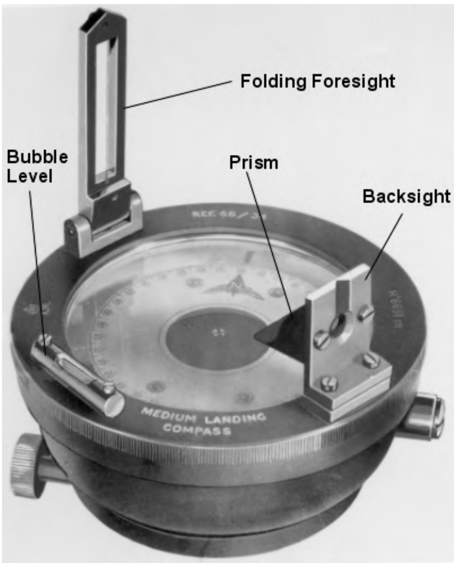
1. In order to calibrate an aircraft compass system, it is necessary to have an accurate heading datum. A Medium Landing Compass provides a sufficiently accurate datum for simple aircraft compasses. However, where the aircraft compass output is used as an input to a navigation system, the heading datum is provided by the more accurate Watts Datum Compass. In order to overcome the accuracy limitations of the aircraft compass display, the aircraft compass heading is read with the aid of a Precise Heading Test Set (PHTS). This permits the aircraft heading to be recorded to an accuracy of 0.05º. This chapter will describe the Medium Landing Compass, the Watts Datum Compass Mk1A and the PHTS.

THE MEDIUM LANDING COMPASS

Description

2. The Medium Landing Compass (Fig 1) is designed for use on a tripod and is fitted with a bubble level. The horizontal compass card is read through a prism mounted on the back sight which, along with the folding foresight, forms the sighting mechanism. The foresight has a fine wire mounted vertically in the centre of the frame to enable accurate sightings to be taken. The back sight, foresight and bubble level are all mounted on the rotating sighting ring. The instrument consists of a fluid-filled metal bowl which houses the float on which the two parallel bar magnets are mounted. The float is supported on a jewelled-bearing pivot. The accuracy of the instrument is quoted as 0.5º, although errors of up to 1º may occur due to pivot friction. The Medium Landing Compass, although relatively simple, is still a delicate instrument and must always be kept in its box when not in use.

5-14 Fig 1 Medium Landing Compass



Operation

3. The procedure for taking a bearing using the Medium Landing Compass is as follows:

a. Extend the tripod legs and firmly attach the compass to the tripod head.

b. Set up the tripod on the extended datum line of the aircraft, with one leg of the tripod pointing away from the operator.

c. Level the compass. This is done in three steps:

(1) Turn the sighting ring until the bubble level is parallel to the aircraft's fore-and-aft line. Level the tripod by moving the forward leg of the tripod until the bubble is central in the level.

(2) Move the sighting ring through 90º and level by moving the rear legs of the tripod to centralize the bubble.

(3) Realign the bubble level with the fore-and-aft axis and check the level.

d. Take the sighting by looking through the slot and prism. Adjust the sighting ring until the object (the aircraft sighting rods), the foresight wire and the back sight slot are in line.

e. Read the compass card scale through the prism where the foresight wire appears to touch the compass card.

THE WATTS DATUM COMPASS MK1A

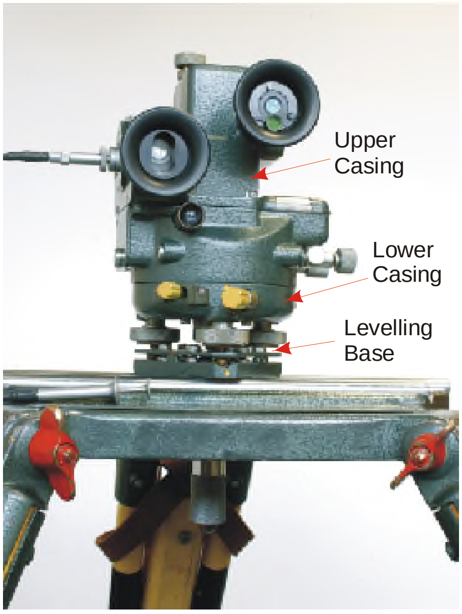
Principle

4. The Watts Datum Compass consists of a compass system, an azimuth circle (or bearing plate), and a sighting telescope. The compass system accurately defines the magnetic meridian and the azimuth circle is aligned with, and locked to, this meridian. The telescope is sighted on the aircraft, along the datum line, and the bearing of the line of sight is read off from the azimuth circle through a microscope.

Description

5. The three parts of the instrument, the compass, azimuth circle, and telescope, are enclosed within an aluminium body, with the necessary controls mounted externally. A three-screw levelling base supports the body and provides the tripod mounting point for the instrument. A general view of the Watts Datum Compass, on its tripod, is shown at Fig 2.

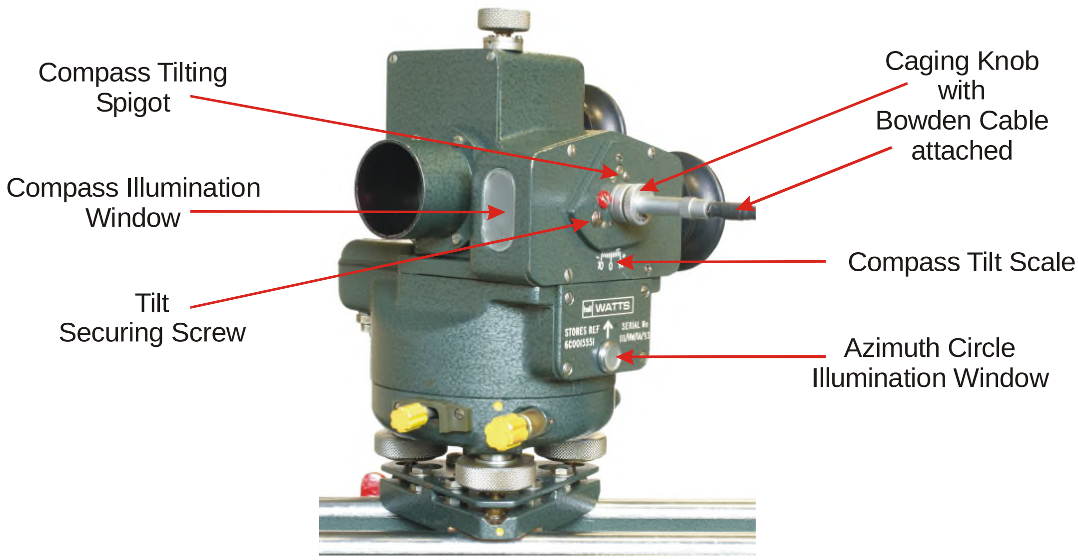
5-14 Fig 2 General View of the Watts Datum Compass on its Tripod



6. **Accuracy**. When all systematic errors have been eliminated, the Watts Datum Compass can be aligned to the magnetic meridian to an accuracy of ± 0.02º. This accuracy will deteriorate in windy conditions since a surface wind speed in excess of about 15kt will cause vibration of the uncaged compass system. The accuracy of aircraft compass deviation measurements using the Watts Datum Compass depends not only on the accuracy of the datum instrument, but also upon the precision of the instrument alignment with the aircraft's datum points (see Volume 5, Chapter 16), and the accuracy with which the aircraft compass can be read. This last factor is independent of the datum equipment and is likely to cause the largest error.

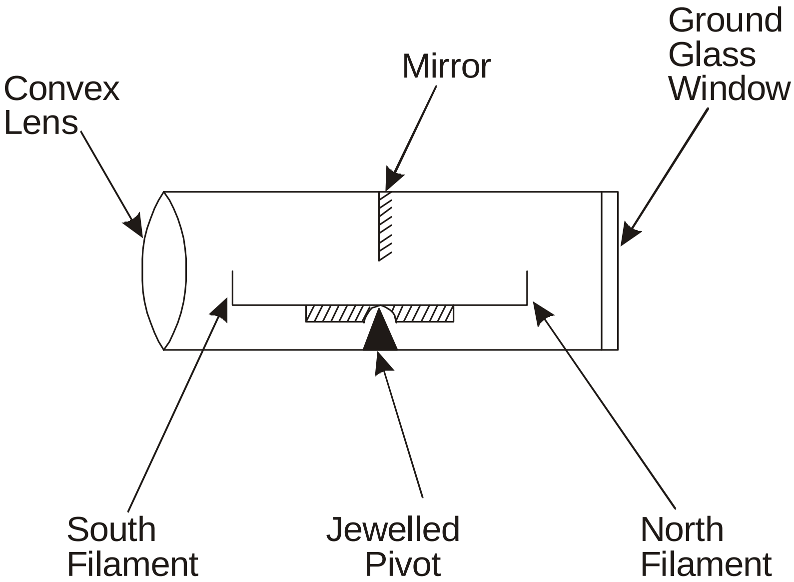
7. **Compass**. The compass consists of a magnet, fitted with a jewelled bearing, in a containing box. A leaf spring normally keeps the magnet lifted off its pivot in the safe or 'caged' position. The compass can be uncaged either by pressing the caging knob, or by operating a Bowden cable release, which can be screwed into the centre of the knob (see Fig 3). In either case, the pressure operates a lever, which depresses the leaf spring and lowers the magnet on to its pivot. A safety lock on the caging knob, which prevents it from being pressed in, is engaged by turning the knob anti-clockwise. The lock does not prevent the use of the Bowden cable release.

5-14 Fig 3 Watts Datum Instrument – Compass-related Features



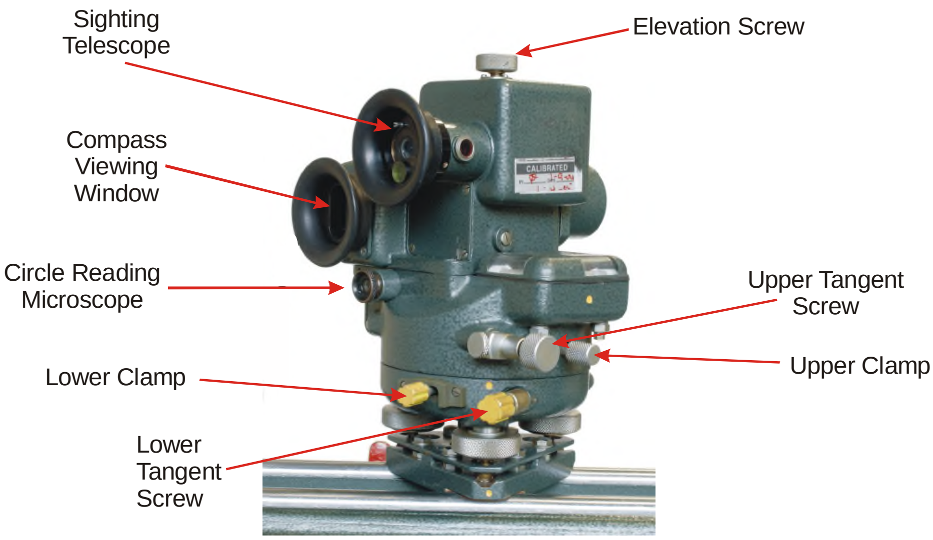
8. **The Compass Box**. The compass box (Fig 4) is closed at its North end by a ground glass window and at its South end by a convex lens. A mirror above the magnet pivot faces the lens and can, for collimation purposes, be moved about its vertical axis by a small adjustment screw. The compass box is mounted on a horizontal spigot so that it may be tilted to allow for dip. It can be tilted up to 10º either side of the horizontal and locked in position by two screws (see Fig 3).

5-14 Fig 4 Schematic Diagram of Compass Box (side view)



9. **Compass Viewing Window**. The convex lens in the compass viewing window (see Fig 5) is focused on the North filament and so the South filament, being out of focus, is not directly visible to the observer; only its image in the mirror is apparent. The compass is aligned with the magnetic meridian when, with the magnet on its pivot (uncaged), the North filament and the image of the South filament reflected in the mirror form one continuous vertical line seen through the convex lens (see Fig 6).

5-14 Fig 5 Clamps and Sighting/Reading Windows



5-14 Fig 6 View through Compass Viewing Window

Fig 6a Approaching Alignment

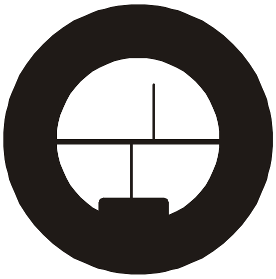
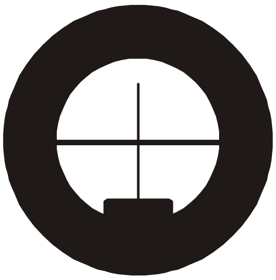
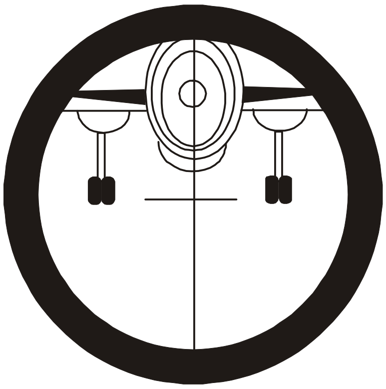


Fig 6b Perfectly Aligned



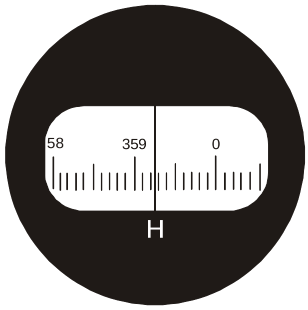
10. **Sighting Telescope**. A fixed-focus prismatic telescope is used to define the line of sight (see Fig 5). A mirror within the optical system can be tilted by the elevation screw on top of the casing and will allow the line of sight to be varied in the vertical plane. A sighting graticule, consisting 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. The telescope gives an erect image with six times magnification and a field of view of 8º. A soft rubber eyepiece is provided for comfort. An example of the view through the sighting telescope is shown at Fig 7.

5-14 Fig 7 Example View through the Sighting Telescope



11. **Azimuth Circle**. The azimuth circle is made of glass and is graduated at intervals of 0.1º with every degree mark numbered. The azimuth circle is read against a fixed index line through a variable focus microscope (see Figs 5 and 8). The azimuth circle and the upper casing are mounted independently, and each is provided with a clamp and a tangent screw (the clamp is the smaller of the two). With the upper clamp loose and the lower clamp tightened, the azimuth circle is fixed to the base of the instrument and the upper casing can be rotated relative to it. The tangent screws enable fine adjustments to be made to the locked positions after their respective clamps have been tightened. In order to differentiate between the two sets of clamps and tangent screws, the lower set has fluted screw-heads coloured yellow, while the upper set has milled screw-heads coloured silver (see Fig 5).

5-14 Fig 8 Reading the bearing on the Azimuth Circle through the Microscope



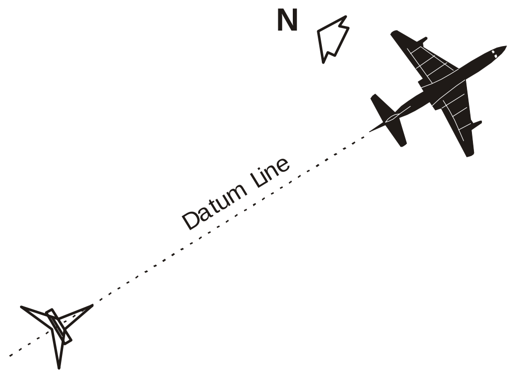
Operation

12. The Watts Datum compass can only be used in good light conditions and should not be used in rain. Operation in strong winds (greater than 15kt) may result in inaccurate readings due to rocking of both the aircraft and the compass system. It is essential that all magnetic materials are kept 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 not be wearing or carrying any magnetic materials, e.g. watches, screwdrivers or spanners (see also Volume 5, Chapter 16, para 10).

13. **Setting Up**. Instrument set-up is in two steps:

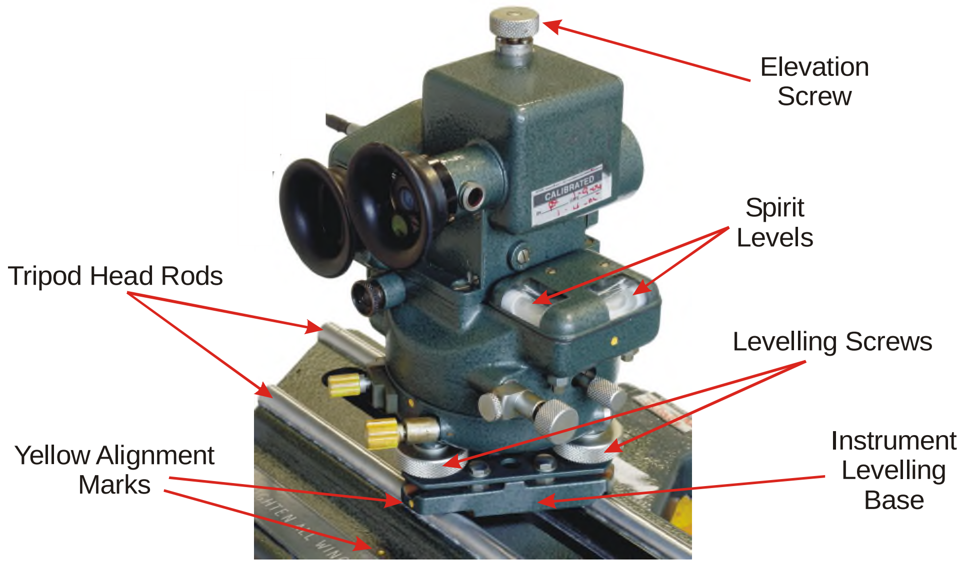
a. **Positioning and Levelling of the Tripod**. Set up the tripod in the desired position. The centre of the tripod head should be approximately on the Datum Line, with the centre leg of the tripod pointing towards the aircraft (see Fig 9). By manipulating and adjusting the legs, set the tripod at a convenient height and level it by reference to its circular spirit level. Remove the tripod protective cap by unscrewing the head bolt (the large silver bolt underneath the tripod platform).

5-14 Fig 9 Tripod Set-up



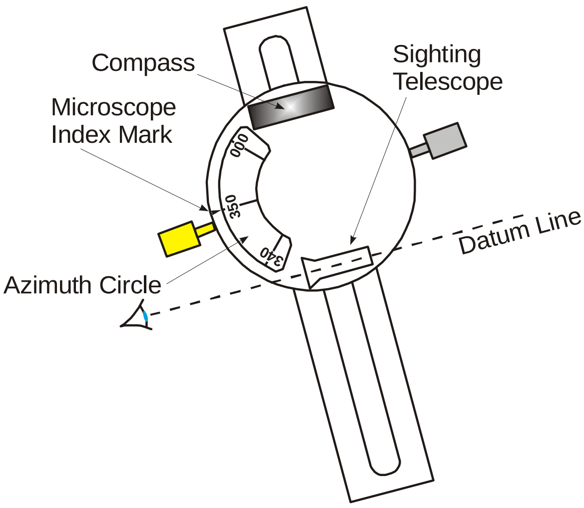
b. **Attaching the Instrument**. Stand behind the tripod looking towards the aircraft. Hold the instrument with the yellow alignment marks towards you. Attach the instrument by locating the grooves on the lower face of the base plate with the tripod head rods and then screwing the tripod head bolt into the threaded hole in the centre of the base plate (see Fig 10). Attach the Bowden cable to the centre of the caging knob. Now, level the instrument by adjusting the levelling screws, with reference to the spirit levels on the upper casing. Having levelled the instrument, loosen the lower (yellow) clamp and tighten the upper (silver) clamp.

5-14 Fig 10 Instrument Set-up



Turn the upper casing to align the sighting telescope with the aircraft's datum points (Fig 11). 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 tripod rods. This stage is only an approximate alignment to ensure that the aircraft datum is within sight, ready for the next stage of the procedure. Finally, tighten the lower (yellow) clamp and check that the instrument is still level (adjust if necessary).

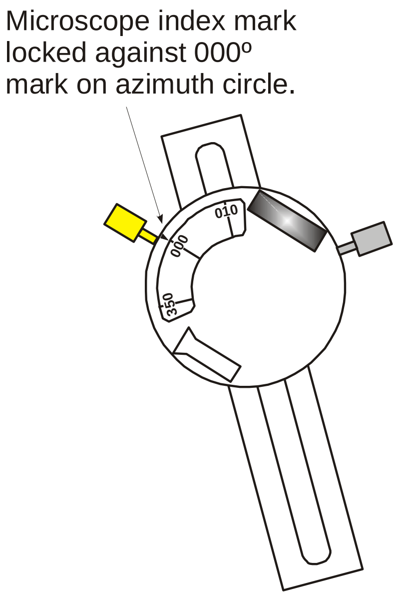
5-14 Fig 11 Telescope aligned on Datum Line



14. **Taking a Bearing**. Taking a bearing consists of three separate stages:

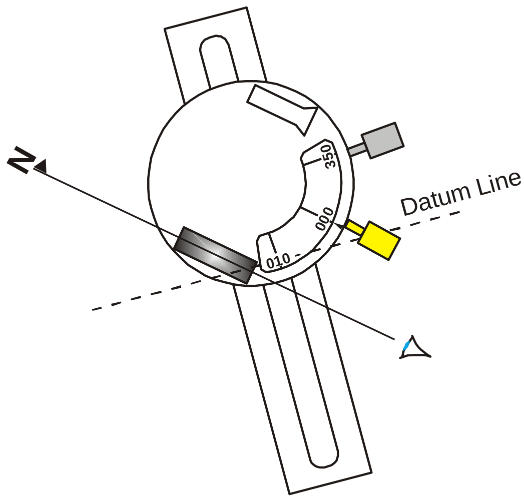
a. **Locking the Microscope Index Mark against the Azimuth Circle**. Adjust the focus on the circle-reading microscope, as required, by turning the knurled cover on the end of the eyepiece. Loosen the upper (silver) clamp. Rotate the upper casing until the index seen through the microscope reads approximately zero. Tighten the upper (silver) clamp and make fine adjustments with the upper tangent screw until the alignment is exactly on 000.00º (see Fig 12). Note that alignment with 000º is for readings taken from behind the aircraft. For abeam sightings, the index mark should be aligned with 090º or 270º, while shots from the front of the aircraft demand that 180º is the mark to be used.

5-14 Fig 12 Azimuth Circle Indexed



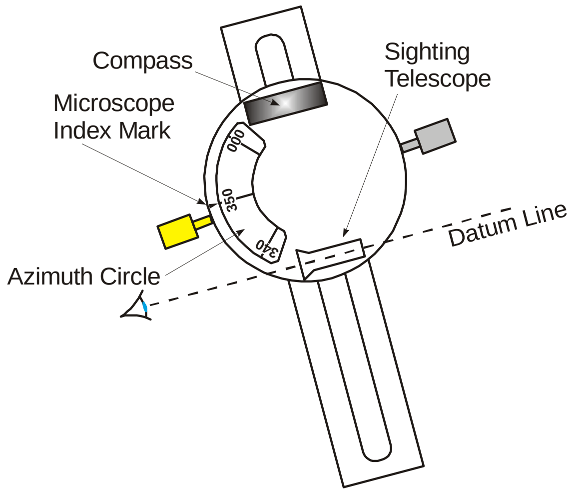
b. **Aligning the Compass with Magnetic North**. Loosen the lower (yellow) clamp and rotate the upper casing to align the compass approximately with the magnetic meridian. Uncage the compass by using the Bowden cable release. When the magnet settles down, tighten the lower (yellow) clamp and align the compass filaments accurately, using the lower (yellow) tangent screw. Cage the compass and check that the azimuth circle indexing is still accurate, ie it still reads 000.0º, 090.0º, 180.0º or 270º, as appropriate (see Fig 13).

5-14 Fig 13 Compass Aligned with Magnetic North



c. **Taking the Sighting on the Aircraft**. Loosen the upper (silver) clamp and rotate the upper casing to align the sighting telescope with the aircraft's datum points. Tighten the upper (silver) clamp and make fine adjustments with the upper tangent screw (see Fig 14). Take care not to disturb the lower (yellow) clamp or tangent screw during this operation. Read the magnetic bearing from the azimuth circle through the microscope.

5-14 Fig 14 Sighting Telescope accurately aligned on Aircraft Datum



15. '**Mustard Sandwich'**. Although operation of the Watts Datum Compass is fairly simple, it is still a daunting task for the inexperienced operator. The words 'Mustard Sandwich' were suggested many years ago as a method of remembering the correct sequence of operation of the clamps and tangent screws. Examination of para 14 will show the origin of this somewhat strange expression. The three stages of taking a bearing, paras 14a, b and c, require the operator to adjust the silver, yellow and silver controls respectively. This has been translated to 'bread' for silver and 'mustard' for yellow, hence 'Mustard Sandwich'.

16. **Packing the Instrument**. The Watts Datum Compass is a very delicate, precise instrument and is expensive. It is provided with a special storage case, in which it must be stored when not in use. Before packing the instrument in its case, ensure that both clamps have been loosened. Line up the yellow dots on the instrument then, with the dots facing upwards, insert the instrument into its protective mount inside the case. Close the case and ensure it is fastened securely before carriage or storage.

THE PRECISE HEADING TEST SET

Introduction

17. In carrying out a compass swing, in addition to the actual magnetic heading of the aircraft, which is determined using the Watts Datum Compass, it is necessary to know the magnetic heading indicated by the aircraft compass to a high degree of accuracy. It is also essential that the compass system is synchronized before any readings are recorded.

18. Most compass displays are only graduated at 1º intervals and this is unsatisfactory for compass calibration purposes, where a precision of ± 0.1º is needed. In addition, most compasses are synchronized by reference to a •/+ annunciator or to a rudimentary centre-reading voltmeter, neither of which is sufficiently accurate.

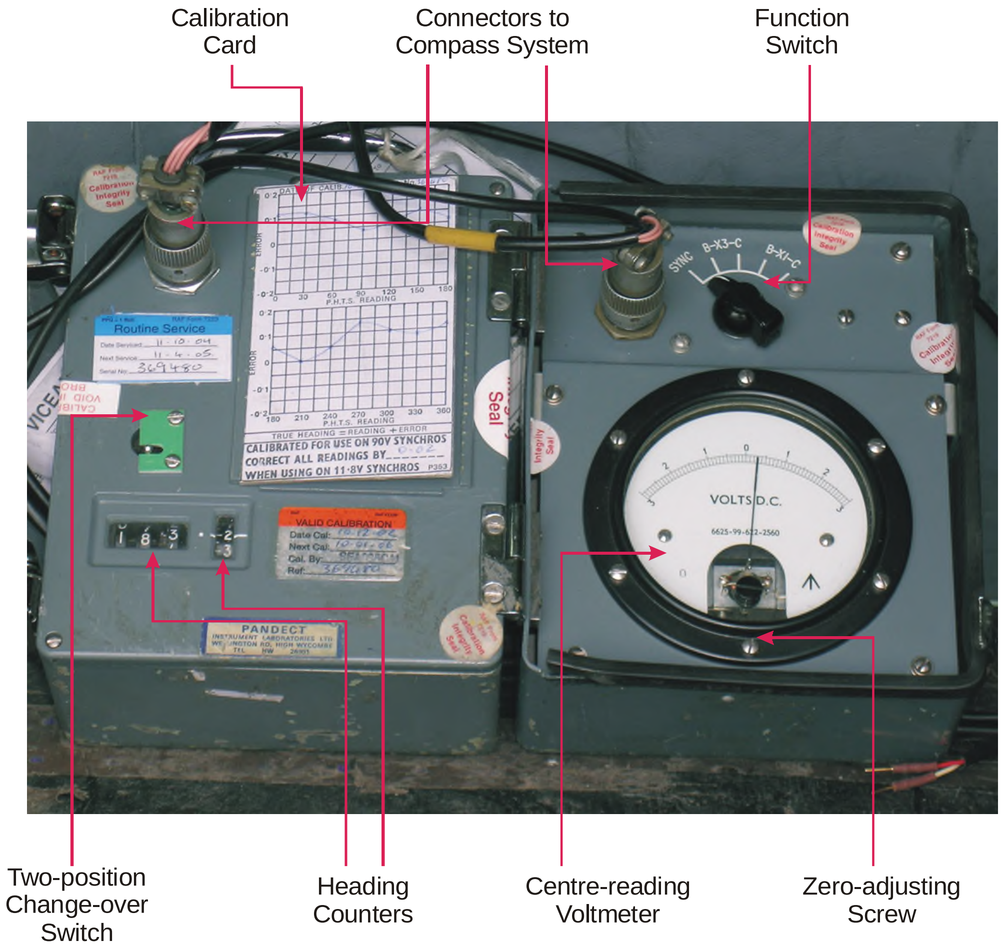
19. The Precise Heading Test Set (PHTS) is designed to overcome these shortcomings by providing:

a. A display of compass heading by means of veeder counters which can be read to 0.05º**.**

b. An accurate centre-reading voltmeter.

The PHTS is in the form of a hinged rectangular box which opens to reveal the controls and indicators (Fig 15).

5-14 Fig 15 Precise Heading Test Set



Controls and Indicators

20. **Heading Counters**. The left half of the PHTS has two windows displaying a veeder counter indication of compass heading. The left-hand window indicates whole degrees of compass heading from 000º to 359º**.** The right-hand window indicates tenths of a degree and can be read to an accuracy of at least 0.05º. There is also a calibration certificate and a calibration graph which allows corrections for instrument error to be made to the heading counter readings.

21. **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 compensator coils). The voltmeter can be centred by turning a zero-adjuster screw.

22. **Function Switch**. A five-position function selector switch is mounted above the voltmeter. The positions are marked SYNC, B-X3-C and B-X1-C. The facilities provided by these positions are:

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-X3-C**. There are two switch positions against the B-X3-C marking and the use of this switch depends on the type of compass being calibrated. On some compass systems, the two positions, B and C, allow the display of the respective DC voltage corrections to the flux valve compensator coils set in at the B and C potentiometers of the remote corrector unit. On other compass systems, only the left-hand, B position is used and selection between B and C voltage displays 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. In either case the voltage indicated is one third of the actual voltage (as implied by the X3 marking), thus, for example, an indication of 2 volts represents an actual measurement of 6 volts.

c. **B-X1-C**. These two switch positions operate in the same manner as the B-X3-C function except that the displayed voltage equates to the actual correction voltage set in at the potentiometers rather than one third of the value.

23. **Change-over Switch**. On some compass systems, because of the design of the test socket, the heading readouts on the PHTS would be 180º removed from the actual heading. The two-position change-over switch permits this anomaly to be corrected if necessary.

24. **Test Cable Harness**. Two sockets, one on each half of the PHTS, are provided to allow connection of the set 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 procedures for the particular aircraft/compass system to ensure that the correct cable harness is used.

Calibration

25. 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 of the set. Corrections to be applied to the PHTS heading counter readings should be extracted from this graph and applied to each reading.

CHAPTER 15 - MAGNETIC COMPASS DEVIATIONS

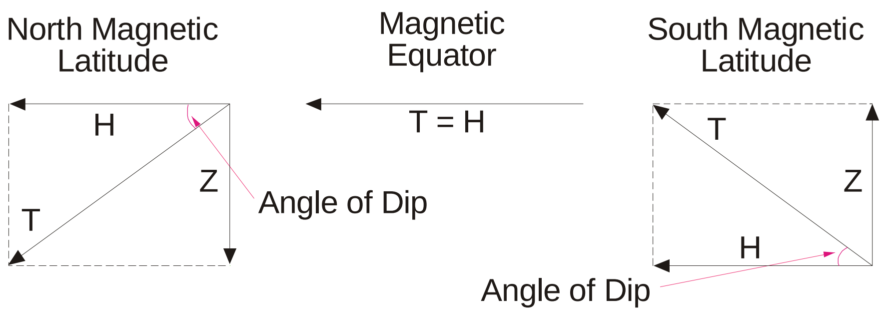
Introduction

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 aircraft. It will as a result indicate the direction of the resultant of the Earth's magnetic field and the magnetic field produced by the aircraft and experienced at the sensor position. The difference between the direction of the horizontal component of the Earth's field, and the direction of the horizontal component of the resultant field, is known as deviation. It is annotated 'East (positive)' or 'West (negative)', depending on whether the resultant field direction is to the East or West of the Earth's field. Deviation 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. This chapter will review the causes of deviation.

The Earth's Magnetic 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 dip. The total field (T) can be resolved into two components, a horizontal component (H) and a vertical component (Z) as shown in Fig 1. Fluxvalve units use only the H component to sense the direction of the local magnetic meridian, and H can therefore be considered to be the directive force acting upon the sensor. Other horizontal magnetic fields will increase, decrease, or act to deviate this directive force. The Z component is significant only in that it contributes to the magnetism induced in the magnetic material of the aircraft.

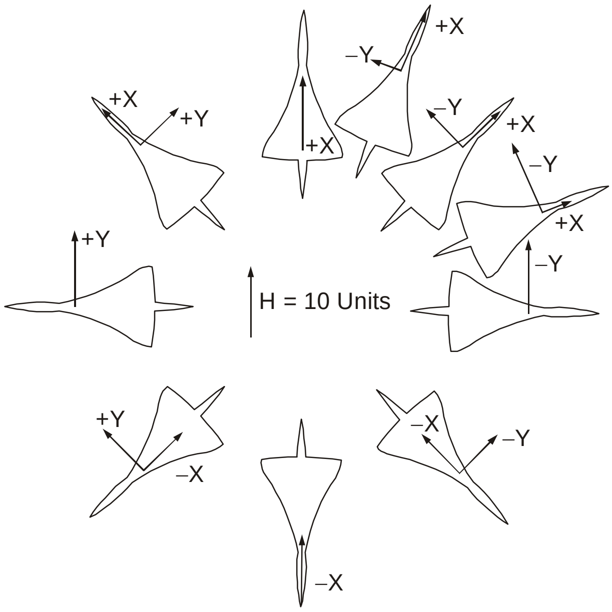
5-15 Fig 1 The Earth's Magnetic Field Resolved



3. The H component can itself be resolved into two components relative to the aircraft axes; an X component along the fore-and-aft axis and a Y component acting athwartships. It is usual and satisfactory to consider only the situation of the aircraft in a level attitude in which case the three components, X, Y, and Z, correspond to the three major aircraft axes. By convention the X, Y and Z components are considered positive when acting forward, starboard and downward respectively.

4. The values of H and Z vary with magnetic latitude, and for any given geographical location the X and Y components vary with aircraft heading (eg the whole of the H component will equate to a positive or negative X component when the aircraft is aligned with the magnetic meridian, or to a positive or negative Y component when the aircraft is at 90º to the meridian - see Fig 2).

5-15 Fig 2 Change in the Magnitudes of the X and Y Components with Change of Heading



The Aircraft's Magnetic Field

5. An aircraft's magnetic field is derived from innumerable pieces of magnetic material, each of which will have a different intensity of magnetization and a different capacity to retain magnetism. However, in order to make a reasonable analysis of the effect of aircraft magnetism on the Earth's field, it is convenient to make a somewhat arbitrary division of the magnetism into two constituents, a permanent field and a temporary field, due to what is known as hard iron and soft iron respectively.

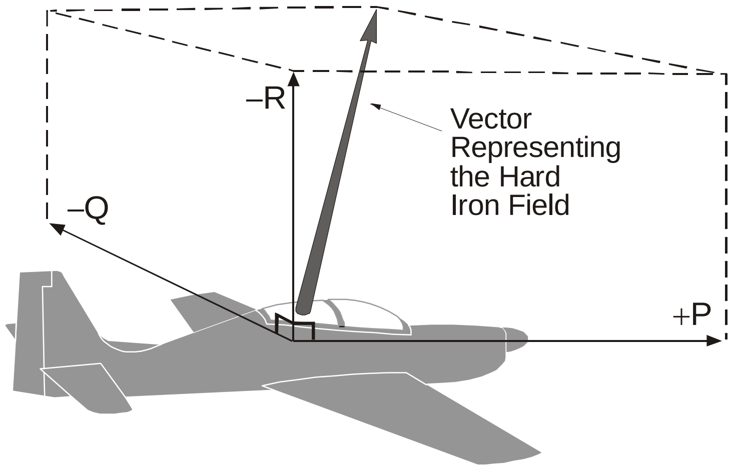
6. **Hard Iron**. Magnetic material of the aircraft structure which has acquired permanent magnetism is described as hard iron. This magnetism may have been acquired during manufacture, or during the flying, servicing, or structural testing of the aircraft. Magnetic components of instruments permanently installed in the aircraft are included in the general designation hard iron. Although permanent magnetism can change slowly with time, and rather more rapidly as the result of a lightning strike, these changes are ignored in the general consideration of compass deviation.

7. **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 the Earth's field, the hard iron, electrical currents, and weapons or cargo. The effects of electrical currents and payload are reduced to negligible proportions by the careful selection of the sensor position.

The Effect of the Hard Iron Field

8. The many elements of hard iron together form a permanent magnetic field of irregular shape, but with an orientation relative to the aircraft axes that does not change with heading; the effect is as if a permanent magnet were fixed to the aircraft. The hard iron field at the sensor position is therefore constant in strength and direction relative to the aircraft axes. This field can be resolved into three component vectors, P, Q and R, aligned with the aircraft axes as shown in Fig 3, analogous to the X, Y and Z components of the Earth's field.

5-15 Fig 3 Resolution to the Hard Iron Field



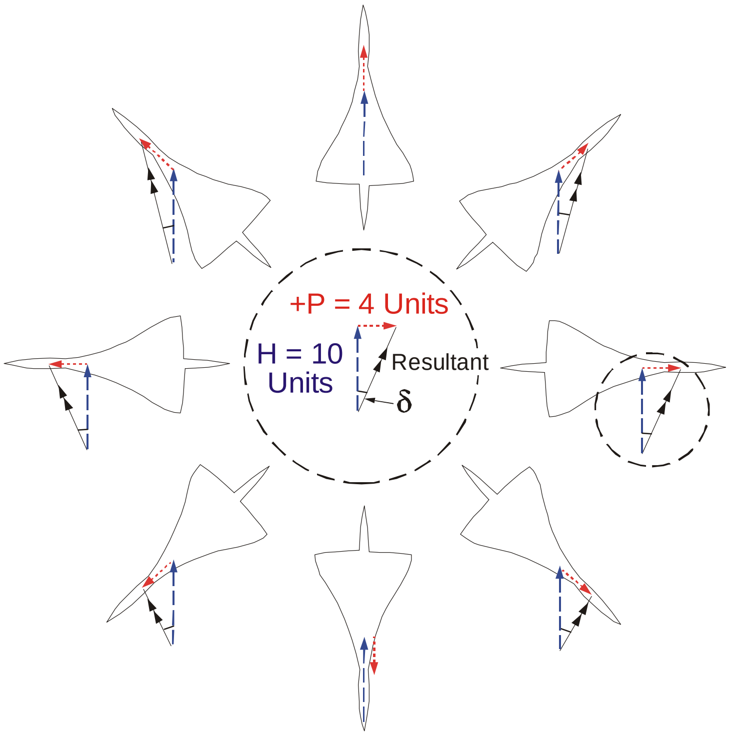
9. The fore-and-aft vector, P, will have the greatest deviating effect on H when the aircraft is on an East or West heading; on North or South the vector merely changes the magnitude of the directive force (Fig 4). The variation of the deviation due to P is in the form of a sine function as shown in Fig 5, i.e.:

$$\delta\theta = \delta_{max} \times \sin\theta$$

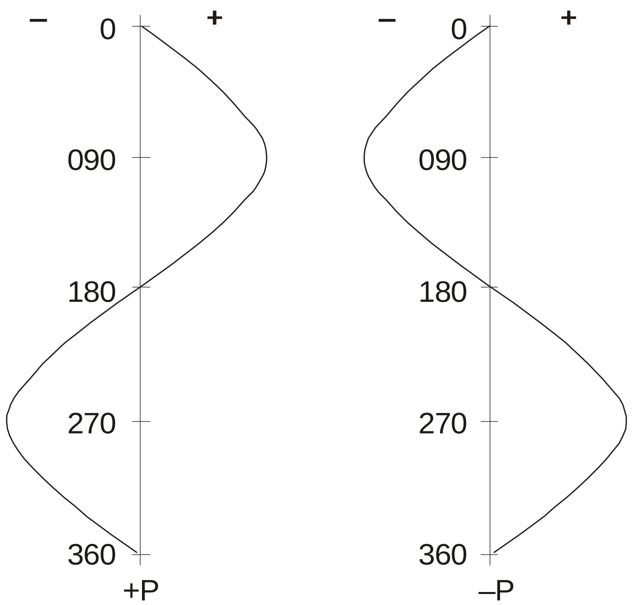
where $\delta\theta$ = deviation on heading θ

and δ_{max} = deviation on heading 090º or 270º

5-15 Fig 4 Deviating Effect of +P



5-15 Fig 5 Graphs of Deviation due to P

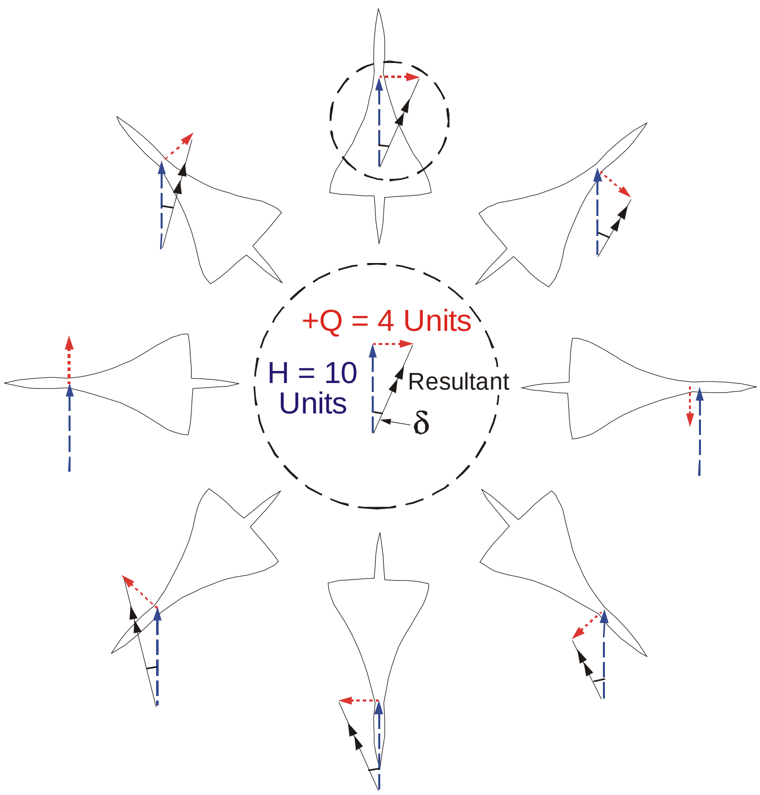


10. The effect of the athwartships vector, Q, is to produce zero deviation on East and West and maximum deviation on North and South (Fig 6), i.e. in the form of a cosine function (Fig 7):

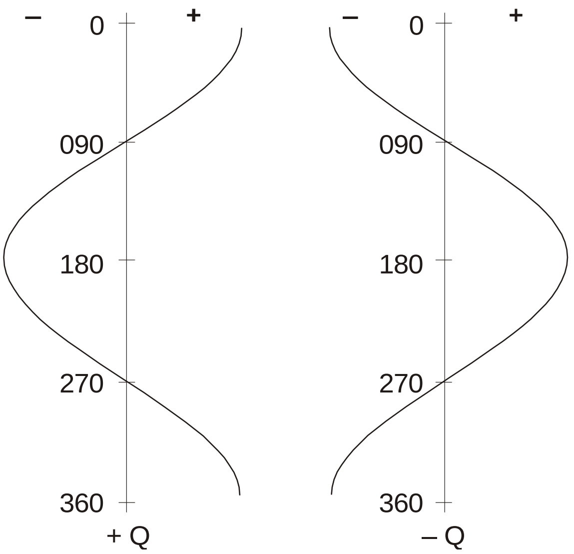
$$\delta\theta = \delta_{max} \times \cos\theta$$

where $\delta\theta$ = deviation on heading θ and δ_{max} = deviation on heading 000º or 180º

5-15 Fig 6 Deviating Effect of +Q



5-15 Fig 7 Graphs of Deviation due to Q



11. The vertical component, R, exercises no deviating effect when the aircraft is in a level attitude.

The Soft Iron Field

12. Magnetism will be induced in the aircraft's soft iron both by the Earth's field, which is the dominant effect, and by the hard iron field. As the hard iron field is constant relative to the airframe, and the soft iron can be considered as a single fixed block, the field induced by the hard iron in the soft iron is constant. However the soft iron field will distort the hard iron field, i.e. the two sources of deviation are in reality inseparable. The hard iron thus has an element of magnetism affected by the soft iron.

13. Soft iron magnetism will be induced by all three components, X, Y and Z, of the Earth's total field. Each component will induce a three-dimensional field in the soft iron, and the horizontal components of these fields will act as deviating forces at the sensor. The amount of deviation depends upon:

a. The amount, permeability, and location in relation to the sensor, of the soft iron. These are constant for any given aircraft.

b. The geographical location. As the inclination and total Earth field strength (T) vary with position, the components X, Y, and Z will vary.

c. The heading of the aircraft, for components X and Y.

14. As each of the three components, X, Y and Z, of the Earth's field is considered to induce a soft iron field, and as the vector representing each of these fields can be resolved into three component vectors coincident with the aircraft axes, there are a total of nine soft iron components. Each is given a two-letter designator as shown in Table 1. The direction in which the soft iron deviating field acts determines the sign convention of the components; the component is annotated positive if it acts forward or starboard on aircraft headings in the North-West quadrant.

Table 1 Soft Iron Components

Inducing Field	Soft Iron Field Components		
	Fore-and-Aft	Athwartships	Vertical
X	aX	dX	gX
Y	bY	eY	hY
Z	cZ	fZ	kZ

Coefficients

15. As the magnetic sensor only detects the horizontal components, the vertical hard iron component (R), and the vertical soft iron components (gX, hY, and kZ) need no further consideration. The two hard iron horizontal components (P and Q), together with the six soft iron horizontal components (aX, bY, cZ, dX, eY, and fZ), can be grouped into four pairs, the members of each group producing deviations which vary as a sine or cosine function of heading. The size of the deviation for any particular pair of components is a maximum on the headings for which the appropriate trigonometrical function is a maximum. The product of this maximum deviation in degrees and the appropriate trigonometrical function of heading will give the deviation produced by that pair on that heading. The maximum deviation is termed a coefficient and is assigned an identifying letter to indicate the pair of components to which it refers.

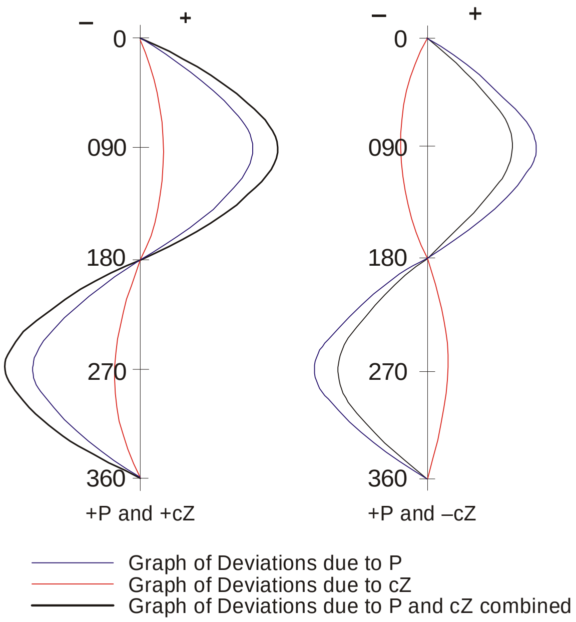
16. **Coefficient B**. Coefficient B is due to components P and cZ, each of which exhibits a sinusoidal variation with heading. The total deviation due to P and cZ is the algebraic sum of the deviation due to P and cZ separately. Thus, the total deviation will depend on the magnitude and sign of the constituents; this is illustrated in Fig 8. If the deviations δE and δW due to P and cZ are measured on East and West, the value of coefficient can be determined from:

$$\text{Coefficient B} = \frac{\delta E - \delta W}{2}$$

The deviations must be given their correct signs. Once coefficient B has been determined, the deviation due to P and cZ on any compass heading can be obtained from the equation:

$$\delta\theta = B \sin\theta$$

5-15 Fig 8 Combined Graphs of Deviation due to P and cZ



17. **Coefficient C**. Coefficient C is the resultant of components Q and fZ, the variation of each with heading being a cosine function. In a similar manner to coefficient B, it can be shown that:

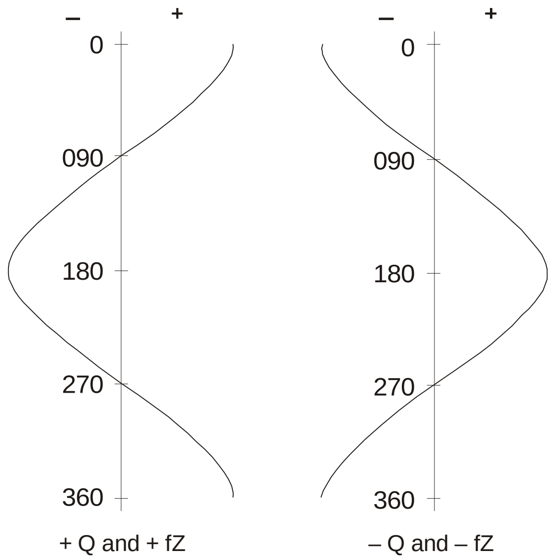
$$\text{Coefficient C} = \frac{\delta N - \delta S}{2}$$

and that the deviation due to Q and fZ on a heading θ is given by:

$$\delta\theta = C \cos\theta$$

Graphs showing the variation of total deviation due to positive and negative values of Q and fZ are shown in Fig 9.

5-15 Fig 9 Combined Graphs of Deviation due to Q and fZ



18. **Coefficient D**. Coefficient D is due to components aX and eY. Each component varies as a function of the sine of twice the compass heading as illustrated in Fig 10; the maximum deviations occur on the intercardinal headings. If the deviations δNE, δSE, δSW, δNW due to aX and eY on the intercardinal headings are measured then the value of coefficient D can be found from:

$$\text{Coefficient D} = \frac{(\delta NE + \delta SW) - (\delta NW + \delta SE)}{4}$$

The deviation on a heading θ due to aX and eY can be obtained from the equation:

$$\delta\theta = D \sin 2\theta$$

5-15 Fig 10 The Components of Coefficient D: aX and eY

Fig 10a Graphs of Deviation due to aX

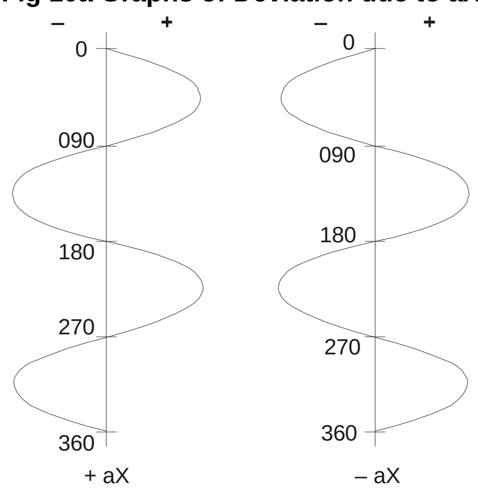
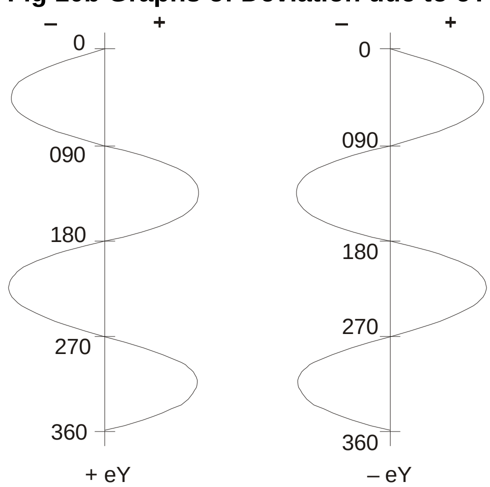


Fig 10b Graphs of Deviation due to eY



19. **Coefficients E and A**. Coefficients E and A are due to components bY and dX. Each component produces a deviation which varies with heading in the form shown in Fig 11; this function is the cosine of twice the heading, displaced to one side or the other of the zero axis.

5-15 Fig 11 The Components of Coefficients E and A: bY and dX

Fig 11a Graphs of Deviation due to bY

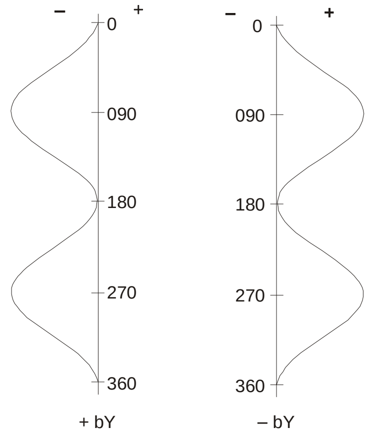
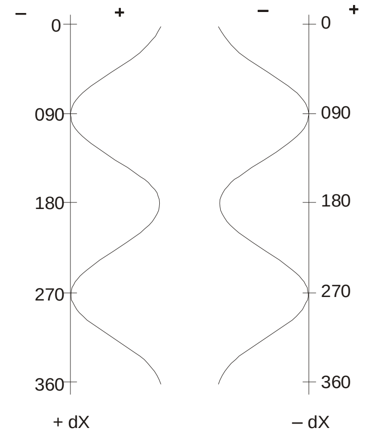


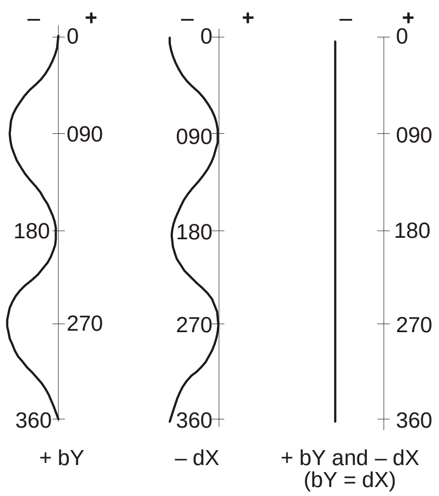
Fig 11b Graphs of Deviation due to dX



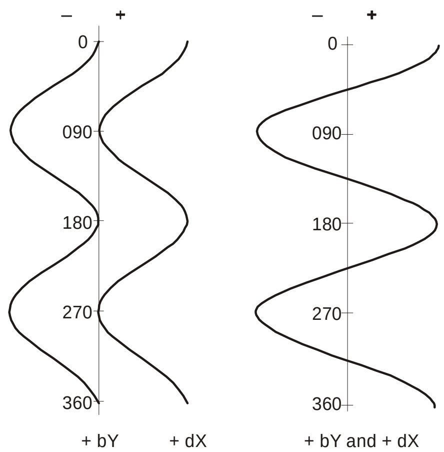
The result of adding the two components depends on their equality or otherwise as follows:

a. If equal, the deviation is constant, or varies as the cosine of twice the heading (Figs 12 and 13).

5-15 Fig 12 Combination of Deviation due to Equal Components of +bY and –d

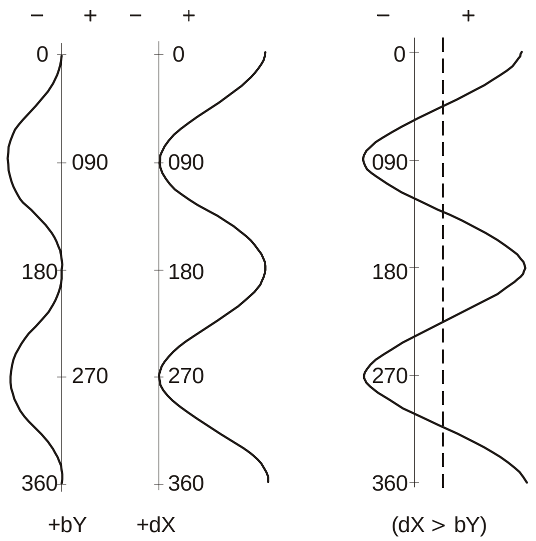


5-15 Fig 13 Combination of Deviation due to Equal Components of +bY and +dX

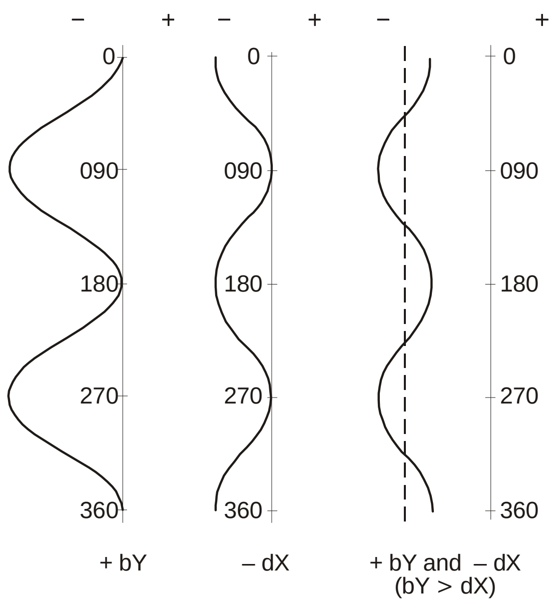


b. If unequal, there is a constant deviation and one which varies as the cosine of twice the heading (Figs 14 and 15).

5-15 Fig 14 Combination of Deviations due to Unequal Components of +bY and +dX



5-15 Fig 15 Combination of Deviations due to Unequal Components of +bY and –dX



The variable part of the deviation is represented by the coefficient E and the constant part by the coefficient A.

20. **Coefficient E**. The maximum values of deviation occur on the cardinal headings. If the deviations δN, δE, δS, and δW 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 variable deviation due to bY and dX on any compass heading can be found from:

$$\delta\theta = E \cos 2\theta$$

21. **Coefficient A**. Coefficient A represents the constant deviation due to the vectors bY and dX. It can be determined by taking the average of the deviations measured on any number of equally spaced headings; the more headings, the greater the accuracy. For the initial correction of a compass before calibration coefficient A is normally determined from observations on four headings, but for deviation analysis it is calculated from observations on eight or twelve headings, thus:

$$\text{Coefficient A} = 1/8(\delta N + \delta NE + \delta E + \delta SE + \delta S + \delta SW + \delta W + \delta NW)$$

Other Sources of Deviation

22. In addition to deviations due to the permanent and induced magnetism of the aircraft, deviations may be caused by the following:

a. **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 is identical to that of coefficient A, and although the errors may be distinguished by the term Apparent A, with magnetic effects being termed Real A, in practice it is not necessary to distinguish between them; they are both included in the term coefficient A.

b. **Electrical Fields**. Direct currents will create fields which have a similar effect to hard iron magnetism. Although the effects can be determined by calibrating the aircraft with and without the appropriate circuits operating, in practice, providing the sensor is in a remote part of the aircraft, the effects of any field will be negligible.

c. **Transmission Errors**. With remote indicating compasses, impedance and voltage imbalances in the flux valve and synchros can cause errors of the sin 2q or cos 2q form. These errors are usually greater than those due to induced magnetism, but it is unnecessary to differentiate between the sources of error and both are included in coefficients D and E.

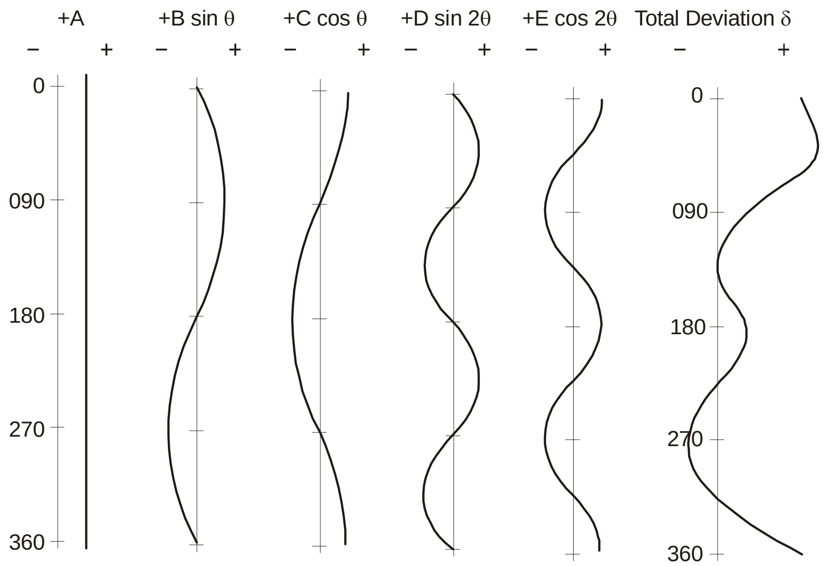
Total Deviation

23. The two hard iron horizontal components, P and Q, and the six soft iron horizontal components, aX, bY, cZ, dX, eY, and fZ, can be grouped according to their similarity of effect to produce five coefficients, A, B, C, D, and E, which represent the maximum deviations caused by the individual sets of components. The deviation due to any set on a compass heading θ can then be determined by multiplying the coefficient by the appropriate trigonometric function of the heading, eg B sin θ for P and cZ. The total deviation (δ) on any heading (θ) is then the sum of these individual expressions, thus:

$$\delta = A + B \sin\theta + C \cos\theta + D \sin 2\theta + E \cos 2\theta$$

This addition is shown graphically in the example of Fig 16.

5-15 Fig 16 Graphs of Component Deviations and Total Deviation



24. Although the previous discussion has considered the components of total deviation separately, they cannot in practice be measured individually as they act simultaneously. However, if the total deviation is measured on the eight headings at which the individual maxima occur, the values of all of the coefficients can be obtained by analysis of the total deviation equation.

25. An expression for the total deviation on each cardinal and intercardinal heading can be obtained by substituting the value of the heading into the total deviation equation, thus:

$$\begin{aligned}
\delta N &= A + C + E \\
\delta NE &= A + B \sin 45^\circ + C \cos 45^\circ + D \\
\delta E &= A + B - E \\
\delta SE &= A + B \sin 45^\circ - C \cos 45^\circ - D \\
\delta S &= A - C + E \\
\delta SW &= A - B \sin 45^\circ - C \cos 45^\circ + D \\
\delta W &= A - B - E \\
\delta NW &= A - B \sin 45^\circ + C \cos 45^\circ - D
\end{aligned}$$

There are therefore eight independent equations from which to determine the five unknown coefficients. Expressions for the coefficients can be deduced as:

$$\begin{aligned}
A &= \tfrac{1}{8}\Sigma\delta \\
B &= \tfrac{1}{2}(\delta E - \delta W) \\
C &= \tfrac{1}{2}(\delta N - \delta S) \\
D &= \tfrac{1}{4}[(\delta NE - \delta SE) + (\delta SW - \delta NW)] \\
E &= \tfrac{1}{4}[(\delta N - \delta E) + (\delta S - \delta W)]
\end{aligned}$$

Having determined the five coefficients, it is possible to calculate the total deviation for any compass heading.

26. **Example**. Suppose the value for total deviation on a compass heading of 060º is required given that the coefficients are: A = +2.0º, B = – 1.5º, C = + 3.0º, D = + 0.5º and E = – 1.0º.

$$\delta\theta = A + B \sin\theta + C \cos\theta + D \sin 2\theta + E \cos 2\theta$$

$$\text{i.e. } \delta 60 = +2.0 + (-1.5 \sin 60^\circ) + (3.0 \cos 60^\circ) + (0.5 \sin 120^\circ) + (-1.0 \cos 120^\circ)$$

$$= +2.0 - (1.5 \times 0.87) + (3.0 \times 0.5) + (0.5 \times 0.87) - (1.0 \times -0.5)$$

$$= +2.0 - 1.3 + 1.5 + 0.4 + 0.5$$

$$= +3.1$$

Thus, on compass heading 060º the total deviation is taken as +3.1º and the magnetic heading of the aircraft will be 063.1º. Fig 17a shows the graphs of the individual coefficients and Fig 17b shows the total deviation curve, from which the value of the total deviation on heading 060º can be confirmed as + 3.1º

Changes in Deviation

27. The examination of aircraft magnetism in this chapter has assumed a constant Earth field and a constant hard iron component of aircraft magnetism. If the magnetic latitude of the aircraft is changed the directive force, H, will change. Over a long period of time, or if for example the aircraft is left on one heading for some weeks, 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, resulting in a change to the deviation angle. Soft iron components will also change with latitude as the horizontal and vertical components of the Earth's field vary. Finally, a lightning strike can radically alter an aircraft's magnetism.

5-15 Fig 17 Graph of Total Deviation

Fig 17a Graphs of The Individual Coefficients

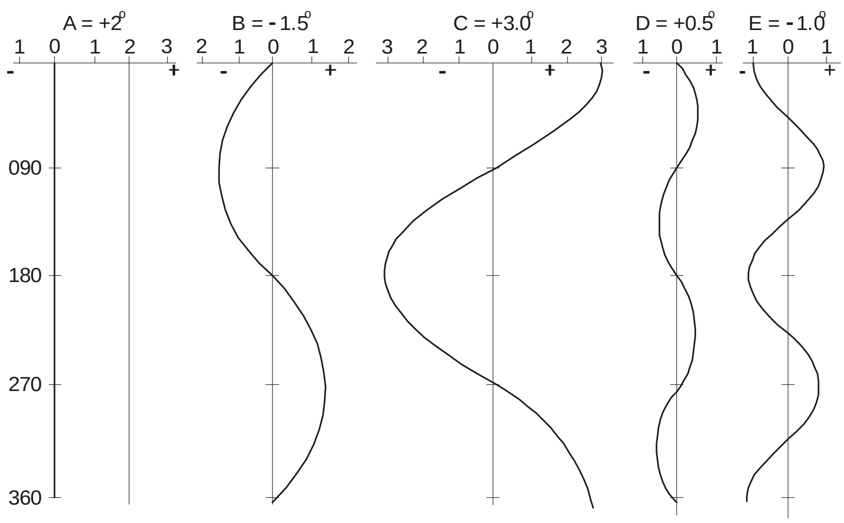
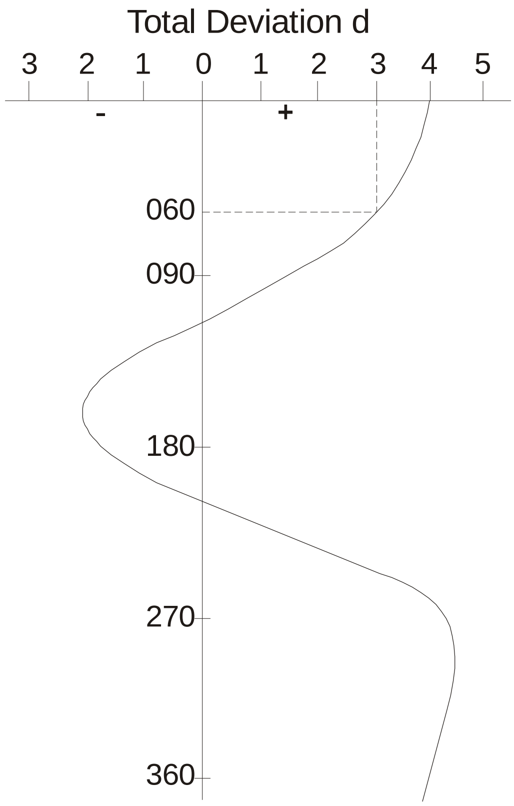


Fig 17b The Total Deviation Curve

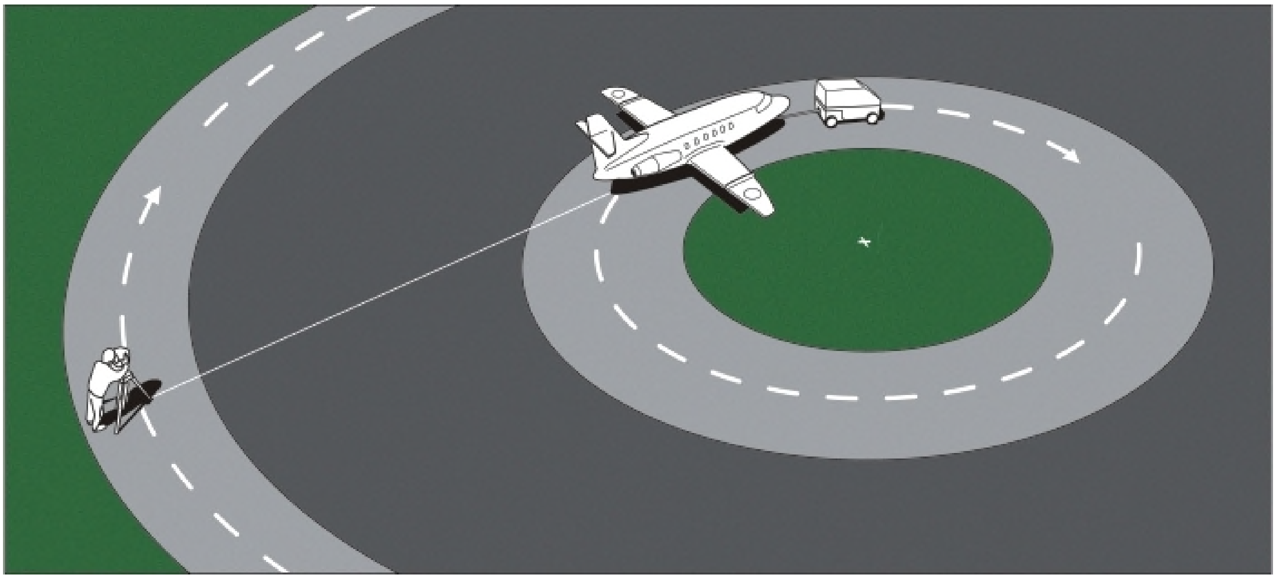


CHAPTER 16 - COMPASS SWINGING PROCEDURES

Introduction

1. Compass swinging is a special procedure used to ensure that magnetic compasses are as accurate as possible. It derives its name from the fact that the aircraft containing the compass system is 'swung' through a series of headings. During this process, readings are taken of the aircraft compass systems and these are compared to an accurate datum compass which is independent of the aircraft system. Compass swings are carried out on a specially prepared part of an airfield known as a 'Compass Base'. Fig 1 shows a typical scenario during a compass swing, with the datum compass operator in position to take a bearing on the fore/aft axis of the aircraft at the compass base. This chapter will concentrate on the practical aspects of carrying out a compass swing, but a knowledge of aircraft magnetism, covered in Volume 5, Chapter 15, would be beneficial. The engineering aspects of compass swings are covered in MAA Manual of Maintenance and Airworthiness Processes (MAP-01), Chapter 12.9, and instructions for use of the MOD Form 712A (the Compass Calibration Log) are detailed in MAA Manual of Maintenance and Airworthiness Processes - Supplement, Chapter 2.

5-16 Fig 1 An Operator ready to take a Bearing on the Fore/Aft Axis of the Aircraft



2. A full compass swing usually consists of a 'correcting swing' followed by a 'calibration swing'.

a. **The Correcting Swing**. In Volume 5, Chapter 15 it was shown that the amplitude of the deviation curve is directly dependent on the value of the deviation coefficients. Ideally, if all the coefficients could be reduced to zero, the deviation curve would become a straight line coincident with the central axis, i.e. there would be no deviation. The purpose of the correcting swing is to approach this condition of zero deviation as closely as possible by reducing the values of the coefficients. This may be achieved by the use of corrector devices, which generate magnetic fields equal in magnitude, but opposite in direction, to those caused by the components of aircraft magnetism. In practice, it is impossible to eliminate the coefficients entirely, and indeed, in most cases only coefficients A, B and C are corrected (see para 16).

b. **The Calibration Swing**. After correction, the compass is calibrated so that the residual deviations can be determined and recorded. These residual deviations can then be used in flight to correct readings taken from the compass indicators.

3. The accuracy with which deviations are measured and corrected depends upon:

a. The accuracy to which it is possible to read both the compass and datum instrument during the swing.

b. The accuracy requirements stipulated by the user, which will depend on how important the magnetic compass is to the aircraft's primary navigation system.

The Compass Base

4. 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. All major UK military airfields are provided with such an area, known as a 'Compass Base'. Full details of compass bases are contained in MAA Manual of Maintenance and Airworthiness Processes (MAP-01), Chapter 12.9.

5. In addition to the need to be free from extraneous magnetic fields, a compass base should be sited such that its use does not interfere with normal aircraft movements on the airfield and its surface should not preclude its use in wet weather. It must be large enough (and strong enough) to take all types of aircraft likely to use it, bearing in mind the radii of the aircrafts' turning circles and the position of any sighting rods (and their path during the swing). The compass base must be clearly and permanently marked to show:

a. The base centre.

b. The central area within which the aircraft's sensor should remain.

c. The datum compass circle.

d. Areas of magnetic anomalies.

6. Compass bases are subject to periodic re-survey to ensure their continued suitability. Responsibility for surveying compass bases is vested in QinetiQ, Land Magnetic Facilities at MOD Portland Bill. QinetiQ are also the authority on many other aspects of compass swinging, including calibration of datum compasses.

7. **Magnetic Anomalies**. Compass bases are classified as Class 1 if there are no known magnetic anomalies in excess of $\pm$ 0.1º at 1.5m above ground level, or Class 2 provided any anomalies are less than $\pm$ 0.25º at 1.5m above ground level. If a base is to be used for aircraft which have magnetic detectors significantly below 1.5m, a special survey is required. It is unusual for there to be any natural ferrous deposits on an airfield, and any magnetic interference is therefore most likely due to buried scrap metal, reinforced concrete, drainage systems, wire fences or conduit for electrical cabling. Electro-magnetic interference may be caused by electrical cabling and, if such cables cannot be avoided or re-routed, their effect, with and without current flowing, must be assessed.

8. **Changes in Variation**. Changes in variation may occur through diurnal changes and magnetic storms. 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 0.25º in the summer to about 0.07º in the winter. Magnetic storms are usually associated with sunspot activity. Although the frequency of such storms is only about once per year, they may last several hours or even days, and can alter the variation by up to 0.5º in the UK.

Occasions for a Compass Swing

9. The engineering responsibility for the calibration and adjustment of aircraft compasses is promulgated in MAA Manual of Maintenance and Airworthiness Processes (MAP-01), Chap 12.9. This document defines the occasions on which a compass swing is required, and these are summarized as follows:

a. As directed in aircraft maintenance schedules.

b. On acceptance by a user unit, if a new-build aircraft has been delivered direct from a contractor.

c. After an aircraft has been in long-term storage.

d. When an aircraft has been subjected to severe static electricity, eg a lightning strike. In this case, particular attention must be paid to compass accuracy and, if necessary, the appropriate demagnetization procedures laid down by the operating authority must be followed.

e. On transfer of an aircraft from one theatre of operations to another, if this entails a large change of magnetic latitude. This need not apply to aircraft on detachment of less than four weeks, unless higher order accuracy is operationally required.

f. Whenever a compass system has been subjected to shock, e.g. after a heavy landing.

g. After an aircraft has been repaired or subjected to conditions likely to affect the accuracy of the compass systems. Examples of such repairs and conditions are:

(1) A change of a component within the compass system likely to create a significant change in deviation, eg flux valve or magnetic compass.

(2) A 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 a detector unit of a remote reading compass.

h. If it is considered likely that a specific freight load will cause magnetic influence and thereby affect compass readings.

i. Whenever the accuracy of the compass system is in doubt.

Preparing to do a Compass Swing

10. Before starting a compass swing, the following general points should be checked to prevent embarrassment and delay:

a. Ensure that the weather conditions are suitable for carrying out the swing. Except under exceptional circumstances, ground-based compass swinging is only carried out in weather conditions clear of persistent rain and with wind speeds of 15 kt or less.

b. Ensure that the compass base is available for use.

c. Check that the aircraft compasses are fully serviceable.

d. Collect all of the items required for the calibration of the compass, eg Precise Heading Test Set, Watts Datum Compass, corrector keys, sighting rods and Compass Calibration Log (MOD Form 712A).

e. Ensure that non-magnetic tools are available.

f. Remove any items of moveable equipment which may affect the magnetic sensor, eg tool kits or spares in the vicinity of the sensor.

g. Obtain permission from Air Traffic Control to tow or taxi the aircraft to the compass base.

h. Check that the appropriate power source is available and that both the towing vehicle and power set have sufficient fuel.

i. Brief the compass swinging team on the procedures to be followed. Ensure that personnel remove from their person all metallic objects likely to interfere with the swing, e.g. watches, pens, spanners, headgear with metallic badges (see also Volume 5, Chapter 14, para 12). If the swing is to be conducted with engines running, ensure that the datum compass operator uses non-metallic ear defenders rather than a headset.

j. If sighting rods are to be used to act as the datum points of the aircraft (see Volume 5, Chapter 14), ensure that they are attached to the appropriate mountings.

k. Before starting the swing, switch on any aircraft electrical equipment which is likely to cause magnetic deviation in flight, but observe restrictions on ground operation of equipment.

Types of Compass Swing

11. There are four types of compass swing. The 'Standard' and 'Refined' swings are the most commonly encountered since they are used by many different aircraft types. Both will be considered in detail in this chapter.

a. **Standard Swing**. The standard swing is used where the compass system is not used as an input to other navigation or weapon aiming systems. Although a Watts Datum compass can be used as the datum, a Medium Landing compass is sufficient (for details of datum compasses see Volume 5, Chapter 14). A standard swing can be carried out on either a Class 1 or Class 2 compass base and uses eight headings during the calibration phase.

b. **Refined Swing**. The refined swing is used when the compass is used as a heading input to produce navigation or weapon aiming solutions. A more accurate reference (Watts Datum compass or Inertial Navigation System (INS)) must be used to provide the datum headings, and the calibration swing is carried out on twelve headings. A refined swing can only be carried out on a Class 1 compass base (but see Volume 5, Chapter 18).

c. **Electrical Swing**. The electrical swing is essentially the same as the refined swing except that, instead of physically moving the aircraft onto the appropriate headings, the headings are simulated by a Compass Calibrator, which applies a DC current to the secondary coils of the detector unit. During set-up, a Watts Datum compass is used to align the aircraft with the magnetic meridian. Unlike conventional swings, the area used for calibration need not be free from magnetic disturbances, provided there is magnetic stability.

d. **Air Swing**. Although compass swinging is usually carried out on the ground, it is possible to carry out an airborne compass swing, normally using an INS as the source of datum heading. Air swings are seldom carried out, but they are discussed in Volume 5, Chapter 17.

The four different types of swing are summarized in Table 1.

Table 1 Summary of the Four Types of Compass Swing

	Type of Compass Swing			
	Standard	**Refined**	**Electrical**	**Air**
Compass Base	Class 1 or 2	Class 1	Not required	Not required
Datum Compass	MLC or WD	WD or INS	WD during set-up	INS
Correcting Swing Needed	Yes	Yes	Yes	Yes
Calibration Swing	8 points	12 points	12 points	12 points
Refer to:	This Chapter paras 14 - 18	This Chapter paras 14 - 18 and Vol 5, Chap 19	Aircraft-specific Manuals	Vol 5, Chap 17

12. Before detailed examination of the different types of swing, there are some general rules which must be adhered to at all times, to ensure an accurate swing:

a. The aircraft must be positioned within ± 5º of each of the nominal headings.

b. After each change of heading, chocks should be inserted to ensure that the aircraft does not move whilst readings are being taken.

c. Sighting rods, where used, should be checked for verticality after each change of heading, before sighting with the datum compass.

d. The aircraft compass system must be allowed to settle after each change of heading, before the reading is taken.

e. Great care must be taken to ensure that both the datum and aircraft compasses are read simultaneously, and to their respective limits of accuracy.

The necessary accuracy and limits of the swing are stipulated by operating authorities, but, in general, coefficients should be reduced to less than 1.0º for a standard swing and 0.5º for a refined swing. Specific limits for each aircraft type are contained in the aircraft engineering documents.

13. As mentioned in para 2, compass swings consist of two distinct phases: correction and calibration. The rest of this chapter will describe these two procedures as they apply to the standard and refined swings. Aircraft which use the electrical swing will have the procedures documented in their type-specific manuals.

The Correcting Swing

14. **Purpose**. The purpose of the correcting swing is to reduce all the correctable coefficients to within limits. The correcting swing may have to be repeated several times to achieve the required accuracy. The data from each correcting swing is entered in the appropriate block on the first page of the Compass Calibration Log (MOD Form 712A).

15. **Procedure**. The correction procedure is common to both Standard and Refined swings, and is as described in the following sub-paras. Fig 2 is an extract from the MOD Form 712A and illustrates, by way of an example, how the results of the correcting swing are recorded.

a. Turn the aircraft onto South, and record the aircraft and datum compass readings.

b. Turn the aircraft onto West, and record the aircraft and datum compass readings.

c. Turn the aircraft onto North, and record the aircraft and datum compass readings.

d. Turn the aircraft onto East, and record the aircraft and datum compass readings.

e. Calculate the deviations (datum heading minus aircraft compass heading).

f. Sum the deviations algebraically and divide by four to find coefficient A.

g. Apply coefficient A to the compass reading (with sign unchanged) and correct the compass (see para 16).

h. Calculate coefficient B using the formula:

$$\text{Coefficient B} = \frac{\delta E - \delta W}{2}$$

i. Apply coefficient B (sign unchanged) to the resultant compass heading after correcting for coefficient A, and, with the aircraft still on East, correct the compass (see para 16).

j. Calculate coefficient C using the formula:

$$\text{Coefficient C} = \frac{\delta N - \delta S}{2}$$

k. Turn the aircraft onto South, record the new aircraft compass heading, apply coefficient C (sign changed) to this reading and correct the compass (see para 16).

l. Repeat this correcting procedure until coefficients A, B and C are all within limits, ie no further corrections have to be made. When this condition is reached, the calibration swing may start.

5-16 Fig 2 Extract from Form 712A showing the Recording of the Correcting Swing

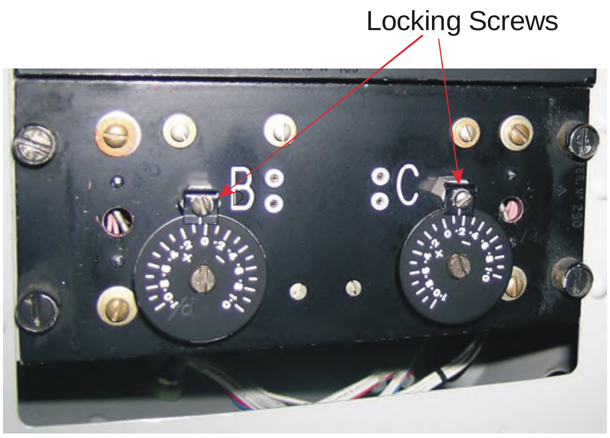
Approx Heading	Mag Hdg + Cor'n or Ins Hdg – Var'n (see Note)		Datum Heading (a)	Compass Heading (b)	Deviation (a – b)
South			180.5	181.5	− 1.0
West			270.0	268.5	+ 1.5
North			359.5	358.5	+ 1.0
East			091.0	090.5	+ 0.5
Coefficient A				+ 0.5	$A = \frac{+2.0}{4}$
Make Compass Read				091.0	$= +0.5$
Coefficient B				− 0.5	$B = \frac{0.5 - (+1.5)}{2}$
Make Compass Read				090.5	$= -0.5$
South				179.5	
Coefficient C Sign Changed				− 1.0	$C = \frac{1.0 - (-1.0)}{2}$
Make Compass Read				178.5	$= +1.0$

16. **Correcting the Compass**. In the previous paragraph, there were three references to correcting the compass during the swing. Although specific details vary according to aircraft and compass type, the two most common methods of applying the corrections to remote indicating compass systems are:

a. **Removing Coefficient A**. Small amounts (up to 1º) of coefficient A can be removed by offsetting the zero line of the Variation Setting Control (VSC) on the master indicator using a special tool. For values greater than 1º, it is normally necessary to physically rotate the detector unit.

b. **Removing Coefficients B and C**. Coefficients B and C are usually removed by means of corrector dials which are located beneath a cover on the compass controller (see Fig 3). Having removed the cover, the locking screw on the appropriate dial is loosened and the dial is turned progressively until the compass reads the required heading. Care must be taken when tightening the locking screw after adjustment to ensure that the dial is not turned in the process.

5-16 Fig 3 Corrector Dials for removing Coefficients B and C



The Calibration Swing

17. **Purpose**. The purpose of the calibration swing is to check that coefficient A has been removed and that the residual deviation is within the limits prescribed in the relevant aircraft documentation. Although the detailed procedure differs slightly between Standard and Refined swings, the final step is essentially the same – the construction of deviation cards for installation in the aircraft.

18. **Procedure**.

a. **Standard Swing**. The calibration phase of the Standard Swing proceeds as described in the following sub-paras. Fig 4 is an extract from the MOD Form 712A and illustrates, by way of an example, how the results of the calibration swing are recorded.

(1) Turn the aircraft onto South-West (225º), record the aircraft and datum compass readings.

(2) Turn the aircraft to the right in steps of 45º. At each step, record the aircraft and datum compass readings. The eighth, and final, heading will be South.

(3) Calculate the deviations on each heading and compute coefficient A by dividing their algebraic sum by eight.

(4) If the deviation figures are all within limits, and no further adjustment to coefficient A is required, the swing can be terminated.

Note: In the example at Fig 4, the algebraic sum of the residual deviations on all eight headings is + 4.0º, therefore the value of A is + 0.5º. As this value is within limits, the swing is terminated.

b. **Refined Swing**. The aircraft is moved through a twelve-point swing starting on 210º. The datum and compass readings are recorded every 30º (see the example in Fig 8). The deviations obtained from this swing form the basis of the Fourier and accuracy analyses, which are described in Volume 5, Chapter 19.

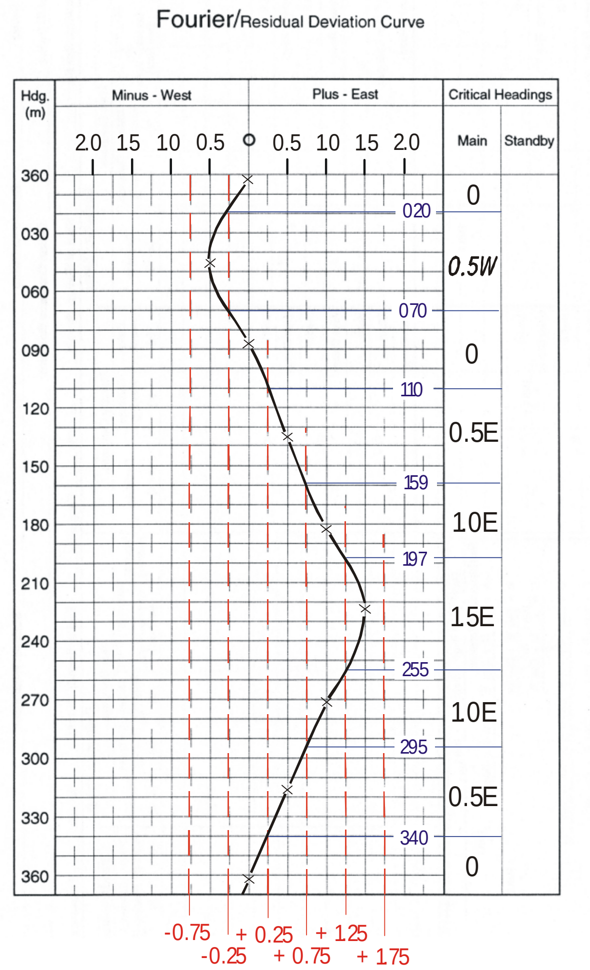
5-16 Fig 4 Extract from Form 712A showing Calibration Phase of Standard Swing

Approx Heading	Mag Hdg + Cor'n or Ins Hdg – Var'n (see Note)	Datum Heading (d)	Compass Heading (e)	Deviation (d – e)
225		*223.5*	*222.0*	*+ 1.5*
270		*272.0*	*271.0*	*+ 1.0*
315		*317.0*	*316.5*	*+ 0.5*
360		*002.0*	*002.0*	*0.0*
045		*046.0*	*046.5*	*− 0.5*
090		*089.0*	*089.0*	*0.0*
135		*135.5*	*135.0*	*+ 0.5*
180		*181.5*	*180.5*	*+ 1.0*

19. **Output**. As mentioned in para 17, the final step of the calibration swing is the production of deviation cards, which show the corrections to be made to the compass indications for all headings. These deviation cards are located in the aircraft, in purpose-designed holders, next to the compass systems to which they apply.

a. **The Deviation Curve**. A deviation curve is plotted from the data derived during the calibration swing. This curve is then used to produce the deviation card. Fig 5 shows a deviation curve plotted from the data at Fig 4. The first step in producing the curve is to choose a suitable scale for the x-axis. The deviation value for each of the headings is then plotted against the corresponding **compass** heading (column 'e' in Fig 4). Having joined the plotted points with a smooth curve, intermediate vertical lines are drawn (the dotted red lines in Fig 5), to intersect the curve at the critical headings. For example, the lines drawn at 1.25E and 1.75E delineate the band where the applied deviation is 1.5E. To ascertain the critical headings, horizontal lines are drawn from the intersections of the intermediate verticals with the curve (the blue lines in Fig 5). Critical headings and the deviation within the band are then annotated on the graph. In the case shown in Fig 5, the critical headings for the 1.5E band previously discussed, are 197º and 255º.

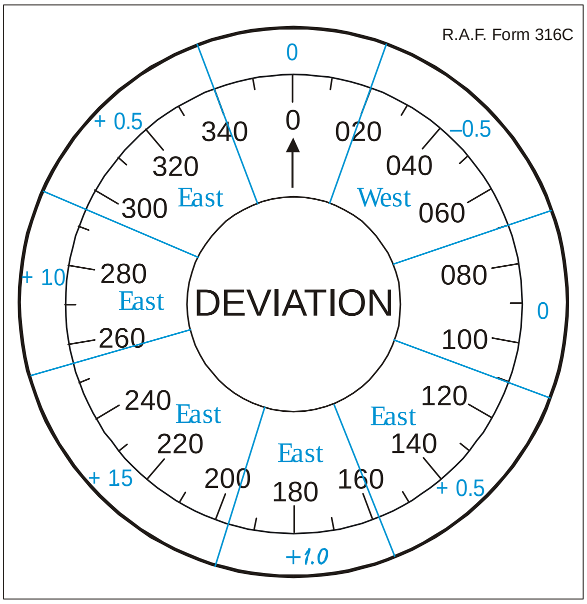
5-16 Fig 5 Residual Deviation Graph constructed from the data in Fig 4



b. **The Deviation Card**. The critical headings and deviation bands from the deviation curve are noted and transposed onto a blank deviation card (see Fig 6). The exact way in which this is done depends on the deviation card in use. It must be remembered that the sign convention used for deviation is that the sign used indicates how deviation is applied to **compass** heading to convert it to **magnetic** heading (see Volume 9, Chapter 1). For example, if deviation is 2º West (negative) then a compass heading of 093º would equate to a magnetic heading of 091º.

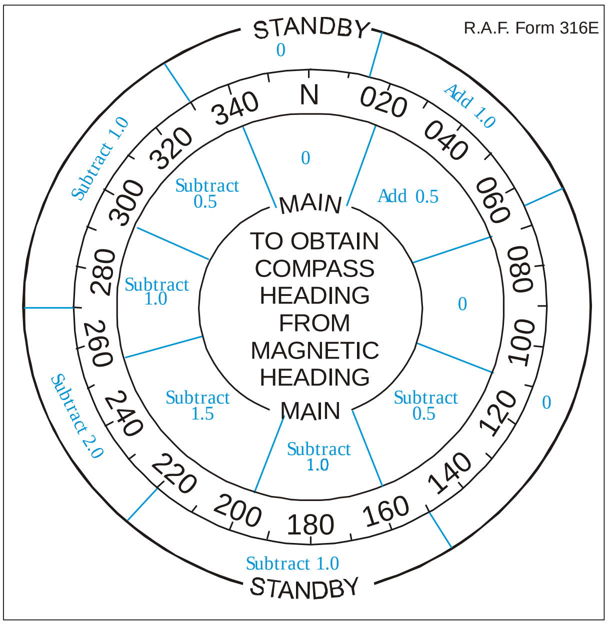
(1) **The Navigator's Card**. The deviation card shown at Fig 6 is intended for use at the navigator's position and has deviation presented with its correct sign, since this is the way the navigator will use it.

5-16 Fig 6 Deviation Card constructed from the Deviation Curve at Fig 5



(2) **The Pilot's Card**. Deviation cards for use in the pilot's position should have the sign of deviation reversed since the pilot will need to know what compass heading to steer from a given magnetic heading. The deviation card shown at Fig 7 is for use at the pilot's position and is constructed from the same data as the Navigator's card at Fig 6 (note that it also shows deviation for the Standby Compass). As an example of its use, consider the situation where Air Traffic Control tell the pilot to steer a heading of 180º magnetic; reference to the card shows that deviation is 1º East (positive), and the required compass heading is 179º.

5-16 Fig 7 Deviation Card for use at the Pilot's Position



20. **Documentation**. It is important to remember that the compass swinging process is part of an engineering procedure and, as such, must be documented and recorded in accordance with relevant orders. The compass swing is not complete until the paperwork is done and the relevant sections of the aircraft documentation signed.

5-16 Fig 8 Compass Calibration Log Entries for a Refined Swing (Correcting and Calibration)

Correcting Swing

	Main Compass				Standby Compass		
Approx Heading	Mag Hdg + Cor'n or Ins Hdg - Var'n (See Note)	Datum Heading (a)	Compass Heading (b)	Deviation (a-b)	Datum Heading (a)	Compass Heading (c)	Deviation (a - c)
South		180.75	181.41	-0.66			
West		270.30	271.54	-1.24			
North		001.13	002.13	-1.00			
East		090.57	090.99	-0.42			
Coefficient A			-0.83	A = -3.32 / 4	Coeff A		A = ____ / 4
Make Compass Read			090.16	= -0.83	Make Comp		=
Coefficient B			0.41	B = -0.42 +1.24 / 2	Coeff B		B = ____ / 2
Make Compass Read			090.57	= 0.41	Make Comp		=
South			179.18	C = -1.00 + 0.66 / 2	South		C = ____ / 2
Coefficient C Sign Changed			0.17		C Sign Ch		
Make Compass Read			179.35	= -0.17	Make Comp		=
South		179.01	179.35	-0.34			
West		271.15	271.00	+0.15			
North		358.77	359.26	-0.49			
East		089.11	089.23	-0.12			
Coefficient A	NOT APPLIED			A = -0.80 / 4	Coeff A		A = ____ / 4
Make Compass Read				= -0.20	Make Comp		=
Coefficient B	NOT APPLIED			B = -0.12 - 0.15 / 2	Coeff B		B = ____ / 2
Make Compass Read				= -0.14	Make Comp		=
South				C = -0.49+0.34 / 2	South		C = ____ / 2
Coefficient C Sign Changed	NOT APPLIED				C Sign Ch		
Make Compass Read				= -0.08	Make Comp		=

Calibration Swing

	Main Compass				Standby Compass		
Approx Heading	Mag Hdg + Cor'n or Ins Hdg - Var'n (see note)	Datum Heading (d)	Compass Heading (e)	Deviation (d-e)	Datum Heading (d)	Compass Heading (f)	Deviation (d-f)
360		358.77	359.26	-0.49			
030		031.10	031.57	-0.47			
060		059.48	059.63	-0.15			
090		089.11	089.23	-0.12			
120		120.18	120.47	-0.29			
150		149.55	149.73	-0.18			
180		179.01	179.35	-0.34			
210		209.73	210.15	-0.42			
240		239.90	239.83	+0.07			
270		271.15	271.00	+0.15			
300		300.41	300.16	+0.25			
330		330.44	330.63	-0.19			

Note: Datum Headings obtained from Watts Datum Compass are to be entered in the **Datum Headings** columns

Residual Coefficients:

$A = \frac{\text{Dev N} + \text{Dev E} + \text{Dev S} + \text{Dev W}}{4}$

$B = \frac{\text{Dev E} - \text{Dev W}}{2}$

$C = \frac{\text{Dev N} - \text{Dev S}}{2}$

$D = \frac{(\text{Dev NE} + \text{Dev SW}) - (\text{Dev NW} + \text{Dev SE})}{4}$

$E = \frac{(\text{Dev N} + \text{Dev S}) - (\text{Dev E} + \text{Dev W})}{4}$

Corrector Current / Voltage as applicable	
'B'	'C'

CHAPTER 17 - THE AIR SWING

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 (see Volume 5, Chapter 12):

   a. Error in measuring coefficient C due to coriolis acceleration.

   b. Settling time after turns, which is a function of hang-off error.

3. If a compass swing is required, prior to a transit and recovery to base, an airborne swing may be an option. An airborne swing could provide sufficient data for a standard swing.

Methods of Determining Heading in the Air

4. **Use of an Inertial Datum Bearing**. The most likely source of a steady, accurate heading datum is the inertial navigation system, where available.

5. **Use of a Gyro Datum Bearing**. A low-drift gyro is a suitable source for datum headings. If such a gyro is not already fitted in the aircraft, it may be feasible to incorporate minor modifications to fit one.

6. **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. One observer is employed to extract a bearing every 15 seconds, for one minute, indicating to the crew when the bearing is taken. These readings are then averaged.

   b. At each indication, another observer records the compass readings every 15 seconds for one minute, and averages these readings.

   c. At the third reading, an accurate fix and the time are recorded.

7. **Magnetic Bearing from the Gyro Datum**. The magnetic heading of the aircraft is found before flight, using the best datum available (this may be an external datum such as a Watts Datum Compass). At the same time, the gyro reading is recorded. The difference, eg Watts 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 can be done quite simply by comparing the gyro heading against the

Watts Datum Compass again after flight. The difference in corrections is due to gyro drift. The gyro drift is applied proportionally to the airborne gyro readings. An example of such calculations is shown in Tables 1 and 2.

Table 1 Calculation of Gyro Drift Rate

	Before Take-off	After Landing
Time	1000 hours	1030 hours
Datum Compass	030.45º	073.92º
Gyro	302.96º	346.62º
Correction	+ 087.49º	+ 087.30º
Gyro drift = – 0.19º Therefore gyro drift rate = – 0.38º per hour		

The correction for difference in variation (Table 2) arises from the difference in position between the point of ground reference, and the aircraft at 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 ground and in-flight variation. Assessment of gyro drift rate should be made before and after the correcting swing, and before and after the calibration swing.

Table 2 Application of Gyro Drift Rate

Time	Gyro	Correction for drift rate	Correction to gyro	Datum Gyro Bearing	Correction for Variation diff	Final Datum
1010	273.25º	– 0.06º	+ 087.43º	000.68º	+ 0.4º	001.08º
1015	001.85º	– 0.10º	+ 087.39º	089.24º	0	089.24º
1018	094.00º	– 0.12º	+ 087.37º	181.37º	– 0.3º	181.07º
1022	182.20º	– 0.14º	+ 087.35º	269.55º	– 0.1º	269.45º

8. **The Accuracy of the Datum Bearings**. Some factors affecting the accuracy of each type of datum bearing are mentioned below:

a. **Inertial Datum**. The inertial system offers high accuracy of true heading readings. Variation must then be applied, to give a magnetic datum.

b. **Gyro Datum**. The calculations shown in Tables 1 & 2 assume that the gyro drift rate is constant, which may not be the case. Also, the possibility of errors due to change of variation with height has not been eliminated. Ignoring the variation factor, the quality of the gyro will dictate the accuracy of the datum headings.

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 considerations include:

a. Air Traffic Control considerations, for area of operation and flight patterns.

b. Selection of a suitable height to fly, bearing in mind the avoidance of turbulence.

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 designated area, and measure magnetic heading of the aircraft. Record the heading, the gyro reading and the time.

c. Take-off and climb to the operating 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. Land at base, and obtain the new gyro correction; then calculate the gyro 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 MOD Form 712A.

h. Calculate and correct for coefficients A, B and C, as for the ground correcting swing.

i. 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 0.5º. 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 0.5º, the calibration swing may proceed as follows:

j. Obtain the readings every 30º. The aircraft may be flown on chords to a circle roughly centred on the airfield.

k. After landing, re-assess the gyro drift rate.

l. Complete the MOD Form 712A and the appropriate deviation card.

11. **Airborne Swings against Inertial/Other Datums**. Airborne swings against other datums differ from the procedure in para 10 in that it is not necessary to spend time on the ground to determine the

gyro corrections and drift. The swing will also only require one flight if the coefficients are less than 0.5º.

CHAPTER 18 - A REFINED SWING ON A CLASS 2 BASE

Introduction

1. Under normal circumstances, a refined swing can only be carried out on a Class 1 compass base (MAA Manual of Maintenance and Airworthiness Processes (MAP-01), Chapter 12.9), ie a compass base where the known magnetic anomalies are less than ± 0.1º. However, it is possible to carry out a refined swing on a Class 2 base (i.e. one where any anomalies are less than ± 0.25º) using a special procedure involving the use of two Watts Datum Compasses.

Principle

2. This procedure, although it uses a Class 2 base, depends on having an area close to the base which meets the requirements of a Class 1 base. This area need only be relatively small, sufficient that it can house a Watts Datum Compass without causing magnetic interference. The area will need to be surveyed and approved by the same authorities responsible for compass bases.

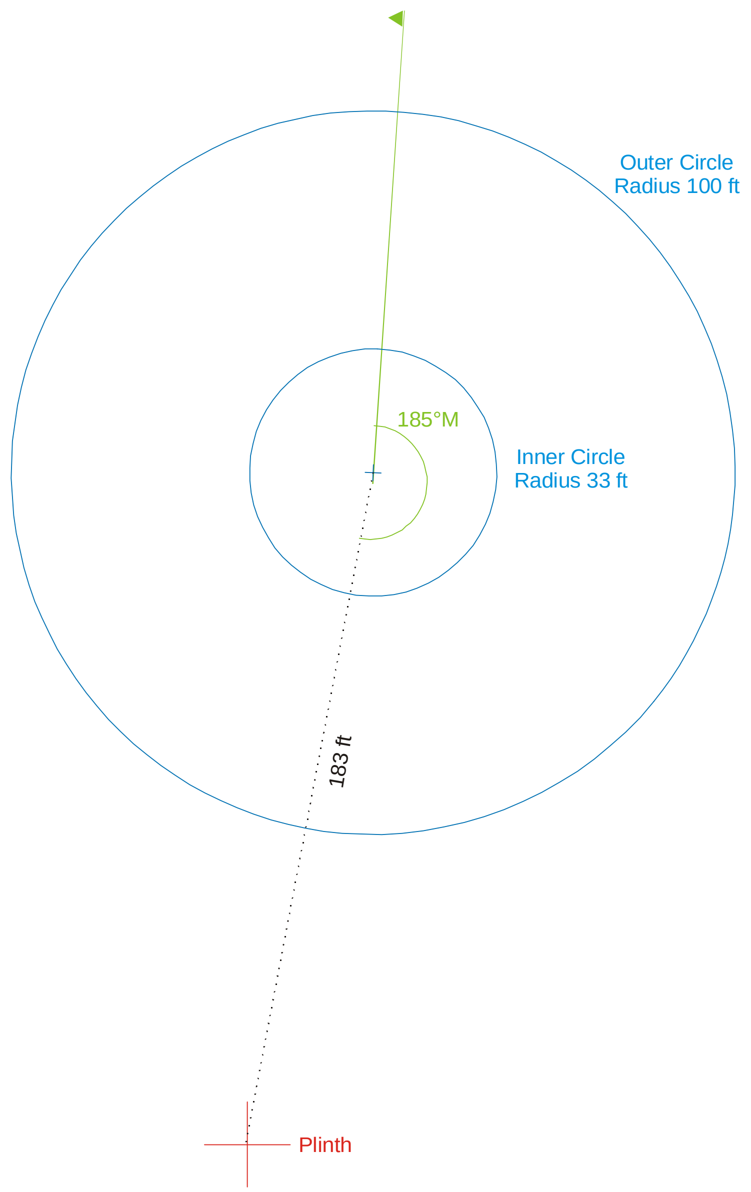
3. Having established and marked this special area (known hereafter as 'the plinth'), the principle used to carry out the swing relies on the use of relative bearings. The Watts Datum on the plinth is used as the master reference. It is aligned with magnetic north and used to take bearings on the mobile Watts Datum.

4. The mobile Watts Datum is used in the same way as for a normal swing, except that instead of aligning it with magnetic north each time it is moved, it is aligned with the plinth Watts Datum and used to take a relative bearing on the aircraft. The magnetic heading of the aircraft is computed by summating the two bearings (subtracting 360 if the sum is greater than 360), as explained later in this chapter.

Swing Procedure

5. The procedure described in the following paragraphs assumes that the class 1 plinth is to the south of the compass base. In reality, the position of the plinth in relation to the base is immaterial as the procedure can be used from any relative position. A plan view of a typical compass base is shown at Fig 1.

5-18 Fig 1 Dimensions of a Typical Compass Base



6. The Aircraft Maintenance Manual (AMM) should contain a form, similar to that shown at Fig 2, to assist in determining the magnetic heading of the aircraft. This form acts as both a checklist and a record of readings.

5-18 Fig 2 Completed Offset Bearing Swing Form

OFFSET BEARING COMPASS SWING PROCEDURE **ALIGN PLINTH WATTS DATUM WITH MAGNETIC NORTH** **(Set to 180 if sighting from behind or 360 if sighting from the front)**				
Approx Heading	**MWDC** Relative Bearing of the Aircraft A	**PWDC** Magnetic Bearing of Mobile from Plinth B	A + B	Magnetic Heading of Aircraft (If A + B > 360, subtract 360)
180	*198.8*	*339.6*	*538.4*	*178.4*
210	*245.3*	*325.1*	*570.4*	*210.4*
240	*272.1*	*327.1*	*599.2*	*239.2*
270	*289.6*	*341.5*	*631.1*	*271.1*
300	*314.2*	*343.8*	*658.0*	*298.0*
330	*333.6*	*357.9*	*691.5*	*331.5*
360	*349.7*	*010.0*	*359.7*	*359.7*
030	*011.2*	*018.8*	*030.0*	*030.0*
060	*031.35*	*030.75*	*062.1*	*062.1*
090	*049.0*	*041.3*	*090.3*	*090.3*
120	*076.8*	*040.9*	*117.7*	*117.7*
150	*115.7*	*033.8*	*149.5*	*149.5*

Note: In this example, the aircraft is being sighted from the front and the PWDC will be set to 360º.

7. The procedure to be used is as follows:

a. Position the Plinth Watts Datum Compass (PWDC) on the plinth and align with magnetic north. Set the zero to 180º if the aircraft is being sighted through the tail, or to 360º if it is being sighted through the nose (the AMM will specify which method is to be used).

b. Position the aircraft on the required heading (see the AMM for details).

c. Position the Mobile Watts Datum Compass (MWDC) on the outer circle of the compass base, in line with the centreline of the aircraft (either nose or tail, as detailed in the AMM).

d. Align the MWDC with the PWDC, i.e. make the '000' graticule point to the PWDC. Now sight the aircraft and, using the compass scale, read off the relative bearing of the aircraft from the MWDC. This is reading 'A'. Record this reading on the form.

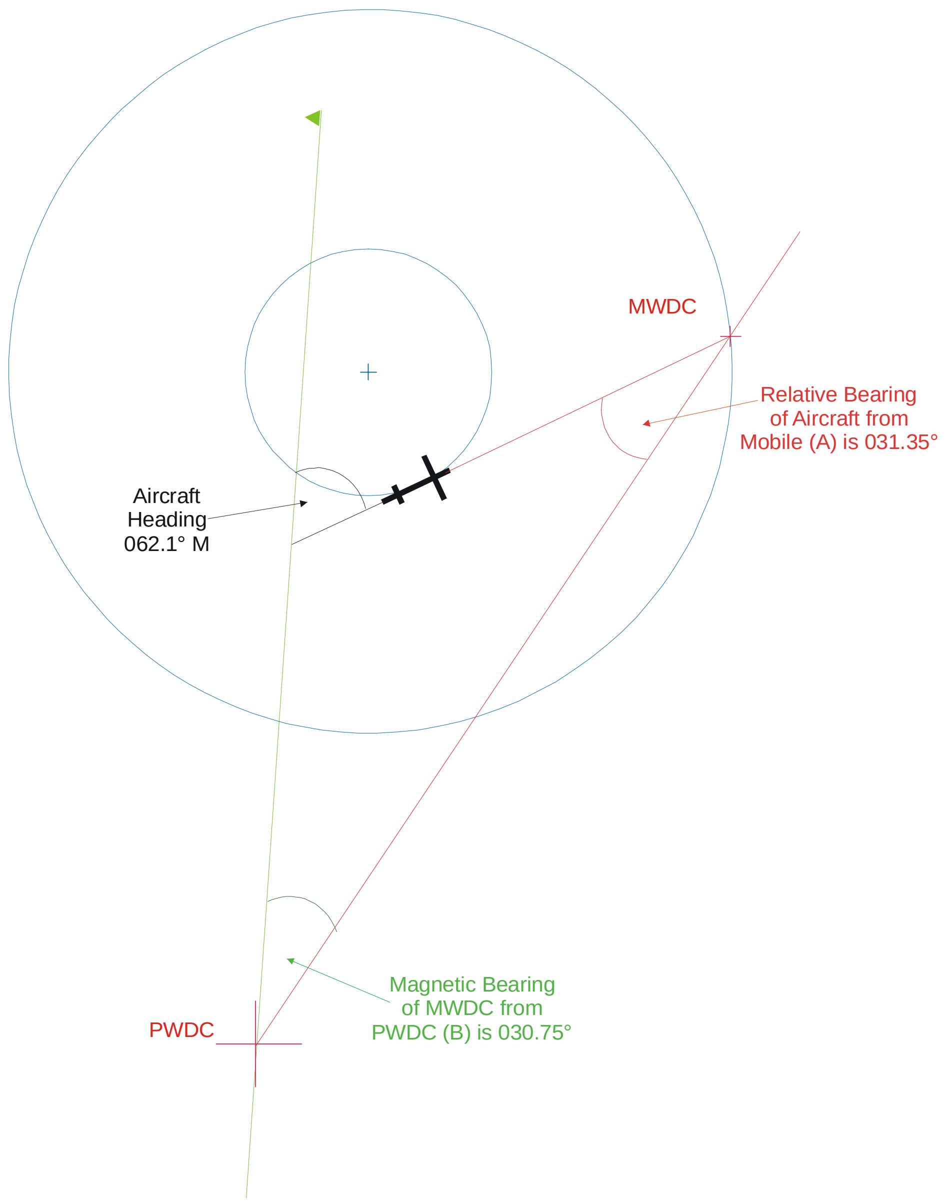
e. At the PWDC, take a bearing on the MWDC. This is reading 'B'. Record this reading on the form.

f. Calculate the aircraft magnetic heading by adding together readings 'A' and 'B'. If the resultant is greater than 360, then subtract 360 to arrive at the correct result (see Fig 2). The aircraft heading thus obtained will be used as the entry argument on the compass calibration proforma.

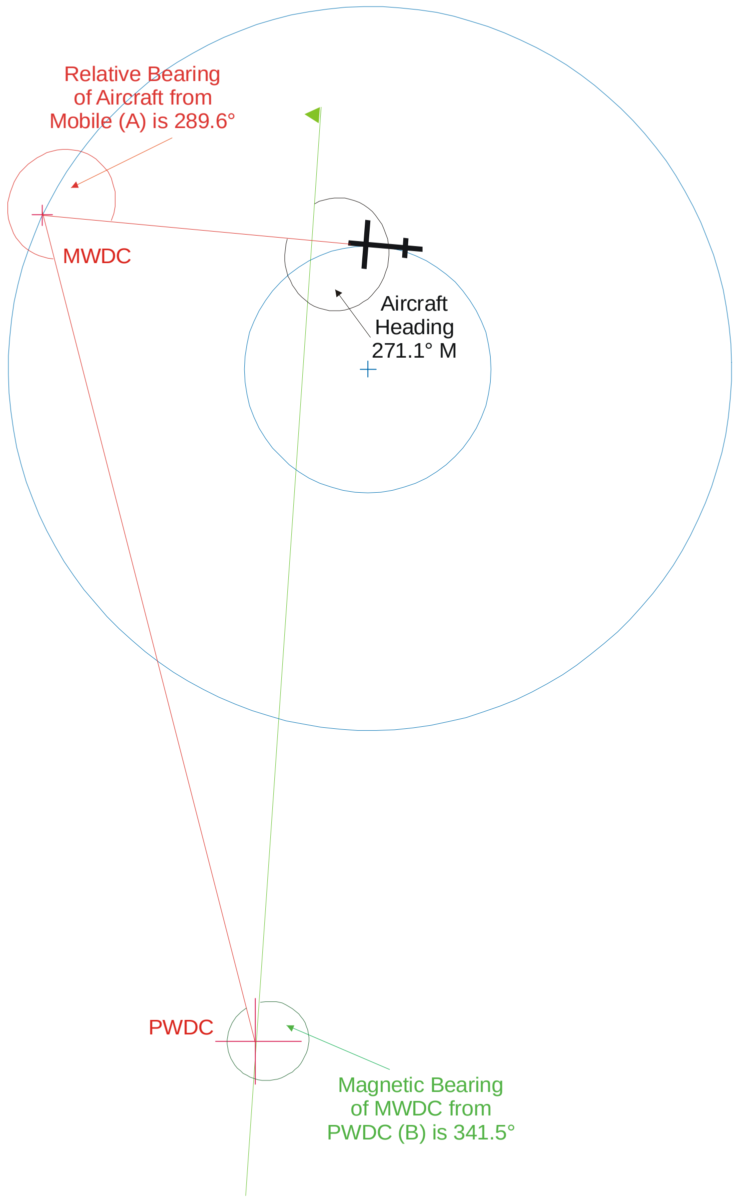
g. Repeat steps c to f each time the heading of the aircraft is changed.

8. Figure 3 shows the observed bearings from the two Watts Datum compasses with the aircraft heading 060º. A similar diagram, with the aircraft on a heading of 270º is shown at Fig 4.

5-18 Fig 3 Observed Bearings from Watts Datum Compasses with Aircraft Heading 060º



5-18 Fig 4 Observed Bearings from Watts Datum Compasses with Aircraft Heading 270º



CHAPTER 19 - THE ANALYSIS OF THE COMPASS SWING

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. Volume 5, Chapter 16 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 Volume 5, Chapter 15 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:

$$A = \frac{\Sigma\delta}{n} \qquad B = \frac{2\Sigma\delta \sin\ \theta}{n}$$

$$C = \frac{2\Sigma\delta \cos\ 2\theta}{n} \qquad D = \frac{2\Sigma\delta \sin\ 2\theta}{n}$$

$$E = \frac{2\Sigma\delta \cos\ 2\theta}{n}$$

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 the Compass Calibration Log which is used for the Fourier Analysis. For convenience these values have been extracted and are listed at Table 1.

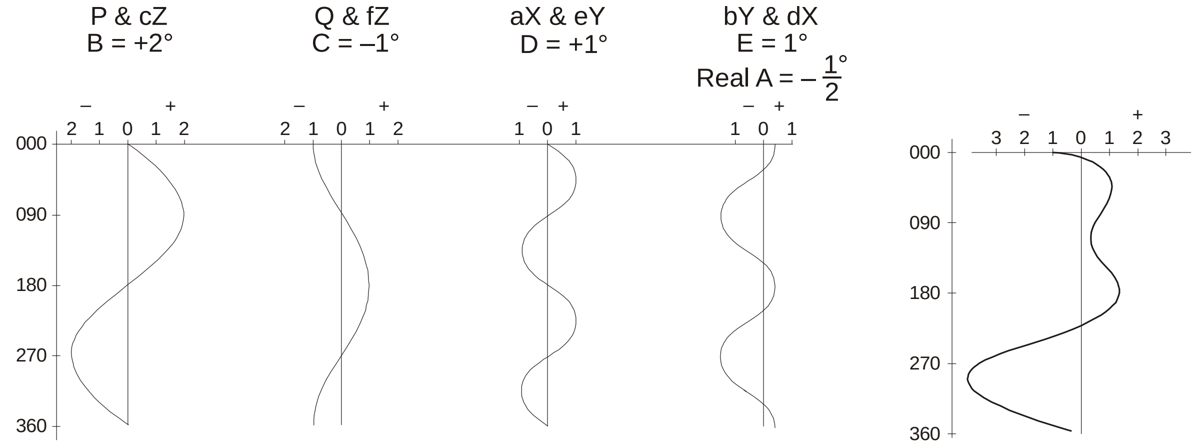
Table 1 Values of Functions of theta

Hdg (θ) a	sin θ b	cos θ c	sin 2θ d	cos 2θ e
0	0	+1.00	0	+1.00
30	+0.50	+0.87	+0.87	+0.50
60	+0.87	+0.50	+0.87	–0.50
90	+1.00	0	0	–1.00
120	+0.87	–0.50	–0.87	–0.50
150	+0.5	–0.87	–0.87	+0.50
180	0	–1.00	0	+1.00
210	–0.50	–0.87	+0.87	+0.50
240	–0.87	–0.50	+0.87	–0.50
270	–1.00	0	0	–1.00
300	–0.87	+0.50	–0.87	–0.50
330	–0.50	+0.87	–0.87	+0.50

4. **Observed Deviations**. At Fig 1b is the total deviation curve derived from the component curves at Fig la. From Fig lb, the observed deviations every 30º, starting at 0º, are: –0.5º, +1.1º, +1.0º, +0.5º, +0.3º, +1.0º, +1.5º, +0.6º, –1.4º, –3.5º, –3.9º and –2.6º. These deviations are used in the Fourier Analysis.

5-19 Fig 1 Deviation Graphs

Fig 1a Component Deviations **Fig 1b Total Deviation**



5. **To Calculate the Coefficients**. Table 2 is an extract of those columns of the Compass Calibration Log 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 the form, and the columns are then totalled to obtain, $\Sigma\delta$, $\Sigma\delta \sin\theta$, $\Sigma\delta \cos\theta$, $\Sigma\delta \sin 2\theta$ and $\Sigma\delta \cos 2\theta$. Dividing column 2 by n, and columns 7, 10, 13 and 16 by $\frac{n}{2}$, gives the calculated coefficients:

A = – 0.49, B = + 1.97, C = – 0.93, D = + 0.94, E = + 1.01.

Table 2 The Derived Coefficients

Hdg (θ) **1**	**Observed Deviation (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

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 Volume 5, Chapter 15 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 the Compass Calibration Log 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.

Table 3 The Calculated Deviations

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

Summary of the Fourier Analysis

8. Any periodic function (the compass swing period is 2π) 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 the Compass Calibration Log (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 0.674σ, 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: 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 (i.e. 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: $\varepsilon_{B\text{ to }E} = \varepsilon\sqrt{\frac{2}{n}}$, or $1.4\varepsilon_A$.

Statistical Analysis of the Swing

11. Fig 2 shows a completed form for the swing used in the Fourier Analysis. Column 4 is D, Column 5 is D^2. Thus, for the figures used:

$$\varepsilon = \pm 0.05^\circ$$

$$\varepsilon_A = \pm 0.014^\circ \quad \text{i.e. } A = -0.49 \pm 0.014^\circ$$

$$\varepsilon_{B\text{ to }E} = \pm 0.02^\circ \quad \text{i.e. B, C, D and E are within } \pm 0.02^\circ\ \varepsilon \text{ of their stated figures.}$$

12. **The Meaning of the Probable Errors**. The figure for ε of ± 0.05º means that any single observed deviation has an evens chance of being within .05º of the calculated deviation, and one would therefore expect half of the differences to be within ± 0.05º. Column 4 confirms this. The coefficient's probable errors provide a means of comparing one compass swing with another form of correlation test.

Further Applications of Statistics

13. Fig 3 also shows a completed form. No observed deviation differs from the next by more than 1º - 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 1.5º - 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 ε = ± 0.20, and coefficients which can be corrected are less than the accepted maximum of 0.5º.

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:

Table 4 The Effect of Carriage of Stores

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	
$\varepsilon_{\text{B to E}}$ =	± 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:

$$\varepsilon_D = \sqrt{{\varepsilon_1}^2 + {\varepsilon_2}^2}$$

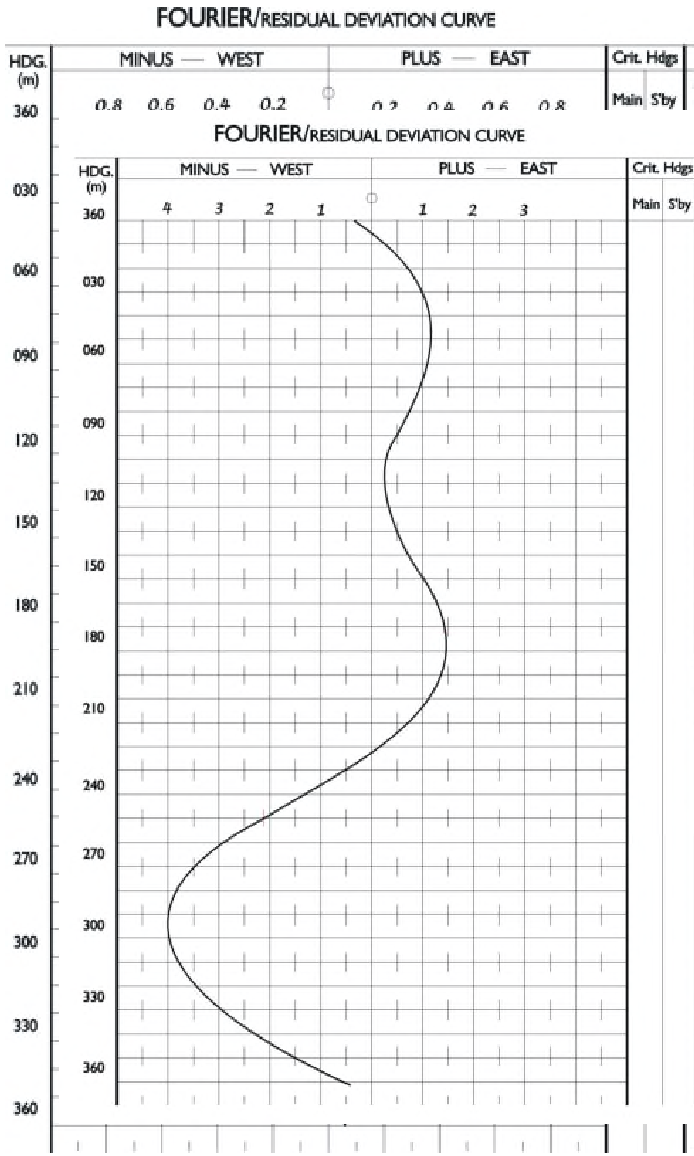
To use the figures shown:

$$\varepsilon_D = \pm\sqrt{0.114^2 + 0.102^2}$$

$$= \pm 0.153^{\circ}$$

This figure becomes significant at the 3ε level i.e. 0.459º. Therefore, it can be said that the bombs have no effect on the aircraft's magnetism because no difference exceeds 0.459º, and there is a better than 19 to 1 chance of being right.

5-19 Fig 2 Compass Calibration Log – Example 1



FOURIER ANALYSIS

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18
COMP HDG.	DEV. OBS.	DEV. CALC.	DIFF.	DIFF. SQR'D	A	d_o (Col. 2)	B	Sin Ⓗ	d_o (Col. 2)	C	Cos Ⓗ	d_o (Col. 2)	D	Sin 2Ⓗ	d_o (Col. 2)	E	Cos 2Ⓗ
Ⓗ	$\pm d_o$	$\pm d_c$	$d_o - d_c$	COL 4 SQRD.		× Sin Ⓗ			× Cos Ⓗ			× Sin 2Ⓗ			× Cos 2Ⓗ		
360	−0.50	0.41	−0.09	.008	−0.49			0	−0.50	−0.93	+1.0			0	−0.50	+1.01	+1.0
030	+1.10	+1.01	+0.09	.008	−0.49	+0.55	+0.99	+0.5	+0.96	−0.81	+0.87	+0.96	+0.82	+0.87	+0.55	+0.50	+0.5
060	+1.00	+1.07	−0.07	.005	−0.49	+0.87	+1.71	+0.87	+0.50	−0.47	+0.5	+0.87	+0.82	+0.87	−0.50	−0.50	−0.5
090	+0.50	+0.47	+0.03	.001	−0.49	+0.50	+1.97	+1.0			0			0	−0.50	−1.01	−1.0
120	+0.30	+0.37	−0.07	.005	−0.49	+0.26	+1.71	+0.87	−0.15	+0.47	−0.5	−0.26	−0.82	−0.87	−0.15	−0.50	−0.5
150	+1.00	+0.99	+0.01	0	−0.49	+0.50	+0.99	+0.5	−0.87	+0.81	−0.87	−0.87	−0.82	−0.87	+0.50	+0.50	+0.5
180	+1.50	+1.45	+0.05	.003	−0.49			0	−1.50	+0.93	−1.0			0	+1.50	+1.01	+1.0
210	+0.60	+0.65	−0.05	.003	−0.49	−0.30	−0.99	−0.5	−0.52	+0.81	−0.87	+0.52	+0.82	+0.87	+0.30	+0.50	+0.5
240	−1.40	−1.41	+0.01	0	−0.49	+1.22	−1.71	−0.87	+0.70	+0.47	−0.5	−1.22	+0.82	+0.87	+0.70	−0.50	−0.5
270	−3.50	−3.47	−0.03	.001	−0.49	+3.50	−1.97	−1.0			0			0	+3.50	−1.01	−1.0
300	−3.90	−3.99	+0.09	.008	−0.49	+3.39	−1.71	−0.87	−1.95	−0.47	+0.5	+3.39	0.82	−0.87	+1.95	−0.50	−0.5
330	−2.60	−2.61	+0.01	0	−0.49	+1.30	−0.99	−0.5	−2.26	−0.81	+0.87	+2.26	−0.82	−0.87	−1.30	+0.50	+0.5
Sums	−5.90	−5.88	−0.02	.042	−5.88	+11.79			−5.56			+5.65			+6.05		
Divi.	12					6			6			6			6		
Coeff.	A = ± 0.49					B = ± 1.97			C = ± 0.93			D = ± 0.94			E = ± 1.01		

Notes:

1. Complete col. 2 from the calibration 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 sin Ⓗ, cos Ⓗ, sin 2Ⓗ and cos 2Ⓗ 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 sin Ⓗ, cos Ⓗ, sin 2Ⓗ and cos 2Ⓗ.
7. Line by line for each heading 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 0.050   50% A Error 0.014   50% B - E Error 0.020

Calculated Coefficients

A	B	C	D	E
−0.49	+1.97	−0.93	+0.94	+1.01

Comments

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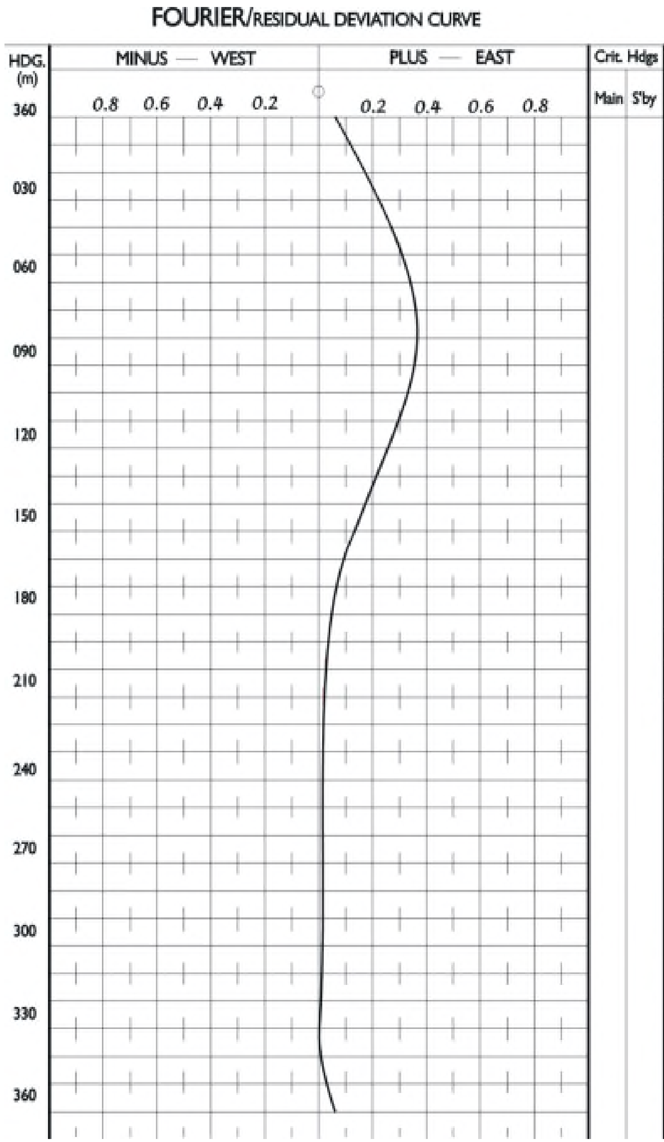
2-2-4 Fig 2 Compass Calibration Log - Example 1

5-19 Fig 3 Compass Calibration Log – Example 2



FOURIER ANALYSIS

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18
COMP HDG.	DEV. OBS.	DEV. CALC.	DIFF.	DIFF. SQR'D	A	d_o (Col. 2)	B	Sin (H)	d_o (Col. 2)	C	Cos (H)	d_o (Col. 2)	D	Sin 2(H)	d_o (Col. 2)	E	Cos 2(H)
(H)	$\pm d_o$	$\pm d_c$	$d_o - d_c$	COL 4 SQRD.		× Sin (H)			× Cos (H)			× Sin 2(H)			× Cos 2(H)		
360	− 0.2	+ 0.06	− 0.26	.068	+ 0.12			0	−0.20	+ 0.01	+ 1.0			0	−0.20	− 0.07	+ 1.0
030	− 0.2	+ 0.20	− 0.40	.160	+ 0.12	− 0.10	+ 0.08	+ 0.5	− 0.17	+ 0.01	+ 0.87	− 0.17	+ 0.03	+ 0.87	− 0.10	− 0.04	+ 0.5
060	+ 0.8	+ 0.33	+ 0.47	.221	+ 0.12	+ 0.69	+ 0.14	+ 0.87	+ 0.40	0	+ 0.5	+ 0.69	+ 0.03	+ 0.87	−0.40	+ 0.04	− 0.5
090	+ 0.7	+ 0.35	+ 0.35	.123	+ 0.12	+ 0.70	+ 0.16	+ 1.0			0			0	−0.70	+ 0.07	− 1.0
120	− 0.2	+ 0.27	− 0.47	.221	+ 0.12	−0.17	+ 0.14	+ 0.87	+ 0.10	0	− 0.5	+ 0.17	−0.03	− 0.87	+ 0.20	+ 0.04	− 0.5
150	− 0.3	+ 0.12	−0.42	.176	+ 0.12	− 0.15	+ 0.08	+ 0.5	+ 0.26	− 0.01	− 0.87	+ 0.26	−0.03	− 0.87	−0.30	− 0.04	+ 0.5
180	+ 0.6	+ 0.04	+ 0.56	.314	+ 0.12			0	−0.60	− 0.01	− 1.0			0	+ 0.60	−0.07	+ 1.0
210	+ 0.4	+ 0.02	+ 0.38	.144	+ 0.12	− 0.20	−0.08	− 0.5	−0.35	−0.01	− 0.87	+ 0.35	+ 0.03	+ 0.87	+ 0.20	−0.04	+ 0.5
240	− 0.6	+ 0.05	−0.65	.423	+ 0.12	+ 0.52	−0.14	− 0.87	+ 0.30	0	− 0.5	−0.52	+ 0.03	+ 0.87	+ 0.30	+ 0.04	− 0.5
270	− 0.3	+ 0.03	−0.33	.109	+ 0.12	+ 0.30	−0.16	− 1.0			0			0	+ 0.30	+ 0.07	− 1.0
300	+ 0.7	−0.01	+ 0.71	.504	+ 0.12	−0.61	− 0.14	− 0.87	+ 0.35	0	+ 0.5	− 0.61	−0.03	− 0.87	− 0.35	+ 0.04	− 0.5
330	0	−0.02	+ 0.02	0	+ 0.12	0	− 0.08	− 0.5	0	+ 0.01	+ 0.87	0	− 0.03	− 0.87	0	−0.04	+ 0.5
Sums	+ 1.4	+ 1.44	0.04	2.463	+ 1.44	+ 0.98			+ 0.09			+ 0.17			−0.45		
Divi.	12					6			6			6			6		
Coeff.	A = ± 0.12					B = ± 0.16			C = ± 0.01			D = ±			E = ± 0.07		

Notes:

1. Complete col. 2 from the calibration 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 sin (H), cos (H), sin 2(H) and cos 2(H) 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 sin (H), cos (H), sin 2(H) and cos 2(H).
7. Line by line for each heading 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 ± 0.40 50% A Error ± 0.12 50% B - E Error ± 0.17

Calculated Coefficients

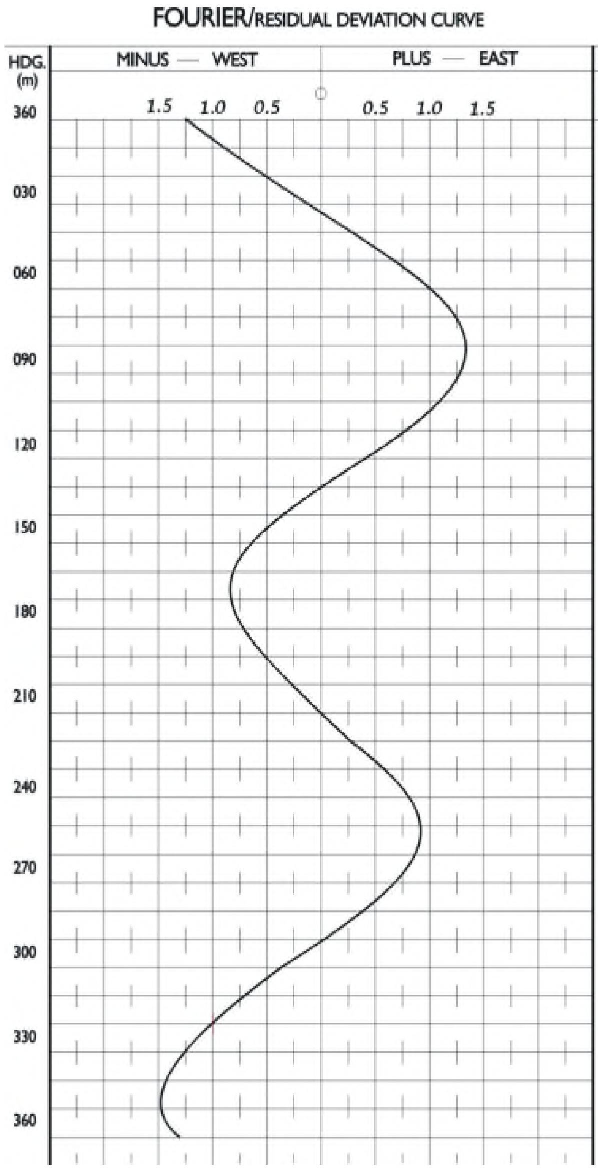
A	B	C	D	E
+ 0.12	+ 0.16	+ 0.01	+ 0.03	−0.07

Comments

Checked by

2-2-4 Fig 3 Compass Calibration Log - Example 2

5-19 Fig 4 Compass Calibration Log – Example 3



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FOURIER ANALYSIS

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18
COMP HDG.	DEV. OBS.	DEV. CALC.	DIFF.	DIFF. SQR'D	A	d_o (Col. 2)	B	Sin Ⓗ	d_o (Col. 2)	C	Cos Ⓗ	d_o (Col. 2)	D	Sin 2Ⓗ	d_o (Col. 2)	E	Cos 2Ⓗ
Ⓗ	$\pm d_o$	$\pm d_c$	$d_o - d_c$	COL 4 SQRD.		× Sin Ⓗ			× Cos Ⓗ			× Sin 2Ⓗ			× Cos 2Ⓗ		
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	.035	−0.03	0	+0.15	+0.5	0	−0.24	+0.87	0	+0.44	+0.87	0	−0.51	+0.5
060	+1.00	+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 = ± 0.3			C = ± 0.28			D = ± 0.51			E = ± 1.02		

Notes:

1. Complete col. 2 from the calibration 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 sinⒽ, cosⒽ, sin 2Ⓗ and cos 2Ⓗ 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 sinⒽ, cosⒽ, sin 2Ⓗ and cos 2Ⓗ.
7. Line by line for each heading 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

2-2-4 Fig 4 Compass Calibration Log - Example 3

CHAPTER 20 - TURN AND SLIP INDICATOR

Introduction

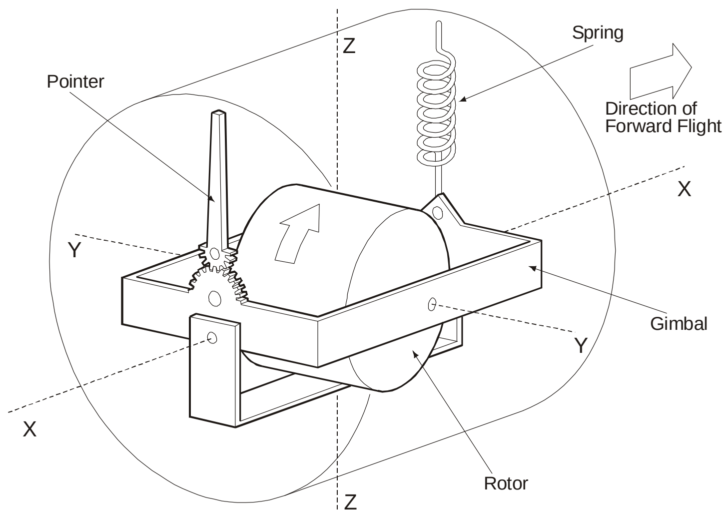
1. The turn and slip indicator comprises two instruments in the same case. The turn indicator is used to indicate the rate at which the aircraft changes heading; a rate one turn (of 3 degrees per second) is the standard turn during procedural instrument flying. The instrument is also invaluable while spinning; in a spin, it always indicates the direction of yaw. The slip indicator shows whether or not corrective rudder is required to achieve balanced flight.

THE TURN INDICATOR

Construction

2. The construction of a basic turn indicator is illustrated in Fig 1, where the X (roll), Y (pitch), and Z (yaw) gyro axes are shown. The instrument consists of a rate gyro mounted with its spin axis arthwartships, and with only one gimbal, such that it has freedom in roll only. This freedom is, however, limited by a restraining spring connecting the gimbal to the outer casing. Movement of the gimbal is transmitted to a pointer on the instrument face via a reverse gearing. This gearing is so arranged that gimbal tilt to the right causes the pointer to move to the left and vice versa. The gimbal is damped, and gimbal stops prevent instrument damage at high turn rates. The scale is non-linear, the calibrations representing standard rate turns (Rate 1, 180º per min; Rate 2, 360º per min; Rate 3, 540º per min). As gyro speed is critical to accuracy, any change is sensed by centrifugal switches which control the DC motor.

3-16 Fig 1 Turn Indicator - Simplified Construction



Principle of Operation

3. Consider, in respect of Fig 1, a banked turn to the left. When the instrument casing rotates around the X-axis, the gyro's rigidity causes it to remain spatially fixed, extending the spring which exerts an anti-clockwise torque. This torque in turn produces a secondary (or indirect) precessionary force about the Z-axis in the direction in which the aircraft is starting to turn. In fact, the gyroscope has no freedom to move independently about its Z-axis, but a state of equilibrium will be reached when the rate of turn of the aircraft equals the rate of secondary precession induced in the gyroscope. If the aircraft rate of turn becomes faster than the secondary precession rate of the gyroscope, primary (or direct) bearing-induced precession will tilt the gyro further to the right with respect to the casing, so increasing the spring tension and causing increased secondary precession until a balance is once again restored. Conversely, if the

aircraft rate of turn decreases, primary bearing-induced precession will tilt the gyro to the left with respect to the casing, reducing spring tension and reducing the rate of secondary precession until it matches the aircraft turn rate. At equilibrium the tilt angle of the gyroscope within its casing is related to the aircraft rate of turn and the dial can be calibrated accordingly.

Errors

4. **Pitching Error**. The turn indicator measures the rate of turn about the yaw axis. Movement about the pitch axis (which usually coincides with the gyro spin axis) would not normally produce any gyro precession. However, if the aircraft is simultaneously yawed and pitched nose-up, gyro axis cross-coupling (see Volume 1, Chapter 8) can cause a torque to be applied such that the indicated rate of turn will exceed the true rate of turn. The error is dependent on the rate of yaw and the rate of pitch, and in some circumstances can cause full-scale deflection of the indicator. The error is unlikely to be corrected until the rate of pitch is significantly reduced, consequently it may continue to indicate a turn in one direction while the aircraft turns in the other. Conversely, with nose-down pitch, the instrument will under-read, and in extreme cases may indicate zero regardless of actual turn rate.

5. **Gyro Speed Errors**. The angle of tilt (θ) of a rate gyro is given by:

$$\theta = \text{Rate of turn} \times I\omega$$

where I = Moment of inertia of the gyro
and ω = Angular velocity of the gyro

Thus, if the angular velocity (rotor speed) of the gyro is altered, a different angle of tilt is generated by the same rate of turn. An over-speeding gyro is uncommon because of the speed governing system, but electrical faults or excessive bearing friction may produce under-speeding. This will be manifested as under-reading and an oversensitive, badly damped, needle.

THE SLIP INDICATOR

Operation

6. The slip indicator consists of a ball mounted in a curved, clear tube filled with a damping liquid. When the aircraft is in straight, balanced flight the only force acting on the ball is gravity, and the ball will be in the centre of the tube (Fig 2a).

7. If the aircraft is in straight but unbalanced flight, the pilot will be countering the rudder-induced yaw with opposite bank. The tube is tilted with respect to the outside world and gravity takes the ball to the lowest position of the tube (Fig 2b).

3-16 Fig 2 Operation of Slip Indicator

Fig 2a Straight & Balanced Flight

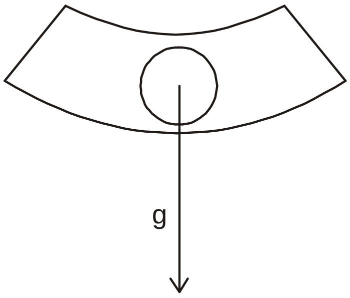


Fig 2b Side-slip

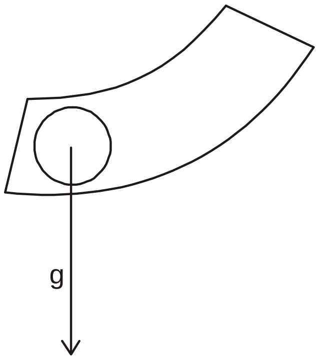
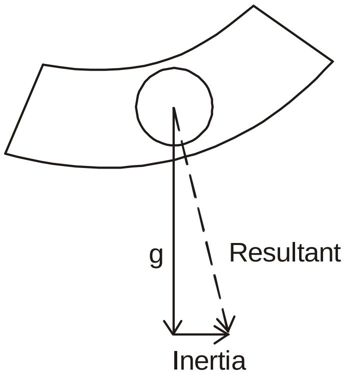


Fig 2c Balanced Turn



8. In a properly balanced turn, inertia acts on the ball in addition to gravity, and the resultant of these two forces causes the ball to remain in the centre of the tube (Fig 2c). If the aircraft is slipping inwards (ie the relative airflow is coming from the inside of the turn), the ball will be displaced towards the low wing. Conversely, if the aircraft is skidding outwards (with the relative airflow coming from the outside of the turn), the ball will be displaced towards the high wing. In all cases, the corrective action is to move the feet to 'kick' the ball back to the centre (i.e. if the ball is to the right of the tube, more right rudder is needed).

Summary of Turn and Slip Indications

9. Fig 3 summarizes various situations that the turn and slip indicator can show, together with the correct sense of rudder movement required.

3-16 Fig 3 Indications of Turn and Slip in Flight

Rate 2 Right Turn
Balanced

No Turn
Balanced

Rate 1 Left Turn
Balanced

Rate 2 Left Turn
with Skid
More Right Rudder

Rate 2 Right Turn
with Slip
More Right Rudder

Rate 2 Right Turn
with Skid
More Left Rudder

CHAPTER 21 - ATTITUDE INDICATORS

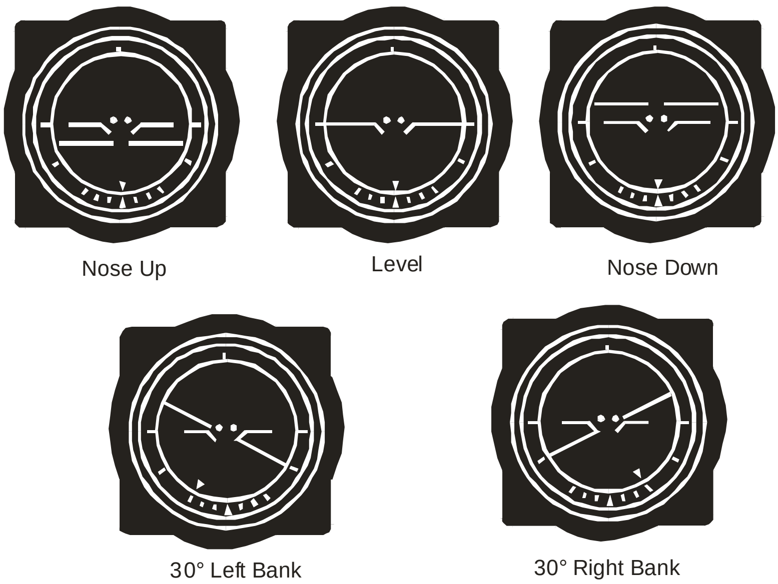
Introduction

1. In order for an aircraft to be flown accurately and safely, without reference to the natural horizon, some form of attitude reference is required. This reference may be provided either by a direct reading attitude indicator (or artificial horizon), or by displays driven by outputs from other aircraft equipment such as inertial systems. This chapter deals with the self-contained, gyro based, direct reading instruments.

2. The principle component of an attitude indicator is an air or electrically driven displacement gyro with its spin axis maintained vertical to the Earth by gravity sensing devices. Latest attitude indicators include the elements of the Turn and Slip indicator (see Volume 5, Chapter 20) within the instrument case.

3. Indication of pitch and bank attitude may be presented in one of two ways. In older instruments (the artificial horizon), the aircraft is represented by a fixed symbol and the horizon by a bar stabilized parallel to the Earth's surface. In more modern displays (the attitude indicator), the horizon bar is replaced by a moving 'ball' marked with a horizon line and with graduated pitch angle markings. The areas of the ball above and below the horizon are typically coloured blue and brown respectively. In both cases, supplementary indication of bank angle is presented by the position of a gyro stabilized pointer against a fixed bank angle scale at the bottom of the display (see the Roll Pointer and Roll Scale in Fig 2a). Some attitude indicators have the Roll Pointer and Roll Scale at the top of the instrument (see Fig 2b). This arrangement is known as 'Skypointer', since the Roll Pointer points towards the sky, and thus gives the direction of roll required to regain wings level flight. Fig 1 shows an artificial horizon in various attitudes; Figs 2a and 2b illustrate attitude indicators (note the integrated Turn and Slip indication).

5-21 Fig 1 Artificial Horizon Indicators



5-21 Fig 2 Attitude Indicators

Fig 2a Attitude Indicator

(Roll information at the bottom of the instrument)

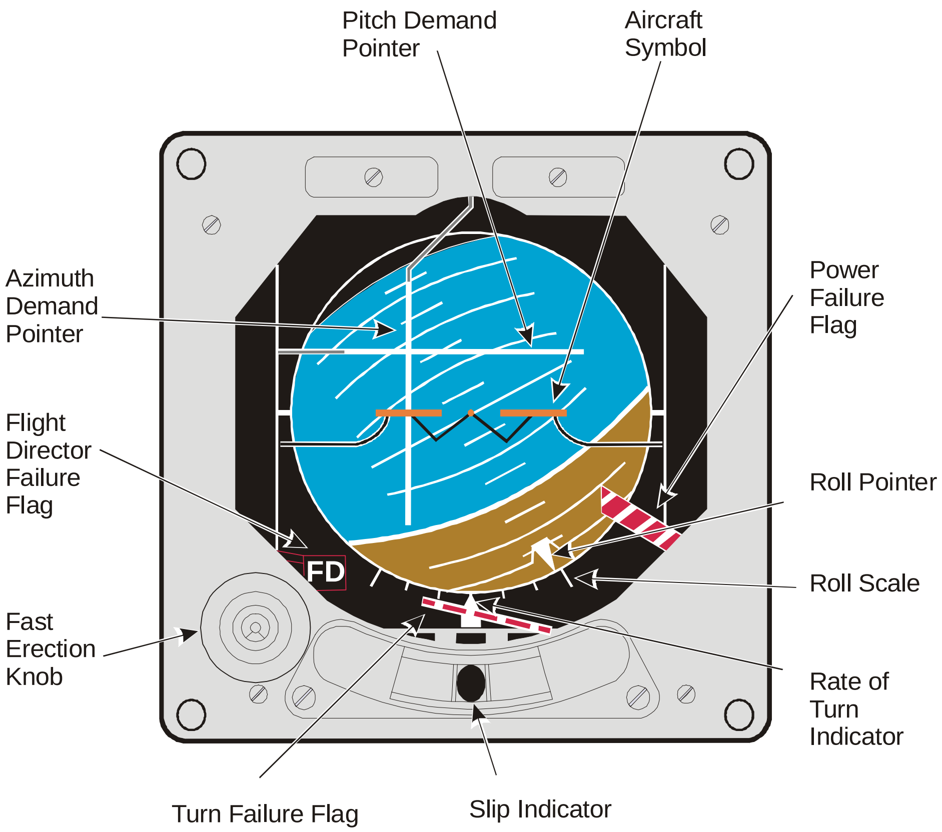
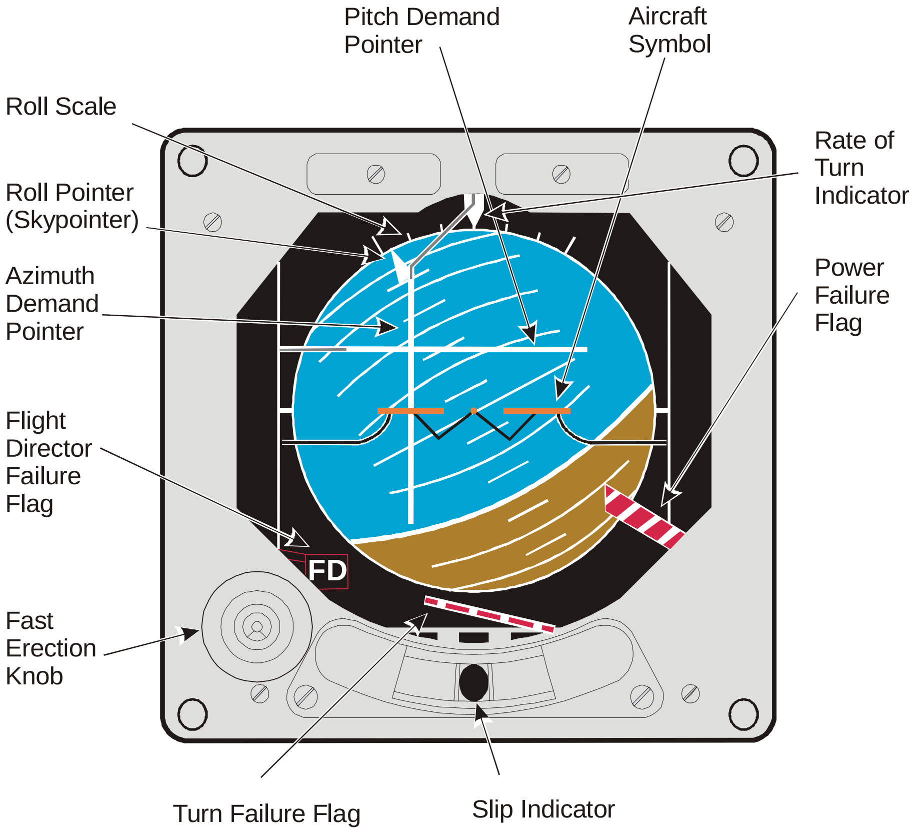


Fig 2b Attitude Indicator

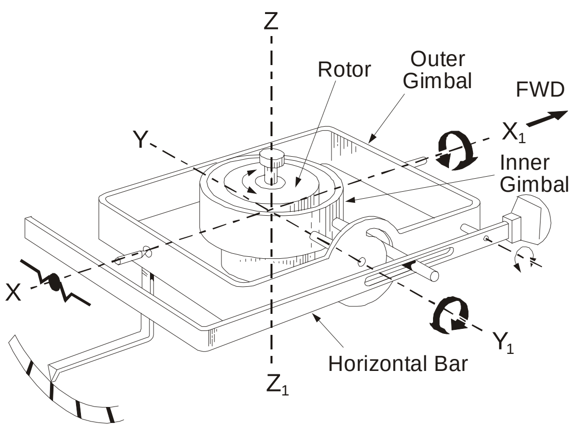
(Roll information (Skypointer) at the top of the instrument)



Principle of Operation

4. The principle of operation of both instruments is similar. Fig 3 shows the arrangement of the artificial horizon gyro and its gimbals. The inner gimbal forms the rotor casing and is pivoted to the outer gimbal ring parallel to the aircraft's pitch axis (YY_1). The outer gimbal is pivoted to the front and rear of the instrument case parallel to the aircraft's roll axis (XX_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. The aircraft symbol is fixed to the front glass of the instrument.

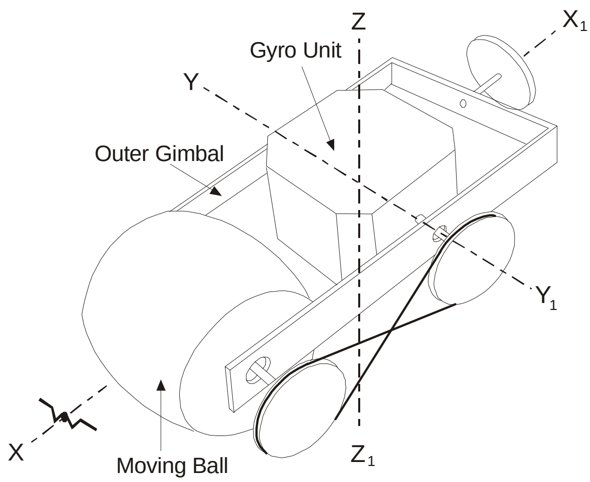
5-21 Fig 3 Artificial Horizon Principle



5. Any change in pitch attitude will result in the instrument case and the outer gimbal rotating around the YY_1 axis of the gyro. A pin attached to the gyro housing moves in a slot in the horizon bar, producing the correct sense of horizon bar movement, relative to the aircraft symbol. The major drawback to this arrangement is that it results in a non-linear scaling in pitch, with reduced sensitivity at high pitch angles. Any change in bank attitude will result in the instrument case rotating around the XX_1 axis, taking the aircraft symbol with it. In some instruments, the movements are also sensed electrically so that attitude information can be transmitted to other aircraft equipment.

6. Fig 4 shows the general construction of the attitude indicator. Bank is indicated in exactly the same manner as described for the artificial horizon, but the pitch mechanism is greatly improved - movement of the gyro unit relative to the outer gimbal producing correct sense movement of the ball by a direct drive consisting of either a wire loop (as shown), or gearwheels. Attitude indicators are therefore equally sensitive at both low and extreme pitch angles.

5-21 Fig 4 Attitude Indicator Principle



Control of the Gyro Spin Axis

7. To provide accurate indications the gyro spin axis must be maintained vertical to the Earth and therefore correcting torques must be applied to compensate for Earth rotation, transport error, and internally generated torques due to any gimbal imbalance or bearing friction.

8. The gyro spin axis is kept vertical by a pendulous system which responds to Earth's gravity and thus initiates the necessary correcting torques (which can be compensated for axis cross-coupling during turns) either mechanically or by controlling the operation of air jets or electric torque motors.

9. **Turning and Acceleration Errors**. A pendulous system responds not only to Earth's gravity but also to any acceleration force that the aircraft experiences. This force will displace the pendulous system and the torque generated by the erecting system will cause a misalignment of the spin axis to produce a verticality error. To prevent the generation of significant gyro verticality error with high acceleration, a cut-off device can be incorporated to inhibit the erecting system above a pre-determined level of horizontal acceleration. However, if accelerated flight below the cut-off limit is maintained, considerable verticality error can be built up, which may affect indications in both pitch and roll, influenced by factors such as axis cross-coupling and gyro precession. Acceleration errors cannot be fully eliminated, but they can be reduced by the use of compensating design and construction features.

10. **Fast Erection**. After extended periods of manoeuvring, the gyro may have very large verticality error or it may be toppled. To restore the gyro to its normal operating position as quickly as possible, a fast erection mechanism is fitted which applies high-rate precessing torques to erect the gyro with respect to the instrument casing. The aircraft must therefore be in unbanked, level, flight when this facility is used. It may also be selected following start-up when errors in attitude indications are apparent.

Controlled Precession

11. It is necessary that the attitude indication should be consistent and coherent over the full flight envelope of an unrestricted manoeuvrability aircraft, irrespective of how any particular attitude is achieved. For example, in inverted flight, zero pitch attitude may result from a 180º roll or from a 180º pitch manoeuvre, starting from wings level flight.

12. In both cases, the outer gimbal is required to rotate 180º relative to the airframe. During a 180º roll manoeuvre, the rotation of the airframe and instrument case around the gyro-stabilized outer gimbal provides the necessary inverted flight indication. To retain the correct attitude display during a 180º pitch manoeuvre however, a rapid 180º rotation of the outer gimbal has to occur just before passing the vertical (otherwise the outer gimbal axis would become coincident with the gyro spin axis, the condition of gimbal lock, leading to topple). The means of achieving this is known as 'Controlled Precession'.

13. Inner gimbal resilient 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 YY_1 axis, at about 85º of pitch. This causes the gyro to precess, forcing the outer gimbal to rotate. After 180º of rotation, the continuing pitch rate of the aircraft results in the inner gimbal moving away from the stop. However, if the aircraft is held in the vertical, during a stall turn or climbing roll for example, the instrument may topple.

Geometric Error

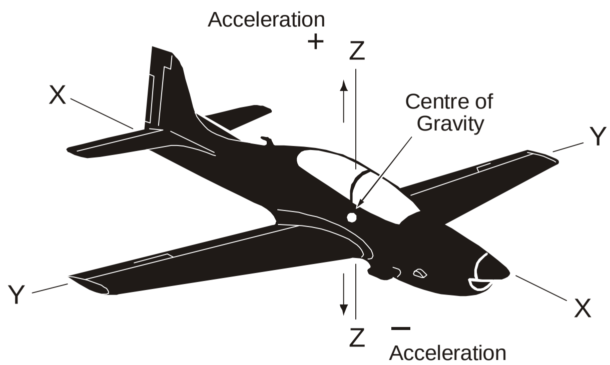
14. Many aircraft have instrument panels inclined from the vertical in normal cruise flight. Zero pitch attitude indication is restored by adjusting the linkage to the indicator to correct for the tilt. However, the inner gimbal stops are intercepted early in dive and late in climb relative to the attitude of the airframe. This geometric offset produces errors, known as geometric errors, in the displayed attitude which vary as a function of the true pitch and bank angles. If the aircraft is looped or rolled inverted, the pitch error is twice the panel tilt angle. At intermediate bank angles the geometric error in pitch increases from zero at 0º bank angle, to tilt angle at ±90º bank angle, and twice tilt angle at 180º bank angle. The geometric error in roll cannot be expressed so simply but in any case is <5º at pitch angles less than ±30º. In manoeuvres involving large pitch or bank angles geometric errors can combine with limitations in the controlled precession system to produce a significant error in the displayed attitude.

CHAPTER 22 - ACCELEROMETERS

Introduction

1. An indicating accelerometer is an instrument used in aircraft to provide a visual indication of acceleration components in the direction of the aircraft Z-axis (Fig 1). In addition, auxiliary pointers are provided which preserve a reading of the maximum and minimum accelerations sustained during any period; these can be reset as required.

5-22 Fig 1 Aircraft Axes



2. The purpose of the instrument is to indicate loadings due to manoeuvre and turbulence, so that excessive loadings may be avoided.

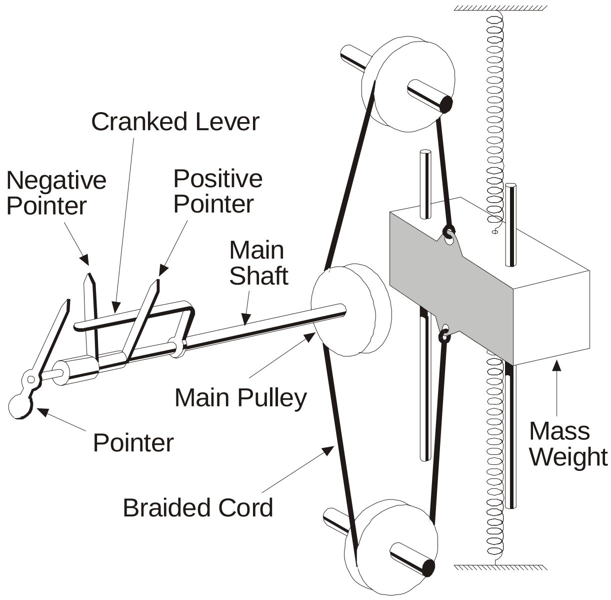
3. Although the accelerometer gives a reasonably accurate indication of the accelerations encountered in flight, indications of the instrument with respect to accelerations of extremely short duration, such as landing shocks, should be treated with caution since the accuracy under these conditions is dependent on the damping characteristics and no generalization is possible.

4. The instrument 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.

Principle of Operation

5. An accelerometer depends upon the displacement of a mass under the influence of an acceleration. Fig 2 illustrates the principle of operation, although other mechanisms may be used. The mass-weight, suspended between 2 springs, is free to move along the aircraft Z-axis and is coupled to a main shaft so that when vertical acceleration forces along the Z-axis are imposed on the mass, the main shaft is caused to rotate. The linear movement is thus converted to the rotary movement of a set of three pointers, one to indicate instantaneous acceleration and the other two to remain at the maximum indications, plus or minus, until reset manually.

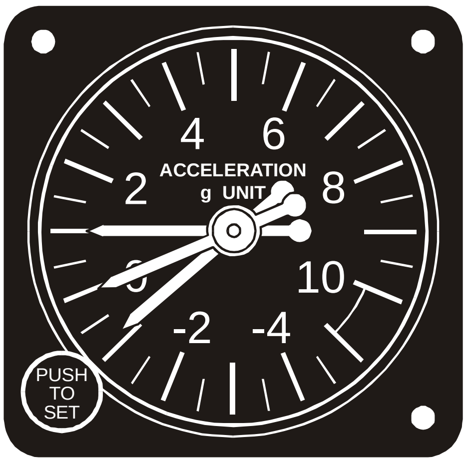
5-22 Fig 2 Accelerometer Mechanism



6. A cranked lever is attached to the shaft, and the horizontal arm of this lever is interposed between positive and 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. The recording pointers may be reset to the neutral position when desired. A device is fitted to damp out vibrations and prevent violent pointer fluctuations under short period accelerations.

7. Fig 3 shows a typical cockpit display.

5-22 Fig 3 Accelerometer Display



8. The cockpit accelerometer should not be confused with the aircraft fatigue meter. This instrument will normally be installed outside the cockpit (often in the undercarriage bay) to monitor cumulative acceleration forces on the airframe. Details of the fatigue meter may be found in Volume 5, Chapter 27, Para 15 and in Volume 1, Chapter 19, Para 12.

CHAPTER 23 - STALL WARNING AND ANGLE OF ATTACK INDICATION

Introduction

1. For any given configuration, an aircraft will stall or depart from controlled flight once a specific angle of attack (AOA) is exceeded. In straight and level flight this angle of attack will be reached at a particular airspeed for a given aircraft weight, but since there will be variations in aircraft weight both during and between flights, there is no simple correlation between airspeed and angle of attack. During manoeuvre, the situation becomes considerably more complex, and the critical angle of attack can be induced by the pilot at almost any airspeed. The airspeed indicator is therefore of limited use in warning the crew of the approach to this potentially dangerous situation, and some other means must be devised.

2. The pilot of an aerodynamically unsophisticated aircraft will usually be given warning by the onset of airframe buffet which can be felt through the control column. However, in modern, more complex, aircraft this is less likely to be the case, and moreover the situation is more difficult to recover if the limit should be exceeded. It is therefore necessary to have a system which will warn the crew of the onset of departure, either by artificially inducing buffet on the controls, or by giving some audio or visual indication, or both, once a designated AOA is exceeded.

3. Whereas a simple stall-warning device can give adequate warning to the crew, it cannot indicate the margin of safety that exists at any time. Furthermore, in high performance aircraft it is usually desirable to fly at the optimum angle of attack for any stage of flight. Such aircraft are often therefore fitted with an AOA indexer to indicate when the aircraft is flying at the optimum approach AOA regardless of aircraft weight, and this may be replaced by or supplemented with an AOA gauge to enable the aircraft to be flown efficiently during other stages of flight.

Simple Stall Warner

4. A typical simple stall warning device comprises a forward-facing vane, edge on to the airflow, mounted on the leading edge of the wing. The vane is spring loaded to the central position and in flight, the vane is held in place by air pressure when the AOA is safe but is pushed upwards when it is not. This upward movement operates a microswitch which triggers an audio or visual stall warning device in the cockpit or can be used to initiate a stick shaker or pusher.

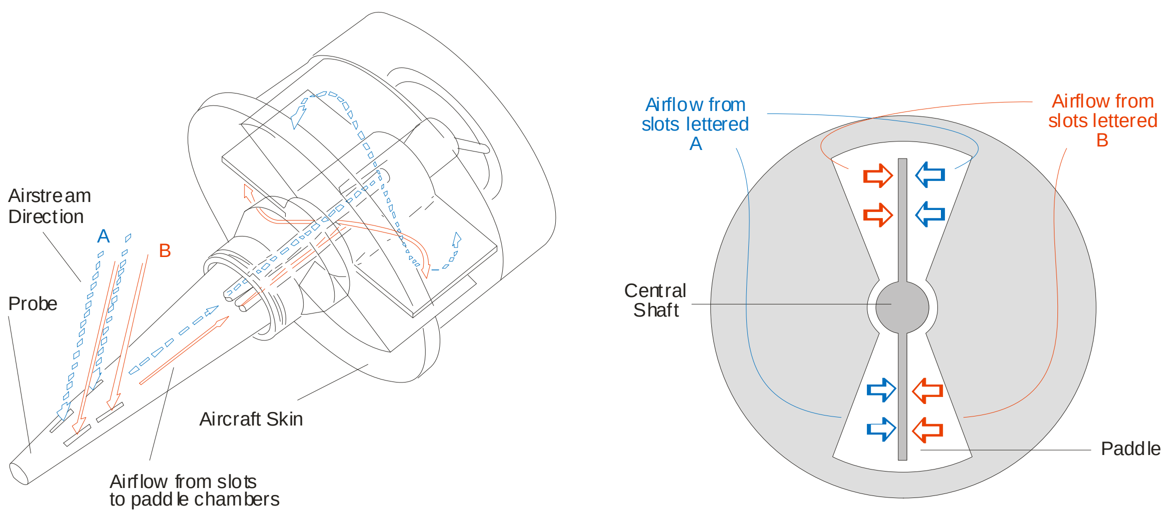
Airstream Direction Detector (ADD)

5. Where a more sophisticated system is needed, some form of airstream direction detector (ADD) is employed which measures the direction of the localized airflow striking it, and relays this information to an indexer, gauge, warning device, or any combination of these. A simple example may consist of a trailing aerofoil, mounted on the outside of the aircraft, which aligns itself with the direction of the local airflow.

6. A paddle type of ADD is illustrated in Fig 1. A cylindrical casing carries a central shaft which is free to rotate through a restricted angular range (typically 50º). The shaft protrudes through one end of the casing and through the aircraft skin to form a probe into the local airflow. Two rows of forward facing slots are cut near the outer end of the probe, and each row is connected by internal ducts to two paddle chambers located within the casing. Operating in these chambers are paddles which are attached to the central shaft. Pressure from one of the ducts acts on both paddles to induce clockwise rotation of the shaft while pressure from the other duct similarly induces anti-clockwise rotation. Thus, if the pressures in the two ducts are

equal, the probe will not rotate. This equal pressure state can only occur if the two rows of slots are equally disposed about the direction of the local airstream. In conditions of misalignment the pressure in one duct will be greater than in the other, and the paddles will be caused to rotate until the probe is once again aligned with the airstream, whence the pressures will be equalized, and the probe will stop rotating. Thus, providing that there is sufficient airflow to operate the system (typically above 50 kt), the probe will follow any changes in the direction of the local airflow.

5-23 Fig 1 Paddle Type of Airstream Direction Detector (ADD)

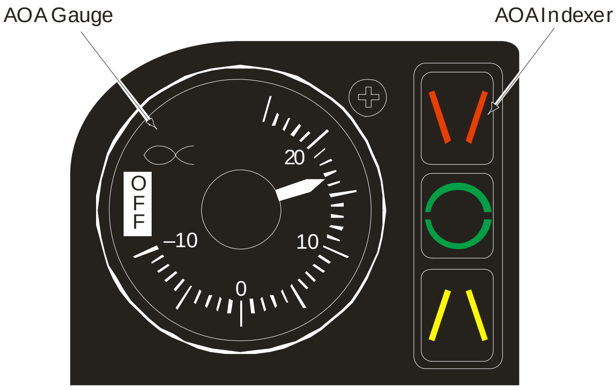


7. The position of the central shaft relative to the casing, and therefore to the aircraft, is transmitted to potentiometer assemblies by means of wiper arms, and the output voltage, which is related to AOA, may be used to operate the particular aircraft indicators or warning devices. In some installations two ADDs are provided to add a measure of redundancy, and so that the output voltage from the two can be compared, and the higher taken, to provide an additional margin of safety.

8. An ADD can only measure the direction of the local airflow, and not the AOA explicitly, although changes in this measured direction reflect changes in AOA. Accordingly, an AOA gauge is marked in arbitrary units (0 to 25 in the example in Fig 2) rather than in angles (of attack). Extremely high AOA is an indicator of impending stall. The AOA gauge illustrated also has a power failure flag.

9. Some ADD-fitted aircraft also have an AOA indexer (Fig 2), which presents a quick reference display of AOA information, primarily for approach to landing. In this example, a green circle will illuminate if the correct AOA is being flown. If too slow (AOA too high), the upper chevron will illuminate (red). If too fast (AOA too low), the lower chevron will illuminate (yellow). In some aircraft, the indexer may be inhibited when the undercarriage is raised.

5-23 Fig 2 Angle of Attack Indicator and Indexer



CHAPTER 24 - REMOTE INDICATION AND CONTROL

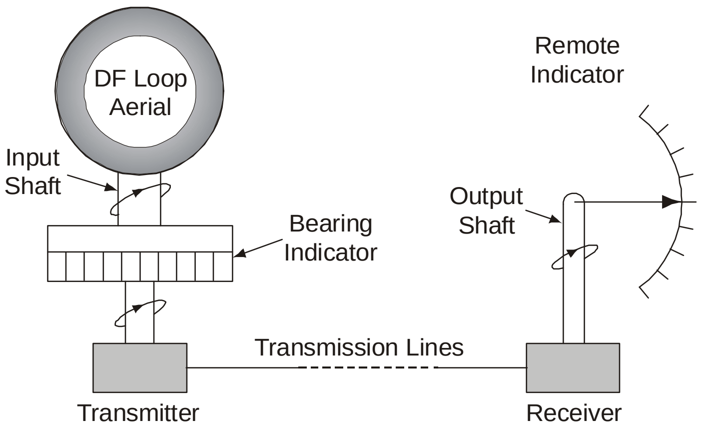
Introduction

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).

5-24 Fig 1 Simple Electrical Remote Indication



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 (i.e. amplifiers and servomotors) 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.

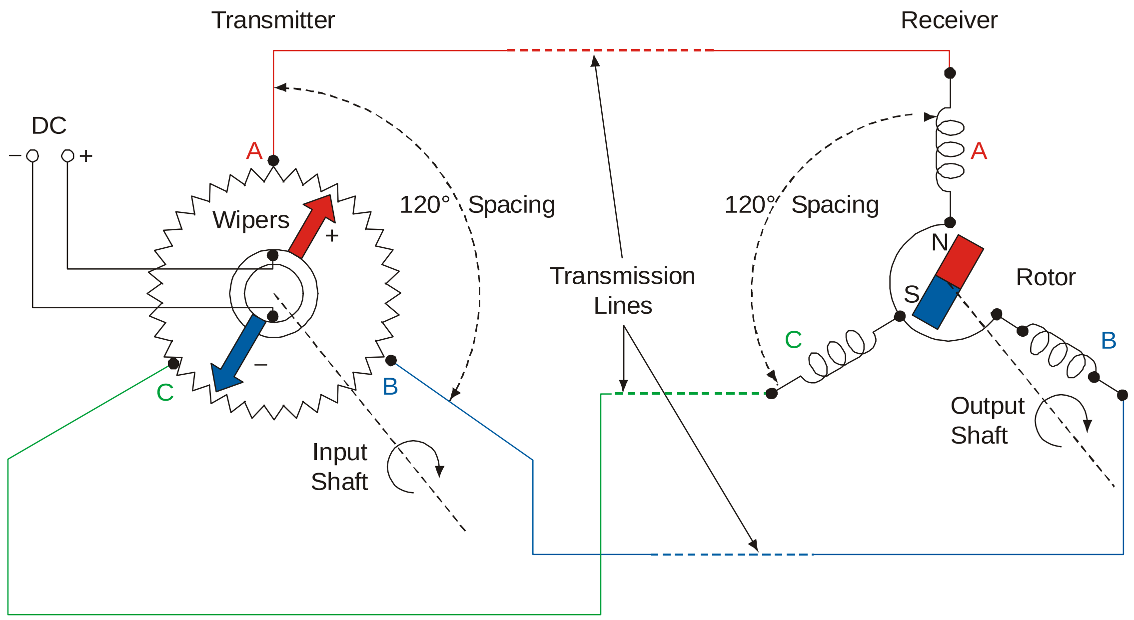
DC SYSTEMS

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 approximately ±2º.

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 120º apart which are connected to the receiver. The input shaft carries two spring loaded sliding contacts or wipers diametrically opposed in contact with the potentiometer. The wipers are fed via slip rings and brushes with DC.

5-24 Fig 2 Desynn Transmission System

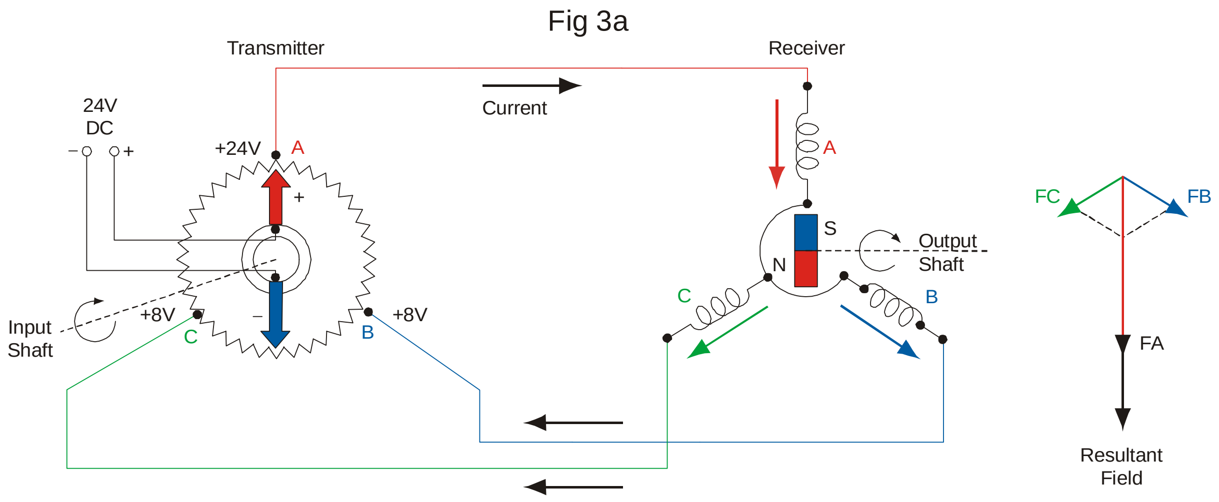


8. **The Receiver**. The receiver (see Fig 2) consists of three high resistance coils whose axes are spaced 120º apart, with a permanent magnet rotor pivoted at their centre carrying a pointer. The three coils are connected to the tapping points A, B, and C in the transmitter.

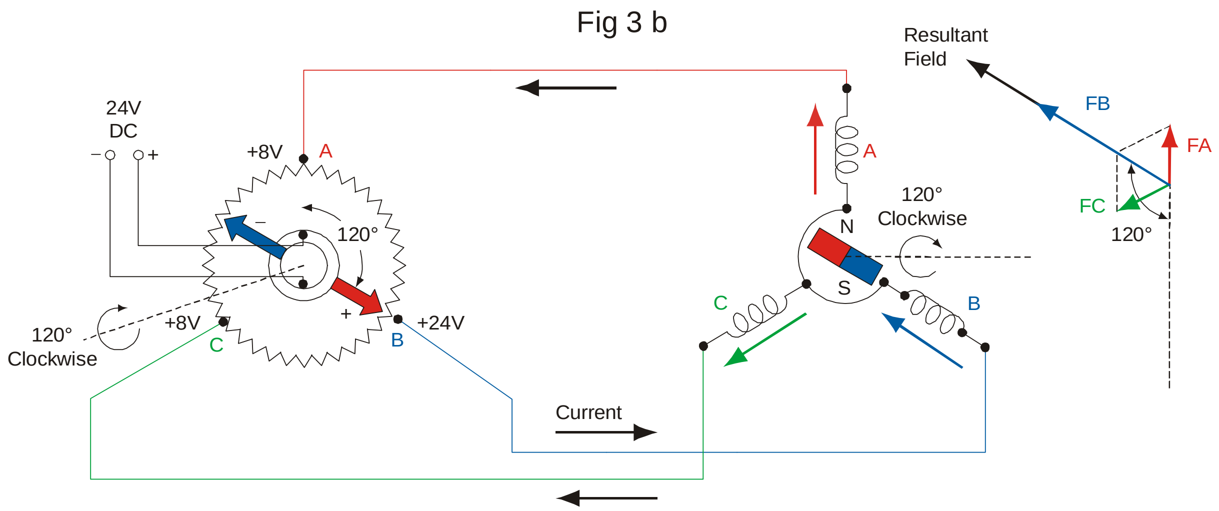
9. **Desynn Operation**. When DC is applied to the transmitter wipers, the voltages at the tapping points A, B and C produce a current flow in the three stators of the receiver and a resultant magnetic field is produced. The rotor magnet aligns itself with this magnetic field. The magnitude and polarity 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 3a 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.

5-24 Fig 3 Operation of Desynn Transmission System



b. If the input shaft is rotated through 120º 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 120º clockwise and the rotor shaft aligns itself along this new axis.



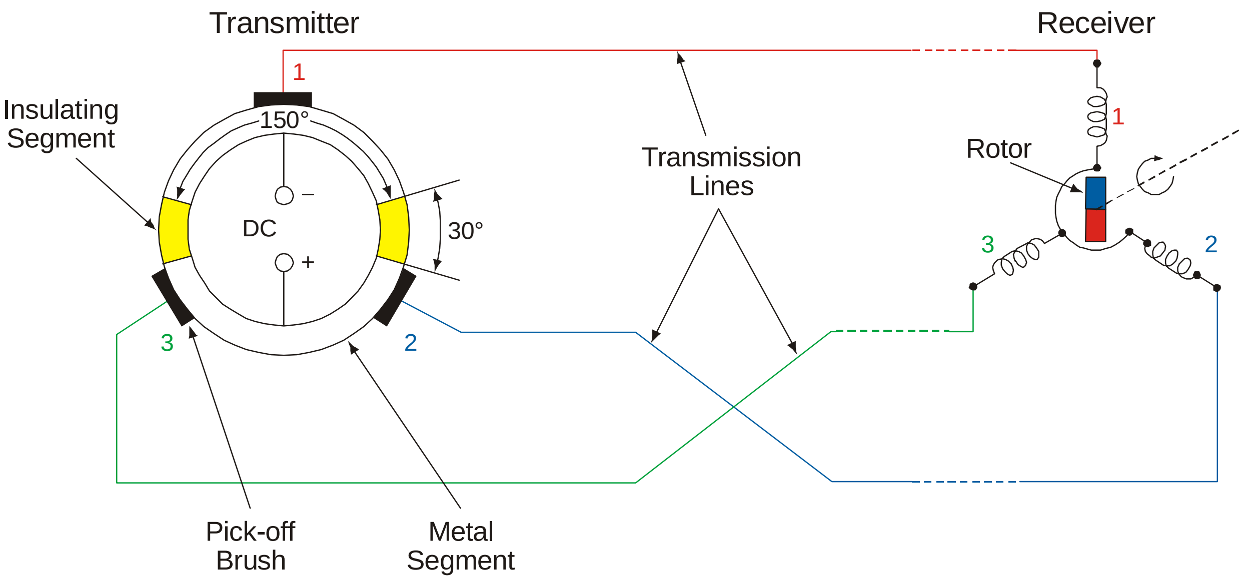
Thus, if the wipers in the transmitter are placed in any position by the input shaft, the resultant field at the receiver and hence the rotor magnet take up corresponding positions. If a pointer, moving over a calibrated scale, is attached to the rotor, remote indication of the position of the input shaft is immediately available.

M-Type Transmission System

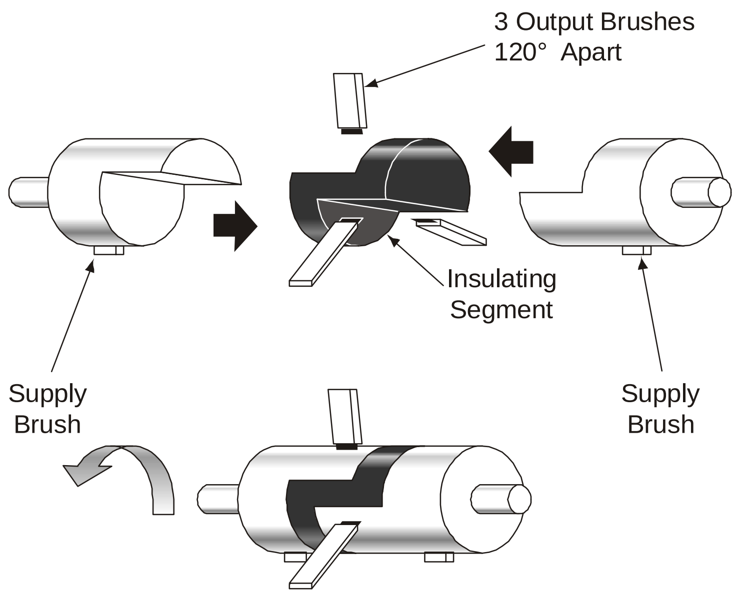
10. The amount of torque produced by the Desynn system is limited by the amount of current which can be taken by the low resistance toroidal potentiometer before overheating occurs. Where moderate torque is required to rotate fairly substantial indicators or comparable devices, a step-by-step or M-type 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.

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 150º separated by two sections of insulating material each extending over 30º. 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 120º (see Fig 5).

5-24 Fig 4 M-Type Transmission System



5-24 Fig 5 M-Type Drum Transmitter



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.

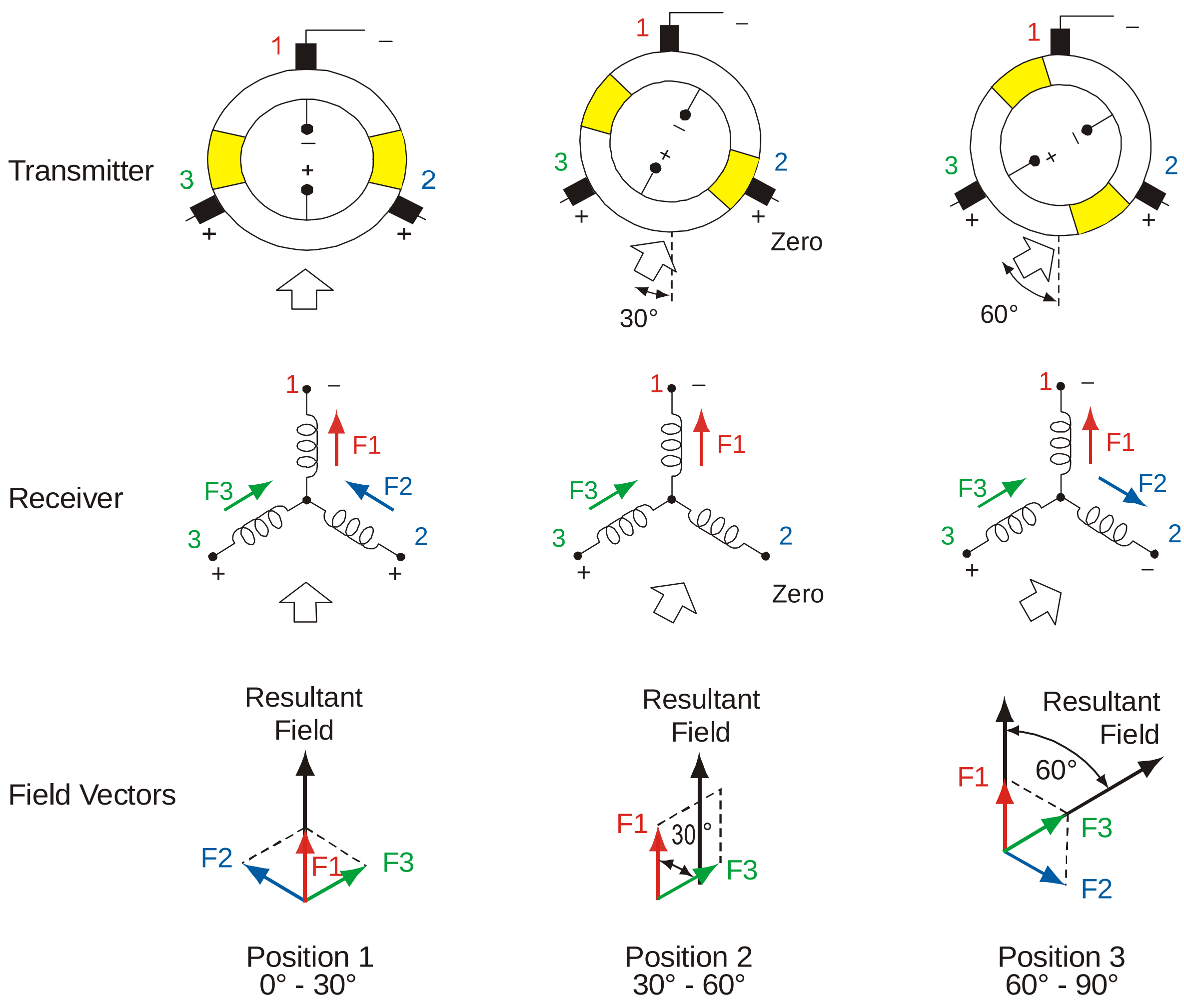
a. In position 1 of the input shaft, brush 1 is connected to the negative supply and brushes 2 and 3 to the positive. These polarities are applied to the three coils in the receiver so that the current divides through coils 2 and 3 with all the current flowing through coil 1. Magnetic fields Fl, F2, and F3 are produced and vector resolution produces the resultant field as shown.

b. Rotation of the input shaft through 30º 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, equal currents flow through coils 1 and 3, while coil 2 carries no current. Resolution of fields Fl, F2, and F3 now produces a resultant field which is seen to have rotated through 30º clockwise in sympathy with the input shaft.

c. The condition after a further 30º rotation of the input shaft is shown at position 3. The resultant field, again following the input shaft, is now rotated 60º 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 30º. There is a change of pick-off brush polarity at one or other of the brushes each time the, input shaft is turned through 30º. For certain purposes, the 30º step is too large and a modified system, giving 24 steps of 15º each, may be used to improve the sensitivity of the system.

5-24 Fig 6 Operation of M-Type Transmission



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 × 15º steps. The latter does not suffer from brush wear and is preferred when the rotation rate is high.

15. **Types of Receiver.** The rotor of 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 180º apart. The permanent 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 an M-type transmission system only moves in 30º (or 1º) 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 6º of the input shaft producing 12 steps of 0.5º each (with a 30º step transmitter) and there are 60 different positions in the full 360º movement of the input shaft, each separated by 6º, 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 ±1º. 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, e.g. 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.

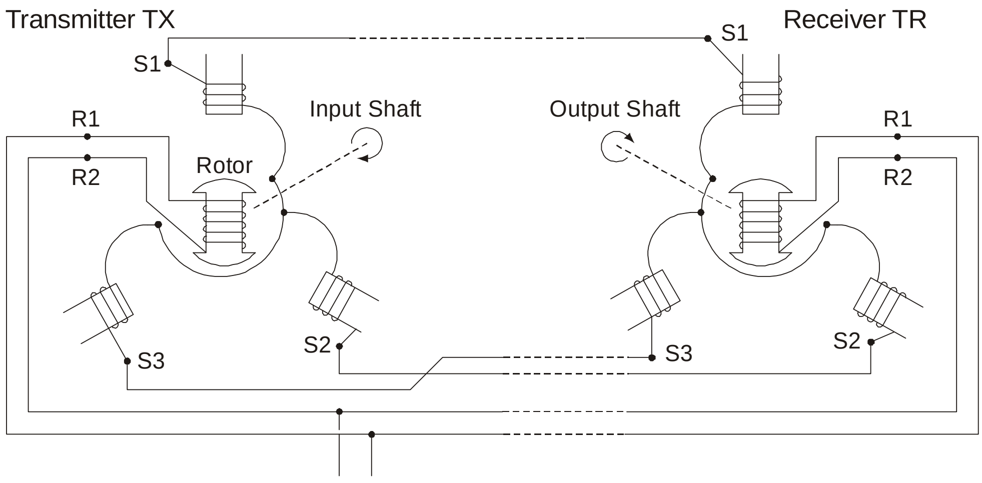
c. Differential synchros.

d. Resolver synchros.

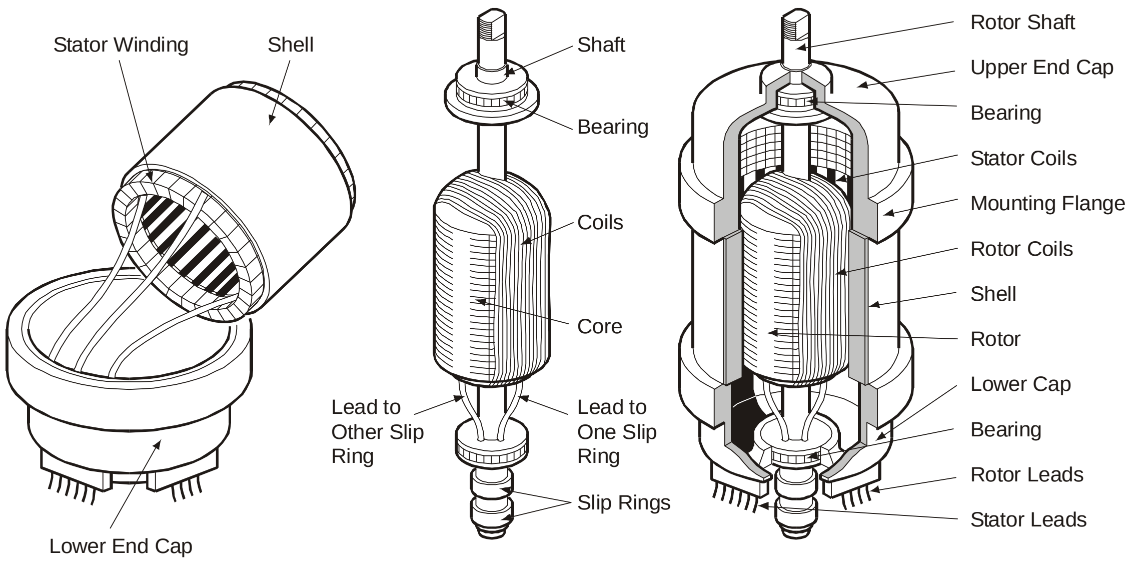
Torque Synchros

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 120º 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 normally fitted with a mechanical damper to prevent oscillation.

5-24 Fig 7 Basic Torque Synchro System



5-24 Fig 8 Construction of Torque Synchro

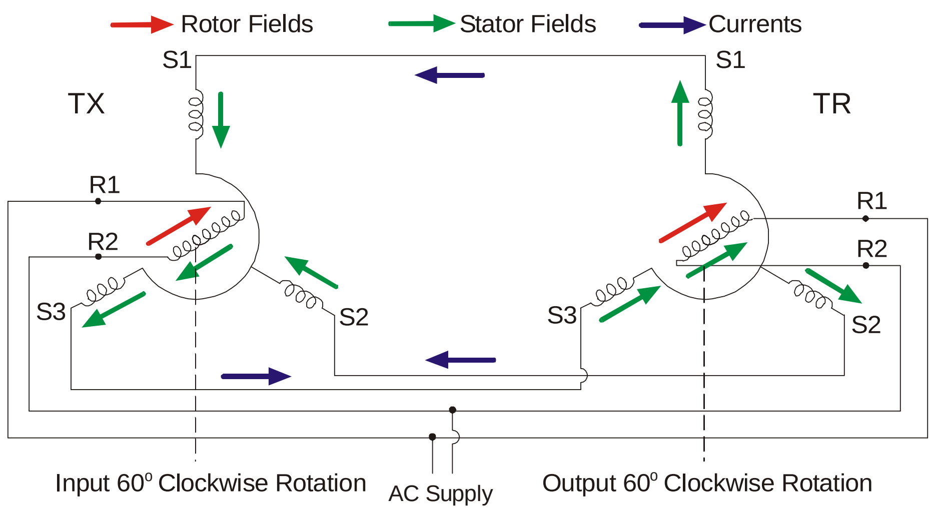


20. **Torque Synchro Operation.** The operation of torque synchro is shown in Fig 9. The TX rotor, energized by the AC supply, has an associated alternating field which cuts the windings of the TX stator coils producing an induced emf. Because the TX stator windings are in closed circuit with the TR 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 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 approximately ± 1º.

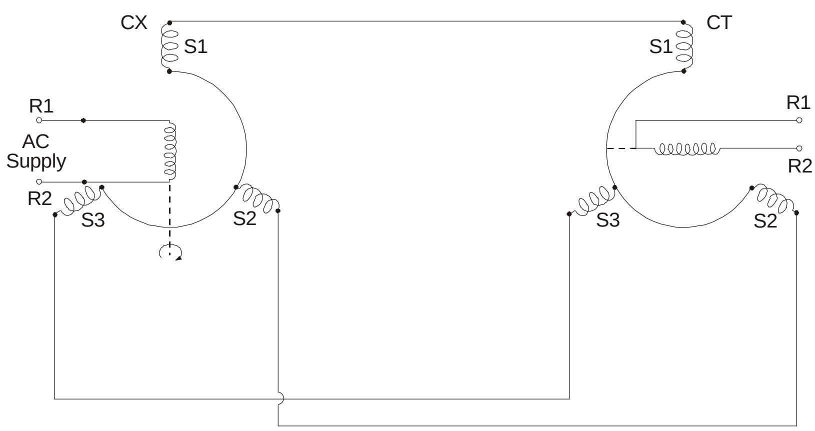
5-24 Fig 9 Operation of the Torque Synchro



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.

5-24 Fig 10 Control Synchro System

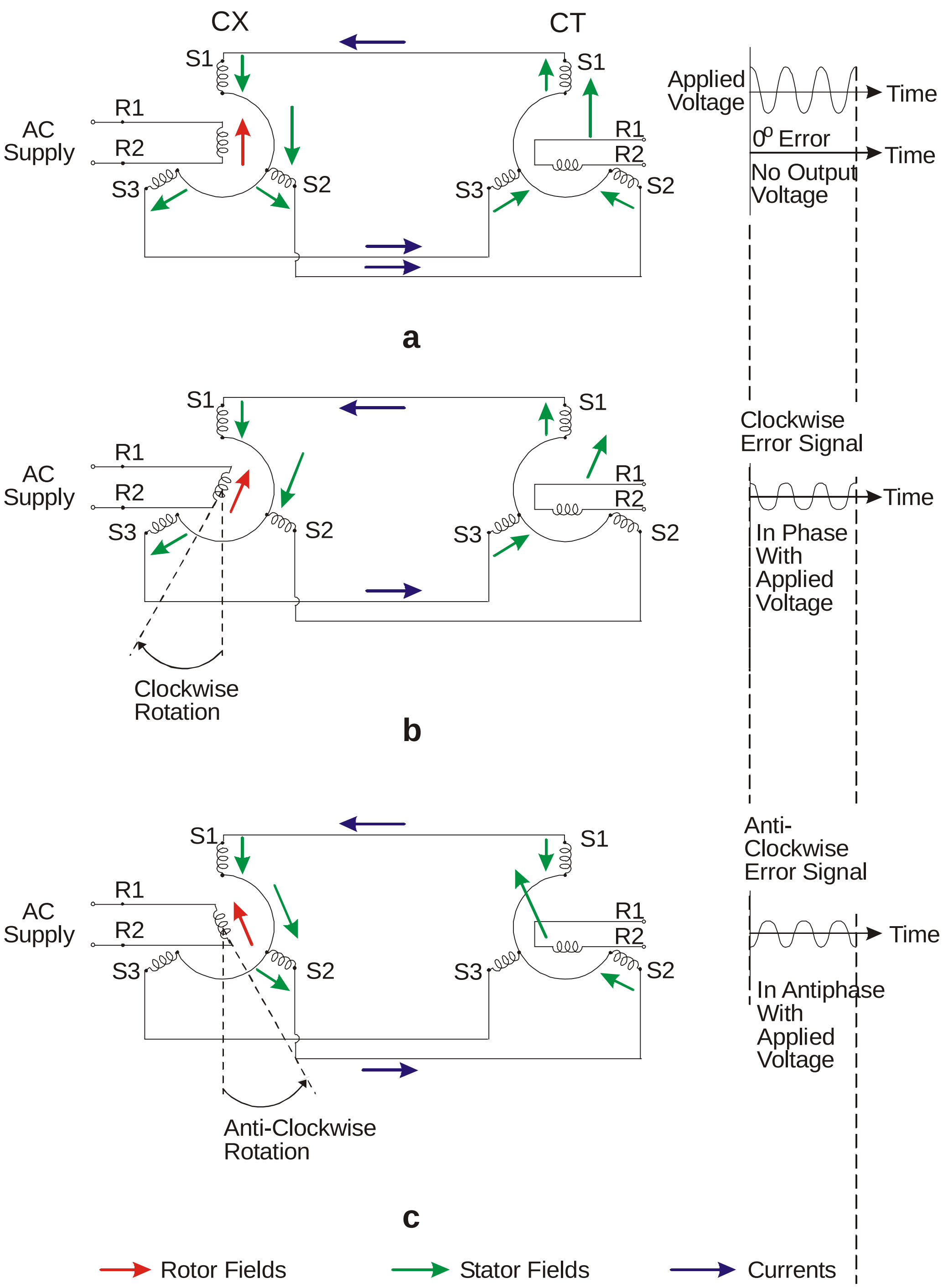


23. **Control Synchro Operation**. The CX rotor, fed from an AC supply, produces an alternating field which, by Henry's Law, induces an opposing field in the CX stator. The circuit current causes a magnetic field associated with the CT stator and parallel to the CX rotor field. When the CT rotor is at 90º 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 CX rotor is displaced, the CT field alignment will change and the CT rotor will

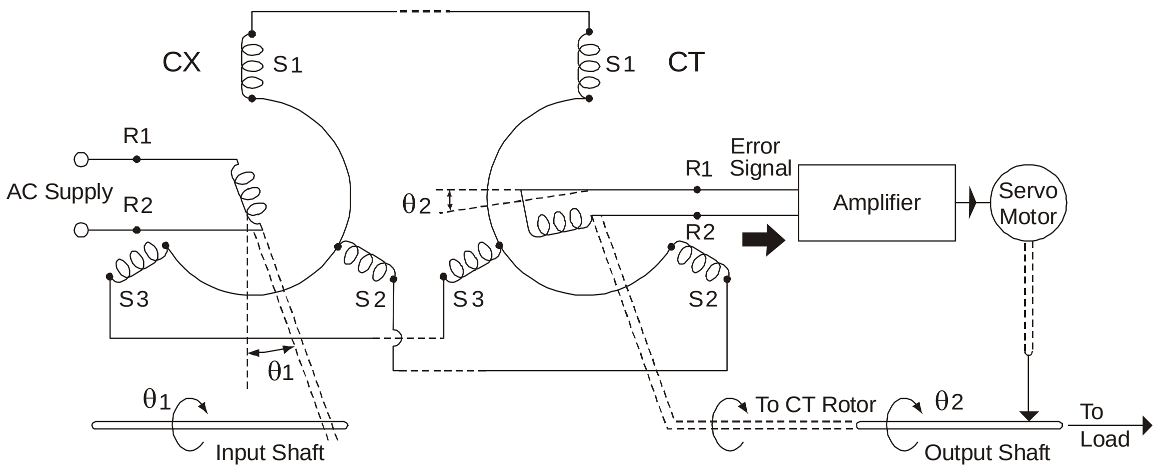
no longer be in the “null” position. An emf, proportional to the angular displacement from the null position, is induced in the rotor, the phase of this induced emf depending upon the direction of displacement. The operation of the control system is shown in Fig 11. The induced error signal is amplified and fed to one phase of a two-phase servomotor 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.

5-24 Fig 11 Operation of a Control Synchro System



5-24 Fig 12 Complete Control Synchro System

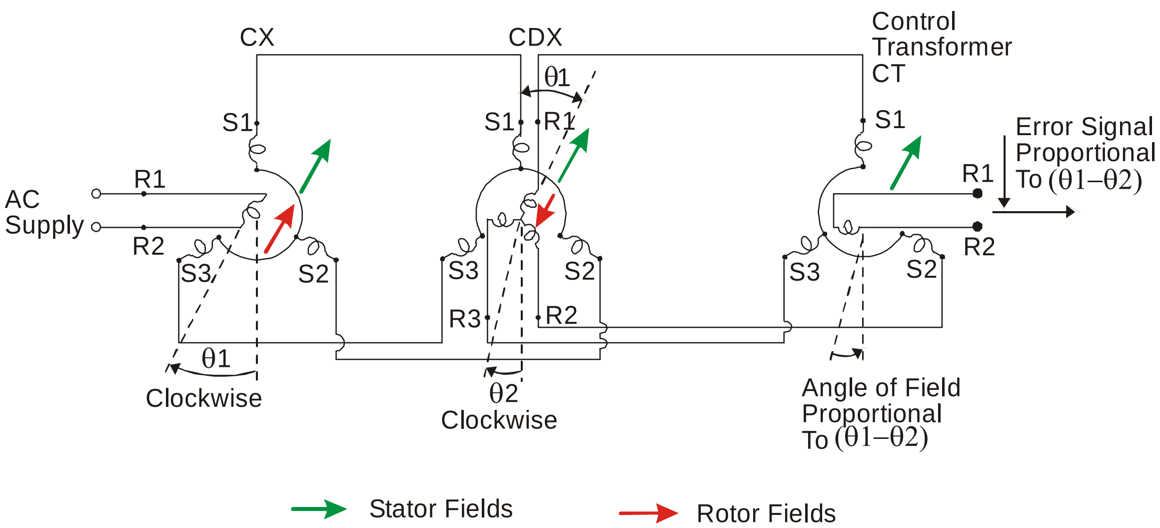


24. **Control Synchro Accuracy**. Using a suitably powered motor, the control synchro accuracy is independent of load. Typical accuracy figures are 16 minutes of arc.

Differential Synchros

25. Differential synchros may be used to add or subtract two shaft rotations. The 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.

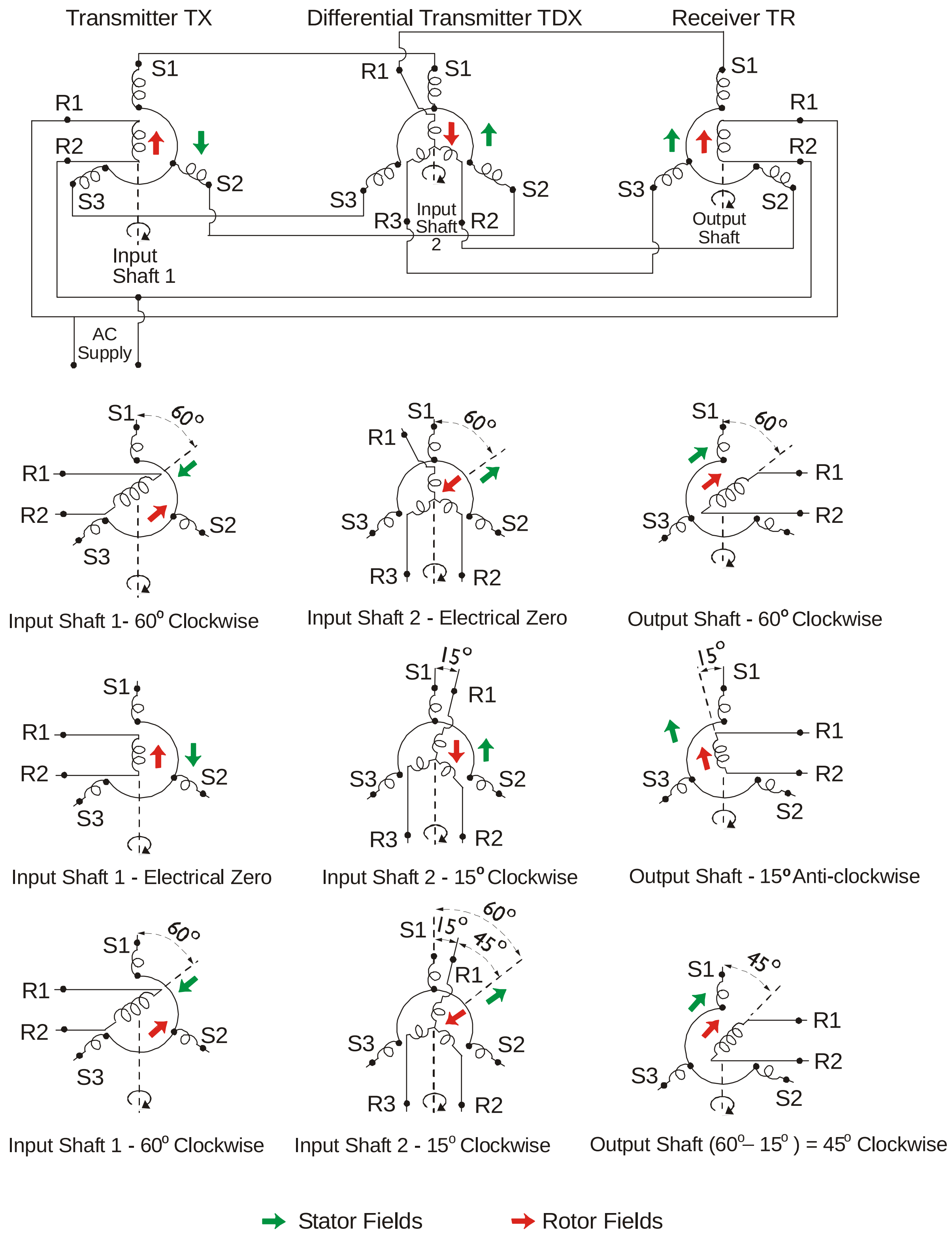
5-24 Fig 13 Differential Synchro System



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 in paragraph 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.

5-24 Fig 14 Action of the Differential Synchro



Resolver Synchros

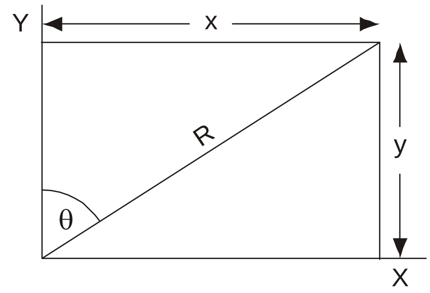
28. **Co-ordinate Systems.** The relationship of one point to another may be defined in either of two ways:

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.

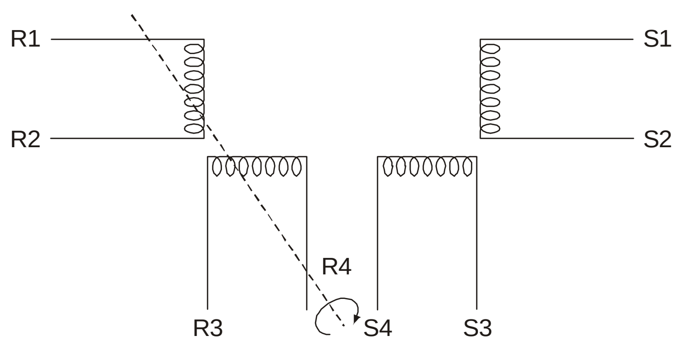
5-24 Fig 15 Relationship Between Polar and Cartesian Co-ordinates



$$x = R \sin \theta$$
$$y = R \cos \theta$$
$$R = \sqrt{x^2 + y^2}$$
$$\theta = \tan^{-1} \frac{x}{y}$$

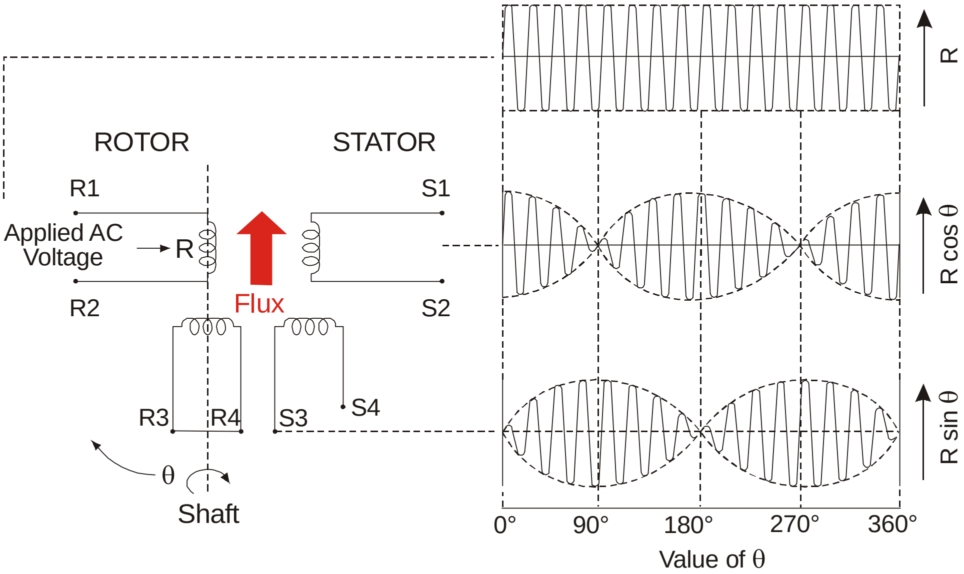
29. **Use of Resolvers.** The resolver synchro is used to convert one co-ordinate system to the other. Ground speed and track define a vector in polar co-ordinates. The same vector may be expressed in northings and eastings in cartesian form. Similar resolvers are used to convert from polar to cartesian 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.

5-24 Fig 16 Resolver Synchro



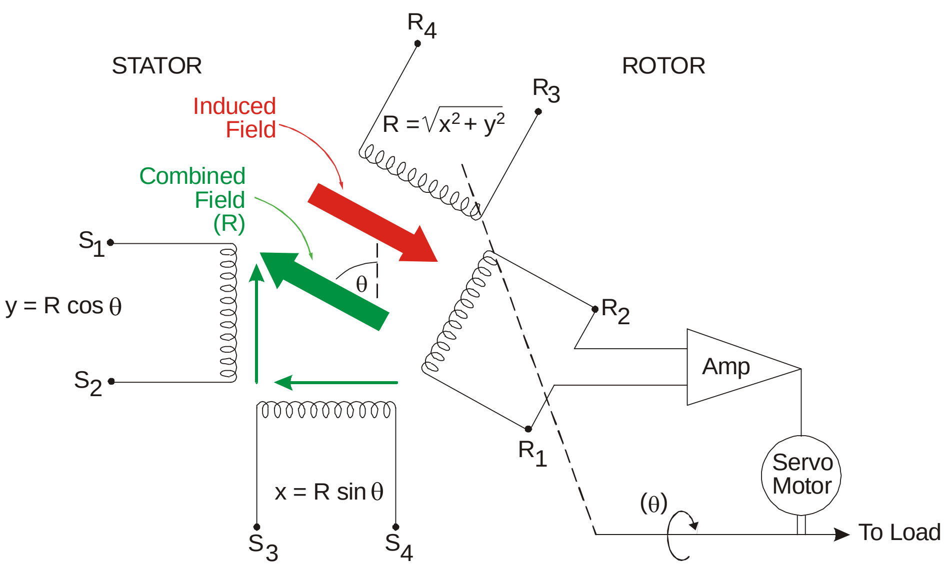
30. **Resolver Synchro (Resolving).** In the resolving mode, the resolver synchro converts polar to cartesian co-ordinates eg ground speed and track to northings and eastings. In this example, R (an electrical analogue of groundspeed) is applied, as an AC voltage, to one rotor winding. The rotor is then turned through angle θ (track). The rotor field has components of R cos θ (northings) and R sin θ (eastings) along the stator winding axes; the voltages induced in the stator windings are proportional to R cos θ and R sin θ. The operation of the resolver synchro (resolving) is shown in Fig 17.

5-24 Fig 17 Conversion of Polar to Cartesian Co-ordinates



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 (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 groundspeed and track). One of the rotor windings is connected to an amplifier and servomotor in the same manner as a control receiver (CT). It is therefore driven to a position at 90º 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 90º 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 groundspeed may be obtained.

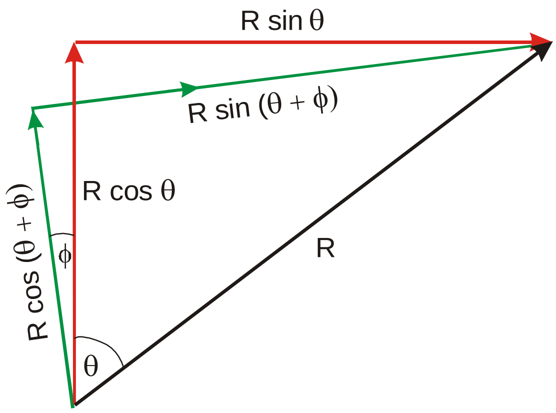
5-24 Fig 18 Conversion of Cartesian to Polar Co-ordinates



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 R cos ($\theta + \phi$) and R sin ($\theta + \phi$) as illustrated in Fig 19.

5-24 Fig 19 Action of a Resolver Synchro (Differential)



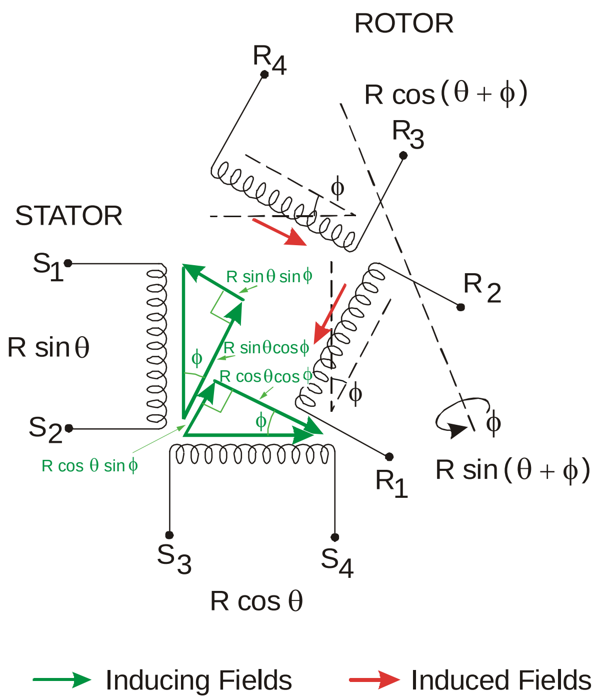
33. **Operation of Differential Resolver Synchro**. The operation of the differential resolver synchro is shown at Fig 20. There are three inputs; R cos θ (true northings) and R sin θ (true eastings) both fed as voltage analogues to the stator coils, ϕ (convergence) is fed as a shaft rotation to position 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. R cos θ cos ϕ – R sin θ sin ϕ = R cos ($\theta + \phi$).

b. R sin θ cos ϕ – R cos θ sin ϕ = R sin ($\theta + \phi$).

Output voltages taken from the rotor coils are analogues of Rcos ($\theta + \phi$) (grid northings) and Rsin ($\theta + \phi$) (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.

5-24 Fig 20 Operation of Differential Resolver Synchro



SUMMARY

Summary Tables

34. Table 1 summarizes the remote indication systems discussed in the preceding paragraphs. Table 2 summarizes the pertinent detail of the various types of synchro mechanisms.

5-24 Table 1 Summary of Remote Indication Systems

System	*Remarks*
Desynn	DC. Provides only sufficient torque to operate small instruments: gives remote indication of dial readings to an accuracy of about ± 2º.
M-Type	DC. Provides moderate torque, sufficient to drive small mechanisms: accurate to about ± 1º. 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 ± 1º: 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.

5-24 Table 2 Synchro Details

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	Control 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

CHAPTER 25 - SERVOMECHANISMS

Introduction

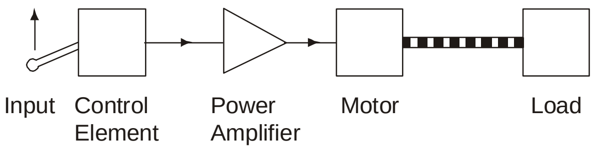
1. The transmission systems described in Volume 5, Chapter 24 included many devices capable only of remote indication on light pointers. At least one, however, the control transmission system, could do more than this. It could not only transmit the information over considerable distances, but its receiving 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. In order to control the position of a radar scanner or other heavy load, an arrangement such as that in Fig 1 could be put together.

5-25 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 between 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:

a. Application of the input angle, θi, to the error detector.

b. Feedback of the load position, θo to the error detector.

c. Subtraction of θ_i from θ_o to produce the error.

d. Production of an error signal proportional to $\theta_o - \theta_i$.

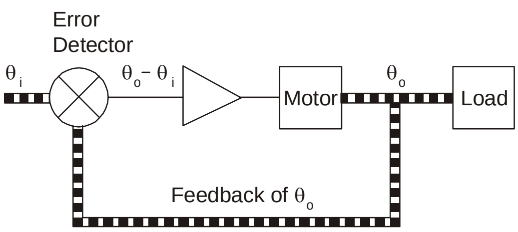
e. Control of the amplifier output by the error signal.

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, at which point the error signal disappears and movement stops.

5-25 Fig 2 Automatic Control System



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 (e.g. Fig 1).

Servomechanisms

12. To be classed as a servomechanism, an automatic control system must be capable of continuous operation and have:

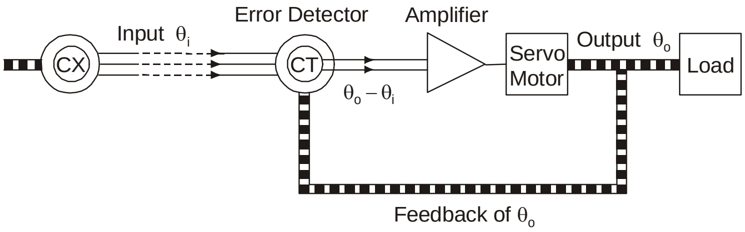
a. Error actuation.

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.

5-25 Fig 3 Servo Elements of Control Synchro



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. The classification into position and speed control servos is a convenient one in view of the applications of the servo principle 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.

16. Servomechanisms of either classification, can be operated by AC or DC power supplies. In general, the AC system is capable of greater accuracy and stability, but is limited to low power applications. The DC servo is used in high torque situations, but more often, a hybrid AC/DC servo, combining the merits of both, will be found when heavy loads are involved.

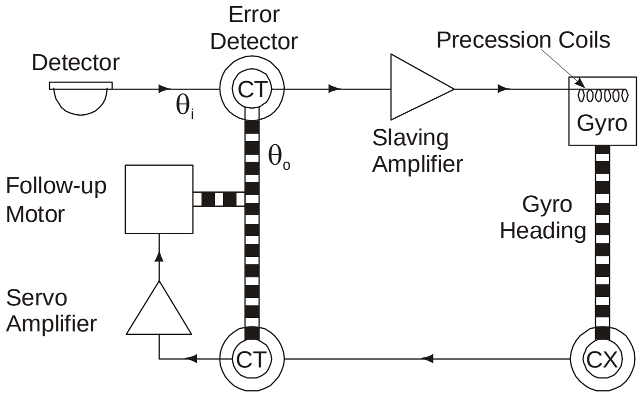
17. The synchro control transmission system has already been mentioned as an example of a servomechanism. It is illustrated in Fig 3 with the servo terms added to assist the reader in identifying the features enumerated in para 10.

EXAMPLES OF SERVOMECHANISMS

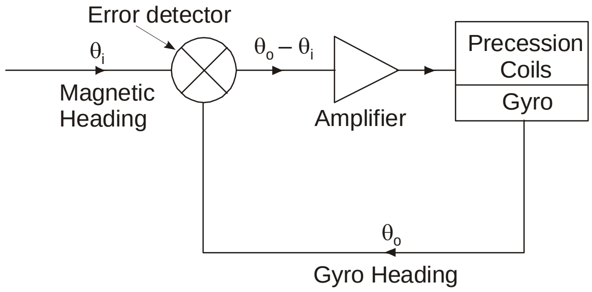
Position Control Servo - The Gyro-magnetic Compass

18. Fig 4 shows the basic components of a typical gyro-magnetic (GM) compass system. By comparing Fig 4 with Fig 3, it can be seen that the servomotor has been replaced by the precession coils and the load is now the gyro. Consequently, gyro heading is the load position (θ_o) which is fed back to the error detector. Magnetic heading is the input (θ_i) and is applied to the CT together with gyro heading. The error in gyro heading actuates the system which is stabilized by magnetic heading. These relationships are emphasized in Fig 5, which shows the GM compass system in servo outline.

5-25 Fig 4 Basic Components of a Typical GM Compass System



5-25 Fig 5 GM Compass System Servo Outline

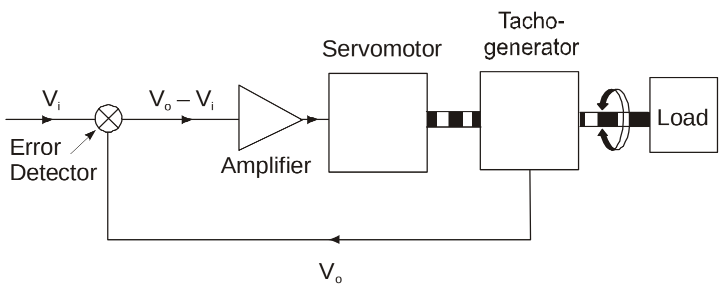


19. The GM compass is therefore essentially a servomechanism, and the fact that a second servo (see Fig 4) is used to provide the load position feedback is a matter of design convenience. Indeed some systems use a direct mechanical link for the feedback. Whatever the feedback method, the servo principle can be identified in all GM compasses.

Speed Control Servo - The Velodyne

20. In navigation equipments, the most common application of the speed control servo is for converting a voltage (representing a speed) into an angular velocity. A shaft, rotating at this angular velocity, can then be used to display distance gone. This requirement is met by a device called a 'velodyne', the component parts of which are illustrated in Fig 6.

5-25 Fig 6 Component Parts of a Velodyne



21. The input voltage (V_i) is applied through a power amplifier to turn a servomotor, 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.

22. The tachogenerator output (V_o) is fed back to the error detector which subtracts the input voltage (V_i) and feeds the resultant voltage to the amplifier. The motor is therefore controlled by the difference in voltages and will speed up or slow down until the difference is zero.

23. 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.

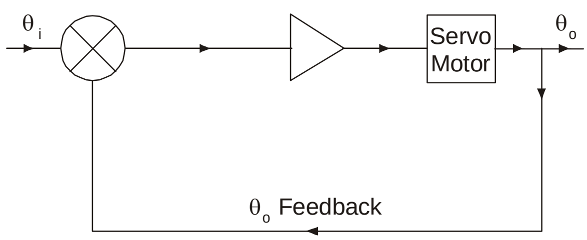
PERFORMANCE OF SERVOMECHANISMS

Introduction

24. 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.

25. The servo illustrated in Fig 7 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 7 will be taken to be a position servo.

5-25 Fig 7 Simple Servomechanism



Response

26. 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 following paragraphs will show that matters are not as simple as this.

27. 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 8. Both are discussed without considering restraints in the first instance.

5-25 Fig 8 Types of Input

Fig 8a Step Input

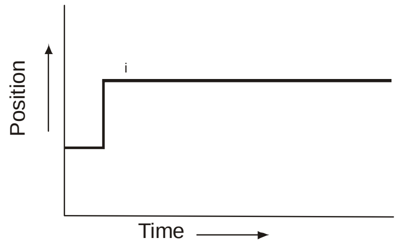
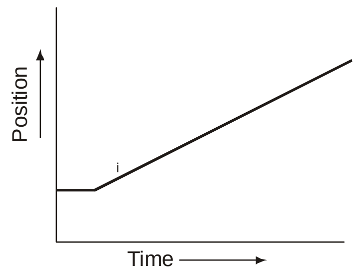


Fig 8b Ramp Input



Step Input - No Friction

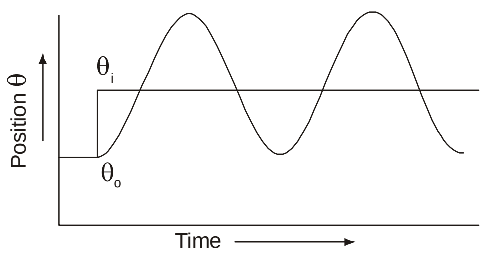
28. For this discussion, we will assume that the input and output were aligned at θ_o, until the input suddenly changed to θ_i. An error signal proportional to $\theta_o - \theta_i$ appears at the amplifier input and the motor is energized to null the error.

29. 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.

30. This is not a satisfactory state of affairs, for the load acceleration is in one sense only and that is 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.

31. 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 9.

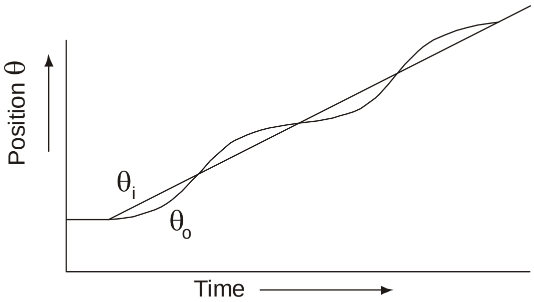
5-25 Fig 9 Oscillating Response to a Step Input



Ramp Input - No Friction

32. The description of the response can be followed in Fig 10. 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.

5-25 Fig 10 Oscillating Response to a Ramp Input



Effect of Restraints

33. 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.

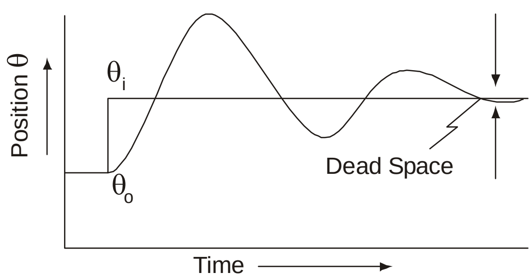
34. While restraints are beneficial in stabilizing, or damping, the response, they do have certain detrimental effects. One of these is that power is wasted; another is the introduction of error in the steady state.

Steady State Errors

35. 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.

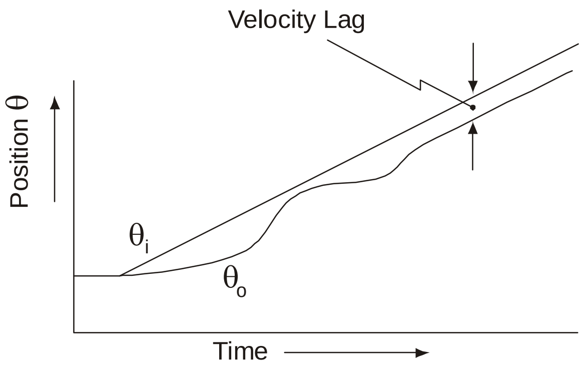
36. The resistance due to coulomb friction tends to degrade the sensitivity of a servo, for a torque which overcomes it must be generated before any movement of the load takes place. To provide this torque the load error must reach some finite size, and any errors less than this will not be corrected. Fig 11 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.

5-25 Fig 11 Response with Coulomb Friction to Step Input



37. 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 12, 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.

5-25 Fig 12 Response with Viscous Friction to Ramp Input



Summary

38. 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

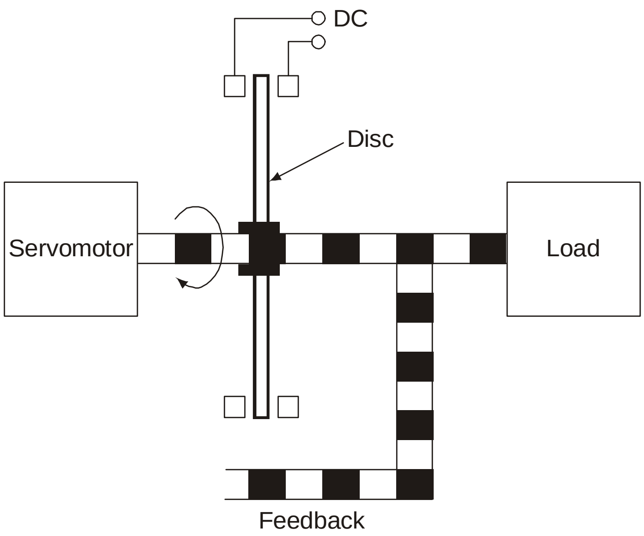
Introduction

39. 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

40. 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 13.

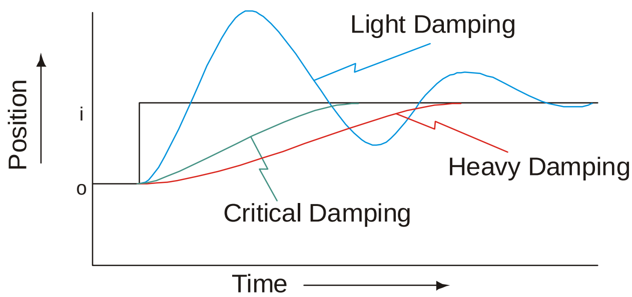
5-25 Fig 13 Eddy Current Damper



41. 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.

42. Varying degrees of damping can be applied. Fig 14 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.

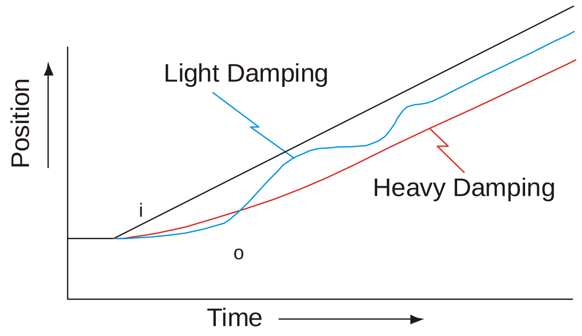
5-25 Fig 14 Degrees of Damping - Step Input



43. 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 increases the velocity lag. Thus to remove the transient oscillations completely a considerable velocity lag must be expected. Fig 15 illustrates the response for two degrees of damping for a ramp input.

44. 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.

5-25 Fig 15 Degrees of Damping - Ramp Input

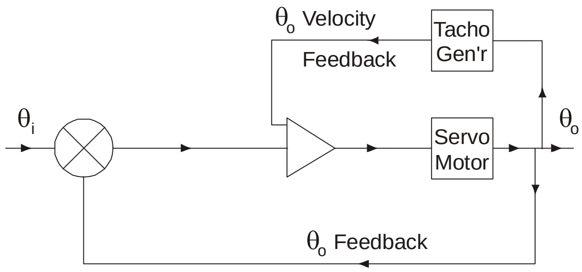


Velocity Feedback Damping

45. 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.

46. 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 16. Since a voltage with negligible current is required, the additional output load can be neglected.

5-25 Fig 16 Velocity Feedback



47. 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.

48. 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

49. 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.

CHAPTER 26 - ENGINE INSTRUMENTS

Introduction

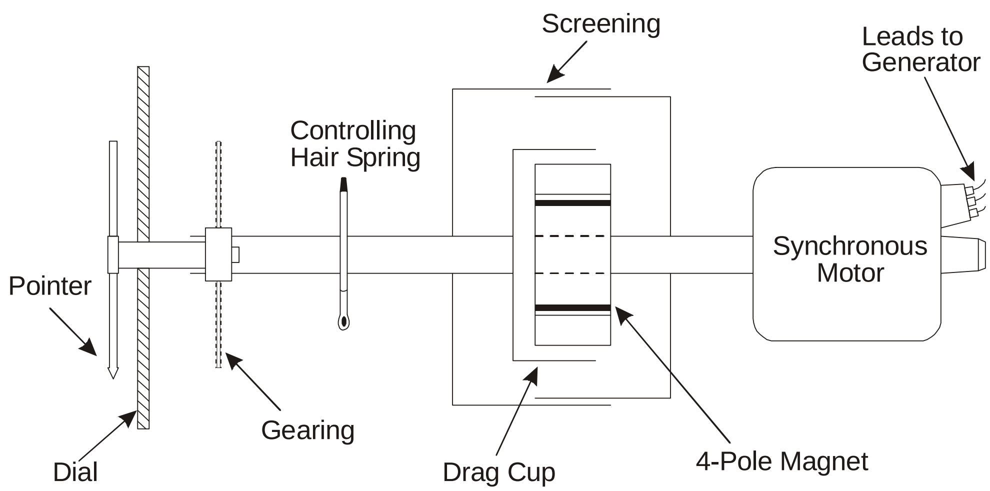
1. An aero-engine is an expensive item and its failure in flight could have serious safety implications. It is therefore most important to have a comprehensive and accurate feedback to the crew of information relating to the performance of the engine. Indications of failures, or of excursions of engine parameters outside limits, are usually in the form of discrete displays, eg warning lights, flags, or audio signals. By contrast, routine monitoring information is usually displayed on gauges. These may be of the conventional, analogue type, although electronic multi-function displays are now more common.

Engine Spool Speed

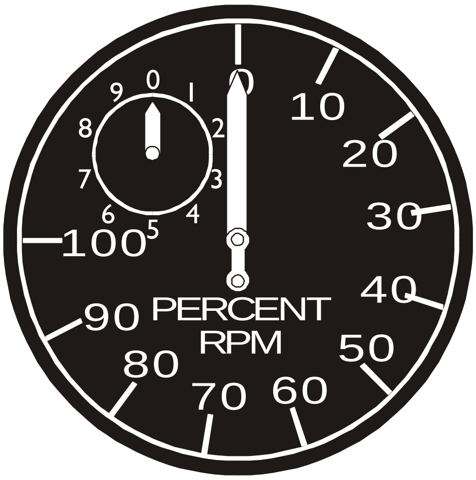
2. The most common method of measuring engine spool speed is by a tachogenerator driven from the external wheelcase. A typical tachogenerator contains a three-phase stator and a two-pole permanent magnet rotor. Rotation of the magnet induces a three-phase voltage in the stator windings, the frequency of this voltage being proportional to the engine speed.

3. The output signal from the tachogenerator is fed to a tachometer indicator mechanism (Fig 1) which contains a synchronous motor, the speed of which is governed by the input frequency from the generator. An extension of the synchronous 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 shaft. The torque is balanced by a hair spring and the shaft rotation is transmitted to the movement of a pointer over a dial through an appropriate gearing system. Engine speed is usually shown as a percentage of maximum rpm (Fig 2).

5-26 Fig 1 Tachometer Indicator Mechanism



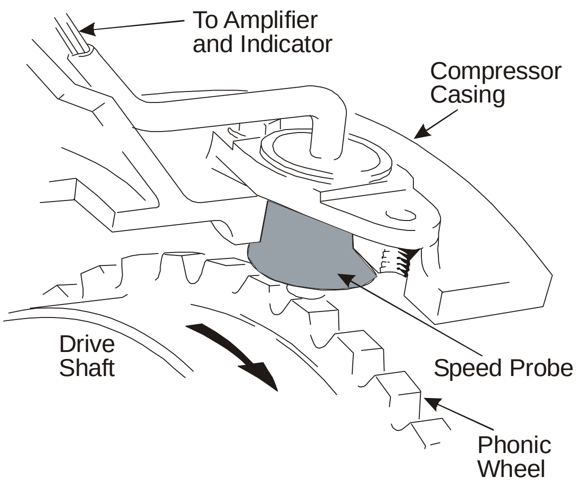
5-26 Fig 2 Percentage rpm Indicator



4. In some multi-spool engines without a gearbox driven from the LP or IP spool, a speed probe, located on the compressor casing, can be used with a phonic wheel to provide an electric current which is proportional to spool rpm (Fig 3 illustrates such an arrangement). Multi-spool engines require gauges to indicate different spool speeds and, in some installations, this is achieved by switching one gauge between separate spool speed generator signals, thus obviating the need for multiple rpm gauges per engine.

Note: The phonic wheel illustrated in Fig 3 consists of a series of evenly spaced notches machined into the outer rim of the spool. The speed probe detects the gaps in the wheel, thereby producing an output which is proportional to the speed of the wheel. It is called a 'phonic' wheel because the frequencies produced are normally in the audio range.

5-26 Fig 3 Speed Probe and Phonic Wheel



Exhaust Gas Temperature

5. The operating temperature of a turbine has a direct effect on its life and it is therefore essential that the temperature is maintained within specified limits. Because of the high temperatures of the gas entering the turbine, it is impractical to make direct measurements. In practice, the temperature of the gas is measured downstream of the turbine inlet either by thermocouples at the exit to the turbine in the jet pipe, or by measuring the blade temperature using a pyrometer. By knowing the behaviour of

the gas through the turbine, the temperature loss can be calculated, and a suitable limit set for the measured value that will ensure that the turbine inlet temperature limit will not be exceeded.

6. When the exhaust gas temperature is measured using thermocouples, several are usually connected in series and positioned in the gas stream to give a representative average temperature. A thermocouple consists of two conductor wires of different metals, usually nickel-chromium and nickel-aluminium, joined at both ends. One end, the 'hot junction', is mounted in a ceramic insulator which is housed in a protective metal sheath. The other end, the 'cold junction', senses ambient air temperature as a reference point. There are two main types of thermocouple in use:

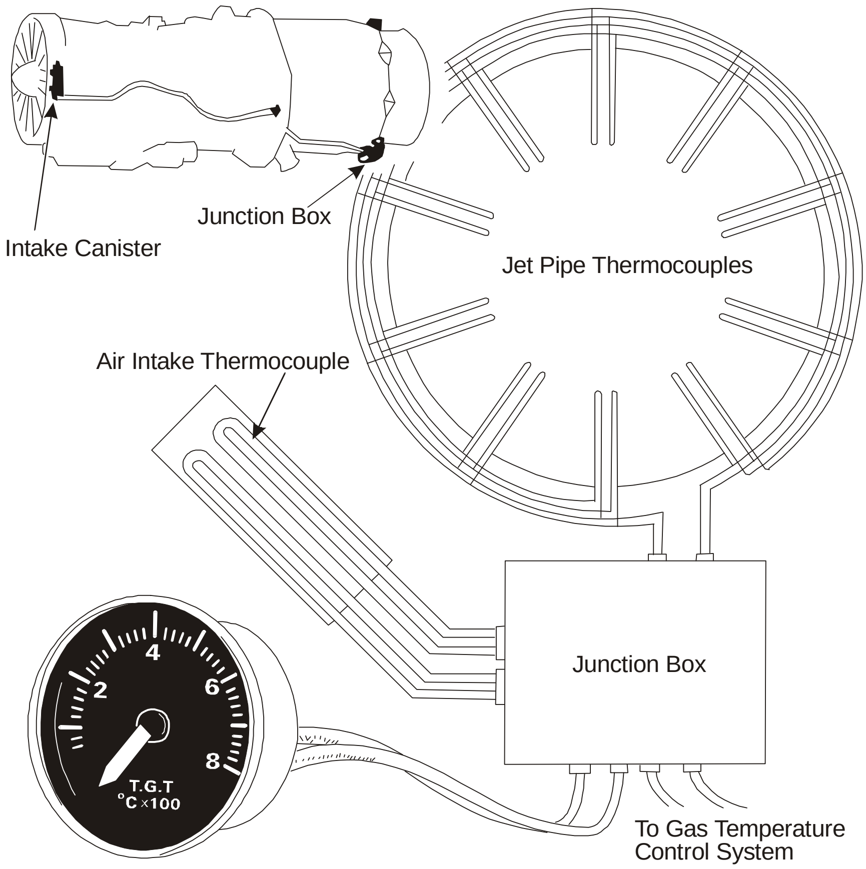
a. **Stagnation Type**. In the stagnation type, the sheath has a hole near its tip for the exhaust gas to enter and an exit hole staggered further away from the tip and of smaller diameter thus forcing the gas to flow through a Z-shaped passage past the elements of the thermocouple. This stagnation chamber reduces the gas velocity past the hot junction and avoids an adiabatic temperature rise on contact with the thermocouple.

b. **Rapid Response Type**. The rapid response type of thermocouple is used on turboprop engines where the exhaust gases have a comparatively low velocity. The gases follow a straight path past the hot junction.

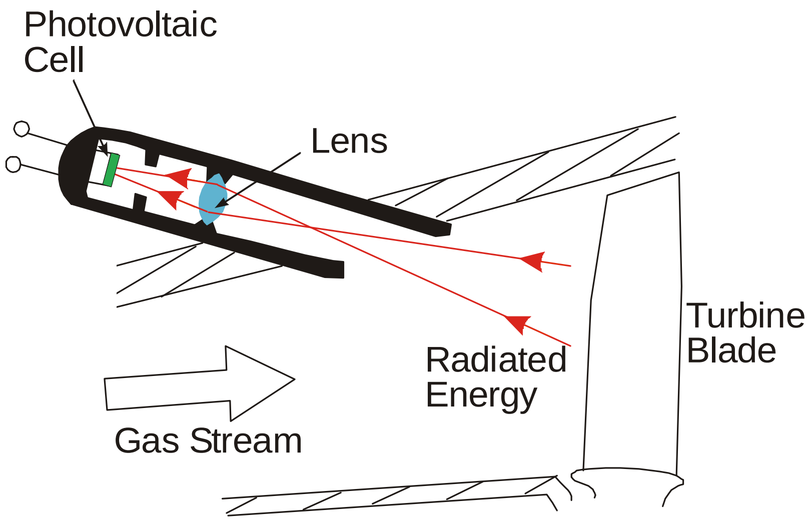
7. To ensure that variations in the temperature of the cold junction do not affect the indicated temperature, an automatic compensator is fitted either to the instrument or elsewhere in the circuit. A typical double element thermocouple installation is illustrated in Fig 4.

8. The optical radiation pyrometer develops an electro-motive force (EMF) proportional to the energy radiated from the surface at which the pyrometer is directed, in this case the turbine blade (Fig 5). The radiated energy is focused onto a photo-voltaic cell and the DC voltage produced is amplified and passed to control and indication circuits. Pyrometers are prone to 'sooting' and require cleaning and calibration at regular intervals. Engines fitted with pyrometers may also have a single thermocouple to measure the exhaust gas temperature during engine start-up as the pyrometer is normally calibrated only for the normal operating range of the engine, i.e. idle to maximum rpm.

5-26 Fig 4 Typical Double Element Thermocouple System



5-26 Fig 5 Optical Radiation Pyrometer



Oil System

9. Oil plays the vital role within the engine of lubricating bearings and it is essential that the oil is cooled and is supplied at the correct pressure if failures are to be avoided. Oil temperature is taken by a temperature sensitive element fitted in the oil system upstream of the bearings. Changes in temperature of the oil cause changes in the electrical resistance of the sensor and thus alterations of current to the indicator. Oil pressure is also sensed upstream of the bearings by a sensor which detects either direct changes in pressure or changes in the difference between engine feed and return

oil pressure. The pressure indicator may be either a dial and pointer type, or a flag type showing the pressure as high, normal, or low.

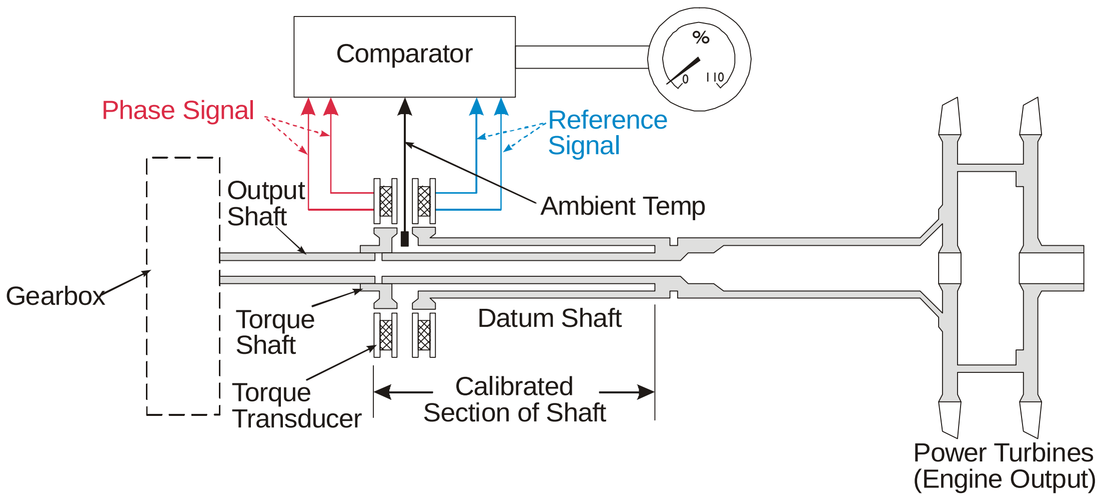
Engine Torque

10. As it is not possible to estimate the power being produced by a turboprop engine from considerations of turbine gas temperature and engine rpm alone, all turboprop and helicopter installations include a system for measuring the torque being delivered to the propeller or rotor. Torquemeters may be electrical or hydraulic.

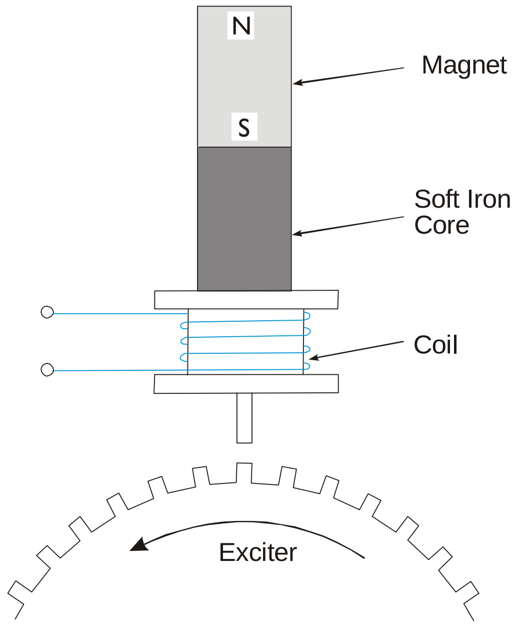
Typical Electrical Torquemeter

11. A typical electrical torquemeter system consists of a torquemeter assembly, a phase detector, and an indicator (Fig 6). The torquemeter assembly comprises two concentric shafts, the inner of which is the shaft connecting the drive from the engine to the propeller reduction gear, and twist in this shaft is proportional to torque. The outer datum shaft is connected to the engine output only. There are toothed gear wheels on both shafts, and above these are situated pick-up assemblies consisting of permanent magnets on top of a coil (Fig 7). As the gear turns, the teeth on the exciter wheels cut the magnetic lines of flux around the magnet, inducing an EMF in the windings of the pick-up coil. Under no-load conditions, the toothed wheels on the two shafts turn with no relative movement between them and there is no difference in the output signals from the pick-ups. As torque is applied and increased, the output shaft will twist along its length. This movement will have no effect on the outer datum shaft and thus its toothed wheel will have an angular displacement relative to that on the drive shaft. This angular displacement is detected by the pick-up assembly as a phase difference in the output signal. A phase comparator generates a signal, dependent on the phase difference, to drive the pilot's indicator.

5-26 Fig 6 A Phase Comparison Torquemeter



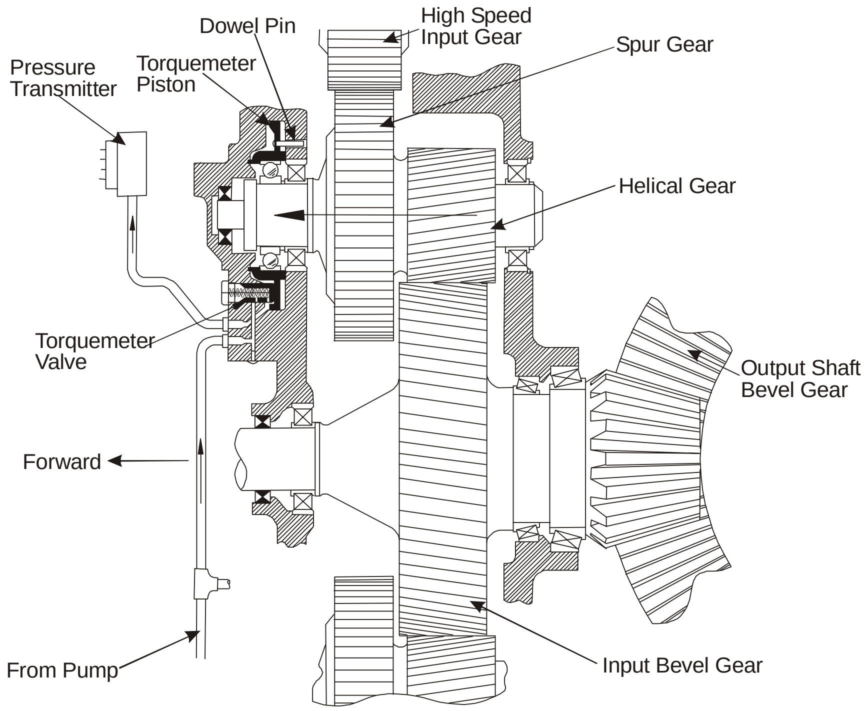
5-26 Fig 7 Torquemeter Transmitter



Typical Hydraulic Torquemeter

12. A hydraulic torquemeter mechanism is built into the main gearbox input section. The power turbine from the engine is connected by way of a drive shaft to the input section of the main gearbox via a high-speed input gear (see Fig 8). The high-speed input gear drives a spur gear on a free-wheel unit which in turn drives a 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 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. The input bevel gear is prevented from moving axially by means of tapered rollers, but movement is allowed on the free-wheel unit helical gear.

5-26 Fig 8 Hydraulic Torquemeter System



13. 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. A torquemeter valve is spring loaded against this piston.

14. A pump supplies oil under pressure to the torquemeter valve. If no torque is being applied, the torquemeter valve will be closed, but as torque is applied, the valve will start to open allowing some of the high-pressure oil to enter the piston chamber. When 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 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 being proportional to the valve movement which in turn is 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 indicating percentage of torque. This system often has input shafts from two engines and two torquemeter mechanisms making it possible to measure and match the torque applied by each engine so that each is carrying an equal load.

Vibration

15. Compared to an internal combustion engine, a gas turbine is an extremely smooth running power generator, and a change in vibration due to the impending failure of a component part may be so slight as to pass unnoticed by the crew. Experience has shown that a vibration monitor installed on an engine is able to detect mechanical defects in rotating parts at a very early stage, thus permitting corrective action to be taken before extensive damage occurs.

16. Although vibration monitoring can be based on measuring acceleration or displacement, current practice tends towards the measurement of velocity amplitude using a seismic accelerometer working on the piezo-electric principle. At least two accelerometers are required per engine so that radial and transverse vibrations can be measured. Cockpit indications may be in the form of gauges, or as warning lights or audio signals triggered when a preset limit is exceeded.

CHAPTER 27 - MISCELLANEOUS INSTRUMENTS

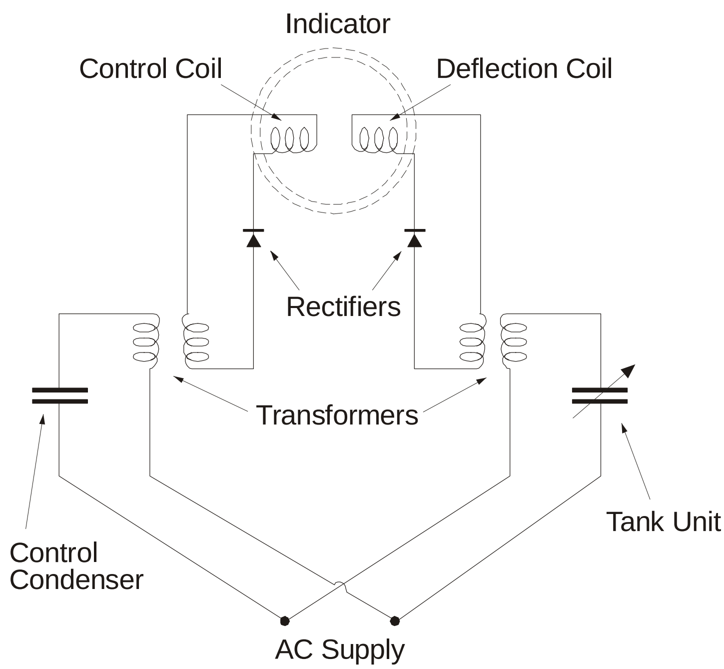
Fuel Contents Gauges

1. Fuel contents gauges indicate the amount of fuel contained in the aircraft tanks. These gauges use sensing elements in the tanks, the majority of which are now of the capacitor type. One of these, the 'Pacitor', is described in this section. The characteristics of any particular aircraft type will be found in the Aircrew Manual. The current practice is for gauges to be calibrated in units of mass (kilograms), although in certain older, and some light piston aircraft, gauges may still show contents in volumetric terms, e.g. gallons. Due to the design of the tanks, and other engineering considerations, not all of the indicated fuel may be available for use.

2. The capacitor type sensors work on the principle that, if all other factors remain constant, the impedance of a capacitive reactance circuit will vary with change in the dielectric of the capacitor. In the case of the fuel-sensing unit, the capacitor plates are concentric, and set a fixed distance apart. The unit is mounted vertically in the fuel tank. Between the plates, the gap is filled with dielectric which is either fuel, air or a combination of both. As fuel is consumed, the ratio of fuel to air changes, thus the dielectric changes, thereby altering the impedance of the tank unit.

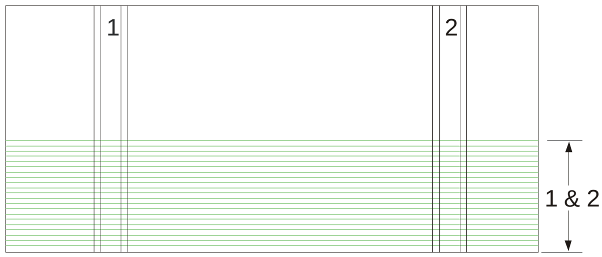
3. Fig 1 shows a diagram of a typical system electrical circuit. The tank unit varies the current flowing in the transformer primary winding to which it is connected. The corresponding alternating current induced in the secondary winding is converted to direct current by a rectifier and then fed to the deflection coil in the indicator. The same process takes place in the control circuit, except that, as the control condenser has a fixed value, the current in the 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.

5-27 Fig 1 Pacitor Fuel Sensor

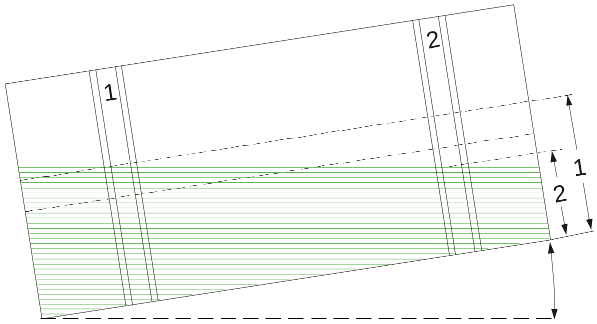


4. Normally, at least two units are fitted in each tank, connected in parallel. This ensures that accuracy is maintained despite aircraft attitude changes (within specified limits) as the fall in fuel level at one end of the tank is compensated for by the rise at the other end (Fig 2).

5-27 Fig 2 Compensation for Tilt



Tank Level (Capacitance of 1 and 2 Equal)



Tank Tilted (Capacitance of 1 Increased and 2 Decreased)

5. Fuel gauges are subject to instrument errors, installation errors, and calibration errors. Whereas instrument and installation errors are virtually constant for any one gauge, calibration error may vary widely since it is caused by inconsistencies in the electrical conducting property of the fuel. Gauges are normally calibrated to a formula using the mean of the highest and lowest values in permittivity found in the range of approved fuels together with an approximate density value. 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 always totally immersed in fuel and its capacitance is determined by the permittivity of the fuel. An increase in density results in an increase in permittivity and so the unit corrects for density error.

Fuel Flowmeters

6. Fuel flowmeters measure the amount of fuel being delivered to the engine. There are two components; a transmitter and an indicator.

7. **Flow Transmitter**. The transmitter may be one of two types, volumetric or gravimetric. In the volumetric type, the fuel flows through a chamber containing a rotor which turns at a rate dependent upon the fuel flow rate. The rotation rate is detected by an electrical pick-off which passes an electrical signal to the indicator. In the gravimetric transmitter, a chamber contains a measuring device consisting of a vane restrained by a calibrated spring. The fuel flows through the chamber and impinges on the vane 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 vane is detected by a pick-off and passed as an electrical signal to the indicator.

8. **Flow Indicator**. The indicator incorporates electrical circuits which convert the signal from the transmitter into either an analogue or digital display of flow rate. If a volumetric transmitter is used a value

of fuel density has to be manually set into the unit so that a mass flow rate can be indicated. Some systems incorporate integrating circuits which enable total fuel used to be displayed on veeder counters.

Undercarriage Indicator

9. The detailed design of undercarriage indicators varies between aircraft type, but the underlying principle is universal. The indicator comprises a series of lights, or electro-mechanical flags, operated by microswitches fitted to the undercarriage locks, each showing the status of an individual undercarriage unit, provided that power is available, as follows:

a. Green light or flag - Unit locked down.

b. Red light or flag - Unit is unlocked.

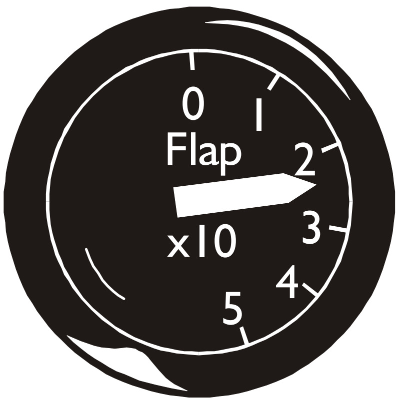
c. No light or flag - Unit is locked up.

10. Some aircraft are fitted with a visual or audio warning system operating in conjunction with the undercarriage indicator. Typically, it will be triggered when the throttle is closed beyond a pre-determined point with the undercarriage not locked down or may be actuated if a particular stage of flap is selected with the undercarriage retracted.

Calibrated Position Indicators

11. Pointer and scale type indicators are used to show the position of flaps, trimming surfaces, tailplanes, etc. The majority of these indicators are actuated by desynn transmission systems (see Volume 5, Chapter 24). A typical indicator for flap position is shown in Fig 3.

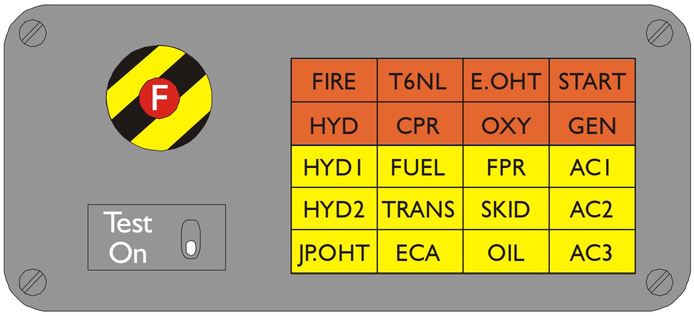
5-27 Fig 3 Desynn Operated Flap Position Indicator



Central Warning System

12. The failure of one of the vital systems in an aircraft can prejudice the success of the flight and may lead to the loss of or damage to an aircraft. Warning systems are therefore incorporated to indicate to the crew if there is a malfunction so that appropriate action can be taken. Although there may be individual warning devices for some aircraft systems, the tendency is to incorporate all warnings on to one central warning panel (CWP). The scope of the warnings and their layout vary from aircraft to aircraft, but a typical example is illustrated in Fig 4.

5-27 Fig 4 Typical Central Warning Panel



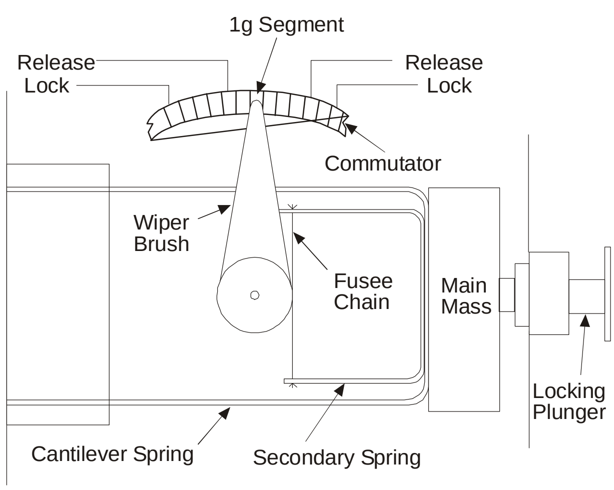
13. The warnings are divided into primary and secondary warnings; the primary, which normally demand immediate action, being illuminated red, and the secondary amber. Failure of a primary system will in addition illuminate red 'attention getter' flashing lights in the cockpit together with an audio warning; failure of a secondary system may or may not initiate these additional warnings dependent on the aircraft type and particular failure. In many installations, the CWP will also house an engine fire warning light and extinguisher operating button (as illustrated in the example at Fig 4). Operation of a test switch tests the warning system, captions, and lamps (but not the systems from which the warnings are derived).

14. CWPs do vary significantly from aircraft to aircraft and it is essential that the specific Aircrew Manual is consulted to clarify the functioning of any system.

Fatigue Meter

15. The effects of acceleration forces on an airframe and the principles of operation of a cockpit accelerometer are covered in Volume 1, Chapter 19 and in Volume 5, Chapter 22 respectively. A fatigue meter is a counting accelerometer, mounted close to the aircraft C of G, the purpose of which is to record the number of times that each of 8 pre-determined values of acceleration normal to the flight path are exceeded. The meter is only required to operate in flight and is normally activated and deactivated by an airspeed switch. The basic mechanism is illustrated in Fig 5. As acceleration forces move the mass up or down, the secondary spring and fuse chain cause the wiper brush to rotate over the commutator. As the wiper passes a selected segment a circuit is completed to 'cock' that counter, and when the acceleration force lessens by a significant amount a release circuit allows the counter to record. This arrangement ensures that only the main acceleration values are taken into account, and not the smaller fluctuations which do not cause fatigue damage. Fig 6 shows the fatigue meter presentation from which details may be entered in a Fatigue Calculation Sheet.

5-27 Fig 5 Basic Fatigue Meter Mechanism



5-27 Fig 6 Basic Fatigue Meter Mechanism