022 INSTRUMENTATION & ELECTRONICS

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Powerplant and System Monitoring Instruments

Pitot and Static Sources

Pitot and Static Sources

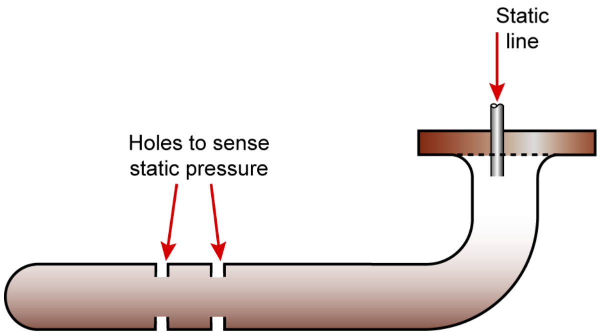
1. Aircraft pressure instruments (such as the pressure altimeter, VSI, ASI and Machmeter) require static and (for the ASI and Machmeter) pitot pressure in order to function.

Static Pressure

2. Static pressure is the **ambient air pressure** at a given point in the atmosphere. Considering an aircraft at rest in still-air conditions, this ambient pressure acts equally on all points of the aircraft.

3. Figure 1-1 shows one method of measuring static pressure, using a **static head**. The head consists of a tube with its forward end sealed and holes or slots cut into the side. The ideal situation is that the head always lies in line with the direction of relative air flow and therefore the pressure sensed is independent of any increase of pressure caused by the aircraft's speed through the air. A static head may be incorporated with the pitot head as shown in Figure 1-3.

FIGURE 1-1
Static Head



4. **Static vents** are more commonly used in modern aircraft to detect static pressure. A static vent consists of a smooth plate with a small hole in the middle. The plate is mounted flush with the aircraft skin at a point where the air flow is relatively undisturbed by the airframe structure itself. This is to ensure that, as far as possible, the static pressure sensed at the vent will be pure ambient pressure, which is free of errors caused by the presence of the aircraft or the speed of the aircraft through the air. It is normal to mount two static vents, one on each side of the aircraft, thereby cancelling errors in the sensed pressure caused by aircraft yaw or side slip; this process is called **static balancing**

5. It is normal to incorporate an **alternate static source** into the static line plumbing. In the event that the static head or the static vents become blocked the emergency static source can be selected by the pilot. This alternate source is located at some sheltered position outside the pressure hull. The pressure sensed at this source is unlikely to represent accurately the ambient air pressure, since it will almost certainly be influenced by the aircraft structure.

6. In some unpressurised aircraft an alternate static source is provided inside the cockpit. It should be noted that unless it is otherwise stated in the flight manual for the aircraft, the static pressure sensed within the cockpit will be **lower** than the true static pressure due to aerodynamic suction. The effect of this artificially low static pressure is that both the pressure altimeter and the airspeed indicator will **over-read** with the emergency static source selected.

Pitot Pressure

7. The composition of pitot pressure (sometimes referred to as **total pressure**), and the use made of it, is fully discussed in the sections dealing with the airspeed indicator and the machmeter. If the aircraft is at rest in still air conditions the pressure sensed at the pitot tube will be the static pressure already discussed. A pitot tube is shown at Figure 1-2 and, like the static head, faces forward into the airflow. In flight the pressure sensed at the pitot tube will be increased due to the aircraft's forward speed. The two elements of the pitot pressure will therefore be:

(a) the **static pressure,** and

(b) the **dynamic pressure**, or **pitot excess pressure.**

It is the **dynamic pressure** which is proportional to the aircraft's forward speed.

8. Since an emergency pitot tube cannot be fitted at a sheltered point of the aircraft, with any hope of success, it is normal to incorporate a heating element into the tube to prevent blockage due to ice formation. Any water ingested by the system is allowed to drain from the tube through drain holes and is prevented from travelling downstream through the plumbing by means of traps and valves

FIGURE 1-2
Pitot Head

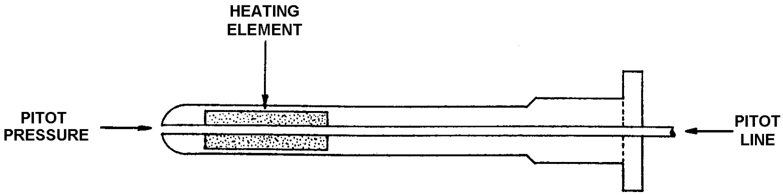
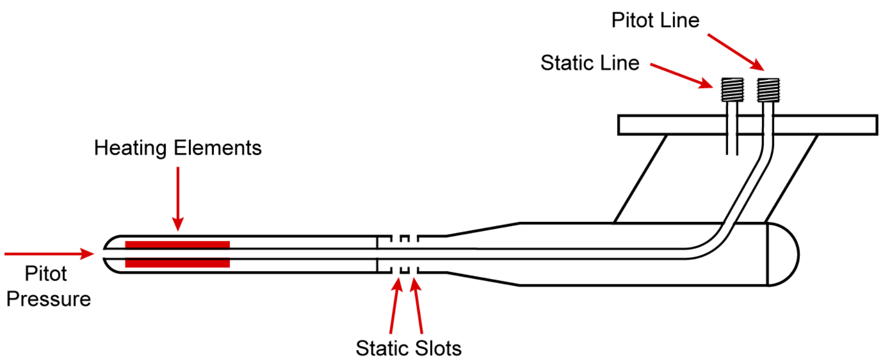


FIGURE 1-3
Pitot/Static Head



9. The incorrect measurement of static pressure is known as **position or pressure error.** The static head and the combined pitot/static head are more prone to this error than are static vents. The magnitude of the error depends on the airspeed and the aircraft attitude. The error is likely to be largest at high angles of attack when some dynamic pressure is generated at the static sensor. Flight manuals will normally provide correction values for this error for different flap settings.

10. **Manoeuvre errors** are the result of temporary fluctuations in static pressure which occur when the angle of attack of the aircraft is changing, principally when flaps and landing gear are raised or lowered. Manoeuvre errors normally cause lag in pressure instruments (including air data computers) and may persist for several seconds after the change of configuration/movement of a control surface has been completed, the higher the aircraft altitude the longer the error will persist. Although changes in pitch attitude are the primary source of manoeuvre errors, rolling and yawing manoeuvres can also give rise to this problem.

Pressure Altimeters

Pressure Altimeters

1. As an aircraft climbs through the atmosphere the static pressure, or the weight of the column of air above the aircraft, **must** decrease. An aneroid barometer may be used to determine the atmospheric pressure at a given point. The instrument which employs an aneroid capsule to determine the pressure of the column of air above the aircraft, but which subsequently indicates a level in the atmosphere at which the sensed pressure should occur, is the pressure altimeter.

2. The rate at which pressure changes with altitude is not linear, and so the calibration of any pressure altimeter is complex. The calibration constants used for pressure altimeters are those of the International Standard Atmosphere (ISA):

- At mean sea level, pressure =1013.25 mb
- At mean sea level, temperature =+15°C
- At mean sea level, density =1225 gm/cubic metre.

Between mean sea level and the tropopause (36,090 feet) temperature is assumed to drop at the rate of 1.98°C/1000 feet (6.5°C/km). Above the tropopause, temperature is assumed to remain constant at -56.5°C up to 66,000 ft (20 km).

3. Since the pressure altimeter is calibrated to these values it follows that only when these conditions pertain will the altimeter indicate the correct altitude. Furthermore, at higher altitudes where pressure reduces more slowly with height, altimeters need to be sufficiently sensitive to convert minute changes in pressure into change of indicated altitude. In round figures, a 1 millibar change of pressure represents a height change of 30 ft at sea level, 50 ft at an altitude of 20,000 ft and 100 ft at an altitude of 40,000 ft.

4. Subject to the errors discussed later, a pressure altimeter will indicate **vertical displacement above a selected pressure datum**, which is set onto the instrument sub-scale. Three sub-scale settings are normally used:

(a) **QFE** - the airfield datum pressure which, when set on the sub-scale, will cause the altimeter to read **height** (the vertical distance between the lowest part of the aircraft and a relevant datum, in an unbanked attitude with the undercarriage extended). The **threshold QFE** is normally given if the elevation of the threshold of the instrument runway in use differs from the published airfield elevation by 7 ft or more.

(b) **QNH** - an approximation of mean sea level pressure which, when set on the sub-scale, will cause the altimeter to read **altitude** (the vertical distance between the lowest part of an aircraft and a specified pressure surface datum. The pressure difference is converted to feet using the International Standard Atmosphere).

(c) **1013.25 mb** - mean sea level pressure in the standard atmosphere. When set on the sub-scale, the instrument will read **pressure altitude** (Flight levels). Pressure altitude is defined as **the altitude of an aircraft above the pressure level of 1013.2 mb**. It can be ascertained by setting the altimeter sub-scale to 1013.2 mb and reading the **indicated altitude**.

NOTE:

The student should also be familiar with the term density altitude. Density altitude is the altitude in the standard atmosphere at which the prevailing density would be found.

5. For standard British altimeters, the range of sub-scale settings is from 800 mb to 1050 mb.

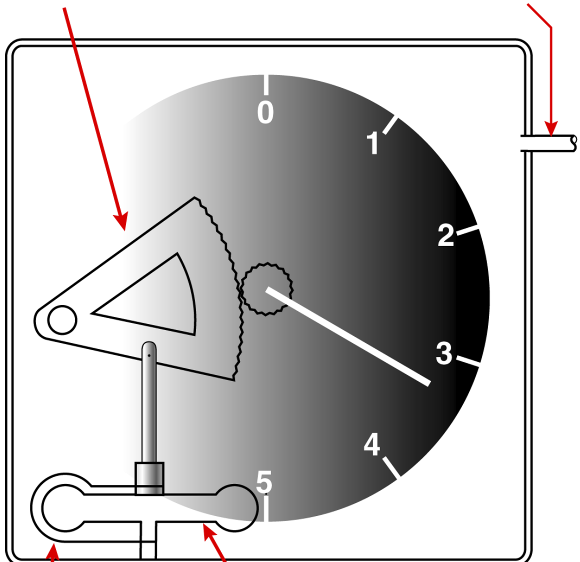
6. Please note that the **millibar** is the unit of air pressure used in most countries and it is therefore the one that is commonly used on altimeter sub scales. In some countries however, for example the USA, inches of Mercury (inch Hg) are used to define a pressure datum (29.92 inches Hg being the equivalent of 1013.25 mb). The current SI unit for pressure is the hectopascal (hPa) and, in theory at least, this should be used instead of the millibar. However, the term hectopascal is not in common use amongst aircrew and we will therefore refer to millibar settings throughout this book. Hectopascals and millibars are in fact identical for all practical purposes.

The Simple Altimeter

7. Figure 2-1 shows schematically the working of a simple altimeter. The sensing element is the partially evacuated or aneroid capsule, which is prevented from collapsing by the action of the leaf spring. Static pressure is fed into the otherwise gas-tight case of the instrument from the static vent. As the aircraft climbs, static pressure decreases and the capsule expands. The movement of the capsule is magnified by the mechanical linkage and a single needle rotates around the instrument face.

Pressure Altimeters

FIGURE 2-1
Simple Altimeter



The Sensitive Altimeter

8. The sensitive altimeter is essentially the same as the simple altimeter but employs a stack of three or four aneroid capsules to achieve a greater degree of movement for a given change of pressure. This has the effect of making the instrument more accurate. Instrument sensitivity is also enhanced, sensitivity being the ability of the instrument to register small changes of altitude.

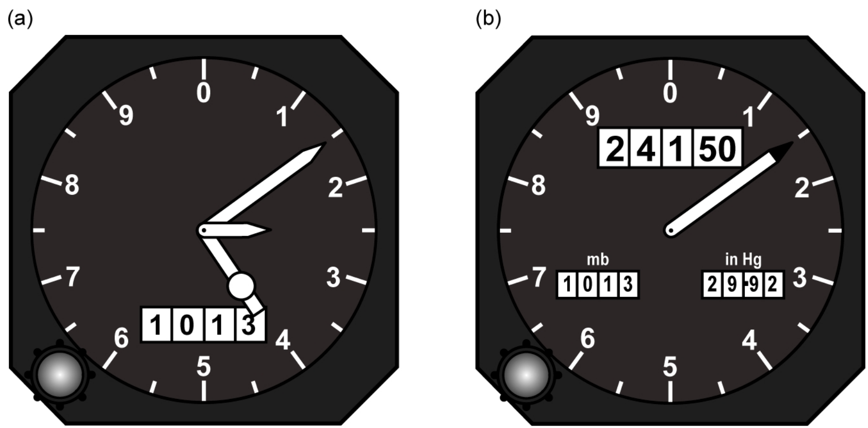
9. Accuracy in the sensitive altimeter is further improved by employing jewelled bearings within the mechanical linkage, and by incorporating a temperature compensating device. This device minimises errors which would otherwise occur as the temperature of the instrument changes, and the various elements of the mechanical linkage expand or contract.

10. The sensitive altimeter incorporates a sub-scale, upon which the required **datum** is set (QFE, QNH or 1013 mb). Finally, the sensitive altimeter normally employs two or three needles at the face of the instrument. A three-needle display is shown at Figure 2-2(a), where the instrument indicates 24,150 ft.

11. You should appreciate that it is **very easy to misread** the type of display shown at Figure 2-2(a). This very important problem is largely overcome by removing two of the three pointers and adding a digital readout, as shown at Figure 2-2(b). This is most easily achieved with a servo-assisted altimeter, which is described in the following paragraphs.

The Servo-Assisted Altimeter

FIGURE 2-2
Altimeter Presentation

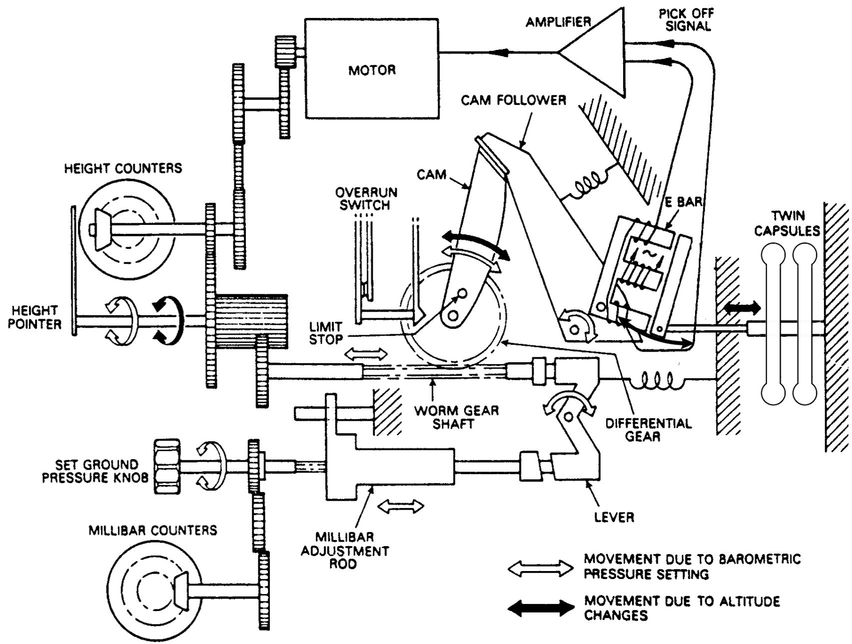


12. Figure 2-3 shows in a schematic form a servo-assisted altimeter. Its principal advantage over the sensitive altimeter is that of increased sensitivity and therefore accuracy resulting from the use of an electro-magnetic pick-off, the **E** and **I** bar, which acts as a transducer, converting capsule movement into electrical current. The **I** bar is connected to the capsule stack, whilst the output of the secondary coils of the **E** bar is amplified and used to drive the motor, which in turn drives the height pointers/height counters and, via a worm gear shaft/cam/cam follower, the **E** bar itself.

13. An alternating current is fed to the primary winding on the centre arm of the **E** bar. If the air gaps between the **E** and **I** bars are equidistant, alternating currents of **equal magnitude** will be induced into the two secondary coils on the upper and lower arms of the **E** bar, and there will therefore be no output.

14. As the aircraft climbs, the stack of capsules will expand, closing the gap at the lower arm of the **E** bar and opening the gap at the upper arm of the **E** bar. Currents of differing magnitudes will now be induced into the secondary circuits and this imbalance in secondary currents is used, after amplification, to energise the two-phase servo motor. The motor drives a system of gears which drive the altimeter needles and digital readout. The same gearing drives a worm gear shaft which causes the cam to rotate. This movement of the cam is transferred via the cam follower to the **E** bar, which moves until such time as the air gaps are again equidistant.

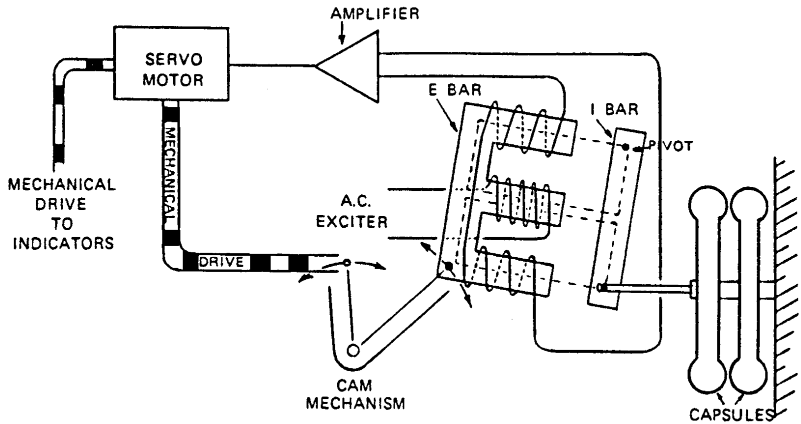
FIGURE 2-3
Servo-Assisted Altimeter



15. With this electro-magnetic linkage, bearing friction is minimal. Consequently, minute movements of the capsule stack are sensed, whereas with a mechanical linkage, such small movements would be masked by friction and/or play in the bearings.

16. Figure 2-4 shows the workings of a servo-assisted altimeter in a simplified format

FIGURE 2-4
Servo-Assisted Altimeter - Simplified Diagram



17. The advantages of the servo-assisted altimeter are that:

(a) It is more accurate at all altitudes.

(b) It is usable to greater altitudes.

(c) Digital readout of altitude is easily incorporated.

(d) Altitude warning devices are easily incorporated.

Usable Altitude and Accuracy

18. Figure 2-5 shows typical values of permissible tolerances and maximum usable altitude for the three types of altimeter previously discussed.

FIGURE 2-5
Typical Altimeter Accuracy Figures

Type	Max. usable altitude (ft)	Permissible error at MSL (ft)	Permissible error at altitude (ft)
Simple	35,000	±100	±600 at 20,000 ±1000 at 35,000
Sensitive	80,000	±70	±1000 at 40,000 ±1500 at 80,000
Servo-assisted	100,000	±30	±300 at 60,000 ±100 at 40,000

19. The figures given in Figure 2-5 above for the servo altimeter equate to an accuracy of ±1 mb at all levels.

Pressure Altimeter Errors

20. It is now necessary to consider the errors of the pressure altimeter.

Position Error

21. As previously discussed, the location of the static vent is important, since the vent is ideally sensing the pressure which would exist at that point in the atmosphere, were the aircraft not there to upset the measurement. Obviously the presence of the aircraft will have some effect on the pressure sensed at the vent, and the difference between the **actual** and the **measured** static pressure is known as **position** or **pressure error.**

22. Position error corrections can be obtained from the **Aircraft Operating Manual** however they are usually very small as shown in the extract from the Skyvan 3C AOM at Figure 2-6.

FIGURE 2-6
Altimeter Position Error Corrections - Skyvan 3C

Flaps Up	- 20 ft
Flaps Take-off (18°)	+ 20 ft
Flaps Take-off (30°)	+ 25 ft
Flaps Land (50°)	+ 30 ft

23. Position Error Corrections for the same aircraft type but when using the **alternative static source** are also given in the AOM as shown at Figure 2-7.

FIGURE 2-7
Alternate Static Position Error Corrections - Skyvan 3C

Flaps Up	- 250 ft @ 160 kts IAS - 100 ft @ 100 kts IAS
Flaps Landing	- 130 ft @ 100 kts IAS - 40 ft @ 70 kts IAS

Manoeuvre Induced Error

24. As the aircraft attitude is changed, particularly in pitch but also in roll and yaw, the magnitude of the position error at the static vent will change. This additional difference between **actual** and **measured** static pressure is known as **manoeuvre induced error**.

Instrument Error

25. Any small manufacturing imperfections in the instrument will result in inaccurate readings, as will bearing friction within the linkage. The instrument is bench-tested and calibrated before installation in order to minimise these errors.

Time Lag

26. Like everything else in the world which obeys Newton's Laws, the capsules and linkages are reluctant to change from a state of rest as the static pressure changes. This results in a small time lapse which tends to cause the instrument to under-read in a climb, and to over-read in a descent. This reluctance is termed **hysteresis error, or lag**. In some altimeters the problem is partially overcome by installing a vibrator inside the case of the instrument. The constant vibration helps to overcome the inertia within the altimeter and therefore it responds earlier to a change of altitude.

Barometric Error

27. The altimeter is calibrated to a MSL pressure of 1013.25 mbs however, the chances of the actual MSL pressure being exactly that value at any particular time are virtually nil. Any error produced as a result of this problem is called **barometric error** and, of course, the solution is to provide a sub scale setting facility on the front of the instrument so that it can be referenced to an alternative pressure.

Temperature Error

28. The pressure altimeter is calibrated to the standard atmosphere. The pressure existing at any point in the **standard** atmosphere will normally differ from the pressure existing at the same point in the **actual** atmosphere. The main reason for this is that the constants of temperature in the standard atmosphere do not often coincide with the ambient temperature conditions.

29. This **temperature error** causes the altimeter to **over-read** if the mean ambient temperature of the air between the surface and the aircraft is **colder** than that of the standard atmosphere; **this is the dangerous case**. Conversely the altimeter will **under-read** if the air is **warmer** than that of the standard atmosphere.

30. Appreciate that the temperature compensating device within the altimeter does nothing to compensate for the temperature error discussed above. The compensating device corrects only for change of temperature at the instrument, and not for variation of the mean ambient temperature from standard conditions.

31. The effects of temperature error are more likely to be critical close to the ground, for example during the final approach phase of a flight. ICAO standard procedures assume that due account is taken of the dangerous low temperature case.

32. The table at Figure 2-8 shows the values that should be added to target or published heights or altitudes to allow for temperature error. The height above the elevation of the altimeter setting source (in other words the airfield or touch down zone elevation) and the aerodrome temperature are the two entering arguments.

33. Remember that, if the approach is made with QFE set on the altimeter sub-scale, it is the decision **height** (DH) for a precision approach, or the minimum descent **height** (MDH) for a non precision approach, which is the target or published height. If however the approach is made with QNH set on the altimeter sub-scale, it is the decision **altitude** (DA) or the minimum descent **altitude** (MDA) which is the target or published altitude.

34. The appropriate correction is then added to the target height (DH or MDH) if flying on QFE or target altitude (DA or MDA) if flying on QNH.

35. For example, at an aerodrome level of 2000 ft and a temperature of -10°C the altimeter is likely to overread by 40ft at 400ft above touchdown. The table of corrections in Figure 2-8 below, are extracted from ICAO DOC 8168.

FIGURE 2-8

Altimeter Correction Table

Aerodrome temp 0°C	Height above the elevation of the altimeter setting source (feet)													
	200	300	400	500	600	700	800	900	1000	1500	2000	3000	4000	5000
0°	0	20	20	20	20	40	40	40	40	60	80	140	180	220
-10°	20	20	40	40	40	60	80	80	80	120	160	260	340	420
-20°	20	40	40	60	80	80	100	120	120	180	240	380	500	620
-30°	40	40	60	80	100	120	140	140	160	240	320	500	660	820
-40°	40	60	80	100	120	140	160	180	200	300	400	620	820	1020
-50°	40	80	100	120	140	180	200	220	240	360	480	740	980	1220
The table is based on aerodrome elevation of 2000 ft, however it can be used operationally at any aerodrome.														

Pressure Altimeters

EXAMPLE

A VOR approach is to be made at an aerodrome with an elevation of 4500 ft amsl. The temperature at the aerodrome is -20°C and the minimum decision height (MDH) for the category of aircraft in question is 800 ft. Determine the pressure altimeter reading that would correspond to the MDH under these conditions.

SOLUTION

Enter Figure 2-8 with the height above the elevation of the altimeter setting source (the aerodrome) of 800 ft and the aerodrome temperature of -20°C to achieve a correction of 100 ft. This is added to the target height (MDH) of 800 ft to give a pressure altimeter reading of 900 ft.

EXAMPLE 2-2

EXAMPLE

An NDB approach is to be made at an aerodrome with an elevation of 3500 ft amsl. The temperature at the aerodrome is -30°C and the MDA for the category of aircraft in question is 4500 ft. Determine the pressure altimeter reading that would correspond to the MDA under these conditions.

SOLUTION

Enter Figure 2-8 with the height above the elevation of the aerodrome (4500 ft MDA - 3500 elevation) of 1000 ft and the aerodrome temperature of -30°C to achieve a correction of 160 ft. This is added to the target altitude (MDA) of 4500 ft to give a pressure altimeter reading of 4660 ft.

Static Line Blockages and Leaks

36. Should the static line become blocked, the altimeter will thereafter continue to show the altitude at which the blockage occurred. Conversely, should the static line fracture within the pressure hull of a pressurised aircraft, the altimeter will thereafter show cabin pressure altitude rather than the aircraft altitude. A static line leak within the fuselage of an unpressurised aircraft will probably result in the altimeter over-reading the correct value.

Height Encoding

37. Secondary surveillance radar (SSR) is described in the Radar Aids section of this course, however perhaps now it is appropriate to consider mode C of the SSR, which is the automatic function by means of which the aircraft's SSR transmits the aircraft pressure altitude as a pulse train in response to a ground secondary radar interrogation.

38. Height encoding is achieved using an electro-optical system (or is extracted from the central air data computer).

39. The heart of the electro-optical system is an encoder disc which is made of glass and is designed with a pattern of eleven concentric rings with a mix of transparent and opaque slots. The disc is rotated using an electrical output from a servo-assisted altimeter, one rotation equating to the full height range of the altimeter. A single line of light is arranged to pass through the disc and, depending on the position of the disc, a pattern of light and shade is formed which registers on a bank of photo-electric cells located behind the disc. The light pattern is converted into a series of binary pulses which represents the aircraft's pressure altitude in increments of 100 ft. Note that the height encoder is based on a fixed datum of 1013 mb, regardless of the subscale setting.

Airspeed Indicators

Airspeed Indicators

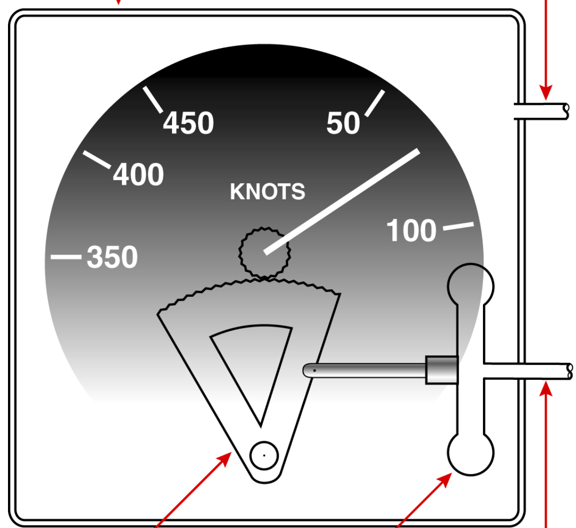
1. As previously discussed the pitot tube senses a pressure in flight which comprises the static pressure plus the dynamic pressure. It is the **dynamic** pressure which is proportional to the aircraft's forward speed through the air.

2. The function of the airspeed indicator (ASI) is to isolate the dynamic pressure, and to use it to move the needle of the instrument across a suitably calibrated scale. A simple ASI is illustrated schematically at Figure 3-1 showing indicated airspeed in knots, which is the most common unit of speed used. However, the ASI can be calibrated in kilometers per hour (KPH).

Airspeed Indicators

FIGURE 3-1

Airspeed Indicator



3. Pitot pressure is fed into the capsule itself, whilst static pressure is fed into the gas-tight instrument casing. Putting it another way, the pressure inside the capsule consists of **dynamic + static pressure**, and pressure outside the capsule is entirely **static** pressure. The **static** pressure is therefore effectively eliminated, and any expansion of the capsule is due entirely to **dynamic** pressure.

4. In other words, the pressure causing the capsule to expand is effectively;

- pitot pressure (P) - static pressure (S)

however;

- pitot pressure (P) = dynamic pressure (D) + static pressure (S)

and therefore;

- $P - S = (D + S) - S = D$

5. Dynamic pressure is directly proportional to the speed of the aircraft through the air, the true airspeed (TAS). However, as dynamic pressure is in effect a measure of the number (mass) of molecules of air entering the capsule in unit time it is also directly dependent on air density (the denser the air the lower the TAS required to produce the same dynamic pressure). This relationship is illustrated by the formula:

$$\text{Dynamic Pressure (D)} = \frac{1}{2}\rho V^2$$

where ρ is density and V is **true** airspeed

6. With appropriate calibration constants dynamic pressure is converted into indicated airspeed (IAS). The calibration of the ASI is based on the density of air at MSL in the international standard atmosphere, that is to say 1225 gm/m^3. When an aircraft is flying in air at this density the IAS and the TAS will be the same and with an aircraft flying at altitude (in air at a density of less than 1225 gm/m^3) the IAS will be less than the TAS.

Square Law Compensation

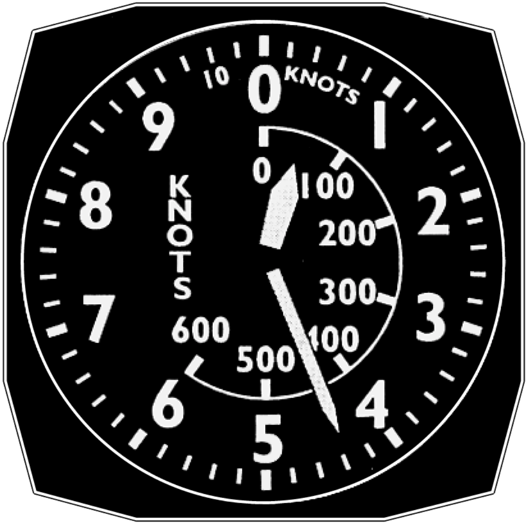
7. The ASI measures dynamic pressure, which varies directly with the square of true airspeed, and the deflection of the capsule is directly proportional to the dynamic pressure. With a straightforward linkage between the capsule and the pointer the speed scale on the face of the instrument would necessarily be distorted, with a much expanded scale at the high speed end. Airspeeds at the low (critical) speed end of the scale would therefore be difficult to read accurately.

8. In fact the architecture of the mechanism linking the capsule to the pointer is arranged to give either a linear presentation, or more usually a logarithmic-type presentation, such that the range is expanded at the lower end of the speed scale where accuracy of presentation is most needed, as shown at Figure 3-1.

Sensitive and Servo Airspeed Indicators

9. In the **Sensitive** and **Servo Airspeed Indicators** extra sensitivity is achieved by using a capsule stack rather than a single capsule, plus additional gearing to drive two pointers over a linearly calibrated dial. A further improvement in the **Servo ASI** is that the mechanical linkages are replaced by an electrically driven transmission system with power amplification. Figure 3-2 illustrates the type of display commonly found on a two pointer sensitive ASI.

FIGURE 3-2
Typical Sensitive ASI



ASI Errors

10. The errors affecting the ASI are considered below in the order in which the appropriate corrections are applied to the indicated reading to achieve true airspeed.

Instrument Error

11. Any small manufacturing imperfections such as bearing friction and play in the bearings as the instrument ages will result in small errors in IAS. As with the altimeter, the instrument is bench-tested and calibrated before installation, and any residual errors noted.

Position Error

12. As already discussed, the static vents suffer from position (or pressure) errors. It is again possible to establish and note the magnitude of the position error across the whole range of airspeeds. If the errors are established for a **clean** aircraft, they will not necessarily apply when the undercarriage and flaps are lowered. The position errors pertinent to **normal operation** will not apply in the event that the emergency/alternate static source is in use. Position Error corrections for different configurations/conditions can be obtained from the **Aircraft Flight Manual.**

13. Manoeuvre induced errors are realistically changes in the magnitude of the position error which occur particularly when the aircraft changes attitude in pitch. Such manoeuvre induced errors cannot be tabulated since they are somewhat random in nature.

Compressibility Error

14. Compressibility error is considered for the purpose of this syllabus only at **true** airspeeds in excess of 300 knots. At high air speeds the air **compresses** when brought to rest in front of the pitot head and consequently enters the tube at an artificially high density. This results in an increased pressure in the capsule, causing the ASI to **over-read.**

Density Error

15. As already established the ASI is calibrated to correctly indicate the aircraft's **true** airspeed only at a density of 1225 gm/m^3. At other air densities the instrument will mis-read, and this is due to **density error**. At the low densities associated with high altitude flight the instrument will **under-read** the **true** airspeed by very significant amounts.

Static Line Blockages And Leaks

16. Should either the pitot or the static line become blocked the ASI will thereafter read incorrectly. With take-off safety speeds and approach speeds being as critical as they are, especially for large and high-performance aircraft, this obviously presents a potentially critical situation.

17. Assume in the first case that the static line has become blocked whilst flying at altitude. As the aircraft descends the static pressure sensed at the static vent and fed to the ASI **should** increase at the same rate as the static element of pitot pressure. Since the static line is blocked the static pressure surrounding the capsule will remain **artificially low**, and the instrument will **over-read** during the descent.

18. Conversely if the static line becomes blocked at low altitude the ASI will **under-read** during a climb to a higher altitude.

19. Should the pitot tube become blocked the instrument would **tend** to behave in the reverse manner to that described above. The problem will however be complicated by the relationship between the airspeed at which the blockage occurred and the airspeed flown during the climb or descent.

20. A static leak inside an unpressurised cabin would tend to result in the ASI over-reading due to the cabin pressure being lower than ambient because of aerodynamic suction.

Airspeed Definitions

21. The definitions which follow use the criteria observed in the majority of British textbooks. As always the Americans have invented their own language, and taken much of the free world with them.

Indicated Airspeed

22. Logically **indicated airspeed** (IAS) is that which is shown on the face of the ASI.

Calibrated Airspeed (CAS)/Rectified Airspeed (RAS)

23. IAS corrected for **instrument error** and **position (pressure) error**, using the correction table adjacent to the instrument or to be found in the pilot's notes for the aircraft, gives **calibrated airspeed** (CAS). Calibrated airspeed is also known as rectified airspeed (RAS).

Equivalent Airspeed (EAS)

24. CAS/RAS corrected for **compressibility error** gives **equivalent airspeed** (EAS).

True Airspeed (TAS)

25. EAS corrected for **density error** gives **true airspeed** (TAS).

26. The calculation of CAS and TAS is fully covered in the Mathematics section of the syllabus. It is not possible to calculate EAS using a standard navigation computer, but fortunately the calculation of EAS is not required in the syllabus.

Other Speed Definitions

V_{SO}. The stalling speed (or if no stalling speed is obtainable, the minimum steady-flight-speed) with the wing-flaps in the landing setting.

V_{S1}. The stalling speed (or if no stalling speed is obtainable, the minimum steady-flight-speed) with the aeroplane in the configuration appropriate to the case under consideration.

V_{FE}. The maximum speed at which it is safe to fly with the flaps in a prescribed extended position.

V_{MO}/M_{MO}. The Maximum Operating Speed (or Mach number, whichever is critical at a particular altitude) which must not be deliberately exceeded in any flight condition. This speed is that which, allowing for moderate upsets, ensures the aircraft will remain free from buffet or other undesirable flying qualities associated with compressibility.

V_{NE}. The speed which must **never** be exceeded.

V_{LO}. The maximum speed at which the undercarriage (landing gear) may be safely extended or retracted.

V_{LE}. The maximum speed at which the aeroplane may be safely flown with the undercarriage extended.

V_{YSE}. The speed used to climb at the maximum rate of climb in a twin-engined aeroplane with one-engine inoperative is referred to as V_{YSE}.

Colour Coding of Airspeed Indicator Speed Scales

27. The critical speed ranges on an airspeed indicator may be identified by coloured arcs (known as range markings) using the convention shown below;

White arc 1.1 x the stall speed (V_{SO}) with full flap, landing gear down, all power units at idle and with the aircraft at maximum certified all up weight to the maximum speed with flap extended (V_{FE}).

Green arc 1.1 x the clean stall speed (V_{S1})at maximum all up weight to the maximum structural cruising speed (V_{MO}/V_{NO} or V_{RA}). This is the normal operating range.

Yellow arc V_{MO}/V_{NO} or V_{RA} to the smooth air never exceed speed (V_{NE}). This is the precautionary range.

A red radial line indicates V_{NE} (the top of the yellow arc).

A blue radial line indicates the best single engine rate of climb speed (V_{YSE}).

Mach/Airspeed Indicators

Mach/Airspeed Indicators

1. The Mach/airspeed indicator combines the functions of the Machmeter and the airspeed indicator and displays both speeds on a single instrument face. It is, in effect, the two conventional instruments in one case.

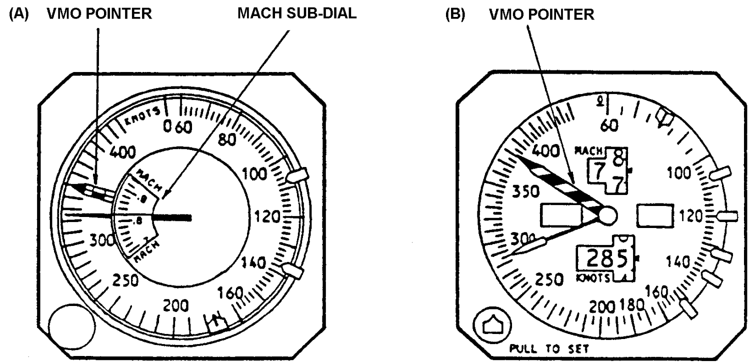
2. On the face of the instrument [see Figure 4-1(a)] the IAS is shown by a conventional dial/pointer system and the Mach number by a rotating dial, the relevant part of which is visible through the window in the face of the instrument (9 o' clock position) and which moves independently of the airspeed pointer as the Mach number changes.

3. A second pointer, the V_{MO}/M_{MO} pointer or barber's pole (so called because it is coloured with red and white) is incorporated to indicate to the pilot the maximum safe indicated speed at which the aircraft can be flown, more of this later.

4. Additionally, a ‘command bug’ is normally incorporated, and again on the basic instrument this is set by the operator using the knob in the 7 o' clock position, either for reference to indicate the target speed or as a datum for the autothrottle in automatic flight. Finally, two or more plastic ‘bugs’ are mounted on the bezel of the instrument (outside of the glass) and these are used to define, for example, V speeds for departure and threshold speeds for arrival.

Mach/Airspeed Indicators

FIGURE 4-1
Mach/Airspeed Indicator



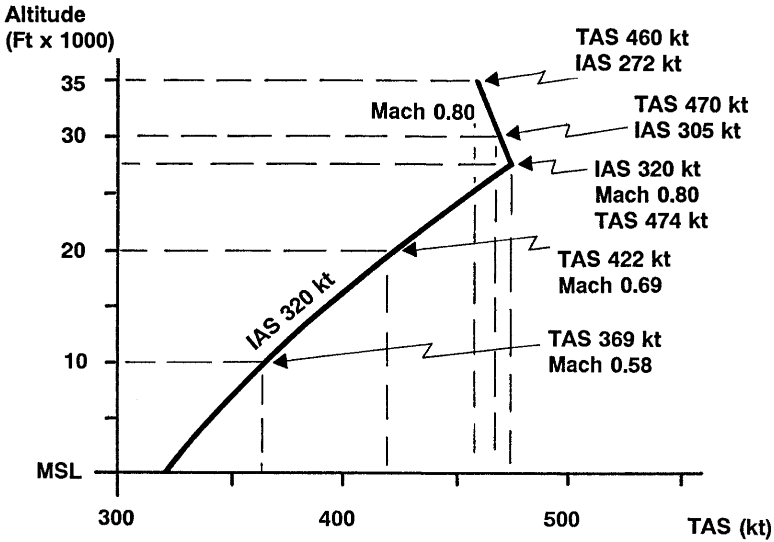
5. As with the conventional ASI, the difference between static and pitot pressure drives the airspeed pointer. Changes in both dynamic and static pressure cause the Mach number dial to rotate beneath the Mach window, anti-clockwise for an increase in altitude at a constant indicated airspeed. The instrument at Figure 4-1(a) shows the aeroplane to be flying at an indicated airspeed of 325 kt which, in this case, equates to a Mach number of 0.82.

6. It is necessary to govern the speed of a jet aeroplane in terms of indicated airspeed at relatively low altitudes and in terms of Mach number at higher altitudes. To illustrate this point, consider an aircraft which has a V_{MO} (maximum safe indicated airspeed) of 350 kt and an M_{MO} (maximum safe Mach number) of 0.84. Assume that this aircraft is climbing to FL 350 in the standard atmosphere at an indicated airspeed of 320 kt until Mach 0.80 is achieved and thereafter at Mach 0.80.

7. As the aircraft passes 10,000 ft in the climb the IAS pointer will show 320 kt and the Mach number indicated against the reference line in the Mach window will be 0.58. As the aircraft passes 20,000 ft in the climb the IAS pointer will still show 320 kt and the Mach reference line will indicate Mach 0.69 on the window scale. As the aircraft passes 27,500 ft the IAS pointer will show 320 kt and the Mach reference line will show Mach 0.80. From this point on in the climb the aircraft is flown at a constant Mach number, and so on passing 30,000 ft at an indicated Mach number of 0.80 the IAS pointer will show 305 kt. On reaching 35,000 ft at an indicated Mach number the IAS pointer will show an IAS of 272 kt. During this climb profile the true airspeed will have increased from 369 kt at 10,000 ft to 474 kt at 27,500 ft and thereafter decreased to 460 kt at 35,000 ft as shown at on the graph at Figure 4-2.

Mach/Airspeed Indicators

FIGURE 4-2
Typical Speed/ Height Profile - Jet Aeroplane



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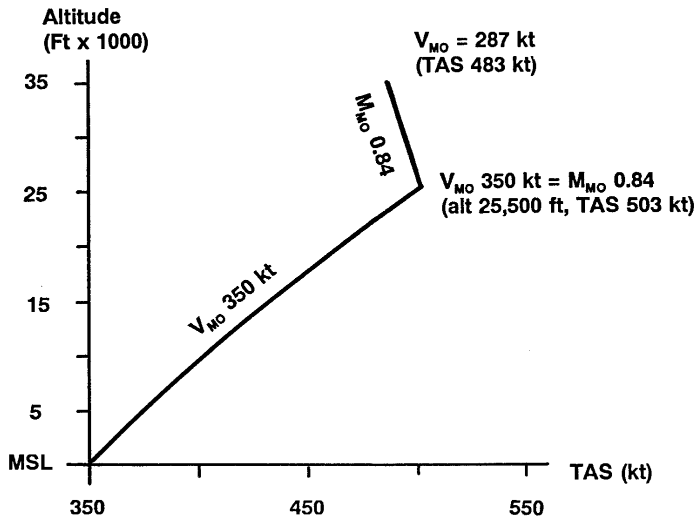
8. Now to consider what has happened to the barber's pole during the climb, again assuming that the climb took place in the standard atmosphere. Appreciate that, although the pole is often termed the V_{MO} pointer, it is also required to indicate the IAS which equates to the M_{MO} once an altitude is reached where M_{MO} becomes the limiting speed. The barber's pole will therefore indicate a fixed speed (V_{MO}) in the climb up to this altitude and a decreasing airspeed, governed by the aircraft's M_{MO}, during the subsequent climb.

9. For our aircraft the V_{MO} of 350 kt is limiting up to an altitude of 25,500 ft. Above this level M_{MO} becomes the limiting factor and, if the barber's pole is to serve its purpose, it must move anti-clockwise over the IAS scale to show a decreasing airspeed. In our case the pole must indicate 350 kt at 25,500 ft, decreasing to 287 kt at 35,000 ft, as shown on the graph at Figure 4-3. This can be achieved quite simply by means of a pin on the rotating Mach scale which picks up the barber's pole as the Mach scale rotates anti-clockwise with increasing altitude. The pole is spring loaded so that, when the aeroplane descends and the Mach scale rotates in a clockwise direction, the pole returns to its original position

FIGURE 4-3
Typical V_{MO}/M_{MO} Limitations



10. Since the instrument is an ASI and a Machmeter combined, it suffers the errors of the two individual instruments, **the ASI errors being numerically predominant**.

11. Mach/airspeed indicators are frequently fitted with actuation switches which are used to trigger automatic audio/visual warnings when selected speeds, such as V_{MO}/M_{MO} or the maximum gear extended speed are exceeded.

12. We have yet to consider the central air data computer or CADC. Most modern jet aircraft are equipped with a CADC, and this means that the Mach/airspeed indicator can by modified to advantage. Since both IAS and Mach number are computed within the CADC and fed to the Mach/airspeed indicator (which now, like all CADC driven instruments contains no capsules) as electrical signals, it is a simple matter to present the both speeds as digital readouts, which are much easier to read (or more difficult to misread). Such a display is shown at Figure 4-1(b).

Machmeters

Machmeters

1. **Mach number** is an expression of the speed of an aircraft **as a ratio of the aircraft's true airspeed to the local speed of sound.**

2. The machmeter shows the aircraft speed as a **Mach number**, where:

$$\text{Mach number} = \frac{\text{true airspeed}}{\text{local speed of sound}}$$

3. With high-speed aircraft the Machmeter is an essential instrument. As the aircraft travels through the air it creates pressure waves which travel ahead of the aircraft at the **local speed of sound.** These pressure waves act as an advance warning to the air ahead of the aircraft, causing it to diverge and giving a streamlined flow past the aircraft. Obviously if the aircraft is itself travelling at the local speed of sound (Mach 1.0) no such warning is given and severe buffeting will occur. This buffeting will result in considerable increase in drag and some loss of lift. Additionally it is possible that structural damage and loss of control may occur.

4. The air over the thickest part of the wing will achieve the local speed of sound at aircraft speeds which are below Mach 1.0. The aircraft speed (expressed as a Mach number) at which this occurs is known as the **critical Mach number.** Thus a knowledge of Mach number is of vital importance.

Local Speed of Sound

5. The value of the local speed of sound varies only with temperature. The formula for calculating the local speed of sound (LSS) is:

$$LSS = 38.94\sqrt{T}$$

where;

LSS is given in knots

38.94 is a constant

T is temperature expressed in degrees absolute or kelvin ($0°C = +273°A$)

EXAMPLE 5-1

EXAMPLE

Using the formula calculate the local speed of sound at a point in the atmosphere where:
(a) The temperature is +15°C.
(b) The temperature is -56.5°C.

SOLUTION

(a)

$$\text{LSS} = 38.94\sqrt{T°A}$$
$$+15°C = 288°A$$
$$\text{LSS} = 38.94\sqrt{288}$$
$$\text{LSS} = 38.94 \times 16.97$$
$$\text{LSS} = 661\text{kt}$$

(b)

$$\text{LSS} = 38.94\sqrt{T°A}$$
$$-56.5°C = 216.5°A$$
$$\text{LSS} = 38.94\sqrt{216.5}$$
$$\text{LSS} = 38.94 \times 14.71$$
$$\text{LSS} = 573\text{kt}$$

6. These calculations can equally well be completed on the navigation computer, using the **Mach number index arrow.** For the computer solution of LSS and Mach number, see the Mathematics section of the syllabus.

7. From the above calculations it can be seen that, as temperature decreases with altitude (in general terms), the local speed of sound also decreases.

8. Going one step further it should be obvious that, for an aircraft climbing at a constant CAS, Mach number will increase on two counts;

(a) as density decreases TAS will increase;

(b) as temperature decreases LSS will decrease and

(c) since Mach number is the ratio of (a) over (b) Mach number will increase.

9. The normal climb profile for a modern subsonic jet is to climb at a constant IAS until the required Mach number (safely below Mach crit) is reached, and thereafter to climb at a constant Mach number.

Principle of Operation

10. As already explained, Mach number is the ratio of TAS to LSS. This is True Mach Number. An instrument which would be capable of measuring this ratio would therefore need to be supplied with the value of TAS and LSS. TAS can be derived from an air data computer, and LSS can be computer-modelled. An instrument which would measure True Mach Number would thus be computer based.

11. The Machmeter cannot measure the local speed of sound. It is an analogue, pressure-driven instrument, which displays indicated Mach number. (From the practical point of view in the operation of an aeroplane it is not necessary to differentiate between True Mach Number and Indicated Mach Number, since they are virtually the same).

12. TAS is a function of pitot excess pressure (the difference between pitot and static pressure), and density. LSS is a function of static pressure and density. Density is common to both functions.

13. Thus TAS and LSS may be expressed as pressure ratios, and it is this that the Machmeter measures. It employs, in combination, an airspeed capsule (CAS) and an altimeter capsule (static pressure).

(a) The airspeed capsule expands only because of **dynamic pressure** (D), and:

$$D = \frac{1}{2}\rho V^2$$

where; ρ is density

V is TAS

therefore; TAS is proportional to $\frac{D}{\rho}$

(b) The aneroid capsule expands only because of static pressure (S). Now:

$$LSS = 38.94\sqrt{T}$$

and therefore; LSS is proportional to T

however; ρ is proportional to $\frac{S}{T}$

and therefore; T is proportional to $\frac{S}{\rho}$

and so; LSS is proportional to $\frac{S}{\rho}$

Going back to the beginning: $\text{Mach number} = \frac{TAS}{LSS}$

and so, combining (a) and (b) above;

(c) Mach number is proportional to $\frac{D}{\rho} \div \frac{S}{\rho}$

In the formula at (c) above, the density ρ cancels out from both parts of the equation, and so;

Mach number is proportional to $\dfrac{D}{S}$

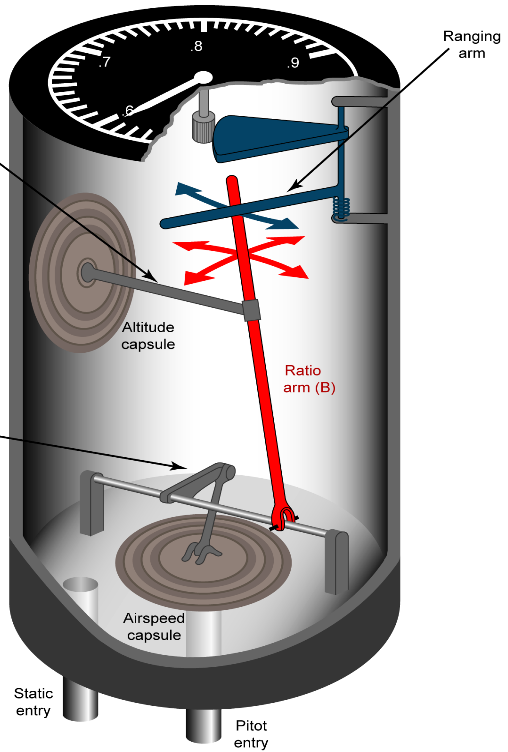
or (since D = P - S);

Mach number is proportional to $\dfrac{P - S}{S}$

14. The machmeter is shown schematically at Figure 5-1

Machmeters

FIGURE 5-1
Machmeter - Schematic



15. Let us start by considering an increase in airspeed at a constant altitude. The airspeed capsule will expand and the movement of arm A will cause the **ratio arm** (B) to move towards the bottom left hand corner of the page; this movement is translated through the **ranging arm** and a mechanical linkage to result in a clockwise rotation of the needle on the face of the instrument (an increase in the indicated Mach number). The spring which is shown at Figure 5-1 ensures that the whole transmission path remains suitably tensioned.

16. Should the aircraft now climb to a greater altitude at a constant CAS there will be no movement of the airspeed capsule, however the indicated Mach number must increase. This is achieved by an expansion of the altitude capsule which moves the mainshaft (from left to right at Figure 5-1). The **ratio arm** consequently moves the **ranging arm** towards the right hand side of the page and this will again result in a clockwise movement of the needle on the face of the instrument.

17. The permissable tolerance figure for a machmeter is ± 0.01M over the range 0.5 to 1.0M.

Machmeter Errors

18. The machmeter suffers only from **instrument** and **position** (**pressure**) errors. In fact the effect of position error on the machmeter is more pronounced than on the ASI, since the machmeter uses the **ratio** of D to S, whereas the ASI uses P - S.

19. The machmeter **does not** suffer from density error. As can be seen at formula (c) on Page 5-4, density equally affects the dynamic and the static pressures, and is therefore cancelled out of the equation.

20. The machmeter **does not** suffer from temperature error, since non-standard temperature will simply change the density of the air, and density error has already been seen to be eliminated.

21. The machmeter **does not** suffer from compressibility error, since compressibility is a function of D divided by S, and it is this ratio which the Machmeter uses to calculate the Mach number.

Static Line Blockages And Leaks

22. In the event that the static line becomes blocked the machmeter will behave in the same way as the ASI, that is to say it will **over-read** in a descent below the altitude at which the blockage occurred, and **under-read** in a climb above the altitude at which the blockage occurred.

23. The reason for this error is twofold, to illustrate the problem take an aircraft descending below the altitude at which the blockage occurred.

$$\text{Mach number is proportional to } \frac{D}{S}$$

or more precisely,

$$\text{Mach number is proportional to } \frac{P - S}{S}$$

24. The static pressure (S) on the top line of the equation immediately above is the static element of the pitot pressure. It **should** be cancelled out by the static pressure surrounding the capsule (S) on the bottom line of the equation, but this is artificially low because of the blockage. The airspeed capsule is therefore expanded.

25. The static pressure (S) on the bottom line of the equation immediately above is also the pressure affecting the altitude capsule. The pressure should be increasing during the descent, but the blockage prevents this from happening and so the altitude capsule remains in its expanded state.

26. Because both capsules are expanded the instrument is, in this case, over-reading.

Mach Number Calculations

27. The calculation of LSS by formula has already been shown at Example 5-1. The calculation of LSS and Mach number by computer is covered in the Mathematics syllabus, but a further example is included here for revision purposes.

28. If the **temperature deviations** given in the questions puzzle you, refer to the PPSC Meteorology Theory notes. The **standard atmosphere** referred to in the following questions assumes a mean sea level temperature of +15°C and a lapse rate of 2°C/1000 ft with no tropopause.

EXAMPLE

At flight level 330 the CAS of an aircraft is 285 kt. The temperature deviation from standard is -12°C.

Use your computer to determine:

(i) The TAS

(ii) The local speed of sound

(iii) The Mach number

SOLUTION

FL 330	standard temperature	=	-51°C
	temperature deviation	=	-12°C
	ambient temperature	=	-63°C

By Computer:

(a)	TAS	=	470 → compressibility → 456 kt
(b)	LSS	=	564 kt
(c)	Mach no.	=	0.81

EXAMPLE 5-3

EXAMPLE

Calculate, **without using a computer**, the altitude in the standard atmosphere at which TAS 470 kt corresponds to Mach 0.82.

SOLUTION

$$\text{Mach no} = \frac{\text{TAS}}{\text{LSS}}$$

therefore;

$$\text{LSS} = \frac{\text{TAS}}{\text{Mach no}}$$

$$\text{LSS} = \frac{470}{0.82}$$

$$\text{LSS} = 573\text{kt}$$

now;

$$\text{LSS} = 38.94\sqrt{\text{T}}$$

$$\sqrt{\text{T}} = \frac{\text{LSS}}{38.94}$$

$$\sqrt{T} = \frac{573}{38.94}$$

$$\sqrt{T} = 14.71$$

$$T = 217°$$

$$T = -56°C$$

-56°C occurs at FL 355 in the standard atmosphere.

Machmeters

EXAMPLE

If a decrease of 0.13 in the Mach number results in a decrease of 77 kt in the TAS what is the local speed of sound?

SOLUTION

$$\text{Mach no} = \frac{\text{TAS}}{\text{LSS}}$$

therefore;

$$\text{LSS} = \frac{\text{TAS}}{\text{Mach no}}$$

$$\text{LSS} = \frac{77}{0.13}$$

$$\text{LSS} = 592\text{kt}$$

EXAMPLE

An aircraft is at FL 350, TAS 463 kt at Mach 0.79 when the temperature deviation from standard is +9°C. **Without using a computer** give the temperature deviation at FL 310 which at Mach 0.79 would give a TAS of 463 kt.

SOLUTION

At FL 350, ISA +9°C, the temperature is -46°C.

At FL 310, since Mach number and TAS remain the same as at FL 350, the LSS must be the same, and the temperature must also be -46°C, **since LSS varies only with temperature.**

At FL 310, temperature -46°C, the deviation from the standard atmosphere temperature of -47°C is + 1°C.

Machmeters

EXAMPLE

If an aircraft climbs from sea level to 30,000 feet in the standard atmosphere at a constant Mach number, would the TAS remain constant, increase or decrease?

SOLUTION

$$\text{Mach no} = \frac{\text{TAS}}{\text{LSS}}$$

As the aircraft climbs LSS decreases.

Mach number remains constant.

Therefore **TAS must decrease.**

EXAMPLE 5-7

EXAMPLE

When climbing in the standard atmosphere at a constant CAS, state whether the Mach number would increase, decrease or remain the same.

SOLUTION

$$\text{Mach no} = \frac{\text{TAS}}{\text{LSS}}$$

In the climb at constant CAS, TAS would increase.

In the climb LSS would decrease.

Therefore the **Mach number must increase.**

or alternatively:

$$\text{Mach no} \; \alpha \frac{\text{D}}{\text{S}}$$

In the climb D is constant (CAS).

In the climb S (static pressure) would decrease.

Therefore the **Mach number must increase.**

EXAMPLE

When descending in a temperature inversion at constant Mach number, state whether the CAS would increase, decrease, or remain the same.

SOLUTION

Mach no $\alpha \frac{D}{S}$

From the question the Mach number remains constant.

Therefore $\frac{D}{S}$ is constant but S (static pressure) is increasing.

D and therefore **CAS must also increase.**

We see from the example above that the temperature inversion is irrelevant. If the question had been asking about TAS, then the reversal of temperature rate would have to be considered, since Mach and TAS are temperature related.

In other words, if you are considering a Mach/CAS relationship, static pressure is important. If you are concerned with a Mach/TAS relationship, temperature is important.

Vertical Speed Indicators

Vertical Speed Indicators

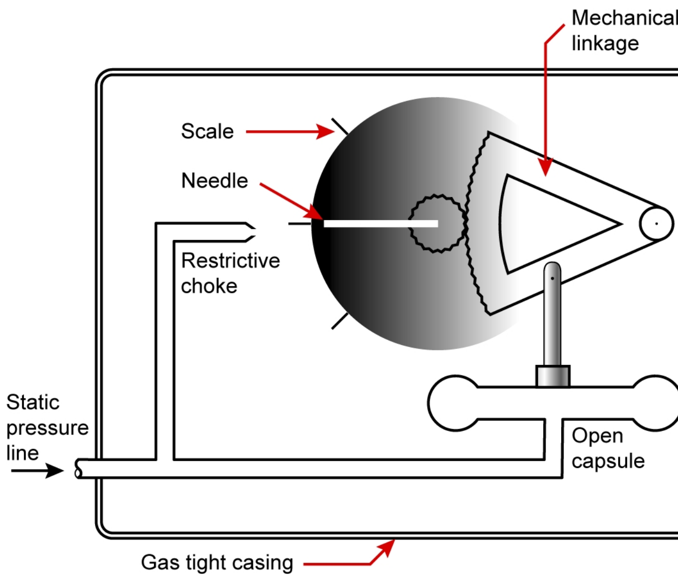
1. The vertical speed indicator (VSI) is otherwise known as the rate of climb and descent indicator (RCDI).

2. A VSI is illustrated schematically at Figure 6-1. Static pressure is fed directly into the capsule, and into the gas-tight instrument casing through a **restrictive choke** or **metering unit**.

3. As the aircraft climbs or descends the static pressure will change, the greater the rate of change of altitude the greater the rate of change of pressure.

4. The changing pressure associated with the climb or descent is sensed immediately within the capsule, however the restrictive choke imposes a small time delay before the change of pressure is sensed outside the capsule. This causes the capsule to **contract** during a **climb** and to **expand** during a **descent**, the greater the rate of change of altitude the greater the distortion of the capsule.

FIGURE 6-1
Vertical Speed Indicator - Schematic



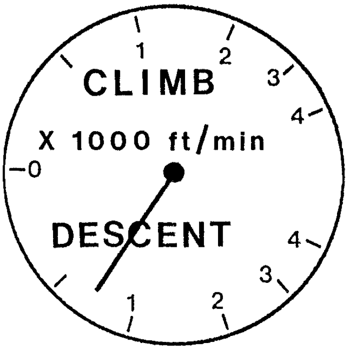
5. The movement of the capsule wall is conveyed via the mechanical linkage to a needle which moves across either a **linear** scale, or more commonly a **logarithmic** scale. An adjuster screw on the front of the instrument facilitates zeroing of the needle before flight should it be necessary. A VSI instrument face with a logarithmic scale is shown at Figure 6-2.

6. Most VSI scales are calibrated in ft/minute however metres/sec is used occasionally as an alternative unit of measurement.

7. It is normal to fit **diaphragm overload stops** to prevent damage in the event that the rate of climb or descent exceeds the maximum design values of the instrument. From the pilot's point of view this means that an aircraft fitted with the VSI illustrated at Figure 6-2 and descending at 6000 ft/min will show an indicated rate of descent of only 4000 ft/min.

FIGURE 6-2
Typical Logarithmic VSI Display

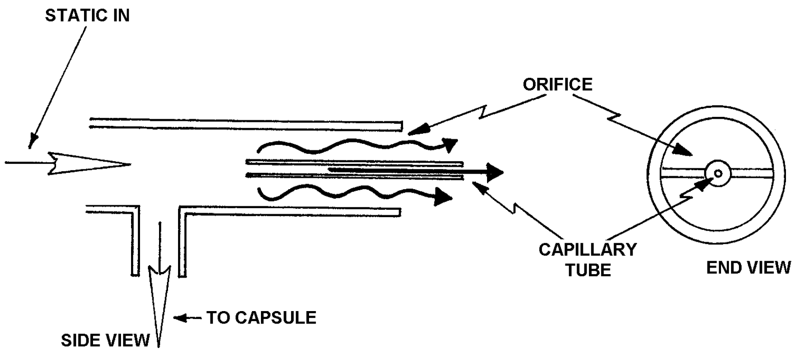
VSI WITH LOGARITHMIC SCALE SHOWING ±4000 FT/MIN AT MAXIMUM NEEDLE DEFLECTION



8. Now for a closer look at the restrictive choke. It has already been established that **the purpose of the choke is to create a pressure difference between the capsule and the instrument casing** which is proportional to the rate of climb or descent. Unfortunately the rate of change of pressure with height varies with altitude and additionally variations in air temperature and viscosity will further affect the operation of the choke. It is therefore necessary to use a metering unit comprising a choke (or chokes) which is capable of correcting for these variations.

9. One type of metering unit which is frequently used comprises a capillary and orifice arrangement, as illustrated at Figure 6-3.

FIGURE 6-3
VSI Metering Unit



10. Airflow through the capillary is laminar (that is to say that it flows in parallel lines), whereas the airflow through the orifice is turbulent.

11. The rate of flow through the capillary varies directly as the differential pressure and therefore the flow slows down with increasing altitude (decreasing pressure) for a given rate of climb. Conversely, the rate of flow through the orifice increases (for a given rate of climb) as the ambient pressure decreases at a given temperature. The result of this is that the two components combine to compensate for pressure variations with height.

12. As temperature (for a given pressure) reduces, the rate of flow through the capillary increases. The reason for this is that laminar airflow is subject to the viscosity of the air (viscosity is the resistance to flow in parallel layers and with gases, unlike liquids, the viscosity increases as temperature increases). Therefore, as the temperature drops so does the viscosity, and consequently the flow rate increases. Turbulent flow through the orifice is not affected by viscosity, in fact the flow of air through the orifice decreases as the temperature falls. Again the two components combine to cancel the effect of temperature changes.

13. The accuracy of the VSI may be checked against the pressure altimeter in straight and level flight and against the altimeter and stop-watch in a steady climb or descent.

14. Apart from its obvious uses during climbing and descending phases of flight, the VSI is invaluable when flying instrument approaches, especially those where there is no positive guidance in the vertical plane.

VSI Errors

15. It is now necessary to consider the errors pertinent to the VSI.

Time Lag

16. A short period of time will necessarily elapse before the **pressure differential** appropriate to the rate of climb or descent is established within the instrument.

Instrument Error

17. As with all instruments this is caused by manufacturing imperfections.

Position Error

18. If the static vent is subject to position (or pressure) error, the VSI will show an erroneous rate of climb or descent when the speed of the aircraft changes suddenly. This may well be noticeable during the take-off ground roll.

Manoeuvre Induced Error

19. A change of attitude of the aircraft, especially in the pitching plane, may well induce a short-term inaccuracy of static pressure, giving a false rate of climb or descent information on the VSI. This error is more marked at high altitude than at low altitude.

20. With a little thought it should be apparent that the three errors outlined above will combine to give maximum erroneous VSI indications when the aircraft initiates a climb or a descent, or levels off following a climb or descent. This error will be most pronounced if the change of speed is significant, and if air brakes, flaps or landing gear are used to achieve the required change of altitude.

Static Line Blockages And Leaks

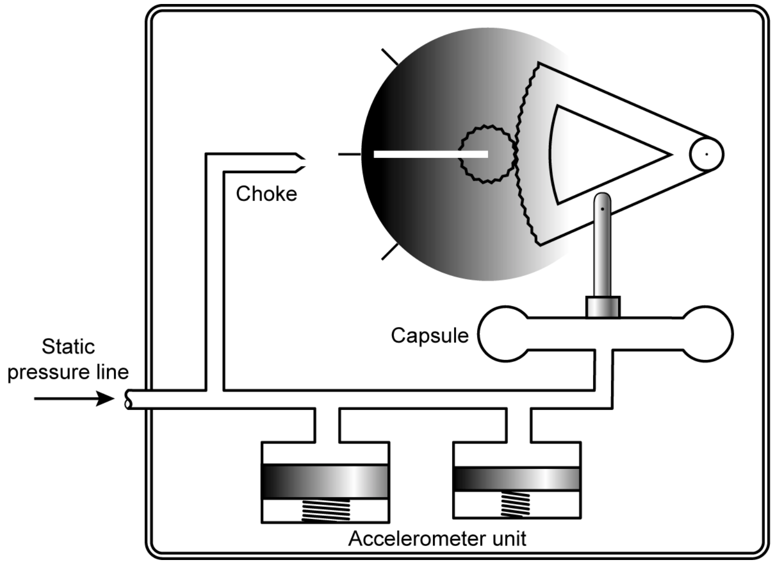
21. Should the static line become blocked, the VSI will cease to operate, showing continuously zero feet per minute rate of climb/descent.

22. Should the static line fracture within the pressure hull the VSI will show rate of change of cabin pressure altitude.

The Inertial Lead or Instantaneous VSI (IVSI)

23. This instrument is designed to overcome the false descent or climb indication which results from a rapid initiation of a climb or descent, and is illustrated at Figure 6-4.

FIGURE 6-4
Instantaneous VSI - Schematic



24. An **accelerometer unit** is added to the conventional VSI. The accelerometer comprises **two dashpots** in the static pressure supply to the capsule. Each dashpot contains an inertial mass in the form of a piston balanced by a spring. A change in vertical speed causes the pistons to be displaced, creating an immediate decrease of capsule pressure in the climb and an immediate increase of capsule pressure in the descent, thereby providing the appropriate indications. After a few seconds the effect of the accelerometer pistons dies away and the indicated rate of climb or descent is now dependant solely on the pressure differential generated by the metering unit. The springs slowly return the pistons to their central steady-state position.

IVSI - Presentation

25. The presentation is the same as for a normal VSI, however the instrument face will normally be marked to the effect that it is an inertial-lead or instantaneous VSI.

IVSI - Errors and Limitations

26. Compared with a normal VSI, lag is virtually eliminated, and manoeuvre-induced error is also significantly reduced. However, the dashpots are not vertically stabilised, and thus during turns, and at high bank angles (>40°), the variations in g acceleration will produce erroneous readings. Flight in turbulence will also have the effect of magnifying vertical movements of the aircraft.

Air Temperature Sources

Air Temperature Sources

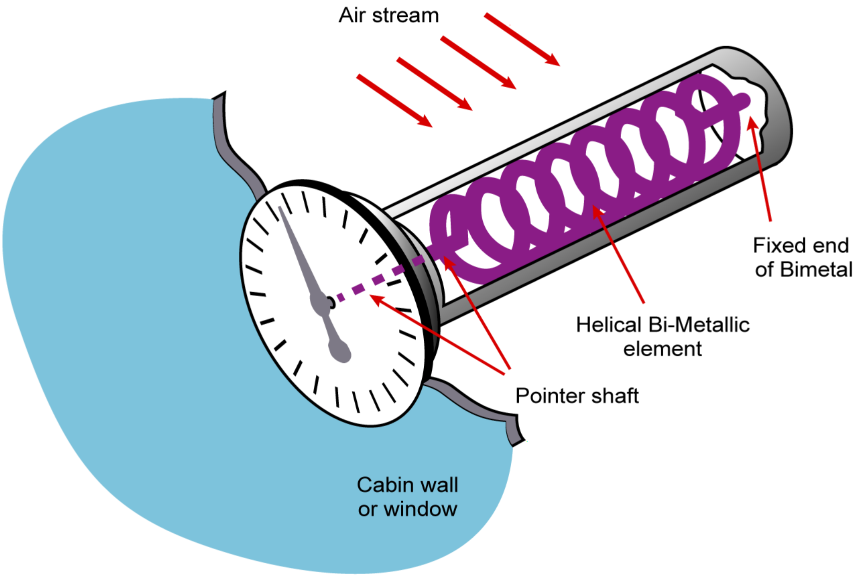
1. It would appear at first glance that measuring the temperature outside the aircraft is a simple matter of introducing a thermometer bulb into the air and noting the reading on a suitable indicator. However as an aircraft speeds increase and air compressibility become significant, the situation becomes complicated by the fact that the temperature sensed at the bulb bears no resemblance to the free air temperature.

Low Speed Thermometers

2. For aircraft flying at true airspeeds below about 150 kt, relatively simple thermometers are sufficient to measure the outside air temperature.

3. The **bimetallic thermometer** is probably the simplest of aircraft thermometers. A helical bimetallic element which twists with temperature changes is encased in a tube. The twisting of the helix drives a pointer over a scale. The thermometer is mounted on the windscreen or fuselage with the tube protruding into the airstream and the dial visible to the pilot. A bimetallic thermometer is shown at Figure 7-1.

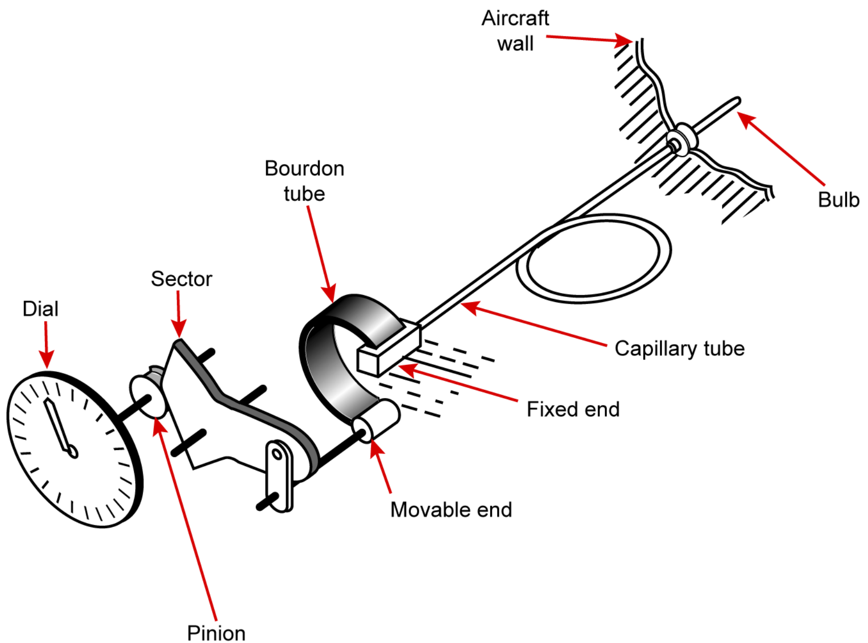
FIGURE 7-1
Bimetallic Thermometer



4. Where it is not feasible to use a bimetallic thermometer, remote bulb thermometers may be used, and these can be either liquid or vapour filled. A remote bulb system is illustrated schematically at Figure 7-2.

5. With a **liquid filled** system the bulb, capillary and Bourdon tube are completely filled with liquid. The expansion of the liquid with increasing temperature causes the Bourdon tube to straighten. This action is transferred to a pointer by a mechanical linkage. A decrease of temperature will have the reverse effect. The bulb is located at a suitable position in the free airstream.

FIGURE 7-2
Liquid/Vapour Thermometer



6. With a **vapour filled** system the bulb is filled with a small quantity of volatile liquid and the capillary and Bourdon tube with vapour. As the fluid in the bulb expands and contracts with change of temperature, the pressure of the vapour in the capillary tube and in the Bourdon tube changes, again modifying the shape of the Bourdon tube. With this system atmospheric pressure changes will cause indicator errors since the Bourdon tube will suffer from a changing inside/outside pressure ratio.

7. At true airspeeds above 150 kt the systems described above are not sufficiently accurate, and it is now necessary to use electrical thermometers. These fall into two categories, depending on the type of sensing element which is employed.

8. With a **coil resistance** sensor, the resistance of a nickel or platinum coil, which changes at a rate which is proportional to the change in absolute temperature, is measured. The change in resistance is small but stable.

9. With a **semiconductor** (**thermistor**) sensor, the change in resistance for a given temperature change is greater than with a coil sensor, but unfortunately it is difficult to obtain constant resistance/temperature characteristics from one thermistor to another.

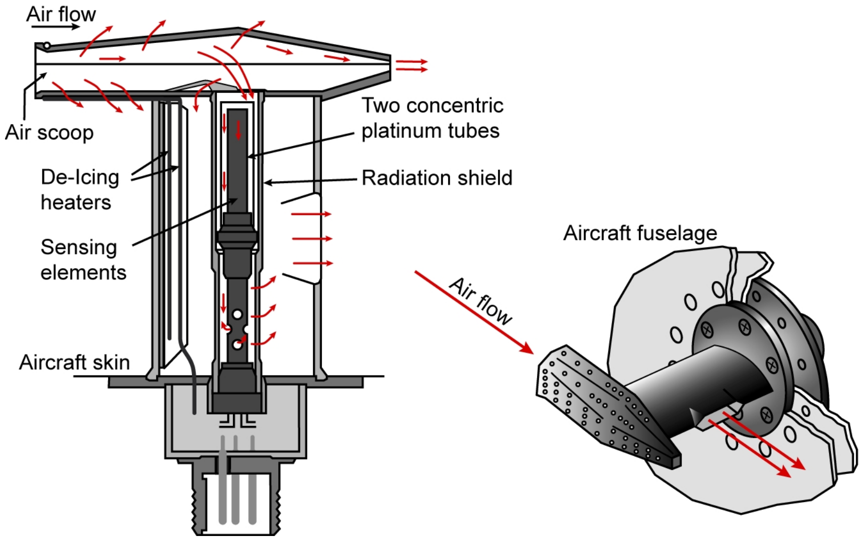
10. A simple method of converting resistance change to an indication of temperature is to include the coil or thermistor in a circuit with a fixed voltage of known value. The current flowing in the circuit will change as the resistance of the sensor changes, and this current is measured by a milli-ammeter. Changing temperature causes changing resistance which causes changing current flow, and therefore the face of the milli-ammeter can be calibrated in °C rather than milli-amps.

11. A more accurate method than that described in the previous paragraph employs a servo indicator. In this system, the resistance of the sensing element is included in a self-balancing (Wheatstone) bridge. The principle of operation is that of balancing resistances. The resistance of the sensor is matched by a variable resistance (a potentiometer) in which the wiper arm is positioned by a servo loop. The amount of movement of the wiper arm is constantly a measure of the temperature change, and it is the movement of the wiper arm which is used to position the needle on the temperature gauge. The advantage of this system is that changes in the sensor circuit voltage will not affect the accuracy of the system.

12. Regardless of whichever of the two systems described above is used, it is important that the current flow through the sensor is minimal, since a current flowing through a circuit within which there is resistance will itself cause a temperature rise.

13. **A Rosemount probe** (**or total air temperature probe**) is illustrated at Figure 7-3. The probe has a small resistance coil surrounded by concentric cylinders and is mounted on a streamlined strut 50 mm or so from the aircraft skin. Being small, the element reacts quickly to temperature changes and being at some distance from the aircraft skin, the coil is not unduly affected by skin temperature. The probe is open at the front end while a smaller orifice at the rear allows the probe to continuously sample new air, although the airflow through the probe is quite slow because of the restrictions within it.

FIGURE 7-3
Rosemount Probe



14. The probe has an integral heating element fitted to it to prevent icing. The heater is of the self-compensating type in that, as the temperature rises, so does the element resistance and therefore the heater current is reduced. The heater obviously affects the temperature sensed by the sensor, however the resultant error is small enough to be acceptable, being in the order of 1°C at Mach 0.1 and 0.15°C at Mach 1.0.

Air Temperature Measurement - Errors

15. **Lag** is usually only a consideration when simple probes, which are basically no more than thermometers, are used. Such installations are usually only found on light aircraft. Lag in a more sophisticated installation will be minimal.

16. **Instrument Error.** There will be small errors due to manufacturing imperfections.

17. **Position Error.** As with any air-data source such as static vents and pitot probes, a temperature probe will be affected by its location on the airframe. By careful design this error can be minimised.

18. The magnitude of instrument and position errors can be determined. Depending on the type of system, these errors may then be largely removed.

True and Measured Temperature

19. **Static air temperature** (SAT) is the **correct** temperature of the ambient air and is sometimes referred to as the corrected or true outside air temperature.

20. Unfortunately, especially at high speeds, the effect of compressibility means that the temperature which is sensed will be considerably higher (warmer) than the static air temperature.

21. **Total air temperature** (TAT) is the temperature of air which has been brought virtually to rest, and which has suffered the full effect of heating due to compressibility. The difference between the static air temperature and the total air temperature is sometimes referred to as the stagnation rise. The proportion of the stagnation temperature which is sensed is known as the **recovery factor or K value.**

22. The type of thermometer employed will govern the percentage of the stagnation temperature rise which is sensed. With a Rosemount probe the K factor is normally assumed to be 1.0, which is why the device is sometimes called the total air temperature probe.

23. With bimetallic or liquid/vapour filled thermometers (and electrical thermometers of a more basic type than the Rosemount probe) the K value is normally in the order of 0.75 to 0.85. In this case the measured temperature is the temperature of air which has been brought only partially to rest. Now the measured temperature is known as the **ram air temperature (RAT)**, and the difference between the static air temperature and the measured temperature (RAT) as the ram rise.

24. Sometimes, incorrectly, the ram air temperature is referred to as the total air temperature. This would only be correct when the ram rise is equal to the full stagnation rise (a thermometer with a K factor of 1).

25. Static air temperature can be derived from the measured temperature using the formula given below:

$$T_S = \frac{T_m}{1 + (0.2 \times K \times M^2)}$$

where T_S = SAT in degrees absolute

T_m = measured temperature in degrees absolute

M = Mach number

K = recovery factor

Air Temperature Sources

EXAMPLE

The measured temperature using a Rosemount probe with a recovery factor of 1 is -30°C and the Mach number is 0.72. Determine the static air temperature.

SOLUTION

In order to solve the problem it is necessary to appreciate the relationship between the absolute (or Kelvin) temperature scale and the celsius temperature scale, and this is dealt with in Meteorology. Suffice for now to say that a change of temperature of 1° absolute is precisely the same as a change of temperature of 1° celsius. The difference lies in the starting point of the scale, 0°C is equal to +273°A (absolute) or, if you prefer, 0°A is equal to -273°C.

$$T_S = \frac{T_m}{1 + (0.2 \times K \times M^2)}$$

$$= \frac{243}{1 + (0.2 \times 1 \times 0.72^2)}$$

$$= \frac{243}{1.10368}$$

$$= 220°A$$

$$= -53°C$$

26. The table at Figure 7-4 gives an easier way of converting Rosemount probe readings to SAT. Similar tables can be produced for other types of thermometers, however they must take account of the lower K value.

27. As the speed of the aircraft increases, the effect of kinetic heating of the sensing element by the airflow over it will tend to increase. Further complications occur when a sensing bulb is mounted flush with the aircraft skin, as the sensor will tend to register a combination of skin temperature and air temperature. Additionally, unless the bulb is shrouded, direct sunlight will give rise to an artificially high reading. Finally, on flying from cloud to clear air, the temperature which is sensed will be artificially low for the period taken for the moisture to evaporate from the bulb or the sensing element within the Rosemount probe. The probe of the bimetallic thermometer cannot be shrouded and is necessarily mounted adjacent to the fuselage skin, this type of thermometer will therefore suffer from all of these errors. The liquid/vapour filled thermometer bulb must again be mounted on the aircraft skin, however the bulb can be shrouded to prevent direct sunlight reaching the bulb. The effect of kinetic heating which is suffered at the low speeds associated with aircraft using bimetallic or liquid/vapour filled thermometers will be small. The Rosemount probe is both shrouded and mounted remotely from the aircraft skin. The effect of kinetic heating within the Rosemount probe is minimised by virtue of the fact that the airflow through the probe is slowed by virtue of the restrictions within it.

FIGURE 7-4
Conversion of TAT to SAT (TOAT)

	INDICATED MACH NUMBER										
	.30	.40	.50	.60	.70	.73	.76	.78	.80	.82	.84
IND TAT °C	TRUE OUTSIDE AIR TEMPERATURE DEGREES C										
70				47	39	37	35	33	31	29	27
65			49	42	35	33	30	28	26	25	23
60		49	44	37	30	28	25	24	22	21	19
55	49	45	40	33	26	24	21	19	18	16	14
50	45	40	35	28	21	19	17	15	13	11	10
45	40	35	30	23	17	15	12	11	9	7	5
40	35	30	25	19	12	10	8	6	4	3	1
35	30	26	20	14	8	6	3	1	0	-2	-3
30	25	21	16	10	3	1	-1	-3	-5	-6	-7
25	20	16	11	5	-2	-3	-6	-7	-9	-11	-12
20	15	11	6	0	-6	-8	-10	-12	-13	-15	-16
15	10	6	2	-5	-11	-13	-15	-16	-18	-19	-21
10	5	1	-3	-9	-15	-17	-19	-21	-22	-24	-25
5	0	-3	-8	-14	-20	-21	-24	-25	-27	-28	-29
0	-5	-8	-13	-18	-24	-26	-28	-30	-31	-33	-34

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-20	-24	-27	-32	-37	-42	-44	-46	-47	-49	-50	-51
-25	-29	-32	-36	-42	-47	-49	-51	-52	-53	-55	-56
-30	-34	-37	-41	-46	-51	-53	-55	-57	-58	-59	-60
-35	-39	-42	-46	-51	-56	-58	-60	-61	-62	-63	-65
-40	-44	-47	-51	-56	-61	-62	-64	-65	-66	-68	-69
NOTE: Probe Recovery Factor is 100%.											

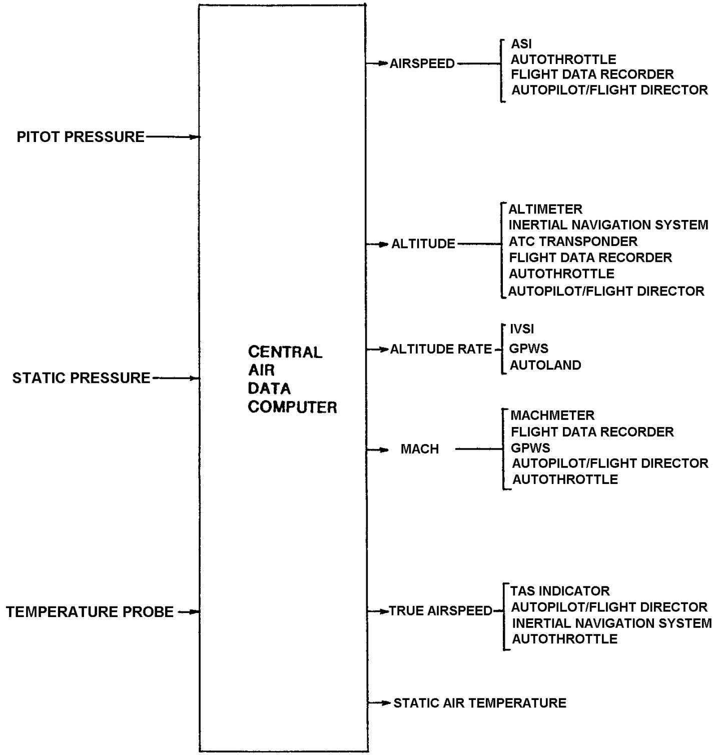
The Central Air Data Computer

The Central Air Data Computer

1. In many aircraft currently in service, altitude, vertical speed, Mach number, indicated airspeed, true airspeed and static air temperature are provided by means of a central air data computer (CADC).

2. The key inputs to the CADC come from the pitot head, the static vent and the temperature probe (normally a Rosemount total air temperature sensor), however, angle of attack may also be an additional input. The outputs from the CADC are supplied to the flight instrument panels and to various other systems, as shown at Figure 8-1.

FIGURE 8-1
Block Schematic of a CADC



3. The system in its entirety is known as the **air data system** (**ADS**). System redundancy is normally achieved by using two independent air data systems with a switching mechanism which allows the pilots to feed all of the flight instruments and other systems from one CADC in the event that the other CADC fails.

4. The principle advantages of such a system are the reductions in weight and bulk which are achieved because of the considerable reduction in the plumbing which would otherwise be required to feed each individual instrument. Space is invariably at a premium in the area behind the pilots' instrument panels, and therefore the ability to utilise CADC fed instruments which contain no capsules is much appreciated by the panel designers. One additional advantage of a CADC is that circuits may be integrated into the system in such a way that corrections for pressure error, barometric pressure changes and compressibility can be automatically applied. Provision can also be made for the calculation of true airspeed from air temperature data inputs.

5. Early examples of CADCs employed two pressure transducers (E and I bar arrangements) and an electro-mechanical analogue computer. New generation CADCs use a digital computer. Shortly we are going to look at an analogue system (because wheels and gears are easier to illustrate and to understand than binary numbers), but first we will consider what we mean when we talk of analogue and digital systems in general.

6. An **analogue** is something which is similar to something else. An analogue computer works by making a model of the real thing or of mathematical values. This model is made by electric circuits, the equations for the circuits being the same as the problem that the computer is designed to solve, or by moving parts such as rotating wheels, gears or cams.

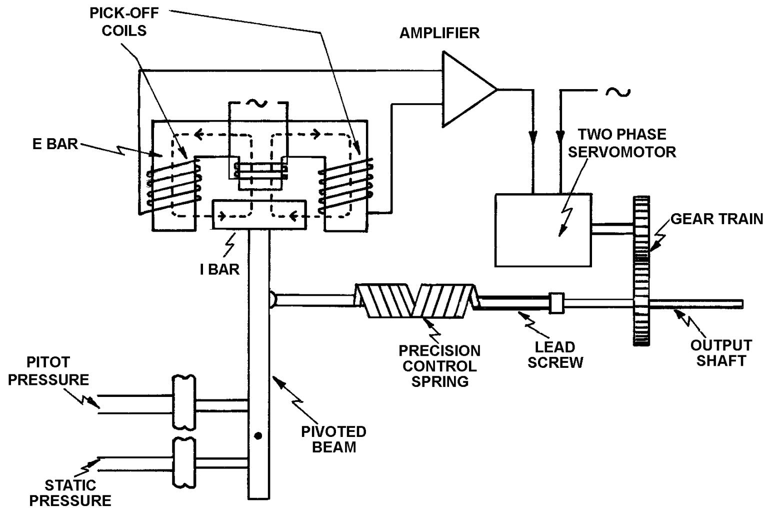
7. A **digital computer** operates with binary numbers (zeros and ones) and has five major components; input, storage, arithmetic, control and output. It is the arithmetic unit which performs the operations required to solve the problem, working with instructions kept within the storage unit (the programme).

The Analogue CADC

8. An analogue CADC comprises 4 modules (**altitude, TAS, computed airspeed, mach modules**) each consisting of a servomechanism and synchros which provide the relevant outputs.

9. In addition to the above 4 modules the analogue system employs two pressure transducers. The first is the dynamic pressure transducer and contains two capsules, one of which is fed with pitot pressure and the other with static pressure. An example of a dynamic pressure transducer is shown schematically at Figure 8-2.

FIGURE 8-2
CADC Dynamic Transducer



10. The first thing that may occur to you is that the system shown at Figure 8-2 will not work, since the pivoted beam is pivoted between the two capsules. Obviously the system does work, on what is termed the force-balance principle.

11. Let us assume that the aircraft accelerates at a constant altitude (constant static pressure). The pitot pressure increases and this forces the pivoted beam to move slightly (top right/bottom left) despite the resistance offered by the static capsule. The I bar now moves relative to the E bar, varying the air gaps and therefore inducing differential current flows in the pick off coils on the outer arms of the E bar. These differential (error) currents are amplified and fed to the servo motor which turns the gear train.

12. One of the rotational outputs of the gear train is connected to a lead screw, which is coupled to the pivoted beam via a precision control spring. As the lead screw turns, the force exerted by the spring is varied (increasing in this case) so as to balance the force which is being exerted on the beam by the increased pitot pressure. This causes the beam to start to return to its central position, reducing the error currents and consequently the activity of the servo motor. The transducer is very sensitive and in fact the pivoted beam moves hardly at all.

13. Since the output shaft is turned at the same time as the lead screw, the angular position of the shaft is proportional to the difference between pitot and static pressure, that is to say dynamic pressure. The angular position of the output shaft is therefore representative of the IAS. When considering dynamic pressure transducers, the term **computed airspeed** (**CAS**) is used instead of IAS, however as far as we are concerned the IAS and the CAS can be considered to be the same.

14. The output shaft is connected via a cam and cam follower arrangement to the rotating coil within the transmitter of a control synchro. The IAS/CAS can now be transmitted as electrical signals to control synchro receivers in the airspeed indicator(s), in other parts of the CADC, in the autothrottle system and so on. The cam and cam follower arrangement is required as an analogue correction to account for the fact that the relationship between the IAS/CAS and the initial angular movement of the output shaft is not linear, in fact the angular movement of the shaft is a function of the dynamic pressure squared.

15. The second transducer within the CADC is the static pressure transducer. It is very similar to the dynamic transducer but now the pitot capsule is replaced with a partially evacuated and sealed (aneroid) capsule. Again the transducer works on the force-balance principle and again a cam and cam follower is included in the output shaft feed to the control synchro transmitter to convert the initial angular movement of the shaft into a movement which is linear in respect of the aircraft altitude.

16. Armed with the outputs of the two pressure transducers, together with the input from the temperature probe, the CADC can easily complete its remaining functions:

(a) The rate at which the cam corrected output of the static pressure transducer is changing is used to determine the **altitude rate**, and this is used to drive the IVSI (instantaneous since the static pressure transducer responds very quickly to static pressure changes).

(b) Using the outputs of both pressure transducers the CADC computes the Mach number.

(c) The total air temperature input is converted into static air temperature using an analogue of the conversion formula.

(d) Armed with the Mach number and the SAT the CADC computes the true airspeed.

System Integrity

17. It is usual for digital air data computers to be equipped with a **built-in-test facility (BIT)**, the function of which is to check correct computer operation. The built-in-test facility operates automatically, and will self-check when power is first applied to the CADC, and then continuously during flight. This self-checking does not interfere with normal CADC operation. A maintenance check facility is also included as part of the BIT programme, for use solely by the ground technicians as an aid to system fault finding. If the switch which activates the maintenance BIT programme is accessible on the flight deck, then the programme can only be activated when the weight of the aircraft is on the wheels (to prevent its activation in flight). Alternatively the switch activating the maintenance BIT programme may be inaccessible in flight. Therefore, as far as the flight crew is concerned, the CADC has no control or test switches. Checking is either automatic or is restricted to ground maintenance personnel.

System Monitoring

18. Each module of a modern CADC incorporates a warning logic circuit network which activates a **warning flag** in the associated indicators in the event of loss of the respective data signals. **Annunciator lights** corresponding to each module are provided on the end panel of the computer and are also illuminated in the event of failures.

Self Assessed Exercise No. 1

QUESTIONS:

QUESTION 1.

In terms of pitot and static pressure, dynamic pressure is:

QUESTION 2.

Which of the following errors will the static source suffer from at high altitude – barometric error, compressibility error, temperature, error, manoeuvre error:

QUESTION 3.

Density varies directly/inversely with pressure and directly/inversely with temperature:

QUESTION 4.

A barometric altimeter is accurate to within 1 mb either way over its whole range. Its approximate readout error at FL200 and FL400 will be:

QUESTION 5.

The sensitive altimeter shown at FIGURE 234 in the Reference Book is indicating an altitude of:

QUESTION 6.

An aircraft departs airfield ‘A’, which is 1200ft AMSL, with QFE 980mb set on the altimeter. The altimeter is not subsequently reset. When landing at airfield ‘B’, which is 2500ft AMSL, where the QNH is 1005mb, the altimeter will read: (assume 1mb = 30ft)

QUESTION 7.

A flight is planned along a track of 235°M which passes over a mountain with an elevation of 3505 metres. The regional QNH is 996mb. Assume 1mb = 30ft. The lowest useable flight level according to the quadrantal rule system which will provide a minimum clearance of 2000ft above the mountain is:

QUESTION 8.

Refer to the diagram of a servo altimeter at FIGURE 237 in the Reference Book. The correct descriptions of the components labelled C, D and E in order are:

QUESTION 9.

The definition of CAS is:

QUESTION 10.

The definition of Equivalent airspeed (EAS) is:

QUESTION 11.

Define VLE:

QUESTION 12.

Name the errors of an Airspeed Indicator:

QUESTION 13.

An aircraft is flying at FL390, temperature 207°A, at mach 0.85. The TAS of the aircraft is:

QUESTION 14.

An aircraft flying at a constant FL and maintaining a constant TAS flies into an area of warmer air. The air density is _____________ causing the CAS to ___________.

QUESTION 15.

If an aircraft descends at constant CAS through an isothermal layer, the TAS will:

QUESTION 16.

If the pitot line becomes blocked in the descent, the ASI will indicate:

QUESTION 17.

An aircraft is climbing at a constant mach No. below the tropopause in ISA conditions. During the climb the CAS will:

QUESTION 18.

An aircraft is flying at FL80 at a constant CAS, and flies from cold air into warm air. The QNH remains constant throughout. The TAS will increase/decrease and the actual height will increase/decrease.

QUESTION 19.

The subscale setting of a pressure altimeter has been wrongly set by 2mb. Over its whole range the readout error will increase with altitude/decrease with altitude/be a constant 60ft.

QUESTION 20.

If you fly in an air of constant density and double your true airspeed, the dynamic pressure will:

QUESTION 21.

On a colour-coded ASI the lower end of the white arc is:

QUESTION 22.

Compressibility error is caused by a false increase/decrease of pitot/dynamic pressure, requiring a positive/negative correction to the CAS to obtain EAS.

QUESTION 23.

When an ASI is calibrated, the elements of ISA which are assumed are:

QUESTION 24.

An aircraft is climbing at a constant TAS. The Mach No will:

QUESTION 25.

An increase of 0.15 in Mach No results in an increase of 93kt in TAS. If the temperature deviation from ISA is +9°C, the FL is:

QUESTION 26.

If the static line becomes blocked during a descent, the Machmeter will:

QUESTION 27.

An aircraft is descending at a constant Mach No. If the aircraft is descending through an inversion layer, the TAS will:

QUESTION 28.

An aircraft is flying at FL650. A change of Mach No. of 0.1 causes a change is TAS of 57 knots. The temperature deviation at FL650, assuming ISA conditions, is:

QUESTION 29.

A VSI metering unit incorporates a capillary tube to compensate for:

QUESTION 30.

The primary purpose of the restricted choke in the VSI is to:

QUESTION 31.

In a climb the pitot line becomes blocked. The VSI will indicate:

QUESTION 32.

Refer to FIGURE 351 in the Reference Book. Figure C is a diagram of:

QUESTION 33.

Refer to FIGURE 352 in the Reference Book. Figure D is a diagram of:

QUESTION 34.

Total Air Temperature is defined as:

QUESTION 35.

What CADC test facilities are there on the flight deck of a modern commercial aircraft, and for use by the aircrew:

QUESTION 36.

What is the formula for obtaining Static Air Temperature from measured temperature:

QUESTION 37.

What are the basic inputs into a CADC in order that it can compute correctly:

QUESTION 38.

An analogue CADC comprises the following modules:

QUESTION 39.

In the event of a module failure in a CADC, what would be the indications:

QUESTION 40.

To obtain SAT from TAT, a positive/negative correction must be applied to TAT for compressibility/ ram rise/pressure/density.

ANSWERS:

ANSWER 1.

Dynamic pressure is pitot pressure minus static pressure.

See Chapter on "Pitot and Static Sources", Page 3 Paragraph 7

ANSWER 2.

At high altitude the static source will suffer from manoeuvre error.

See Chapter on "Pitot and Static Sources", Page 5 Paragraph 10

ANSWER 3.

Density varies directly with pressure and inversely with temperature.

See Chapter on "Pressure Altimeters"

ANSWER 4.

The readout error would be ± 50ft at FL200, and ± 100ft at FL400.

See Paragraph 3

ANSWER 5.

13600ft.

See Chapter on "Pressure Altimeters", Page 6 Figure 2-2

ANSWER 6.

1750ft

See FIGURE 235 in the Reference Book

See Chapter on "Pressure Altimeters"

ANSWER 7.

FL 160

See FIGURE 236 in the Reference Book

See Chapter on "Pressure Altimeters"

ANSWER 8.

Items C, D and E, in order are, amplifier, transducer, capsule stack.

See Chapter on "Pressure Altimeters", Page 8 Figure 2-3

ANSWER 9.

CAS is IAS corrected for Instrument error and Position error.

See Chapter on "Airspeed Indicators", Page 8 Paragraph 23

ANSWER 10.

EAS is CAS corrected for compressibility.

See Chapter on "Airspeed Indicators", Page 8 Paragraph 24

ANSWER 11.

VLE is the maximum speed at which the aeroplane may be safely flown with the undercarriage extended.

See Chapter on "Airspeed Indicators", Page 9 Heading "Other Speed Definitions"

ANSWER 12.

The errors of an Airspeed Indicator are Instrument Error, Position Error, Compressibility and Density.

CH3 P3-3/3-4

ANSWER 13.

476 kt.

See Chapter on "Machmeters", Page 12 Example 5-2

ANSWER 14.

The air density will be decreasing causing the CAS to decrease.

See Chapter on "Airspeed Indicators"

ANSWER 15.

The TAS will decrease.

See Chapter on "Airspeed Indicators"

ANSWER 16.

The ASI will show a decreasing IAS.

See Chapter on "Airspeed Indicators", Page 7 Paragraph 19

ANSWER 17.

The CAS will decrease due to decreasing static pressure.

See Chapter on "Machmeters", Page 19 Example 5-8

ANSWER 18.

The TAS will increase and the actual height will increase.

See Chapter on "Airspeed Indicators"

ANSWER 19.

If the subscale has been wrongly set by 2 mb, the readout error will be a constant 60ft at all altitudes.

See Chapter on "Pressure Altimeters"

ANSWER 20.

If you double your true airspeed in air of fixed density, the dynamic pressure will quadruple.

See Chapter on "Airspeed Indicators", Page 3 Paragraph 5

ANSWER 21.

The lower end of the white arc on an ASI indicates 1.1 times the stall speed with full flap, landing gear down, all power units at idle, and with the aircraft at maximum certified all up weight.

See Chapter on "Airspeed Indicators", Page 10 Paragraph 27

ANSWER 22.

Compressibility error is caused by a false increase of dynamic pressure, requiring a negative correction to the CAS to obtain EAS.

CH3 P3-5

ANSWER 23.

An ASI is only calibrated to a mean sea level density of 1225 gm per cubic metre.

See Chapter on "Airspeed Indicators", Page 7 Paragraph 15

ANSWER 24.

If an aircraft climbs at a constant TAS, the Mach No will increase.

See Chapter on "Machmeters", Page 17 Example 5-6

ANSWER 25.

FL220

Actual Temp –20°C

ISA Dev +9°C

ISA Temp -29°C

See Chapter on "Machmeters"

ANSWER 26.

The machmeter will overread

See Chapter on "Machmeters", Page 10 Paragraph 22

ANSWER 27.

The TAS will decrease.

See Chapter on "Machmeters", Page 17 Example 5-6

ANSWER 28.

The ISA deviation would be –3.5°C (actual temp –60°C; ISA temp –56.5°C).

See Chapter on "Machmeters"

ANSWER 29.

The capillary tube in a VSI metering unit compensates for temperature and pressure changes with height.

CH6 P6-3 paragraphs 11 & 12

ANSWER 30.

The primary purpose of the restricted choke is to create a differential pressure between the capsule and the case of the instrument.

See Chapter on "Vertical Speed Indicators", Page 1 Paragraph 4

ANSWER 31.

A pitot line is NOT connected to a VSI therefore, it will continue to indicate the actual rate of climb.

See Chapter on "Vertical Speed Indicators"

ANSWER 32.

Refer to FIGURE 351 in the Reference Book. Figure C is an airspeed indicator.

See Chapter on "Airspeed Indicators"

ANSWER 33.

Refer to FIGURE 352 in the Reference Book. Figure D is a machmeter.

See Chapter on "Machmeters"

ANSWER 34.

Total Air Temperature is the temperature of air which has suffered the full effect of compression heating.

See Chapter on "Air Temperature Sources", Page 8 Paragraph 21

ANSWER 35.

There are no serviceability test facilities on the flight deck for aircrew use, there is only a ground maintenance BITE.

See Paragraph 17

ANSWER 36.

$$Ts = \frac{T_M}{1 + (0.2 \times K \times M^2)}$$

See Chapter on "Air Temperature Sources", Page 5 Paragraph 25

ANSWER 37.

A CADC requires inputs of Pitot Pressure, Static Pressure and Air Temperature in order to perform the necessary calculations.

See Figure 8-1

ANSWER 38.

An analogue CADC comprises of altitude, TAS, computed airspeed and mach modules.

See Paragraph 8

ANSWER 39.

Should a module fail in a CADC, a warning flag will appear on the relevant indicator plus an annunciator light will show on the CADC unit.

See Paragraph 18

ANSWER 40.

To obtain SAT from TAT, a negative correction must be applied to TAT for compressibility.

See Chapter on "Air Temperature Sources", Page 8 Paragraph 21

Terrestrial Magnetism

Terrestrial Magnetism

1. The Earth is surrounded by a weak magnetic field which culminates at two magnetic poles lying beneath the surface, at points fairly close to the true north and south poles.

2. Presently, the north magnetic pole is situated near Hudson Bay (Canada), and the south magnetic pole near Victoria Land (Antarctica). These poles are not in fact stationary, but rotate very slowly about the true poles in a clockwise direction once every 960 years. This gives rise to the annual change in variation which is termed the secular change. Small and irregular changes also occur daily, annually and over an eleven year cycle.

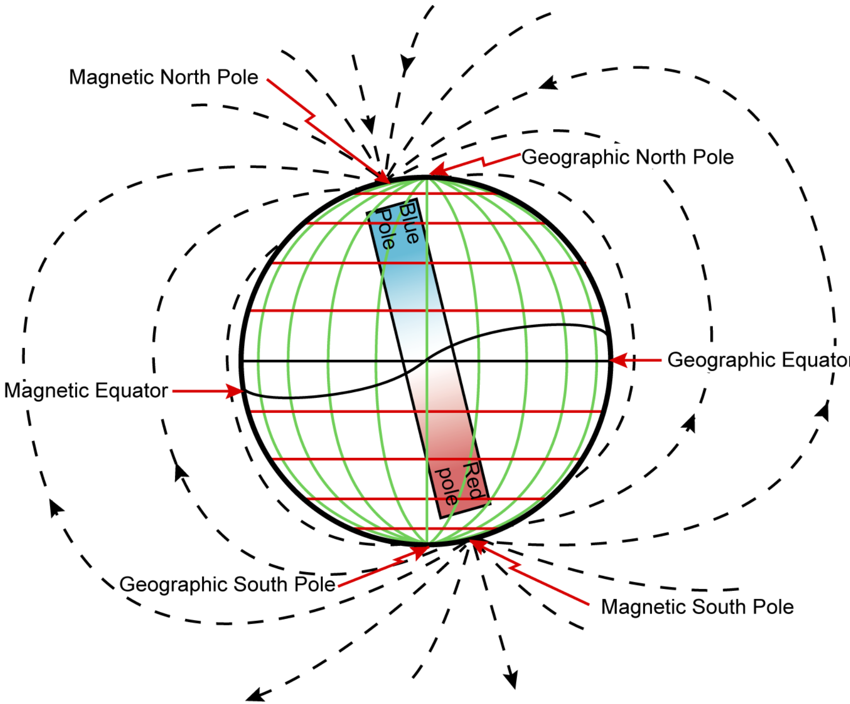
3. The reasons for the existence of the Earth's magnetic field are still under investigation. It is believed that the field is produced electrically within the Earth's core. It is sufficient to accept that the effect is as if an extremely large bar magnet were located within the Earth with its extremities at the magnetic poles, as shown at Figure 9-1.

4. Because of the irregularity of the terrestrial magnetic field, the magnetic lines of force are of varying direction and strength. Figure 9-1 shows diagrammatically the Earth's magnetic field. Note that at the magnetic poles the lines of force are vertical, whilst at points equidistant from the magnetic poles the lines of force are horizontal. A line drawn through points where the lines of force are horizontal is known as the magnetic equator. It is normally acceptable to consider the magnetic equator to be co-incident with the geographic equator.

5. The north magnetic pole is, by convention, normally shown as a blue pole, and the south magnetic pole as a red pole. Remember that unlike poles attract, therefore a freely suspended magnet will come to rest with its red (north-seeking end) pointing towards the north magnetic pole.

Terrestrial Magnetism

FIGURE 9-1
The Earth's Magnetic Field

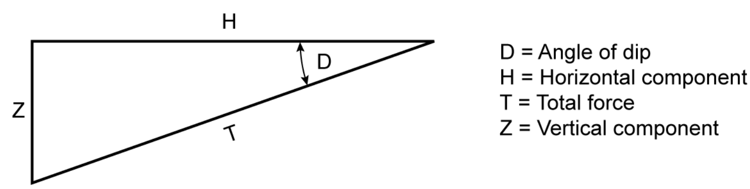


6. A problem arises because a freely suspended magnet will lie in the plane of the terrestrial magnetic lines of force and, as is already known, the magnetic force field approaches the vertical at high magnetic latitudes. Consequently, in the northern hemisphere, the north-seeking end of a freely suspended magnet will **dip** below the horizontal. Conversely, in the southern hemisphere the south-seeking end will **dip** below the horizontal. Another way of defining the magnetic equator is, therefore, as a line joining points of zero dip. This line is otherwise known as the **aclinal** line. A line joining points of equal dip is called an **isoclinal** line.

7. It is the strength of the horizontal component of the Earth's magnetic field which governs the compass needle's ability to point positively towards the magnetic pole.

8. At any given magnetic latitude the Earth's magnetic field may be resolved into horizontal and vertical components, as shown at Figure 9-2. Obviously at the magnetic equator a compass needle will be highly directive, since the horizontal component (H) is equal to the total force (T). As the magnetic latitude is increased the horizontal component decreases in magnitude, and the compass becomes unreliable; it is for this reason that aircraft **magnetic compasses are not generally considered to be usable at latitudes in excess of 70°.**

FIGURE 9-2
Components of the Earth Magnetic Field



Direct Reading Compasses

Direct Reading Compasses

1. A direct reading compass enables the pilot to read the aircraft heading directly in relation to a magnetic assembly.

2. This type of compass basically consists of two or more pivoted magnets, which are free to align themselves with the horizontal component of the Earth's magnetic field. It is desirable that any direct reading compass should possess three elementary properties:

(a) Horizontality

(b) Sensitivity

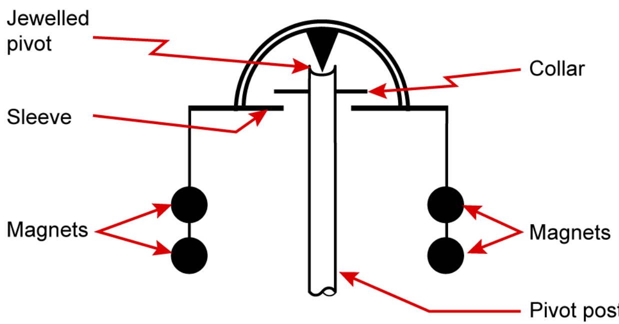
(c) Aperiodicity

Horizontality

3. As already discussed, it is the horizontal component of the Earth's magnetic field which enables the compass magnets to align themselves with north. It is therefore essential that the compass magnets should lie as close as possible to the horizontal plane.

4. If a magnet were pivoted at its centre on a pin, it would dip to lie in the plane of the Earth's total field. Even in mid latitudes, the dip angle would be unacceptably high. To overcome this problem, a system of pendulous suspension is employed which is schematically outlined at Figure 10-1. The success of the system lies in the fact that the centre of gravity of the magnets lies below the pivot point. Thus the dipping effect due to the vertical (Z) component of the Earth's magnetic field is opposed by the weight of the magnets.

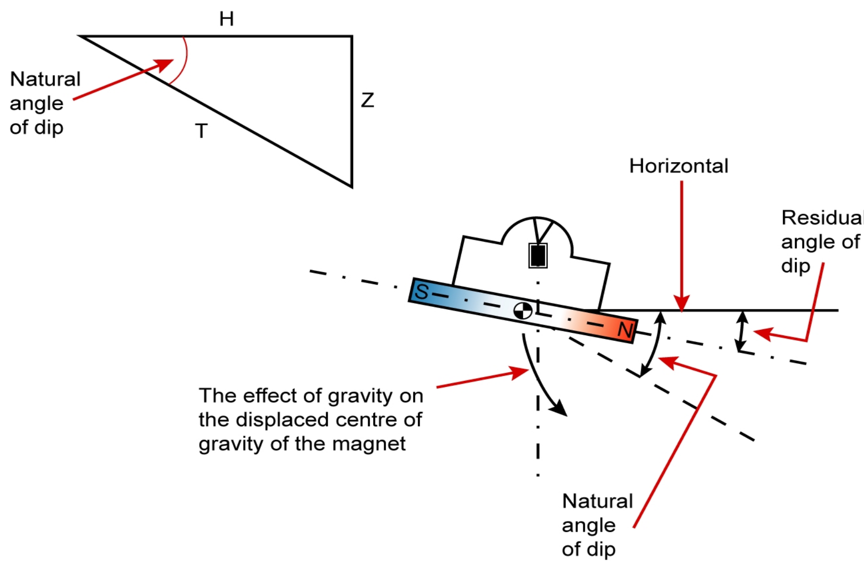
FIGURE 10-1
DRC Magnet Assembly Suspension System



5. The collar and sleeve assembly in Figure 10-1 prevents the magnet assembly from parting company with the pivot during inverted flight.

6. The result of pendulous suspension is that the magnets will lie close to the horizontal. Except at the magnetic equator there will however be a small residual dip, see Figure 10-2. In mid latitudes the residual angle of dip should be less than three degrees.

FIGURE 10-2
Natural Angle of a Dip



Sensitivity

7. Sensitivity is a measure of the ability of the compass magnetic assembly to point accurately towards north.

8. Nothing can be done to increase the strength of the weak terrestrial magnetic field and so it is necessary to use several magnets with high pole strengths. The magnetic assembly is made as light as possible to reduce friction at the pivot.

9. The pivot itself normally incorporates a jewelled bearing which is lubricated by the viscous fluid which fills the bowl. Being fairly dense the fluid effectively lightens the magnet assembly still further, once again reducing friction at the pivot.

Aperiodicity

10. The aperiodic quality of a compass may be defined as the ability of the magnet assembly to settle quickly, pointing towards the magnetic north pole, following displacement during manoeuvres or turbulence.

11. If a compass is not entirely aperiodic, the effect is that the magnets oscillate, or hunt, around magnetic north, coming to rest only slowly.

12. Aperiodicity is achieved by using short length magnets, thereby keeping the mass near to the centre of rotation and reducing the moment of inertia. This is also aided by the use of light materials in the magnetic assembly. The fluid within which the magnets are immersed will tend to dampen any oscillation of the magnet assembly.

13. The compass fluid discussed above must obviously be transparent. Additionally, the liquid must also completely fill the compass bowl in order to prevent liquid swirl during turns, as this would deflect the magnet assembly. To ensure that the bowl is always completely full despite change in temperature, an expansion bellows is fitted which acts as a fluid reservoir.

Serviceability Tests

14. The following tests would normally be carried out after compass installation, following a compass swing, or whenever the accuracy of the compass is in doubt.

(a) Check that the compass liquid is free from discolouration, bubbles and sediment. The movement of bubbles could deflect the magnetic assembly and the presence of sediment could prevent its free movement.

(b) Carry out a damping test. Using a small magnet deflect the compass by 90° and hold for at least 20 seconds to allow the liquid to stabilise. Now remove the magnet. The time taken for the compass reading to return to within 5° of the original reading should be 2 to 3 seconds for an E type compass.

(c) Carry out a pivot friction test. Turn the aircraft so that it is on one of the cardinal headings (N, E, S or W). Again using a small magnet, deflect the compass by 10° and hold for 10 seconds. Remove the magnet and note the reading when the compass settles. Now repeat the procedure but deflect the compass in the opposite direction. For an E type compass the readings after deflection should be within 2½° of the readings prior to deflection. Now repeat the process on each of the remaining cardinal headings.

Compass Safe Distances

15. Having already considered the advantages of the gyro slaved compass, and presumably from experience, we know that one of the major problems with a direct reading compass is that it must necessarily be mounted in the cockpit or on the flight deck. It is therefore surrounded by equipment which is capable of causing deviation, and the compass must be carefully sited.

16. The compass should be sited such that no single item of non electrical equipment causes a compass deviation of more than 1°. The sum of the deviations caused by all such equipment must not exceed 2°. Similarly, no single item of electrical equipment, or its associated wiring, may cause a compass deviation of more than 1°, and again the sum of the deviations caused by all such equipment and wiring may not exceed 2°. The operation of the aircraft controls is not permitted to change the deviation suffered by the compass by more than 1°.

17. Where the compass concerned is the **primary heading reference, the maximum permissible deviation on any heading is 3°** and the maximum permissible value of coefficients B and C are 15°. When a direct reading compass is installed as a **standby heading reference, the maximum permissible deviation on any heading is 10°**.

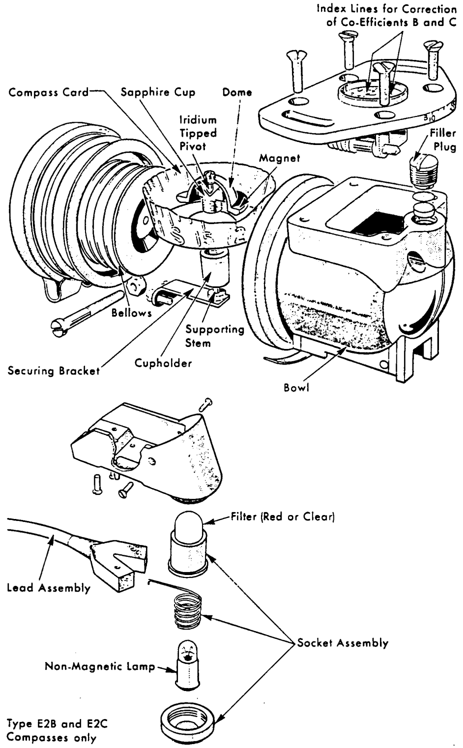
E Type Compass

18. The E Type compass, which is illustrated at Figure 10-3, is otherwise known as the vertical card compass. It is in very common use, as the main magnetic heading reference in modern light aircraft, and as a standby compass in larger aircraft.

Direct Reading Compasses

FIGURE 10-3

E-type Compass



19. Notice that with the E Type compass, the compass card is attached to the magnet assembly. Appreciate that (ignoring all errors) as the aircraft turns the magnets and compass card remain stationary whilst the instrument case and lubber line move around them.

(a) A displacement of the magnet assembly away from north in an anti-clockwise direction will result in an erroneous increase in indicated heading.

(b) Conversely, a displacement of the magnet assembly away from north in a clockwise direction will result in an erroneous decrease in indicated heading.

Acceleration Errors

20. We know from basic flight training that direct reading compasses will give incorrect readings during aircraft acceleration or deceleration. There are two reasons for this, inertia is one, and the effect of the vertical (Z) component on the displaced magnets is the other.

Inertia (Acceleration)

21. Figure 10-4 shows a pendulously suspended magnet (with residual dip) in the northern hemisphere. Notice that the vertical line through the pivot point lies closer to the nearer (north) magnetic pole than does the centre of gravity of the magnet.

22. Figure 10-5 shows the same magnet viewed from above (the diagram is rotated through 90° to put north conventionally at the top of the diagram).

23. Figure 10-6 shows the effect on the magnet of an acceleration on a heading of 090°(M). The acceleration is felt through the pivot but, due to inertia, the magnet wishes to maintain its state of uniform motion and so a reaction is evident which acts through the centre of gravity

Direct Reading Compasses

FIGURE 10-4
Effect of Residual Angle of Dip - Northern Hemisphere

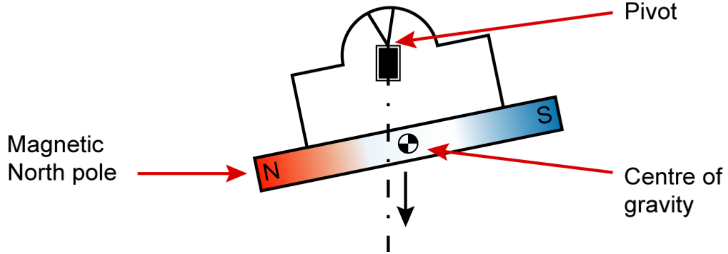


FIGURE 10-5
Relative Positions of Pivot and Centre of Gravity (Northern Hemisphere)

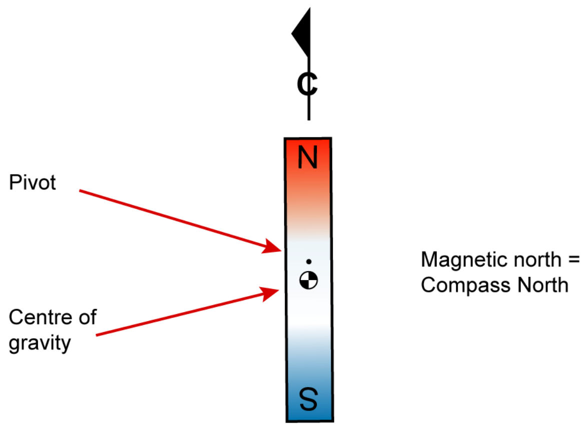
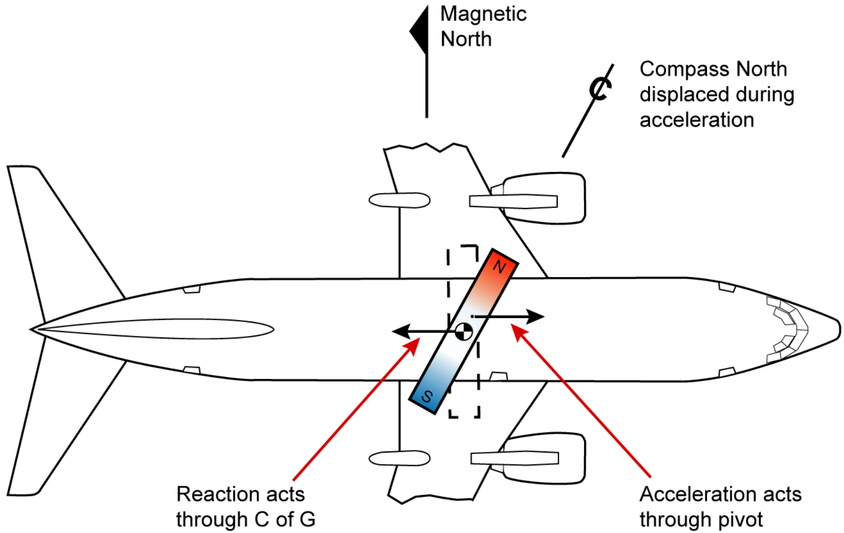


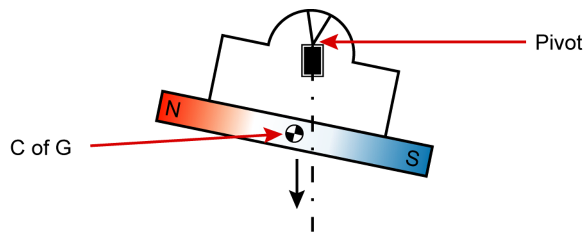
FIGURE 10-6
Inertia Effect - Aircraft Accelerating East in the Northern Hemisphere



24. The needle has swung clockwise when viewed from above. The compass will therefore read less than 090° during the acceleration. A deceleration on a heading of 270°(M) will also cause the magnet to swing clockwise when viewed from above, causing the compass to read less than 270° during the deceleration.

25. Figure 10-7 shows a pendulously suspended magnet in the southern hemisphere. Again the pivot point is nearer than the centre of gravity to the nearest (south) pole, as illustrated at Figure 10-8. An acceleration on a heading of 090°(M) will cause the magnet to swing anti-clockwise when viewed from above, see Figure 10-9. This will cause the compass reading to increase during the acceleration. A deceleration on a heading of 270°(M) will cause the needle to rotate in the same direction as an acceleration on 090°M, again causing an increase in the compass reading.

FIGURE 10-7
Effect of Residual Angle of Dip - Southern Hemisphere



Direct Reading Compasses

FIGURE 10-8
Relative Positions of Pivot and Centre of Gravity (Southern Hemisphere)

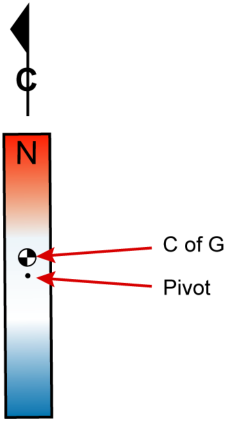
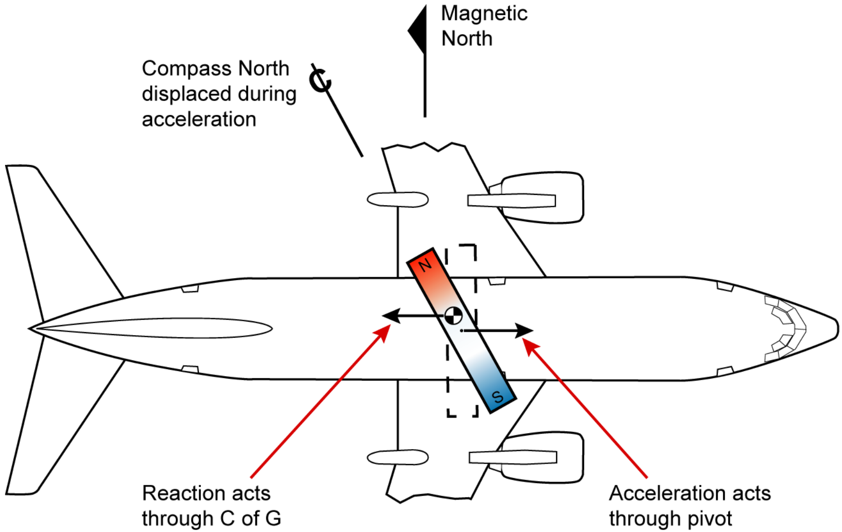


FIGURE 10-9
Inertia Effect - Aircraft Accelerating East in the Southern Hemisphere



26. From Figure 10-6 and Figure 10-9 it is apparent that:

(a) An acceleration **always** produces an **apparent** turn towards the magnetic pole which is physically closest to the aircraft;

(b) A **deceleration always** produces an **apparent** turn towards the magnetic equator.

27. At the magnetic equator there will be no acceleration error. This is because the magnet lies in the horizontal plane (no residual dip) and therefore the pivot point and the centre of gravity are vertically co-incident.

28. On headings of north and south there will again be no acceleration error, since both the centre of gravity and the pivot lie along the aircraft's fore and aft axis. In this situation acceleration or deceleration causes the magnet to move in the vertical plane (causing a change in the residual dip angle) rather than the horizontal plane (which would cause an erroneous heading indication).

The Z Field Effect (Acceleration)

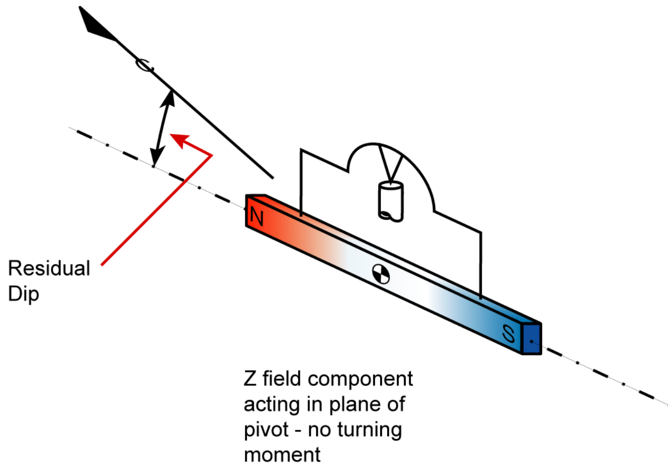
29. The effect of the Earth's vertical field component during change of speed is not as easy to explain as the effect of inertia. Remember, however, that the error caused by the Z field effect **always acts in the same direction as the error caused by inertia.**

30. Figure 10-10 shows a pendulously suspended magnet in the northern hemisphere. Since the magnet lies beneath the pivot, the Z field can have no turning effect. Figure 10-11 shows the effect of an acceleration to the east. The magnet is left behind (because of inertia) and no longer lies vertically beneath the pivot. The Z field can now exert a turning force on the dipped end of the magnet. The magnet can only turn by rotating about the pivot, causing the already dipped north-seeking end to move downwards under the influence of the Z field. The magnet therefore moves in a clockwise direction when viewed from above, as shown at Figure 10-12.

Direct Reading Compasses

FIGURE 10-10

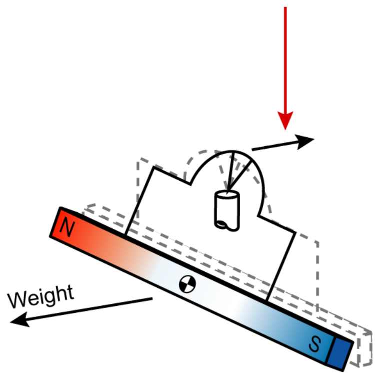
Z Field Effect



Direct Reading Compasses

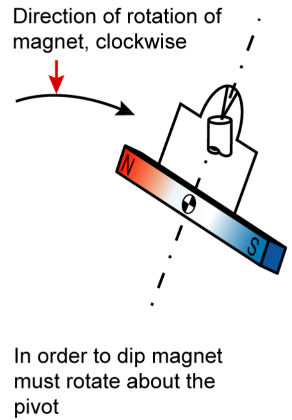
FIGURE 10-11

Z Field Effect (Continued)



Z field component now free to exert a turning influence on magnet

FIGURE 10-12
Z Field Effect (Continued)



31. Refer back to Figure 10-6(c) and check that the inertia induced error and the Z field error are in fact complementary. They are, and this is always so.

Turning Errors

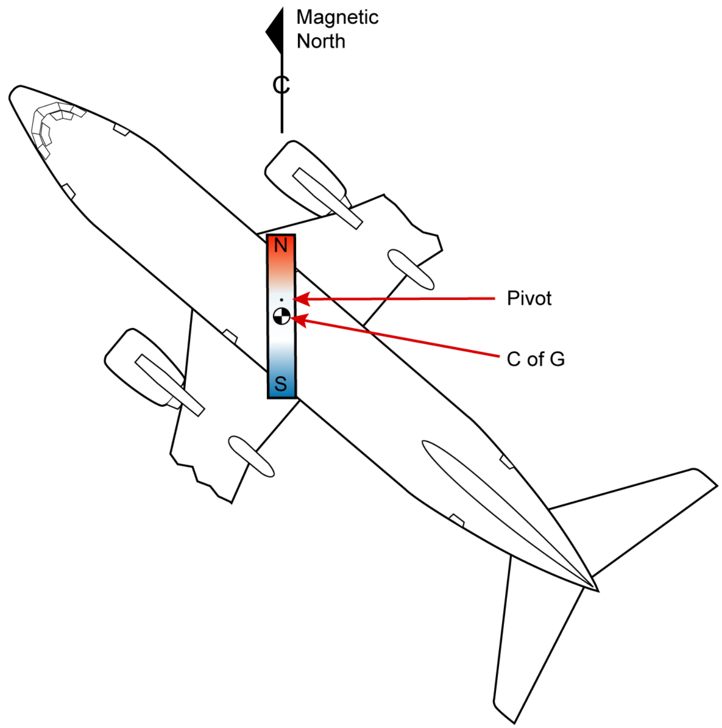
32. Turning errors should be easy to understand if it is appreciated that any turn is effectively an acceleration towards the centre of curvature of the turn. Again it is the effect of inertia and of the Z field which are primarily responsible for errors in a turn.

33. Once again, appreciate that the errors induced by inertia and by the vertical field component are **always** complementary.

Inertia (Turning)

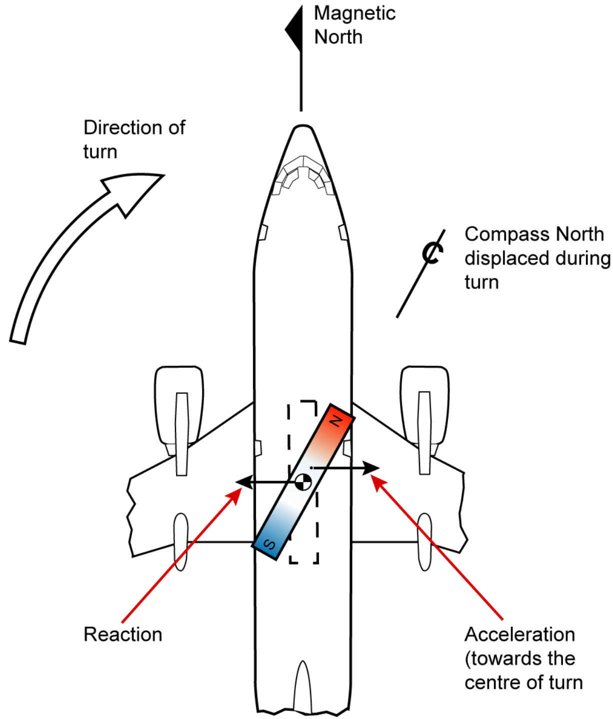
34. Figure 10-13 shows a pendulously suspended magnet (with residual dip) in the northern hemisphere. The aircraft within which the compass is fitted is presently heading 315°(M) and compass deviation is assumed to be zero. Notice that, as always, the pivot point lies closer to the nearer magnetic pole than does the centre of gravity of the magnet. If you are in doubt as to why this is so refer back to Figure 10-5 and Figure 10-8.

FIGURE 10-13
Turning Errors - Northern Hemisphere



Direct Reading Compasses

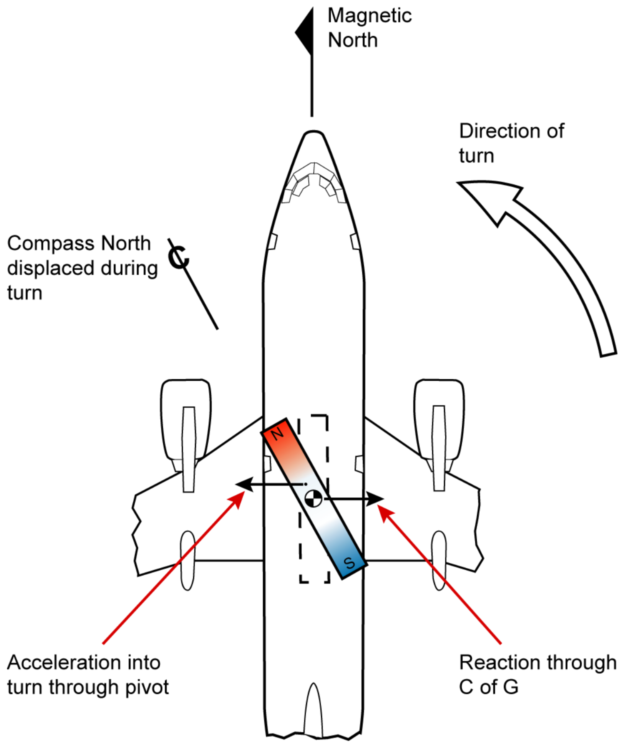
FIGURE 10-14
Aircraft Turning From 315°(M) onto 045°(M) - Northern Hemisphere



35. Figure 10-14 shows the same aircraft during a turn from 315°(M) through north onto 045°(M). The aircraft is presently passing through 360°(M). The acceleration force (towards the centre of curvature of the turn) is acting through the pivot and the reaction to this force is acting through the centre of gravity of the magnet. The result is that, during the turn, the magnet will swing clockwise when viewed from above, and consequently the compass will under-read.

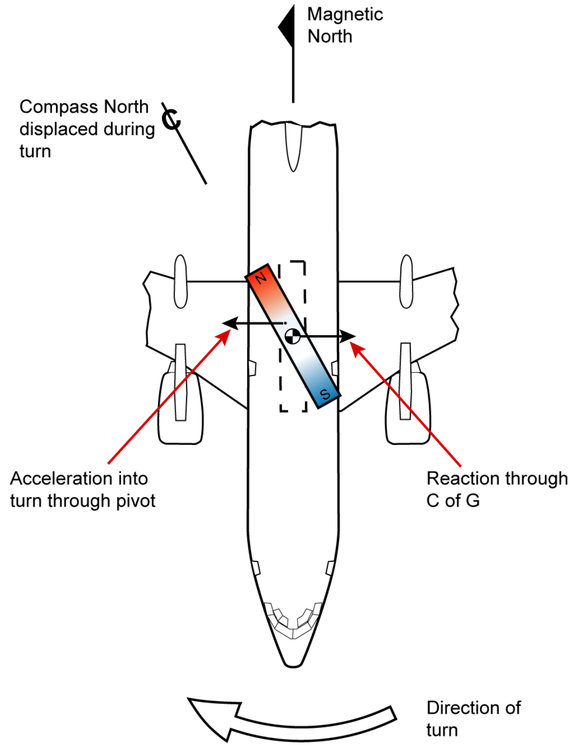
36. Figure 10-15 shows an aircraft in the northern hemisphere turning from 045°(M) through north onto 315°(M). Again, for simplicity, deviation is assumed to be zero during straight and level flight at a constant airspeed. As the aircraft passes through 360°(M), the acceleration force and the reaction to it have caused the magnet to swing anti-clockwise when viewed from above, and the compass will over-read during the turn

FIGURE 10-15
Aircraft Turning From 045°(M) onto 315°(M) - Northern Hemisphere



37. Figure 10-16 shows an aircraft in the northern hemisphere turning from 135°(M) through south onto 225°(M). As the aircraft passes through 180°(M), the magnet has swung anti-clockwise when viewed from above and is causing the compass to over-read during the turn.

FIGURE 10-16
Aircraft Turning From 135°(M) onto 225°(M) - Northern Hemisphere



38. Three of the eight possible conditions where turning errors will exist have now been illustrated and discussed. Figure 10-17 summarises the turning errors in all eight cases. The turning errors listed at Figure 10-17 are for reference only and you shouldn't try to memorise them. It is far better to apply the logic previously discussed to resolve the effect of turning errors.

39. The term **'sluggish'** which appears in the right-hand column of the table denotes that the compass heading is lagging behind the aircraft heading. Conversely, when the term **'lively'** is used, the compass is leading the aircraft around the turn.

FIGURE 10-17
Summary of Turning Errors

Direction of Turn	Hemi-sphere	Displacement of Magnet (viewed from above)	Compass Reading Error	Compass Condition
315°(M) through N to 045°(M)	Northern	Clockwise	Under-read	Sluggish
045°(M) through N to 315°(M)	Northern	Anti-clockwise	Over-read	Sluggish
135°(M) through S to 225°(M)	Northern	Anti-clockwise	Over-read	Lively
225°(M) through S to 135°(M)	Northern	Clockwise	Under-read	Lively
315°(M) through N to 045°(M)	Southern	Anti-clockwise	Over-read	Lively
045°(M) through N to 315°(M)	Southern	Clockwise	Under-read	Lively
135°(M) through S to 225°(M)	Southern	Clockwise	Under-read	Sluggish
225°(M) through S to 135°(M)	Southern	Anti-clockwise	Over-read	Sluggish

From Figure 10-17, the following rules of thumb can be formulated:

(a) During a turn through the pole which is physically nearer to the aircraft, the compass will be **sluggish**. It is therefore necessary to roll out early when using the direct reading compass.

(b) During a turn through the pole which is physically further from the aircraft the compass will be **lively**. It is therefore necessary to roll out late when using the direct reading compass.

40. From the two statements above it can be seen that, at the magnetic equator, there is no turning error (no residual dip). Furthermore, when rolling out on headings near to 090°(M) and 270°(M), the turning error will be minimal as the acceleration force and the reaction to it will lie close to a north-south direction and will result only in a change of the residual angle of dip.

41. Remember that it is a displacement of the magnet in a **clockwise** direction when viewed from above which causes the compass to **under-read,** and a displacement in an **anti-clockwise** direction which causes the compass to **over-read**.

The Z Field Effect (Turning)

42. The effect of the Earth's vertical field component during a turn is not easy to illustrate with two-dimensional diagrams. Remember, however, that the error caused by the Z field effect **always** acts in the same direction as the inertia induced error.

43. Consider an aircraft in the northern hemisphere turning from 315°(M) through north onto 045°(M). During the turn, the magnet will be thrown outwards (towards the high wing) and will no longer be vertically beneath the pivot. The Z field component will now cause the dipped (in this case the north-seeking end) of the magnet to swing downwards and clockwise about the pivot. This situation is as illustrated at Figure 10-12, remembering that the acceleration is into the turn. Check Figure 10-14 and you will see that the clockwise movement of the magnet under the influence of the Z field complements the displacement of the magnet due to inertia.

Liquid Swirl

44. A third factor involved in turning errors is liquid swirl. Ideally the compass bowl will have a smooth internal surface and be full of a low viscosity fluid. If these conditions are not fully met, there will be a tendency for the liquid to be dragged around the bowl during turns. Once in motion, the liquid will continue to swirl under its own momentum. Should it occur, this fluid swirl will carry with it the magnetic assembly, thereby displacing it from its correct orientation. The displacement will therefore be in the direction of the turn and a clockwise turn (135° through south to 225°) with liquid swirl will result in a clockwise turn of the magnets and an under-reading compass.

45. Depending on the hemisphere and the direction of turn, the effect of liquid swirl will either increase or decrease the turning error. At the magnetic equator liquid swirl would be the only source of any turning error.

Compass Deviation

Compass Deviation

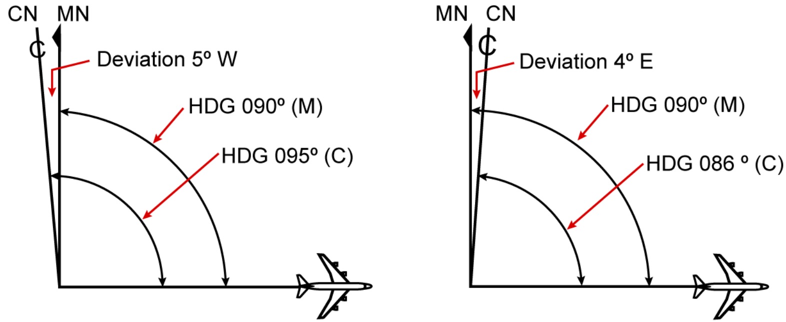
1. It is unlikely that the magnets in an aircraft compass will point exactly towards magnetic north. The aircraft and its equipment (electrical wiring, payload etc) will produce a separate magnetic field which will tend to deflect the magnets by, hopefully, only a few degrees at most.

2. The direction in which the magnets point under the influence of the combined terrestrial and aircraft magnetic fields is termed compass north, and the angle between magnetic north and compass north is termed **compass deviation**.

3. If compass north lies to the west of magnetic north the deviation is westerly (or negative) as shown at Figure 11-1(a). If the compass north lies to the east of magnetic north the deviation is easterly (or positive) as shown at Figure 11-1(b).

4. Note that, unlike magnetic variation, deviation changes with change of aircraft heading.

FIGURE 11-1
The Calculation of Compass Deviation



(A) Deviation West (-)
Compass best

(B) Deviation East (+)
Compass least

5. Deviation in an aircraft compass is reduced to a minimum by conducting a **compass swing. Residual deviations** are then recorded on a **compass deviation card/curve** which is mounted in the aircraft. The compass swing is discussed in another part of this syllabus.

Sources of Aircraft Magnetism

6. Deviation can occur as a result of **hard iron** and **soft iron** magnetism within an aircraft and **also as a result of misalignment of reference lines** from which readings are taken or of the detector element in a remote sensing compass system.

Permeability

7. **Permeability** is the name given to the capacity of a material to become magnetised. In scientific terms it is expressed as the ratio between the flux density (magnetic field strength) within the material compared with the magnetising force. Materials possessing high permeability are easily magnetised whereas materials such as rubber, brass, copper, aluminium, plastic, and carbon fibre have very low or zero permeability and cannot be magnetised. Material such as Permalloy has a high permeability but once the magnetising force is removed it decays very quickly. In the case of iron however, once magnetised the magnetism is likely to decay at a much slower rate. The measure of whether iron is hard or soft is this capacity to retain magnetism. The magnetisation process is called magnetic induction.

Hard Iron Magnetism

8. **Hard iron magnetism** is the name given to magnetism that is virtually **permanent** or decays very slowly. It is caused by the presence of iron and steel components within the aircraft structure that acquire magnetism slowly from the Earth's magnetic field during manufacture (hammering, riveting, vibration assist the process). Also it can form when an aircraft remains on the ground on one heading for a long period of time (weeks). A very powerful electrical shock such as that received in a lightning strike can also establish hard iron magnetism in an aircraft. **Hard iron magnetism is very difficult to remove.**

Soft Iron Magnetism

9. **Soft iron magnetism** is the name given to **temporary** magnetism. Such magnetism is usually induced within highly permeable 'soft' iron or steel components by the presence of magnetic fields, but **decays rapidly when the magnetising force is removed.** Such magnetism is frequently induced by the magnetic fields surrounding electrical components. For example, the operation of an electrical motor (such as a windscreen wiper) in the cockpit of an aircraft can therefore induce temporary soft iron magnetism that disappears when the component is switched off.

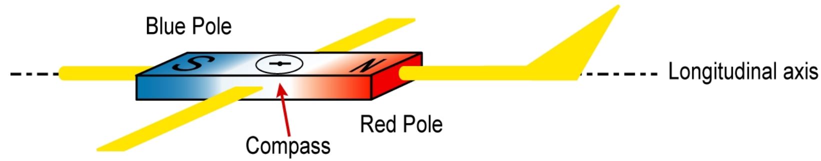
Analysis of Hard and Soft Iron Magnetism

Components of Hard Iron Magnetism

10. In order to analyse and correct for the effect of magnetism within an aircraft it is necessary first to analyse its effect in detail. To achieve this the magnetism is assessed to act in each of three planes within the aircraft structure and thus to have three identifiable components. The three components are labelled with the capital letters **P, Q and R** depending whether they act in the **fore/aft, athwartships (lateral axis) or vertical axis of the aircraft respectively.**

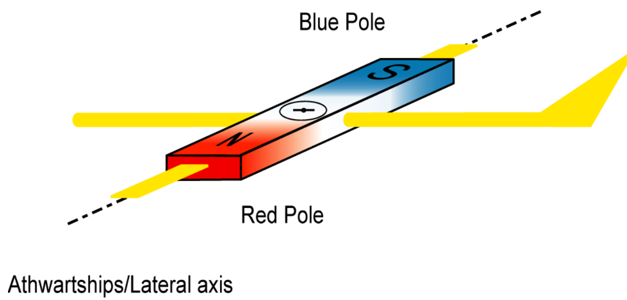
11. **Magnetism component P.** The conventional description of the effect of magnetism component P draws on the analogy of a 'bar magnet' positioned in the fore/aft axis with its poles on either side of the aircraft compass system. Component P is, by convention, positive when the south-seeking or blue pole is ahead of the compass system and negative when behind it. Figure 11-2 illustrates diagrammatically the principle and shows a component +P.

FIGURE 11-2
Hard Iron Magnetism (Component +P)



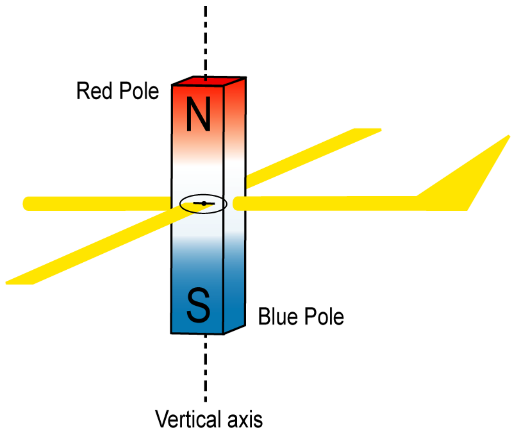
12. **Magnetism component Q.** A similar analogy is used to identify component Q, but this time the 'bar magnet' is positioned so that its poles are ether side of the compass system. By convention, a south-seeking (blue) pole on the starboard side is positive and when on the port side is negative. Figure 11-3 illustrates +Q.

FIGURE 11-3
Hard Iron Magnetism (Component +Q)



13. **Magnetism component R.** This component is assumed to act in the vertical plane of the aircraft above and below the compass system. A south-seeking (blue) pole below the compass system is positive and a red pole negative. Figure 11-4 illustrates +R.

FIGURE 11-4
Hard Iron Magnetism (Component +R)



Effects of Components P,Q and R

14. In an aircraft compass system the compass magnets are mounted horizontally to enable them to align with the horizontal component of the Earth's magnetic field. The effect of component R on the compass magnets should not interfere with this alignment providing the aircraft remains in level flight. Were the aircraft to pitch up or down the R would no-longer remain vertical and would then cause some deviation. In practice the effect on a compass system are negligible compared to other errors likely to be present and are generally ignored.

15. In normal operation the magnets in a direct reading compass remain nearly horizontal and aligned with North. However, the position of components P and Q relative to the compass magnets will change with change of aircraft heading and whilst the strength of the components does not change with heading their deviating effect does. This aspect is covered in detail in the chapter on compass swinging.

16. In principle, the effect of each of the components P and Q is to produce a proportion of the total deviation called a **'coefficient of deviation'**. Each coefficient is also given an identifying letter. **Component +P produces coefficient +B, component +Q produces coefficient +C,** (similarly -P produces –B etc.). The purpose of the compass swing is to identify these coefficients to enable corrections to be made to cancel their effect.

Soft Iron Magnetism

17. **Soft iron magnetism** is temporary, induced magnetism, it is present only when an inducing magnetic field is present. One of the main causes of soft iron magnetism is the Earth's magnetic field in which the horizontal and vertical components may induce horizontal and vertical soft iron magnetism respectively in the aircraft structure near to the compass magnets. The effect of horizontal soft iron is rather complex but as it is strongest when the H component of Earth magnetism is strongest the deviation caused is relatively small. Vertical soft iron magnetism is more important because it is strongest in high latitudes because Z is stronger there, whereas H is weaker and the magnets of the compass are more easily deflected.

18. The bar magnet analogy is continued with respect to vertical soft iron magnetism. In this case the reader must imagine vertical bar magnets positioned ahead and behind the compass magnets on the longitudinal axis giving soft iron component 'cZ'. Also, vertical bar magnets positioned on the lateral (or athwartships) axis causing component 'fZ'.

19. From the positioning of cZ and fZ it can be seen that **these components contribute to coefficients of deviation B and C respectively.**

Summary of Effects of Hard and Vertical Soft Iron Magnetism

20. **Hard iron magnetism is permanent and does not vary with magnetic latitude** but, **the deviation caused by hard iron magnetism increases with latitude** because in high latitudes where H is weaker, the compass magnets are more easily deflected. **The deviating effect of hard iron magnetism varies with the heading of an aircraft.**

21. Vertical soft iron magnetism increases when an aircraft moves to a higher latitude because Z increases. The deviation caused by vertical soft iron magnetism in higher latitudes is doubly increased because H is reduced. The deviating effect of the vertical soft iron component of coefficients B and C varies with the heading of an aircraft.

22. **Co-efficient A results from misalignment.** Co-efficient A errors are therefore removed by re-alignment of the lubber line in the direct reading compass.

Removing Co-efficients A, B and C in an E Type Compass

23. The principles for removing co-efficients are the same, regardless of whether it is a direct reading compass or a gyro magnetic compass (see later) which is considered.

24. Errors due to co-efficients B and C are minimised by deliberately introducing magnetic fields which have an equal but opposite effect to that of the aircraft's own magnetic fields. This is achieved by means of **scissor magnets** in direct reading compasses and **electro magnets** in gyro magnetic compasses.

Adjusting the E Type Compass

25. The E Type compass is fitted to its mountings using slotted channels which enable the entire compass to be rotated once the retaining screws are loosened (using a non-magnetic screwdriver).

26. Co-efficient A can be removed on any heading, since it has the same deviating effect on all headings.

27. If it is necessary to remove a positive (easterly) co-efficient A the compass is physically rotated in a clockwise direction by the required number of degrees. This will cause the compass reading to increase.

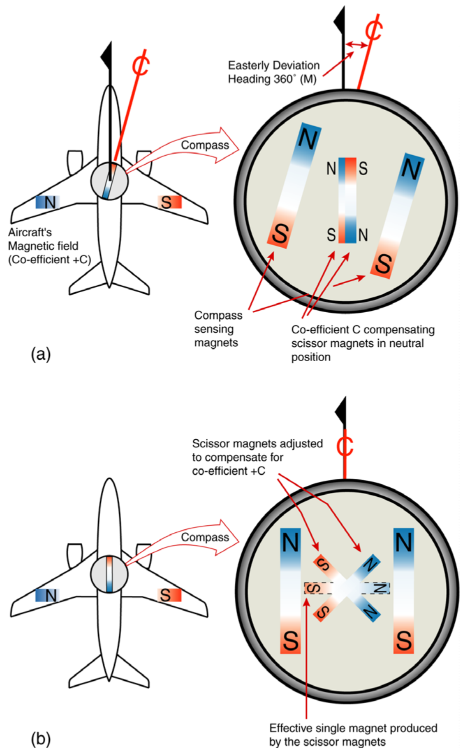
28. **To remove co-efficients B and C scissor magnets are adjusted using grub screws/Allen key on the instrument face.** These scissor magnets are attached to the compass casing and therefore change position relative to the sensing magnets as the aircraft alters heading. Their effect upon the compass reading will therefore depend on the aircraft heading, in much the same way as the aircraft magnetic fields causing deviations B and C.

29. Figure 11-5(a) shows an aircraft with co-efficient + C represented by a magnet with its south-seeking end in the starboard wing. At Figure 11-5(a) the scissor magnets used to compensate for co-efficient C are in their **neutral** position.

30. These two small scissor magnets have equal pole strengths and when set in the neutral position they exert no influence on the pendulously suspended sensing magnets of the compass.

31. Figure 11-5(b) shows the same aircraft, but now the scissor magnets have been adjusted to compensate for deviations caused by the co-efficient +C.

FIGURE 11-5
Co-efficient C Compensation Using Scissor Magnets



32. Compensation is achieved by turning the appropriate grub screw until the required heading beneath the lubber line. In order to avoid turning the grub screw the wrong way remember that **negative** co-efficients are removed by turning the appropriate grub screw **clockwise,** whilst **positive** co-efficients are removed by turning the appropriate grub screw **anti-clockwise.** Co-efficient B is adjusted in the same manner as that described in the foregoing paragraphs.

Gyroscopic Principles

Gyroscopic Principles

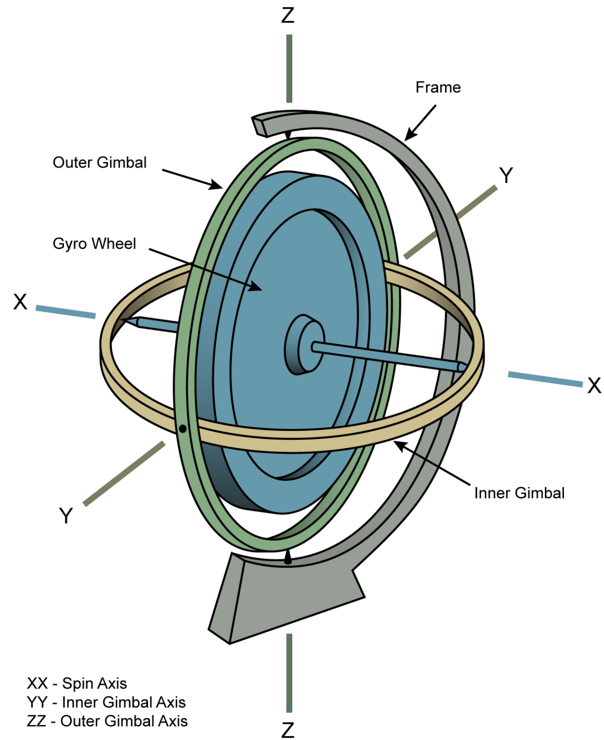
1. Fundamental gyroscopic theory is considered below.

The Gyroscope

2. An unrestricted gyroscope consists of a spinning wheel, mounted within two rings, or **gimbals.** The axle, or **spin axis,** of the gyro wheel is mounted in the inner gimbal. The inner gimbal is itself pivoted at two points on the outer gimbal, these points being 90 degrees removed around the inner gimbal from the spin axis bearings. The outer gimbal is itself pivoted at two points 90 degrees removed from the inner gimbal pivots.

3. The system described above and illustrated at Figure 12-1 allows movement of the gyro in three mutually perpendicular planes. Such a gyro is known as a **free gyro** or, more commonly, as a **space gyro.**

FIGURE 12-1
Space/Free Gyro



XX - Spin Axis
YY - Inner Gimbal Axis
ZZ - Outer Gimbal Axis

Fundamental Properties of a Gyro

4. A gyro possesses two distinct properties:

(a) **Rigidity.** Rigidity, or gyroscopic inertia, is the property of a gyroscope which causes it to continue to rotate in the same plane. It therefore follows that, in the absence of any external force, the spin axis will continue to point to the same position in space to which it was originally set.

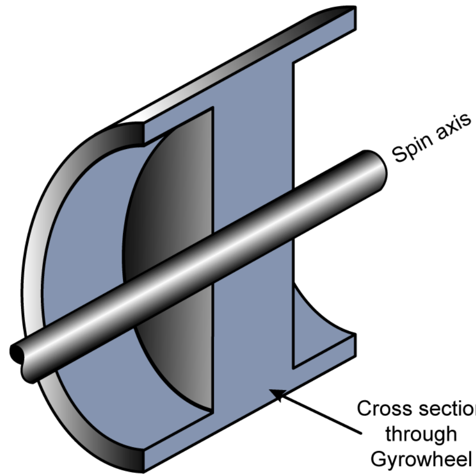
In order to enhance the rigidity of a gyro it is necessary to:

(i) increase the spin speed

(ii) increase the mass of the wheel

(iii) concentrate the mass of the wheel about the circumference, as shown at Figure 12-2

FIGURE 12-2
Construction of a Gyro Rotor



(b) **Precession.** If an external force is applied at the gyro, the gyro will move as if the external force had been applied at a point 90 degrees removed from the actual point of application in the direction of rotation of the gyro. Precession is illustrated at Figure 12-3.

At Figure 12-3 the applied force is considered to act at the rim of the gyro, as is the precessed force. At Figure 12-4 the same applied force is shown acting on the spin axis, the result is the same.

Gyroscopic Principles

FIGURE 12-3
The Effect of Precession on a Gyro

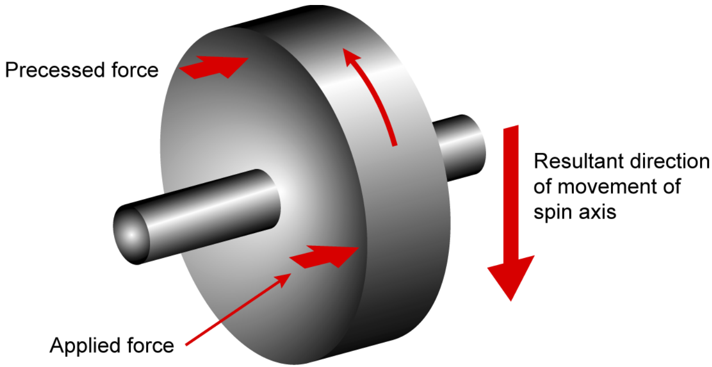
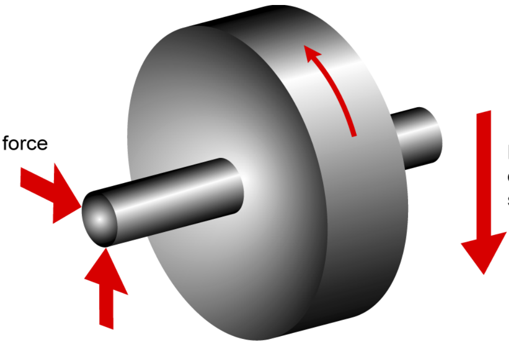


FIGURE 12-4
Alternative Analysis of Figure 12-4



The rate of precession of a gyroscope depends on the magnitude of the applied (deflecting) force and the inherent resistance to that force (rigidity or gyroscopic inertia). Increasing the mass or the speed of rotation of rotor increases rigidity. The rate of precession can therefore be said to be directly proportional to the applied force and inversely proportional to the mass and rotational speed of the rotor.

Gyroscopic Wander

5. Due to its rigidity, the spin axis of a perfect gyro should continue to point in a fixed direction. Any movement of the spin axis away from this fixed direction is known as **gyro wander.** Depending on the direction in which the spin axis moves, the gyro may be said to be **drifting or toppling**.

(a) **Gyro drift** occurs whenever the spin axis moves in a **horizontal plane,** see Figure 12-5.

(b) **Gyro topple** occurs whenever the spin axis moves in a **vertical plane,** see Figure 12-5 and Figure 12-6.

FIGURE 12-5
Drift and Topple in a Horizontal Gyro

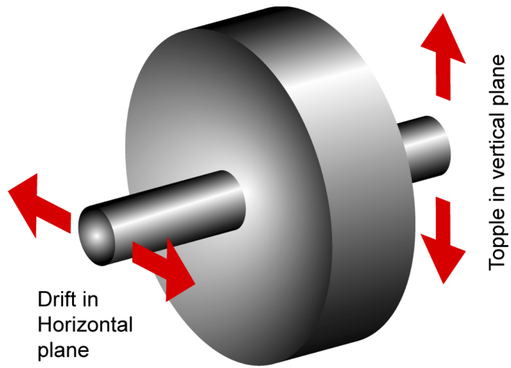
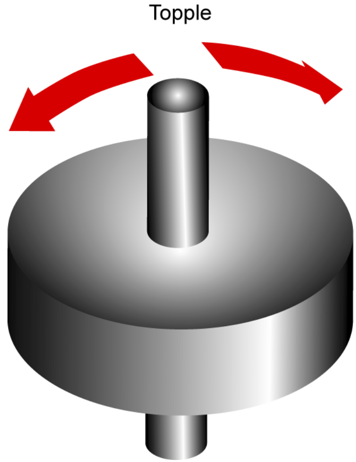


FIGURE 12-6
Topple in a Vertical Gyro



Note at Figure 12-6 that a gyro whose spin axis is vertically mounted **cannot drift but can only topple.**

Real Wander

6. Whenever the spin axis **actually** moves relative to a **fixed point in space**, the gyro is said to be suffering **real wander**, that is to say **real drift, real topple** or a composite of both.

7. Such **real wander** may be deliberately induced (by the latitude nut of a DGI) or may be due to mechanical imperfections in the gyro assembly, for example:

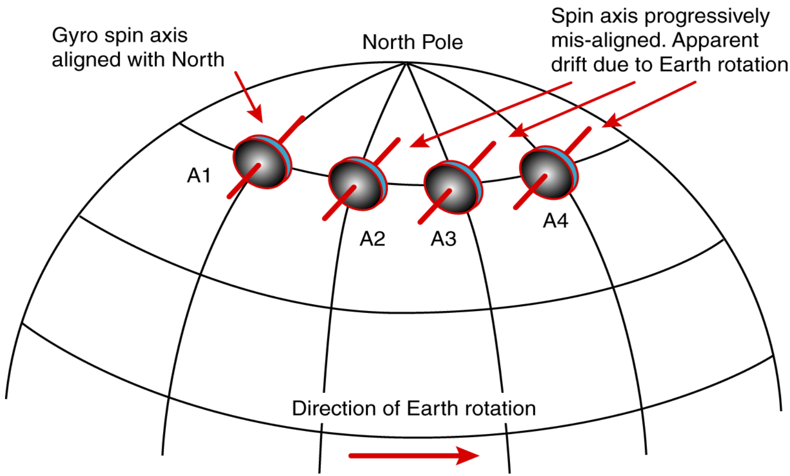
- An imperfectly balanced gyro wheel.
- Imperfectly balanced gimbals.
- Uneven friction loadings at the bearings.

Apparent Wander

8. Whenever the spin axis of a perfect gyro (with no **real wander**) appears to an Earth-bound observer to be changing direction, the gyro is said to be suffering from **apparent wander.**

9. Figure 12-7 shows **apparent drift**. The spin axis of a perfect gyro is aligned with true north at time A1. The gyro continues to remain perfectly rigid relative to a fixed point in space, however with the passage of time (A_2, A_3, A_4) the spin axis **appears** to an Earth-bound observer to be **drifting** away from true north. Appreciate that the gyro is stationary on the Earth, it is the Earth which is moving about its own spin axis.

FIGURE 12-7
Apparent Drift due to Earth Rotation - Mid Latitude

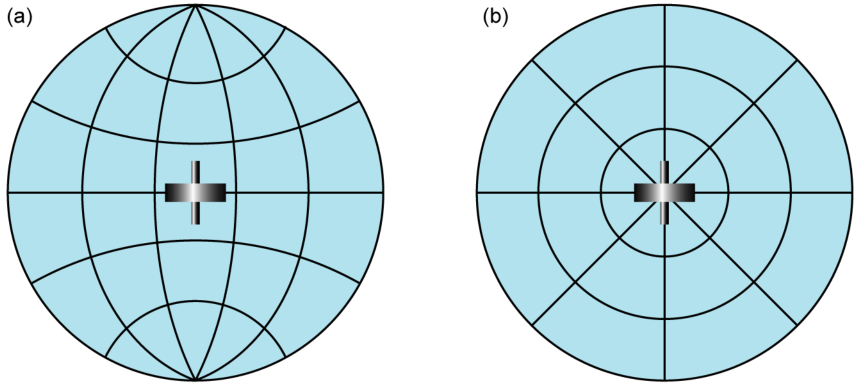


10. Apparent drift does not occur at the equator, since the meridians are parallel. At the poles the rate of apparent drift is equal to the rate of Earth rotation (15° per hour), see Figure 12-8(a) and Figure 12-8(b).

11. The formula for the **apparent drift** (due to Earth rotation) is:

Rate of apparent drift = 15° x the sine of the latitude °/hour

FIGURE 12-8
Apparent Drift due to Earth Rotation - Equator/Poles



Zero Drift at Equator

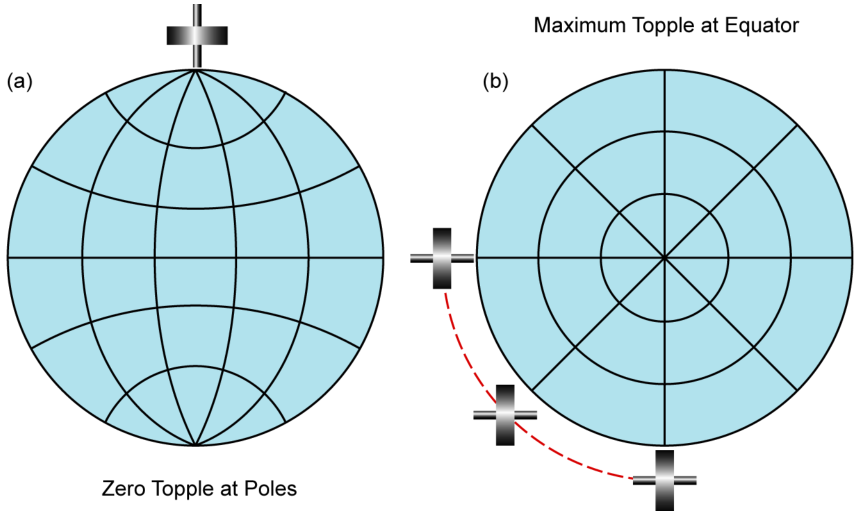
Maximum Drift at Poles

12. **Apparent topple** is calculated using the formula:

Rate of apparent topple = 15° x the cosine of the latitude °/hour

It is therefore zero at the poles, but occurs at the maximum rate of 15° per hour at the equator, see Figure 12-9(a) and Figure 12-9(b).

FIGURE 12-9
Apparent Topple due to Earth Rotation - Equator/Poles



13. Apparent wander (either drift or topple) also occurs whenever the gyro is transported east or west across the surface of the Earth. This apparent wander is specifically termed **transport wander.**

Types of Gyroscopes

14. There are five types of gyros which need to be considered:

The space gyro. As illustrated at Figure 12-1, a space gyro has two gimbals and has total freedom of movement about three axes, one of which is the spin axis.

The tied gyro. A gyro with two gimbals having freedom of movement in three planes, but which is controlled by some external force, is called a tied gyro. An example of a tied gyro is the **directional gyroscopic indicator** (DGI), where the spin axis is constrained to remain within the aircraft's **yawing plane**.

The Earth gyro. A gyro with two gimbals having freedom of movement in three planes, but controlled by gravity, is called an Earth gyro. An example of an Earth gyro is the **artificial horizon**, where the spin axis is constrained by gravity to remain **Earth vertical.**

The rate gyro. A gyro with only one gimbal having freedom of movement about only two axes, one of which is the spin axis, is called a rate gyro, and is designed to show **rate of movement** about the axis which is mutually at right angles to the two axes of freedom. An example of a rate gyro is found in the **turn and slip indicator**, or the **turn co-ordinator.**

The rate integrating gyro. Rate integrating gyros are used in inertial navigation systems. As the aircraft turns about the sensitive axis of the rate gyro a precession results and is used to generate an error signal, the magnitude of which signifies the rate at which the aircraft is turning about the sensitive axis.

Gyroscope Drive Types and Monitoring

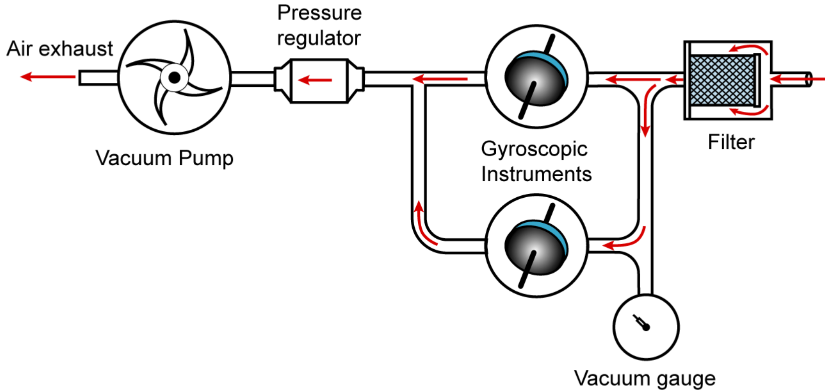
15. Gyroscopic instruments may either be air driven (by an engine-driven vacuum or pressure pump), or electrically driven.

Air Driven Gyros

16. Air driven gyros are used in a number of smaller aircraft, and may be either **vacuum** or **pressure**. A typical vacuum system is shown in Figure 12-10; it consists of an engine-driven pump that is connected by pipes to the appropriate flight instruments. A vacuum gauge, a relief valve, and an air filter also form part of the system. The pump produces a vacuum that is generally regulated by the valve at a value between 3.5 and 4.5 in Hg; however, some types of turn-and-bank indicator may operate at a lower value which is obtained by including an additional relief valve in the main supply line.

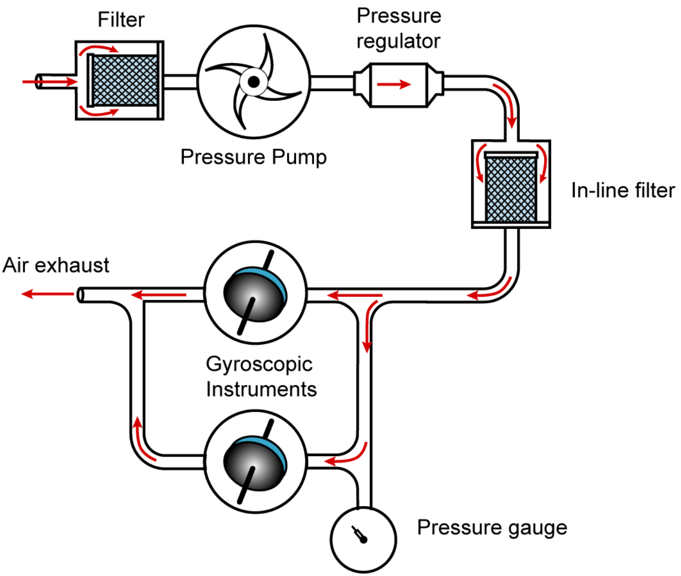
17. As the air is drawn through the filter and then through the gyro instruments it impinges on 'buckets' on the respective rotors, which therefore rotate at high speed.

FIGURE 12-10
Vacuum-operated System



18. During system operation the relief valve remains closed by compression of a spring, the tension of which is pre-adjusted to obtain the required vacuum so that air pressure acting on the outside of the valve is balanced against spring tension. If for some reason the adjusted value should be exceeded, the outside air pressure would overcome spring tension thus opening the valve to allow outside air to flow into the system until the balanced condition is again restored.

FIGURE 12-11
Pressure-operated System



19. A pressure-operated system is similar to a vacuum system apart from the fact that the inlet and outlet connections are reversed on each instrument (Figure 12-11).

Electrically Driven Gyros

20. In **electrically-operated instruments,** the gyroscopes are variations of ac or dc motors that are designed to be driven from the power supply of an aircraft. Typically, ac motors are adopted in artificial horizons, while dc motors are more common in turn-and -bank indicators. Gyroscopes used for the purpose of direction indicating can also be electrically-driven, but they normally form part of a slaved gyro compass or a flight director system. Such systems are covered in later sections. The power supply to an electrically driven gyro can often be monitored by **power failure flags** on the instrument face.

Advantages of Electrically Driven Gyros

21. The advantages of electrically-driven gyros are summarised below:

(a) A faster spin speed, giving greater rigidity, is achieved.

(b) A constant spin speed is easily maintained, since it is independent of engine rpm.

(c) The unit can be sealed to be airtight, thereby eliminating dust and moisture which can accelerate wear at the bearings.

(d) The heat generated by the electric motor can be used to maintain a stable operating temperature, again minimising wear at the bearings.

(e) At high altitudes the rotor speed of an air driven gyro will reduce due to the smaller pressure differential at the altitude. No such problem exists with an electrically driven gyro which will still operate at the same rotation speed.

Solid State or Ring Laser Gyros

Solid State or Ring Laser Gyros

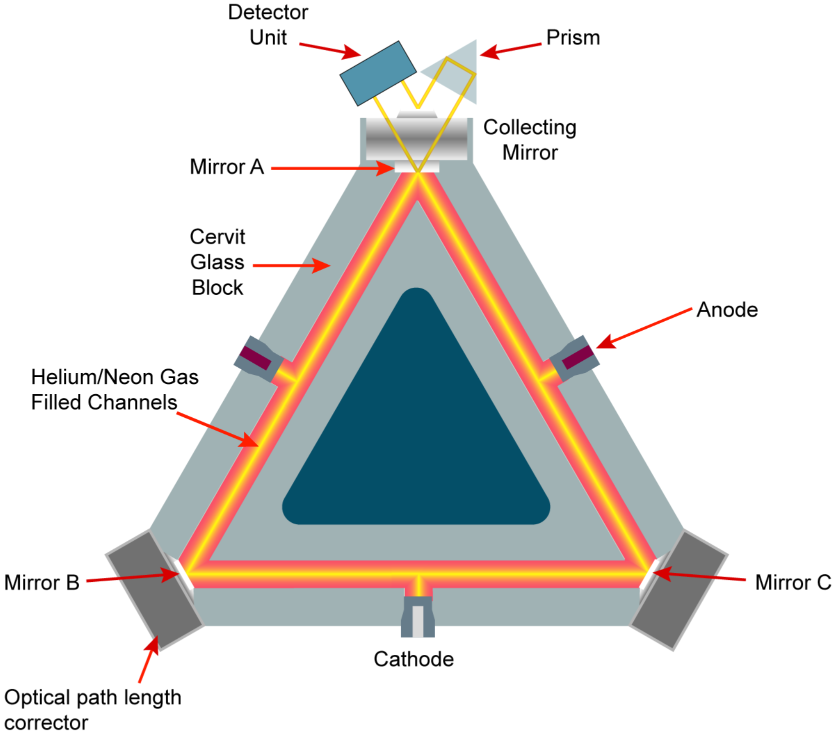
1. The ring laser gyro (RLG) is just about as different from a conventional gyro as it is possible to get. The RLG operates on the principle of the relative movement of two beams of laser light, whereas a conventional gyro operates on the principle of stored mechanical energy (inertia). RLGs are a solid state alternative to the conventional rate integrating gyro.

Construction

2. The construction of a triangular ring laser gyro is shown at Figure 13-1. Figure 13-1 shows a cross section through the sensitive plane of the RLG, the sensitive axis passes perpendicularly through the page. Square RLGs are also manufactured, the principle of operation is much the same.

Solid State or Ring Laser Gyros

FIGURE 13-1
Ring Laser Gyro



3. The RLG is formed from a solid block of cer-vit (glass/ceramic) material which is used because it has a very low temperature co-efficient. It does not distort with age, which would degrade the accuracy and perhaps destroy the gas tight seal necessary to contain the helium and neon gases. Channels are drilled very accurately in the block to form two triangular laser paths which are filled with a low pressure helium/neon mixture.

4. Two anodes and a common cathode are used to create electrical discharges within the gases which cause the channels to act as 'gain tubes', producing laser beams. These beams travel in opposite directions around the triangular block.

5. At each of the three corners of the block are flat mirrors which reflect the laser beams into the next channel. Two of these mirrors (B and C at Figure 13-1) are movable and are controlled by piezo-electric actuators operating in a closed loop system to adjust the optical path length to within precise limits, and so to correct for the small expansions/contractions of the block with changing temperatures within the permitted operating temperature band. It is necessary that the optical path is maintained at a length which is an integral multiple of the lasing wavelength, that is to say an exact number of wavelengths of the light waves at the frequency achieved when the lasers are operating at peak power. The third mirror (A at Figure 13-1) permits a small portion of the laser light beams to pass through the mirror and to be presented at the photoelectric cells of the detector. One of these beams is necessarily routed via a prism and a further mirror (as shown at Figure 13-1) so that the beams approach the detector from the same direction.

Principle of Operation

6. As in many applications of laser oscillators, the RLG makes use of the high sensitivity of the laser's oscillating frequency to variations of the dimensions of the resonant structure. The resonant structure, in this case the triangular block, is designed such that each beam will only sustain oscillation (that is to say resonate) at one particular spot frequency. If the RLG is stationary, the resonant frequency for both clockwise and anti-clockwise laser beams will be identical since the path lengths travelled by the beams is identical. When the beams combine at the detector they interfere with each other to form a fringe pattern of light bars at the photoelectric cells of the detector.

7. Assume now that the RLG shown at Figure 13-1 is rotating in a clockwise direction. The light beam which is travelling in the same direction as the rotation must travel a slightly longer path to complete one revolution, whilst the opposite beam will travel over a correspondingly shorter path. The resonant frequencies of the two beams will therefore be different and there will be a resultant change in the interference pattern, causing the light bars produced at the photoelectric cells of the detector to move. The direction of movement of the light bars will depend upon the direction of rotation of the RLG and the distance by which the light bars move will depend on the rate of rotation.

8. At first glance it might appear that the RLG shown at Figure 13-1 cannot work in the manner described above. The laser beam which is travelling clockwise is travelling further than the laser beam which is travelling anti-clockwise, since it is necessarily routed via the prism and one further mirror. In fact the clockwise beam, having passed through mirror A, travels a precise extra distance which ensures that it arrives at the detector at exactly the same phase as it would have done with a direct path. The effect of the additional path length is therefore cancelled.

9. The output of the photoelectric cells which comprise the detector unit are converted to pulse signals which are representative of the rate and direction of rotation of the RLG in the sensitive plane.

10. One problem remains to be solved. When the RLG is not rotating in its sensitive plane, scatter from the mirrored surfaces causes the opposing beams to lock together in a phenomenon known as **frequency lock**. The beams then tend to remained locked together until the rotation of the RLG reaches a certain rate, preventing small rates of rotation from being detected. To overcome this problem a **dither** is introduced by an external vibration device which vibrates the gyro at a resonant frequency. This breaks the frequency lock and allows much smaller rotations to be measured.

11. If we consider that the sensitive axis of the RLG shown at Figure 13-1 lies parallel to the lateral axis of the aircraft, then the gyro would measure the rate and direction of roll of the aircraft, since the aircraft would be rolling in the sensitive plane of the gyro.

12. The laser gyro is employed to perform the same functions as a conventional rate integrating gyro, however because of the high cost of RLGs it is likely that their use will continue to be restricted to systems requiring high levels of accuracy combined with low weight. In modern inertial navigation/reference systems (strapped down systems), three RLGs mounted orthogonally are used to provide the same outputs as the three mechanical rate integrating gyros which were used in older (stable platform) inertial navigation systems.

Advantages of the RLG

13. The principle advantages of the RLG over its mechanical counterpart are summarised below;

(a) high reliability (up to 60,000 hours before failure has been demonstrated)

(b) sensitive to a wide range of rotation rates (0.004 to 400 degrees/second)

(c) the RLG does not suffer from mechanical precession, and cannot topple

(d) much lower power inputs are required

(e) there is no spin up time required, and the stable operating temperature is quickly achieved.

The Directional Gyro Indicator

The Directional Gyro Indicator

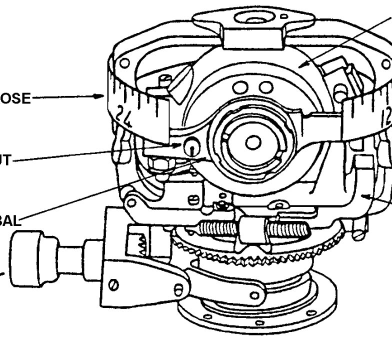
1. The Directional Gyro Indicator (DGI) or Direction Indicator (DI) is used in aircraft to give a stable heading reference which is free from the turning and acceleration errors of the Direct Reading Compass.

Principle of Operation

2. The construction of one of the older types of DGI is shown at Figure 14-1, simply because it is easier to illustrate than the modern type of instrument.

3. The spin axis is **tied** to remain within the aircraft's yawing plane. Because of the inherent **rigidity** of the gyro a stable heading reference is achieved relative to a fixed point in space. The **latitude nut** shown at Figure 14-1 is used to induce **real drift**, hopefully at a rate which **is equal in magnitude** but **opposite in direction** to the **apparent drift due to Earth rotation**. The **apparent drift due to the aircraft's movement across the Earth** (**transport wander**) cannot be compensated for in this basic instrument. Transport wander is considered shortly.

FIGURE 14-1
Directional Gyro Indicator (DGI)

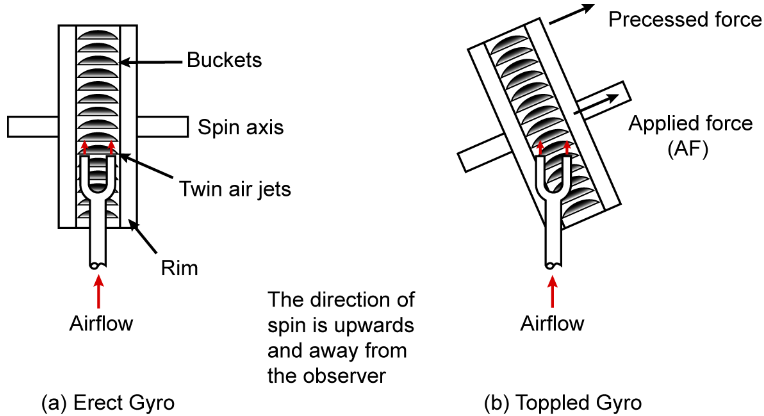


4. As with any gyro the instrument may be air driven or electrically driven. It is the air-driven instrument which is considered in the following paragraphs.

5. In order for the instrument to function correctly it is necessary to ensure that the spin axis of the gyro remains within the yawing plane of the aircraft. The spin axis is **physically clamped** into the yawing plane whenever the heading is reset. As the heading reset control is depressed a caging clamp engages with the inner gimbal, rendering the gyro spin axis **aircraft horizontal.**

6. One technique for maintaining the spin axis in the yawing plane at other times involves the use of two air jets, as shown at Figure 14-2.

FIGURE 14-2
Gyro Erection Mechanism (Air Driven DGI)



7. Providing that the spin axis is already aircraft horizontal [Figure 14-2(a)], the twin air jets both strike the buckets on the circumference of the gyro wheel, causing it to rotate.

8. Should the gyro start to topple **relative to the aircraft's yawing plane,** one or other of the air jets will strike the **rim** of the gyro [applied force AF at Figure 14-2(b)], and will erect the gyro (at a typical rate of 2°/3° per minute), since this applied force will be precessed through 90°.

9. Electrically driven DGIs use a slip ring and commutator arrangement (as described in Chapter 15, paragraphs 38 and 39) to prevent the gyro from toppling.

System Errors

Real Wander

10. The gyro will inevitably suffer from **real wander**, due to mechanical imperfections. Any **real topple** will be corrected as previously described. Any **real drift** will result in erroneous heading indications, which can only be eliminated by periodic resetting of the instrument.

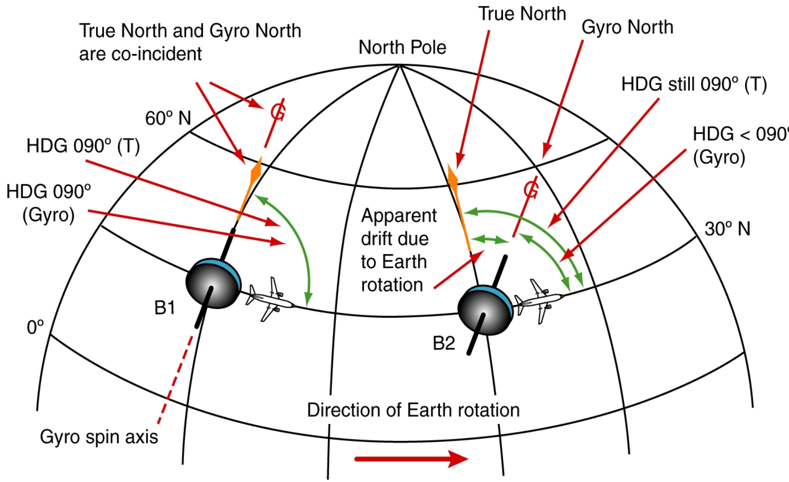
Apparent Wander

11. The gyro will also suffer from **apparent wander. Apparent drift** due to **Earth rotation** was briefly considered in the section entitled Gyroscopic Principles where it was illustrated at Figure 14-2, but now a closer look is necessary.

12. Figure 14-3 shows the spin axis of a DGI correctly aligned with true north at time B_1. The spin axis defines **gyro north**, and so at time B_1 gyro north and true north are coincident, and the DGI is reading correctly.

13. Assume that the gyro is perfectly balanced (no real drift), and that the aircraft is stationary. At time B_2 the apparent drift due to Earth rotation has caused the spin axis, and therefore gyro north, to move to the right of true north. **The DGI is consequently under-reading, and this is caused by apparent drift due to Earth rotation in the northern hemisphere.**

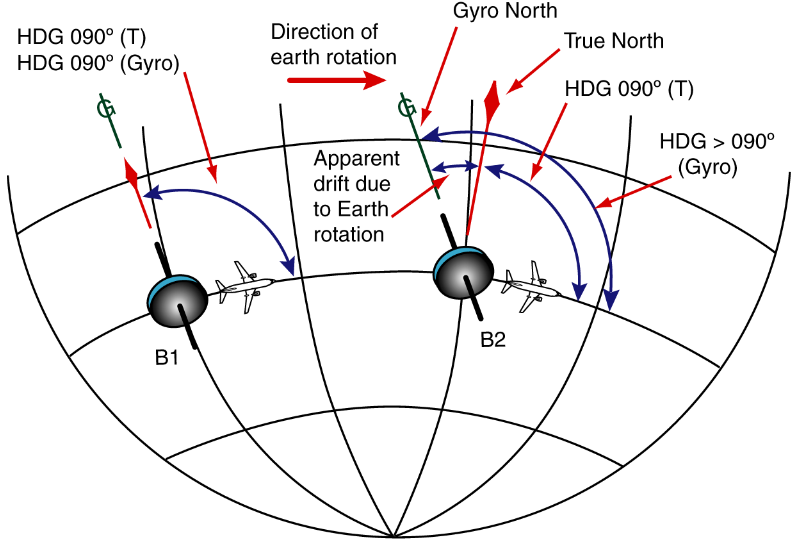
FIGURE 14-3
Apparent Drift due to Earth Rotation - Northern Hemisphere



14. Figure 14-4 shows the same situation but in the southern hemisphere. Now at time B_2 the gyro spin axis, defining gyro north, lies to the left of the meridian defining true north. **The DGI is consequently over-reading, and this is caused by apparent drift due to Earth rotation in the southern hemisphere.**

The Directional Gyro Indicator

FIGURE 14-4
Apparent Drift due to Earth Rotation - Southern Hemisphere



15. The rate of apparent drift due to Earth rotation is given by the formula:

Apparent drift = 15° x the sin of the latitude °/hour

16. Unfortunately the problem of apparent drift is further complicated by the fact that the aircraft will change its position on the Earth during flight. This gives an apparent error known as **transport wander.**

17. Flight on a north-south track will affect the rate at which the DGI readings become erroneous only inasmuch as the **latitude** in the formula for apparent drift due to Earth rotation will be changing. In theoretical calculations this problem is overcome by using **mean latitude** in the formula for Earth rotation.

18. Flight on an easterly track will complement the Earth's rotation (from west to east), and therefore the DGI will be in error by an amount which is greater than 15° sin lat °/hr.

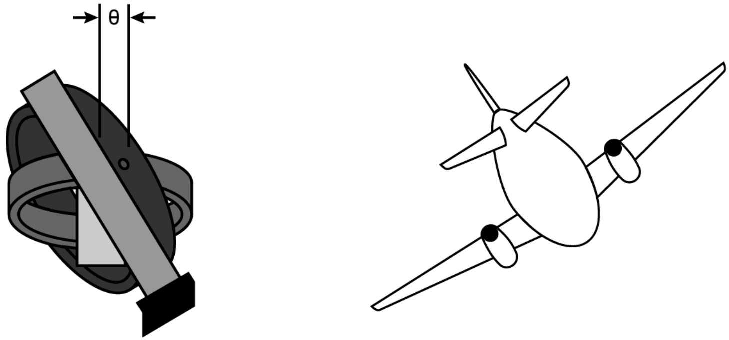
19. Flight on a westerly track will oppose the Earth's rotation, and therefore the DGI will be in error by an amount which is smaller than 15° sin lat °/hr.

20. The mathematics of apparent drift are deliberately vague at this stage, since for now it is the **logic** which is important.

Gimbal Error

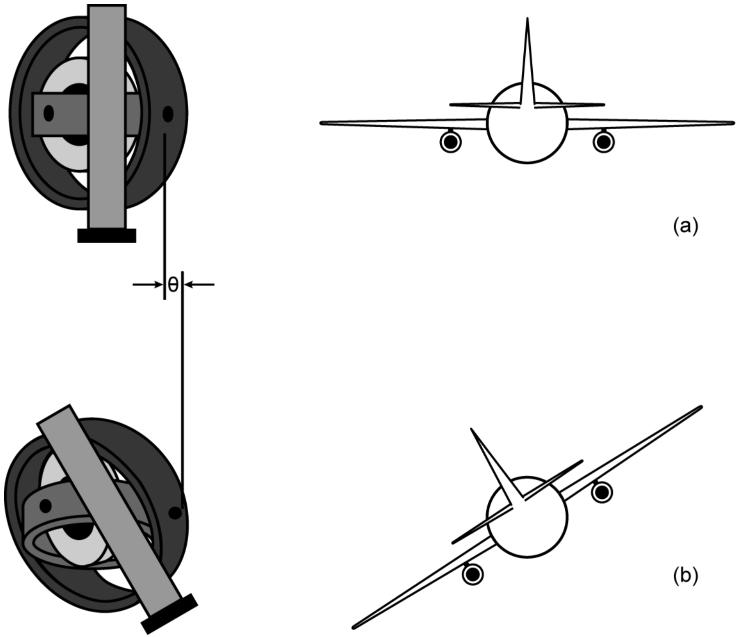
21. A DGI will also suffer from small and transient inaccuracies of the indicated heading due to **gimbal errors** which occur in a gyro whenever the gyro as a whole is displaced such that its inner and outer gimbals are not at right angles to each other. This error is particularly relevant to gyros, such as the DGI, where the spin axis is primarily horizontal.

FIGURE 14-5
Gimbal Error - Descending Banked Turn



22. Gimbal error in a DGI is not easy to visualise. Basically, when an aircraft carries out a manoeuvre which involves changes in both pitch and roll attitudes, for example the descending banked turn shown at Figure 14-5, the outer gimbal is forced to rotate by a small amount about its own axis, in order to maintain the spin axis of the gyro in the aircraft horizontal (yawing) plane. The amount of this rotation ($\theta°$ in Figure 14-5) will be translated into a DGI indicated heading error, for the duration of the manoeuvre. This error will be apparent during turns through all points of the compass.

FIGURE 14-6
Gimbal Error - Non Cardinal Heading



23. Figure 14-6(a) shows an aircraft on a non-cardinal heading and therefore the spin axis of the gyro is neither in line with the aircraft's fore and aft axis, or at right angles to it. Now any turn or change of pitch attitude (or a combination of both as previously described) will result **in a heading readout error, for example an error of θ°** at Figure 14-6(b) which results from a level turn.

24. Appreciate that gimballing errors result in relatively small and transient errors in the DGI reading.

The Latitude Nut

25. It is desirable that the DGI remains as accurate as possible with the passage of time. Since it is possible to calculate the magnitude and direction of apparent drift due to Earth rotation **for a given latitude or mean latitude**, it is also possible to correct for this error. This is done by precessing the gyro so that the spin axis moves in the yawing plane at the same rate, but in the opposite direction, to the apparent drift caused by Earth rotation. This correction is achieved using a device known as a latitude nut.

26. Refer back to Figure 14-1, and note that the latitude nut is located on the inner gimbal. With the latitude nut in its neutral position the gimbal is balanced. By winding out the latitude nut a vertical force is applied by gravity. This force is precessed through 90° into the horizontal plane and caused real drift. Conversely, winding the nut inwards beyond the neutral position will cause real drift in the opposite direction. It should be noted that the latitude nut cannot be adjusted by the pilot, it is set prior to the installation of the instrument into the aircraft.

Resetting the DGI

27. You will already appreciate that due to gimbal error a DGI will show a slight, short-term error during a turn, whereas a magnetic compass will suffer a similar short-term problem during turns and accelerations, due to both inertia and the Z field effect (see later). However, over the long term a magnetic compass will prove to be a fairly accurate heading reference, whilst the combination of apparent drift and real drift in a DGI, if unchecked, will result in large heading errors.

28. Real drift rates associated with early versions of the DGI were in the order of ±16°/hr, later versions managed drift rates of ±12°/hr (electrically driven DGIs achieve considerably better drift rates). The need to reset the DGI regularly should therefore be obvious. Such resets, based on a reliable compass direction, should only be carried out with the aircraft in smooth air and straight and level flight.

DGI Drift Calculations

29. Should it be required to calculate the total drift of a DGI, a comparison of all the known errors should be made as described in the following paragraphs.

To revise:

$$\text{Total Drift} = \text{Real Drift (RD)} + \text{Apparent Drift (AD)}$$

or

$$TD = RD + AD$$

30. Apparent drift will comprise drift due to the Earth's rotation and drift due to the east/west component of the aircraft's track (normally known as **transport wander**), therefore:

Apparent Drift (AD) = Earth Rotation Induced Drift (ER)
+ Transport Wander Induced Drift (TW)

or

$$AD = ER + TW$$

31. Real drift will comprise drift due to mechanical imperfections (normally known as random wander) and intentionally induced real drift due to the latitude nut, therefore:

Real Drift = Drift Due To Random Wander (RW)
+ Drift Due To Latitude Nut (LN)

or

$$RD = RW + LN$$

32. Putting together what has gone before, and using the convenient shorthand introduced above, it should be obvious that:

$$TD = RD + AD$$

and therefore;

$$TD = RW + LN + ER + TW$$

33. Bear in mind that the addition signs in the above formulae imply algebraic addition.

34. As already established, apparent drift due to Earth rotation causes a DGI in the northern hemisphere to under-read, the sign of this error is by convention given as a negative.

35. To illustrate how little it is necessary to remember if the subject is fully understood, Figure 14-7 contains only the one piece of information contained in the previous paragraph. The blank spaces will shortly be filled in using logic rather than memory.

FIGURE 14-7
Determination of Sign Values for DGI Calculations

Line	Cause	Northern Hemisphere	Southern Hemisphere
1	Drift due ER	–	
2	Drift due LN		
3	Drift due TW Eastwards		
4	Drift due TW Westwards		

Now for the logic:

(a) If the apparent drift due ER in the northern hemisphere is negative, in the southern hemisphere it must be positive.

(b) The perfect latitude nut induces a real drift which is equal in magnitude but opposite in direction to the apparent drift due to Earth rotation at the pre-set latitude. The signs in line 2 of Figure 14-7 must therefore by reversed when compared with line 1.

(c) Drift due to transport wander with an easterly track will complement the drift due to Earth rotation since the aircraft is moving in the same direction as the Earth's direction of spin, therefore the signs in line 3 are the same as in line 1 of table 3.

(d) Conversely, drift due to transport wander with a westerly track will oppose the drift caused by Earth rotation, therefore the signs in line 4 are opposite to the signs in line 1 at Figure 14-7.

The completed table is shown at Figure 14-8.

FIGURE 14-8
Complete Table of Sign Values for DGI Calculations

Line	Cause	Northern Hemisphere	Southern Hemisphere
1	Drift due ER	-	+
2	Drift due LN	+	-
3	Drift due TW Eastwards	-	+
4	Drift due TW Westwards	+	-

Drift Due to Earth Rotation

36. As already established, the drift due to Earth rotation is calculated using the formula:

Drift due ER = 15° Sin Mean Latitude °/hr
(- In the Northern Hemisphere)
(+ In the Southern Hemisphere)

Drift Due to Latitude Nut

37. Logically, since the latitude nut compensates for drift due to Earth rotation, the formula is the same as for calculating drift due to Earth rotation, but the signs are reversed for the hemispheres:

Drift due LN	=	15° Sin Latitude °/hr
	(+	In the Northern Hemisphere)
	(-	In the Southern Hemisphere)

38. The latitude in the formula above is of course the latitude to which the nut is set.

Drift Due to Transport Wander

39. The drift caused by transport wander depends upon the change of longitude achieved by the aircraft in a given period. With a little thought it should be apparent that, for an aircraft tracking 090° or 270°(T), the drift due to transport wander will be a function of groundspeed and latitude. If the groundspeed is known it is convenient to use the formula:

$$\textbf{Drift Due Transport Wander} = \frac{\textbf{E/W Component of Groundspeed (kt)} \times \textbf{Tan Latitude}}{\textbf{60}} \ \textbf{°/hour}$$

For the appropriate sign see Figure 14-8.

EXAMPLE 14-1

EXAMPLE

An aircraft is fitted with a DGI containing a perfectly balanced gyro. The latitude nut is set to 54°S and the aircraft is tracking 270°(T) at a groundspeed of 640 kt along the parallel of latitude 54°S. Determine the magnitude and direction of the hourly rate of drift suffered by this DGI.

SOLUTION

$$TD = RD + AD$$

$$TD = RW + LN + ER + TW$$

$$RW = 0 \text{ (perfect gyro)}$$

LN is equal in magnitude but opposite in direction to ER, therefore they cancel.

Drift due to transport wander is the only drift affecting the gyro. Therefore:

$$TD = TW$$

$$TD = -\frac{\text{Groundspeed} \times \tan \text{lat}}{60}\ °/\text{hour}$$

$$= -\frac{640 \times \tan 54°}{60}$$

$$= -14.7°/\text{hour}$$

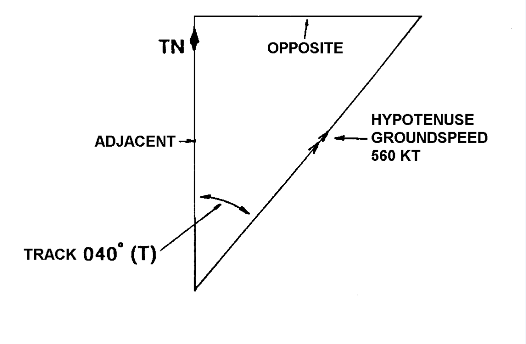
40. In the event that the aircraft is not tracking exactly 090° or 270°(T) it is necessary to isolate the component of the groundspeed which is in an east-west direction. Simple trigonometry is used to determine the east-west component as per Example 1-13.

EXAMPLE 14-2

EXAMPLE

An aircraft is making good a track of 040°(T) at a groundspeed of 560 kt. During one hour of flight the mid latitude was 72°N. Calculate the drift suffered by the DGI due solely to transport wander.

SOLUTION



$$\text{Sin } 40° = \frac{\text{opposite}}{\text{hypotenuse}}$$

$$= \frac{\text{easterly component of groundspeed}}{\text{groundspeed}}$$

$$\text{Sin } 40° \times \text{groundspeed} = \text{easterly component of g/speed}$$

$$0.643 \times 560 = \text{easterly component of g/speed}$$

$$360\text{kt} = \text{easterly component of g/speed}$$

The formula for drift due to transport wander is now used in a slightly modified form:

$$\text{Drift Due TW} = \frac{\text{EW Component of Groundspeed x Tan Mid Lat}}{60} °/\text{hour}$$

which in this case gives:

$$\text{Drift due TW} = \frac{-360 \times \tan 72}{60} °/\text{hour}$$

$$= -18.5°/\text{hr}$$

With some questions the aircraft's groundspeed may not be given, and in this case a different approach is required.

With drift due to Earth rotation the formula used was;

$$\text{Drift due ER} = 15° \text{ Sin Mid Lat } °/\text{ hr}$$

Logically then an alternative formula for drift due to transport wander is;

$$\text{Drift due TW} = \text{Change of Longitude } (°/\text{hr}) \times \text{Sin Mid Lat } °/\text{hr}$$

Drift Due to Random Wander

41. As the name suggests, it is impossible to evolve a formula for drift caused by random wander.

42. The mean drift rate due to random wander can only be calculated if the total drift is given, and the drifts due to Earth rotation, latitude nut and transport wander are calculated and algebraically subtracted from the total drift.

EXAMPLE 14-3

EXAMPLE

A DGI in an aircraft is corrected to give zero drift when the aircraft is stationary on the ground in latitude 45°S. Whilst flying west along the parallel of latitude 40°N at a groundspeed of 540 kt the total drift of the gyro is found to be -10.7°/hr. Calculate the magnitude and direction of the random wander affecting the gyro.

$$TD = RD + AD$$

$$TD = RW + LN + ER + TW$$

$$TD = -10.7°/hr$$

$$RW = \text{the unknown factor}$$

$$LN = -15 \sin 45°/hr = -10.6°/hr$$

$$ER = -15 \sin 40°/hr = -9.6°/hr$$

$$TW = +\frac{540 \tan 40°/hr}{60} = +7.6°/hr$$

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therefore:

$$TD = RW + LN + ER + TW$$

$$-10.7 = RW + (-10.6) + (-9.6) + (+7.6)$$

and;

$$-10.7 + 10.6 + 9.6 - 7.6 = RW$$

$$\text{Drift due to random wander} = +1.9°/\text{hr}$$

EXAMPLE 14-4

EXAMPLE

A DGI in an aircraft is corrected to give zero drift when the aircraft is stationary on the ground in latitude 51°N. Assuming the gyro to be frictionless and perfectly balanced, determine the drift rate that may be expected and state whether readings will increase or decrease when:

(a) The aircraft is stationary on the ground at latitude 42°N.

(b) The aircraft is tracking east along the parallel of 56°N at a groundspeed of 600 kt.

SOLUTION

$$TD = RD + AD$$

$$TD = RW + ER + TW$$

(a)	RW	=	0(perfect gyro)
	LN	=	$+15 \sin 51 = +11.7°/hr$
	ER	=	$-15 \sin 42 = -10.0°/hr$
	TW	=	0(aircraft stationary on the ground)
	TD	=	$+11.7 - 10.0°/hr$
	Total drift	=	$+1.7°/hr$ (readings increasing)
(b)	RW	=	0(perfect gyro)
	LN	=	$+15 \sin 51 = +11.7°/hr$
	ER	=	$-15 \sin 56 = -12.4°/hr$
	TW	=	$-\dfrac{600 \tan 56}{60} = -14.8°/hr$
	TD	=	$+11.7 - 12.4 - 14.8°/hour$
	Total drift	=	$-15.5°/hr$ (readings decreasing)

EXAMPLE 14-5

EXAMPLE

A DGI in an aircraft, on the ground in latitude 33°N, is found to have a drift rate of 5°/hr, the readings decreasing.
Assuming that any real drift present on the ground remains the same in flight, determine the hourly drift rate that may be expected from this gyro in latitude 68°N when flying on a track of 270°(T) at a groundspeed of 385 kt. State whether the readings would be increasing or decreasing.

SOLUTION

$$TD = RD + AD$$

$$TD = RW + LN + ER + TW$$

It is neither possible (since the latitude nut setting is not specified in the question) nor necessary to isolate the two component parts of the real drift.

At 33°N on the ground:

TD	=	–5°/hr
RW	=	the unknown factor
ER	=	– 15 sin 33 = – 8.2°/hr
TW	=	0

therefore;

$$-5 = RD - 8.2$$

$$RD = +3.2°/hr$$

Tracking 270° at 68°N:

$$TD = \text{the unknown factor}$$

$$RD = +3.2°/hr \text{ (from the previous working)}$$

$$ER = -15 \sin 68 = -13.9°/hr$$

$$TW = \frac{+385 \tan 68}{60} = +15.9°/hr$$

therefore;

$$TD = +3.2 - 13.9 + 15.9$$

$$\text{Total drift} = +5.2°/hr \text{ (readings increasing)}$$

The preceding notes and examples are totally adequate and should enable the reader to answer any question concerning gyro drift that may be asked in the examination. One last piece of advice, read the question carefully and answer the question set by the examiner rather than the question that you think he should have set. This advice is not intended to be flippant, too many marks are lost in examinations because the candidate misreads the question.

The Slaved Gyro Compass

The Slaved Gyro Compass

(Also referred to as: The Heading Reference Unit, The Remote Indicating Compass or, The Gyromagnetic Compass)

1. The direct-reading compass (DRC) which forms the primary heading reference in most light aircraft is relegated to the role of a standby system in larger aircraft. Direct reading compasses suffer from significant turning and acceleration errors and therefore have serious limitations. Furthermore, by virtue of its design, the sensing elements of a DRC (the magnets) must necessarily be housed on the flight deck, which is an area rich in deviating materials and components.

2. The directional gyro indicator (DGI) goes some way to solving the problems discussed above. If the DGI is set with reference to the DRC when the aircraft is in straight and level flight at a constant airspeed, and if sensible use is made of the DRC deviation card, then it is reasonable to suppose that the DGI will read the correct value of magnetic heading. Unfortunately the DGI will subsequently drift because of both **real** and **apparent** errors, and so the problem of producing an accurate heading reference, whilst reduced, is still evident.

3. The slaved gyro compass (otherwise known as the heading reference unit, the gyro magnetic compass or remote-reading compass) essentially solves the problem by automatically and continuously comparing the output of a magnetic sensing element with the indicated heading of the gyro indicator, and by resetting the gyro whenever a discrepancy exists. The gyro output is therefore slaved to magnetic north.

4. The pilot is no longer required to reset the gyro indicator periodically, or putting it another way, significant errors of indicated heading do not occur if he fails to do so.

The Slaved Gyro Compass

5. The major components of a slaved gyro compass system are shown at Figure 15-1.

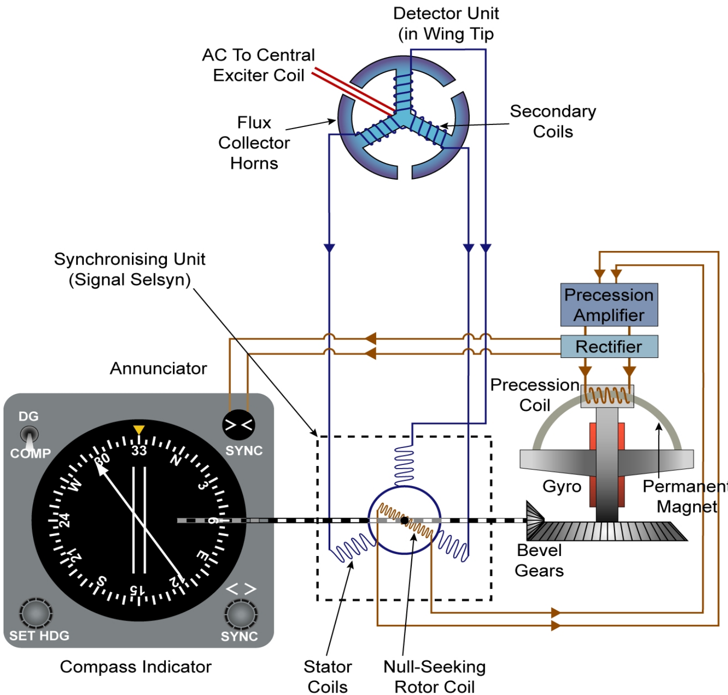

FIGURE 15-1
Slaved Gyro Compass System - Block Schematic

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The Detector Unit

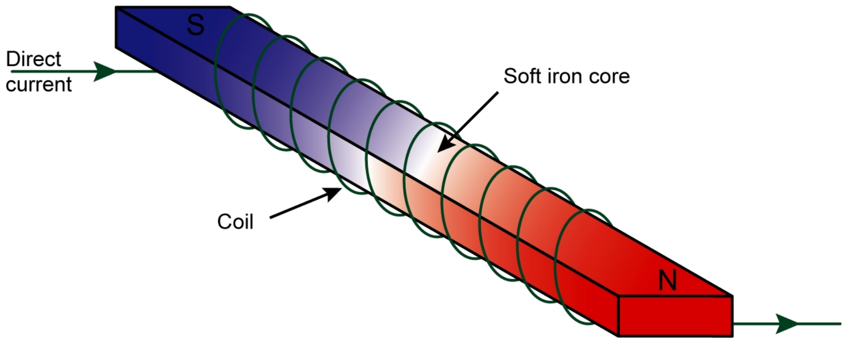
6. The element which senses the direction of magnetic north, the detector unit, is normally mounted in a wing tip or at the top of the fin, in an area where the deviating influence of the aircraft is at an absolute minimum.

7. The output of the detector is a series of electrical currents which **represent** magnetic heading in a manner which will be discussed shortly. The detector unit itself suffers from turning and acceleration errors, however the electrical output currents can be interrupted whenever the aircraft accelerates or turns. When this happens the gyro unit will function as a pure DGI for the duration of the manoeuvre. Once the aircraft returns to constant velocity flight the gyro heading is automatically updated with reference to the detector output.

8. The heart of any detector unit is the **flux valve**. The principle of operation of such a valve is now discussed.

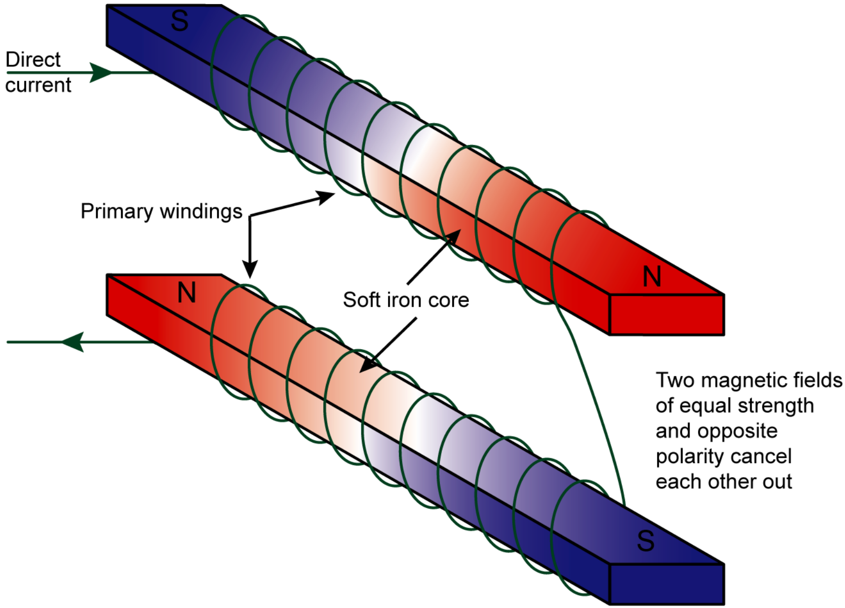
9. If a direct current is passed through a coil wound around a soft iron core the core will become magnetized as shown at Figure 15-2.

FIGURE 15-2
Simple Electromagnet



10. If the soft iron core is now split at the middle and the two halves are laid side by side without disturbing the coil, two magnets of equal strength and opposite polarity are produced as shown at Figure 15-3.

FIGURE 15-3
Self-Cancelling Electromagnet



11. Were the value of the current passed through the coil to be steadily increased a stage would be reached when the soft iron cores would become **saturated.** This is to say that any further increase in coil current would not result in a corresponding increase in the strength of the magnetic fields produced.

12. In fact, in a flux valve it is an alternating current rather than a direct current which is fed to the primary windings shown at Figure 15-3.

13. The effect of the alternating current is to completely reverse the magnetic polarity of both soft iron cores each time that the direction of current flow changes.

14. The peak value of the alternating current fed to the primary coil is just sufficient to saturate the soft iron cores. In other words, ignoring any external magnetic influences, the cores would just saturate at the 90° and 270° phase points of the primary winding alternating current flow, see Figure 15-4.

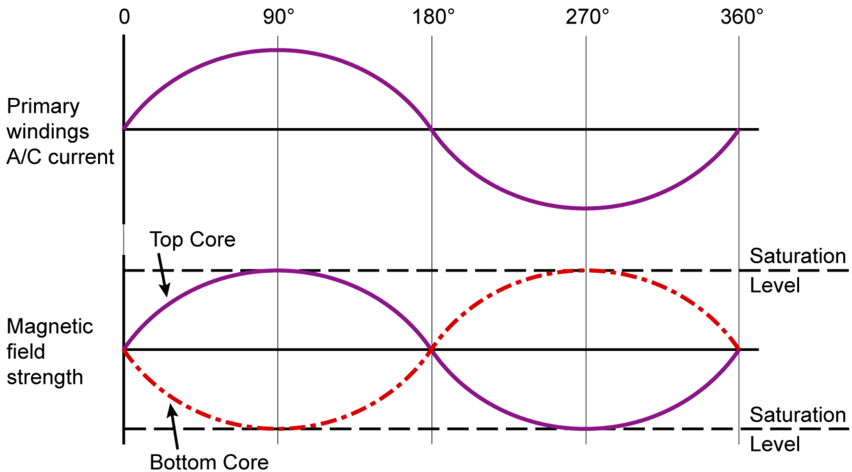
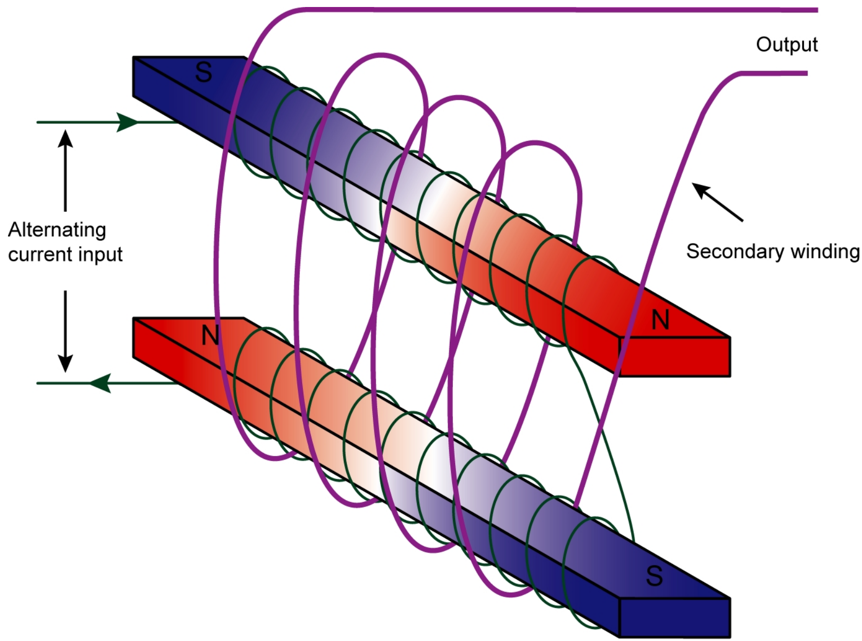


FIGURE 15-4 Alternating Current and Associated Magnetic Flux Density

15. Thus far we have established that an electric current may be used to produce a magnetic field. Of course the reverse is also true. If a magnetic field, of changing field strength and/or polarity, cuts a conducting element an alternating electric current will be induced to flow through the conductor.

16. A secondary coil is wound around the flux valve as shown at Figure 15-5.

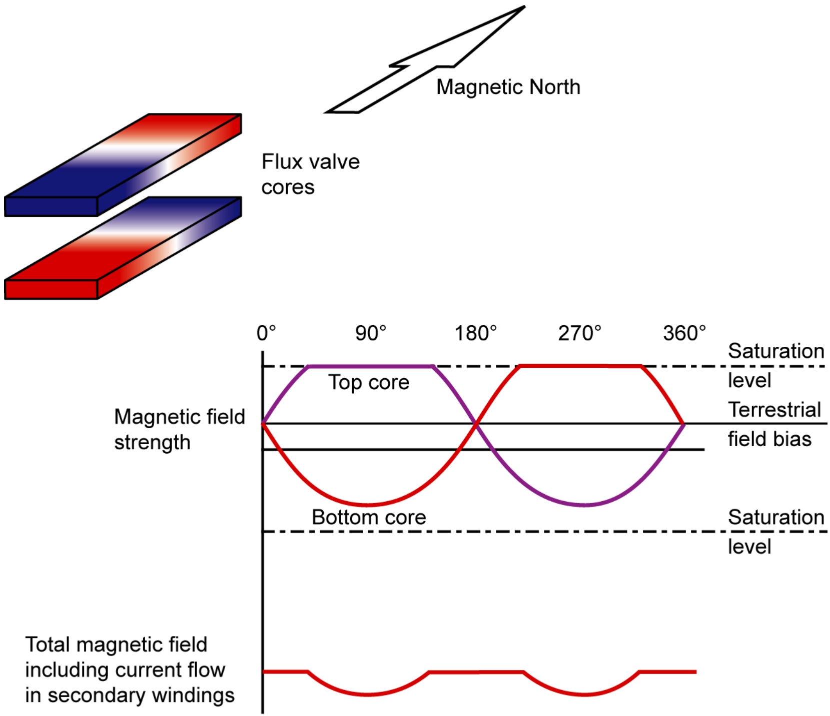
FIGURE 15-5
Simple Flux Valve



17. If the flux valve were to be placed in a totally screened container such that there were no external influences acting upon it, there would be no current induced into the secondary coil. This is because the two soft iron cores are producing equal but opposite magnetic fields which effectively cancel each other.

18. When the flux valve is placed within the terrestrial magnetic field the equality of the magnetic fields produced by the soft iron cores is disturbed, and consequently a current will be induced to flow in the secondary (output) winding, as shown at Figure 15-6.

FIGURE 15-6
Effect of Terrestrial Magnetism on Flux Valve Operation



19. Comparing Figure 15-4 and Figure 15-6 should give some idea of what is happening. At Figure 15-4 no external magnetic influence was evident at the flux valve, and consequently both cores **just** achieved saturation twice during each 360° phase cycle of the alternating current fed to the primary coil. At Figure 15-6 the Earth's own magnetic field is acting upon the flux valve and its effect is to apply a **magnetic bias** to the system. The result is that one core will become totally saturated at the 90° phase point and the other at the 270° phase point. Differing magnetic field strengths now result in a current being induced to flow in the secondary winding.

20. It should be said at this point that the explanation given above does not represent the complete picture, however the syllabus does not require a study of the **hysteresis characteristics** of the soft iron cores. Without such a study it is necessary to accept that the explanation offered above is factually correct, if incomplete.

21. Were the flux valve to be turned through 180° the bias effect of the terrestrial magnetic field would be reversed, as would be the flow of current induced into the secondary coil.

22. Were the flux valve to be placed at 90° to the Earth's own magnetic field there would be no current induced into the secondary coil.

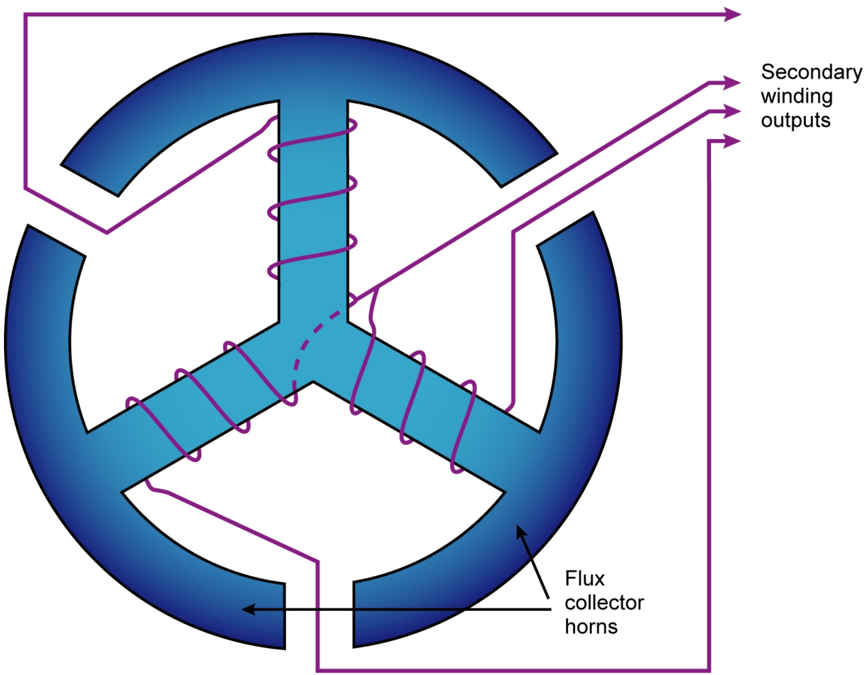
23. A detector unit employs three flux valves, positioned 120° apart, as shown at Figure 15-7. Appreciate that the detector unit is fixed in azimuth with respect to the aircraft. In other words, if the aircraft turns through 90°, so does the detector unit. Therefore the orientation of the detector unit to the Earth's magnetic field (and the currents generated within the secondary windings) vary with aircraft heading.

24. At Figure 15-7 only the secondary windings are shown. The **flux collector horns** are simply extensions of the soft iron cores and are employed to concentrate the terrestrial magnetic field.

25. The whole detector unit is required to lie in the Earth's horizontal plane, so that it is the H component of the terrestrial magnetic field which is sensed rather than the Z component. In order that the detector can remain horizontal when the aircraft is pitching or rolling the unit is suspended by a universal joint knows as a **Hooke's Joint.** This arrangement allows, typically, 25° of freedom in pitch and roll.

26. When the freedom of movement limits imposed by the Hooke's joint are exceeded the electrical outputs of the flux valve are isolated from the gyro unit. Since the detector unit in this condition is no longer in the horizontal plane an element of the Earth's Z component would necessarily be sensed at the detector and the resultant turning/acceleration errors would cause an eventual misalignment of the gyro.

FIGURE 15-7
Detector Unit/
Flux Valve System



27. Even if the freedom of movement limits of the Hooke's joint were not exceeded during a turning or acceleration manoeuvre the detector unit would still depart from the Earth's horizontal plane as its own mass reacted under the effect of inertia. Again a part of the Earth's Z component would be sensed and the gyro would eventually become misaligned. In order to minimise this error the gyro unit is precessed to align itself to the detector output at a slow rate (only 2° per minute typically) during normal operation. During any manoeuvre of limited duration the heading indicated by the gyro magnetic compass will not therefore become significantly in error.

28. The next step is to consider the technique by which the gyro is maintained in alignment with the detector unit output.

The Transmission System

29. The signals generated in the detector must somehow be transmitted to the gyro unit in order to keep the gyro slaved to (or synchronised with) magnetic north as determined by the detector. This is achieved within the synchronising unit by means of a self synchronous control unit (or **selsyn**).

30. The selsyn is effectively a detector unit in reverse. The electrical currents which are produced in the secondary windings of the flux valves within the detector unit, and which synthesise the Earth's magnetic field direction, are fed to the selsyn, as shown at Figure 15-1.

31. The **stator coils** within the synchronising unit are positioned mutually at 120° to each other, as are the flux valves in the detector. The flux valve currents flowing through these stator coils will produce a magnetic field which represents the Earth's magnetic field sensed at the detector. The **null seeking rotor coil** of the selsyn will have a current induced into it whenever it lies at any angle other than 90° to the magnetic lines of flux produced by the stator coils.

The Slaving System

32. The gyro unit itself is mechanically coupled to the null seeking rotor within the selsyn, as is the compass rose of the heading indicator, see Figure 15-1.

33. In the event that the gyro is misaligned with magnetic north as defined by the detector unit, then necessarily the null seeking rotor must be at other than 90° to the magnetic field manufactured within the selsyn, and consequently an error signal will be produced by the rotor. This error signal is amplified, rectified and fed to the **precession coil** on the gyro itself, again as shown at Figure 15-1. The precession coil now generates its own magnetic field which acts upon the semi-circular shaped permanent magnet which is attached to the inner gimbal of the gyro. This exerts a downward force on the inner gimbal, the consequence of which is that the gyro will precess in the aircraft's yawing plane and, because of the mechanical linkage, both the null seeking rotor and the compass rose on the face of the instrument will also rotate. This will continue until the error signal at the null seeking rotor ceases, at which time the rotor will again lie at 90° to the selsyn's magnetic field and the gyro will now be aligned with magnetic north as sensed at the detector unit.

34. An alternative to the precession coil and permanent magnet arrangement described above is a torque motor, which is more commonly found in modern systems.

Annunciator Indicators

35. It is obviously desirable that the pilot should know when an error current is flowing from the null seeking rotor, since at such times the heading indication will be erroneous. The annunciator circuit is located between the precession amplifier and the precession coil. An indicator shows when precession currents are flowing and therefore indicates that the compass rose is not aligned with magnetic north as sensed by the detector unit. Two types of annunciator indicator are shown at Figure 15-8.

36. It has already been mentioned that a typical rate for the gyro to automatically align with the detector unit output is 2° per minute. Remember that this is deliberately kept to a low value so that the system will not become grossly misaligned whenever the detector output is affected by the Z component during turning or acceleration manoeuvres.

37. If the operator notices a gross misalignment, as may well occur when the equipment is first switched on, then the system can be rapidly re-aligned using the **manual synchronisation control.** Figure 15-9 shows the indicator of a G4F compass, the manual resetting control is positioned at bottom right and the annunciator indicator (dot/cross type) is positioned at top right. The set heading knob is used to position the heading 'bug', and the DG/COMP switch gives the operator the option of operating in the pure DGI mode whenever the output of the detector unit is suspect, following a lightning strike, or at very high magnetic latitudes for example.

FIGURE 15-8
Annunciator Indications

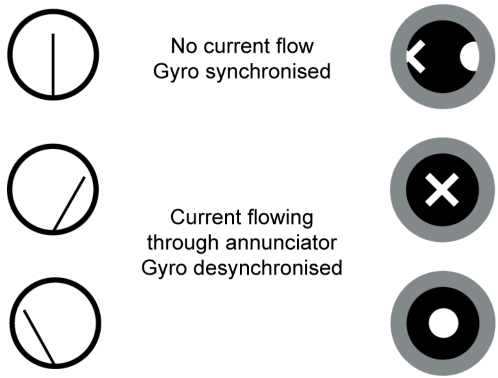
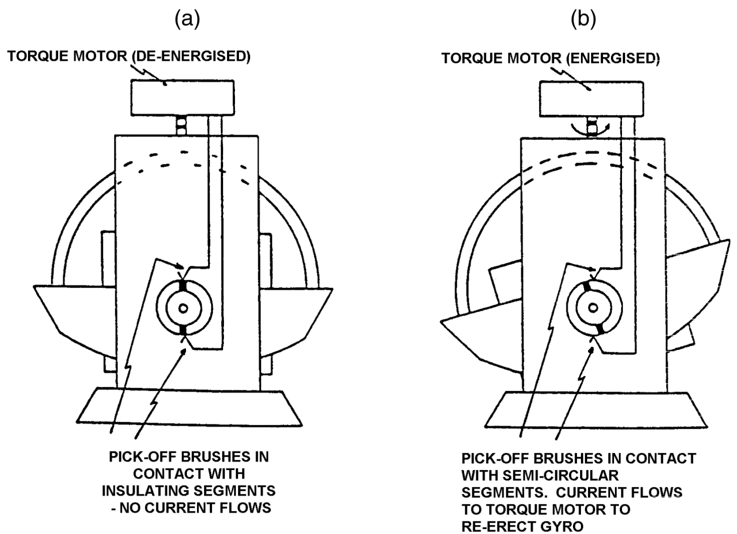


FIGURE 15-9
G4F Compass Indicator

The Gyro Self Levelling System

38. Finally it is necessary to consider the technique employed to maintain the gyro spin axis in the yawing plane of the aircraft. Figure 15-10 shows schematically how this is achieved.

FIGURE 15-10
Gyro Self Levelling System



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39. The commutator switch consists of two semi-circular contacts insulated from each other and attached to the pivot joining the inner and outer gimbals. With the gyro spin axis lying in the yawing plane [Figure 15-10(a)] the pick-off brushes are in contact with the insulating segments and no current flows to the torque motor. With the gyro toppled [Figure 15-10(b)] the commutator contacts have rotated under the pick-off brushes and an electrical current flows to the torque motor. The precessing force from the torque motor is applied around the aircraft's normal (vertical) axis and the resultant movement of the spin axis is back towards the yawing plane. When the spin axis again lies in the correct plane the torque motor is de-energised as the pick-off brushes contact the insulating segments.

Remote-Reading Compass System Errors

40. The remote reading compass system suffers from the following errors:

Fluxvalve Tilt Errors

41. Any horizontal accelerations which cause fluxvalve tilt can cause heading errors in a simple uncompensated remote-reading compass system. Accelerations are caused by coriolis, vehicle movement (rhumb line), aircraft turns, linear changes of velocity and flux valve vibrations. Fluxvalve induced heading errors will not manifest themselves immediately and the rate of any heading error introduction depends on the limiting precession rate and the response time of the system (time constant).

(a) **Turning Error.** Although a high rate of turn in a fast aircraft would show the greatest flux valve 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. Flux valve 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.

(b) **Coriolis Error.** An aircraft flies a curved path in space and in consequence there will be a central force acting to displace the pendulously suspended flux valve. The error is calculable, depending on groundspeed, latitude, dip and track, and can be compensated automatically.

(c) **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. A correction can be applied in a similar manner to the coriolis error.

(d) **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

42. 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. 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 pattern is repeated and the **amplitude can be as great as 6°**. The amplitude of the weave tends to **increase with an increase in dip and aircraft velocity.**

Hang-Off Error

43. 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 any given time there must be an increment of error present. This is known as hang-off error. 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 °/hr. The correction can be supplied through a manually set latitude correction mechanism or through a constantly biased gyro.

(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:

$$\frac{G/S}{60} \tan lat \text{ °/hr} (\text{where G/S = east-west groundspeed})$$

44. 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

45. 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 magnitude 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

46. Overall system accuracy is lowered by the errors in the synchro systems. Typically each synchro might be expected to have an error in the order of 0.1° with an overall system error of perhaps 0.5°. This shows in a compass swing as a Coefficient D or E error (not part of this syllabus).

Compass Swinging Errors

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

Variation and Deviation Errors

48. Charted values of variation may be considered to vary between 0.1° and 2°. Over the UK the uncertainty at height is considered to be within 1° but the value varies both with height and locality. Setting of variation and deviation is likely to be accurate to 0.25°.

Advantages of the Slaved Gyro Compass

49. Some of the advantages of this type of system over a direct reading system have already been mentioned, the main advantages are summarised below:

(a) The detector unit is located in an area which is low in aircraft magnetism.

(b) The compass rose is mechanically driven, giving a stable compass heading, notably in turbulence.

(c) The compass rose is flat faced and located centrally on the flight instrument panel, removing the parallax error which occurs with direct reading compasses.

(d) The electro/mechanical output of the compass system can be used to feed other instruments and systems, for example RMIs or a Flight Management Computer.

(e) The compass rose can be electrically or mechanically corrected with variation to enable true headings to be flown.

(f) Turning and acceleration errors are much reduced.

(g) The system can be operated as a DGI in high latitudes or in the vicinity of thunderstorms, where magnetic compasses are unreliable.

(h) The detector unit senses rather than seeks the magnetic meridian, giving increased sensitivity.

Pre-flight Checks

50. When power is available to the system, operate the synchronising control in the direction indicated by the annunciator. Now check the heading readouts of the primary display and the RMI against the other gyro slaved compass system (if two are fitted), the direct reading compass and the known heading of the aircraft (with reference to the stand centre line markings and the aerodrome manoeuvring area chart. Be aware that if the aircraft is on a stand and is surrounded by ground power units, fuel bowsers and so on, the readings of each of the gyro compasses and the direct reading compass may disagree. Whilst the aircraft is taxiing, compare the compass readouts against each other, especially during the turns. If any doubt still exists, a final check can be conducted with the aircraft aligned with the runway centre line.

51. In an aeroplane with two slaved gyro compasses, it is normal for the captain's primary display (horizontal situation indicator, HSI) and the first officer's RMI to be driven by one gyro compass. The first officer's HSI and the captain's RMI are naturally driven by the second gyro compass. It is normal to include a compass comparator, which will warn the pilot's whenever a discrepancy (of, for example, 5° or more exists for 5 seconds or more) is sensed between one compass system and the other. Finally, it is normally possible for the pilot's to drive all of the HSIs and RMIs from a single gyro compass in the event that the other system fails.

Self Assessed Exercise No. 2

QUESTIONS:

QUESTION 1.

What is the name given to a line joining all points on the Earth's surface where the angle of dip is zero.

QUESTION 2.

The theoretical maximum value of variation is:

QUESTION 3.

Between what latitudes are Direct Reading Compasses generally considered to be useable.

QUESTION 4.

What are the three desirable properties of a magnet system in a Direct Reading Compass.

QUESTION 5.

What are the 2 causes of Turning and Acceleration Errors in a DRC.

QUESTION 6.

In a direct reading compass the maximum errors will occur when turning through N or S/E or W at high/low latitudes

QUESTION 7.

An aircraft accelerates on an easterly heading in the northern hemisphere. The compass magnet assembly will rotate clockwise/anti-clockwise and the compass will underread/overread:

QUESTION 8.

When accelerating on a westerly heading in the northern hemisphere, the magnet assembly of a direct-reading magnetic compass, when viewed from above, will turn clockwise/anti-clockwise, indicating an apparent turn towards north/south.

QUESTION 9.

An aircraft lands on a Southerly runway in the Northern Hemisphere, which way will the magnet system in a DRC rotate on landing.

QUESTION 10.

An aircraft takes off on a Westerly runway in the Southern Hemisphere. Which way will the magnet system of a DRC rotate on take off, and will this cause the compass to underread or overread.

QUESTION 11.

An aircraft in the northern hemisphere makes a rate 1 turn from 135° through south. After 30 seconds the aircraft DRC will read more than/less than/exactly 225°.

QUESTION 12.

An aircraft fitted with a direct-reading compass is turning from 315° through north on to 045° in the Southern Hemisphere. The direction of turn of the magnet system and the effect of liquid swirl on the error due to the turn are: -

QUESTION 13.

Deviation due to Horizontal Hard Iron increases with increase/decrease of magnetic latitude because the Z/H component of earth magnetism is increasing and the Z/H component is decreasing:

QUESTION 14.

The deviating effect of vertical soft iron increases/decreases with decrease of magnetic latitude, due to the increase/decrease of H and the increase/decrease of Z.

QUESTION 15.

The rigidity (gyroscopic inertia) of a gyroscope may be increased by __________ the speed of rotation and __________ the mass of the rotor.

QUESTION 16.

The rate of precession of a gyroscope varies directly/inversely with (a) APPLIED FORCE, (b) ROTOR SPEED, and (c) ROTOR MASS

QUESTION 17.

Increasing the speed of rotation of a gyroscope ____________ its gyroscopic inertia and __________ its rate of precession.

QUESTION 18.

A torque is applied to the gyroscope at FIGURE 239 in the Reference Book. The Gyro will precess clockwise/anti-clockwise about axis ________.

QUESTION 19.

Topple is said to occur in a gyroscope whenever the spin axis moves in a __________ plane.

QUESTION 20.

Name three causes of real wander in a gyroscope.

QUESTION 21.

What is the formula for calculating Apparent Drift due to Earth Rotation in a gyroscope.

QUESTION 22.

A gyroscope which utilises gravity to control a vertical reference is a termed a:

QUESTION 23.

A gyroscope which is sensitive to movement about one axis only is called:

QUESTION 24.

Name the two categories of air driven gyroscopes.

QUESTION 25.

How can 'Frequency lock/ Beam lock' be overcome in a Ring Laser Gyro:

QUESTION 26.

The latitude nut of a directional indicator (DI) is on the ………. gimbal and causes the gyroscope to precess around its ………. axis.

QUESTION 27.

In a directional indicator (DI) the random wander is -4°/hr. The latitude nut is adjusted for latitude 50°S and the aircraft is flying west along the parallel of 30°N at a groundspeed of 240kt. The total hourly rate of change of indication of the DI is:

QUESTION 28.

A directional gyro in an aircraft is corrected to give zero drift when the aircraft is stationary on the ground at latitude 48°N. Assuming the gyro to be free from random wander:

The gyro is set to read 100°. The gyro reading after 45mins when stationary on the ground at latitude 25°S is:

QUESTION 29.

A directional gyro in an aircraft is corrected to give zero drift when the aircraft is stationary on the ground at latitude 48°N. Assuming the gyro to be free from random wander:

The gyro is set to read 090°. The gyro reading after flying for 80 mins, on a track of 090°T along the parallel of 20°N, at a groundspeed of 540kts is:

QUESTION 30.

If a simple fluxvalve lies in line with the H component of the earth's magnetic field, the current induced in the pick-off coil will be maximum/minimum.

QUESTION 31.

The purpose of a fluxvalve detector unit is:

QUESTION 32.

The purpose of the annunciator circuit in a remote indicating compass is:

QUESTION 33.

In FIGURE 240 in the Reference Book of a gyromagnetic Compass, the components labelled B, H and D are:

QUESTION 34.

What are the two main advantages of a Gyromagnetic Compass over a Direct Reading Compass:

QUESTION 35.

What flight parameters would produce the greatest turning errors in a gyromagnetic compass:

QUESTION 36.

Northern instability is a heading oscillation experienced by high speed aircraft attempting to fly straight and level at or near a heading of Magnetic North. What are the two main factors that govern the amplitude of the oscillation.

QUESTION 37.

Name the type of suspension system that is used to secure the 3 fluxvalves inside the detector unit of a gyromagnetic compass.

QUESTION 38.

Typically, what are the limits of freedom, and in what axes, of the fluxvalve system inside the detector unit of a gyromagnetic compass:

QUESTION 39.

What is a typical precession rate for a gyro in a gyromagnetic compass system.

QUESTION 40.

Fluxvalve tilt errors do not manifest themselves as heading errors immediately. What factors govern the rate of heading error introduction.

ANSWERS:

ANSWER 1.

Aclinal line.

CH9 P9-2 Para 6

ANSWER 2.

180°

CH9 Fig 9-1

ANSWER 3.

70°N to 70°S

CH9 P9-2 Para 8

ANSWER 4.

Horizontality, Sensitivity, Aperiodicity.

CH10 P10-1 Para 2

ANSWER 5.

Inertia Effect and Z Field Effect

CH10 P10-6 Para 20

ANSWER 6.

N/S high latitudes

CH10 P10-6/8

ANSWER 7.

clockwise underread

CH10 P10-7 Fig 10-6 and Para 24

ANSWER 8.

turns anticlockwise, indicating an apparent turn towards north

CH10

ANSWER 9.

There is no acceleration/deceleration error when travelling in a Southerly direction.

CH10 P10-9 Para 28

ANSWER 10.

The magnet system will rotate clockwise causing an underread in the Compass heading.

CH10

ANSWER 11.

More than 225° (i.e. overread)

CH10

ANSWER 12.

anti-clockwise reduces the turning error.

CH10

ANSWER 13.

increase Z H

CH10 P11-5 Para 20

ANSWER 14.

decreases increase decrease

CH10 P11-5 Para 21

ANSWER 15.

The rigidity of a gyroscope may be increased by INCREASING the speed of rotation and INCREASING the mass of the rotor.

CH12 P12-2 Para 4

ANSWER 16.

The rate of precession of a gyroscope varies DIRECTLY with APPLIED FORCE, and INVERSELY with ROTOR SPEED and ROTOR MASS.

CH12 P12-3 Para 4

ANSWER 17.

Increasing the speed of a gyroscope INCREASES its gyroscopic inertia and DECREASES its rate of precession.

CH12 P12-3 Para 4

ANSWER 18.

Clockwise \ Y-Y

CH12 P12-3 Fig 12-3

ANSWER 19.

Topple occurs whenever the spin axis of a gyroscope moves in a vertical plane

CH12 P12-3 Para 5

ANSWER 20.

Friction in the bearings.

Imbalance in the gimbals.

Imbalance in the rotor.

CH12 P12-4 Para 7

ANSWER 21.

Apparent Drift due to Earth Rotation = 15 x sine latitude °/hr.

CH12 P12-5 Para 11

ANSWER 22.

earth gyro

CH12 P12-7 Para 14

ANSWER 23.

a rate gyro

CH12 P12-7 Para 14

ANSWER 24.

Air driven gyros are either vacuum or pressure driven.

CH12 P12-7 Para 16

ANSWER 25.

Frequency/Beam lock can be overcome in a Ring Laser Gyro by using an external vibration device to introduce a 'dither' into the unit.

CH13 P13-3 Para 10

ANSWER 26.

inner vertical

CH14 P14-1 Para 3, and Fig 14-1

ANSWER 27.

20.7°/hr readings decreasing

$$\text{Total Drift} = (-4) + (-15 \text{ Sin } 30) + (-15 \text{ Sin } 50) + \left(+\frac{240 \tan 30}{60}\right)$$

$= -4 - 7.5 - 11.5 + 2.3$

$= -20.7°/\text{hr}$

CH14 P14-12 EX 14-4

ANSWER 28.

113°

TD = (+15 Sin 25) + (+15 Sin 48)

$= +6.35 + 11.15$

$= +17.5°/\text{hr}$ therefore, 45mins = +13.12°

CH14 P14-12 EX 14-4

ANSWER 29.

093.5°

$$TD = (-15 \sin 20) + (+15 \sin 48) + \left(-\frac{540 \tan 20}{60}\right)$$

$= -5.1 + 11.1 - 3.3$

$= +2.7$ °/hr therefore, 80 mins = + 3.6°

CH14 P14-12 EX 14-4

ANSWER 30.

If a fluxvalve lies in line with the H component of the earth's magnetic field, the current induced in the pick-off coil will be at a maximum value.

CH15 P15-6 paragraph 19 to paragraph 22

ANSWER 31.

A fluxvalve detector unit transmits an electrical signal proportional to the aircrafts magnetic heading.

CH15 P15-7 paragraph 23

ANSWER 32.

The annunciator circuit has two functions - it indicates the correct way to turn the synchronising knob to initially synchronise the compass system and, thereafter, it indicates the correct operation of the system.

CH15 P15-9 paragraph 35 to paragraph 37

ANSWER 33.

B - Signal selsyn

H - Rotor

D - Amplifier

CH15 P15-2 Figure 15-1

ANSWER 34.

The two main advantages of a Gyromagnetic Compass over a DRC are that it reduces deviation, and it also reduces turning and acceleration errors.

CH15 P15-1 paragraph 1 + paragraph 2

ANSWER 35.

Slow prolonged turns at high speed.

CH15 P15-12 paragraph 41

ANSWER 36.

Angle of dip and aircraft velocity.

CH15 P15-13 paragraph 42

ANSWER 37.

Hookes joint

CH15 P15-7 paragraph 25

ANSWER 38.

plus or minus 25° in pitch and roll.

CH15 P15-7 paragraph 25

ANSWER 39.

2° per minute

CH15 P15-8 paragraph 27

ANSWER 40.

The limiting precession rate and the response time of the system.

CH15 P15-11 paragraph 41

The Artificial Horizon

The Artificial Horizon

(Also referred to as the Classic Artificial Horizon)

1. The artificial horizon, otherwise known as the **attitude indicator**, provides the pilot with information in terms of the aircraft attitude both in **pitch** and **roll**. It should be appreciated that the artificial horizon does not necessarily show climb or descent, or a turn, as is often thought. An aircraft with bank applied does not have to be turning, and a swept-wing transport aeroplane may well descend on an ILS glidepath at 700 feet per minute with the nose 10° or more above the horizon. Instruments which show ascent/descent are altimeters and vertical speed indicators. A rate-of-turn indicator and a compass show that the aeroplane is turning. The artificial horizon shows only attitude and is used in conjunction with other instruments to confirm a flight path.

2. Obviously in a modern flight director system, the artificial horizon is somewhat different in appearance to the instruments considered here, however the basic instrument inside is the same, and we are not concerned with the command bars and other ancillaries. In order to meet airworthiness requirements, public transport aeroplanes are necessarily equipped with a standby attitude indicator (normally with its own battery pack as a reserve power source), and this instrument would be similar to that discussed below.

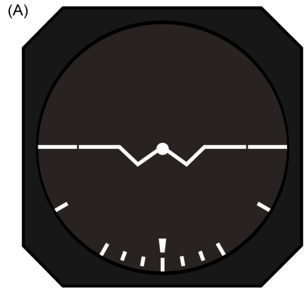
3. The illustrations at Figure 16-1 show the face of a basic artificial horizon. Notice that the aircraft symbol is fixed (usually to the inside of the glass face) and that the horizon bar moves in pitch and roll behind the symbol. The type shown indicates angle of bank at the bottom of the instrument, at increments of 10°, 20°, 30°, 60°, and 90°. Pitch attitude is not shown in degrees, and must be learned and interpreted from experience with the instrument and the particular type of aircraft to which it is fitted.

4. The illustrations at Figure 16-2 show the face of a more modern artificial horizon. Now the angle of bank is displayed at the top of the instrument, and in addition pitch attitude in degrees is displayed.

5. A superior presentation is achieved with the use of colour, one colour above the horizon line (the 'sky' portion) and another below the horizon line (the 'earth' portion). Many combinations are used, such as, (top and bottom respectively), grey/black, light blue/black, light blue/brown.

The Artificial Horizon

FIGURE 16-1
Simple Artificial Horizon Presentation



(A) Aircraft nose level with horizon and wings level
(B) Aircraft nose low with 35° right bank
(C) Aircraft nose high with 45° left bank

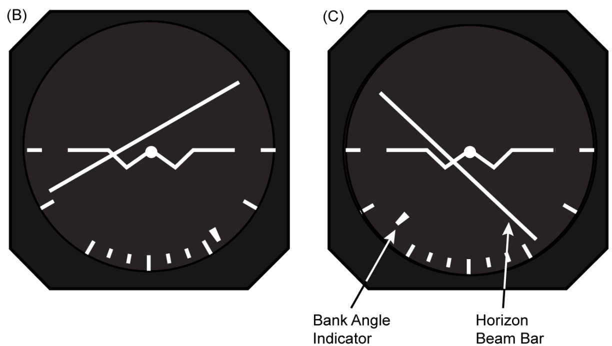
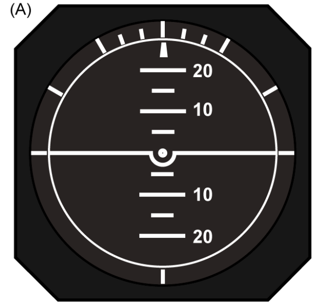
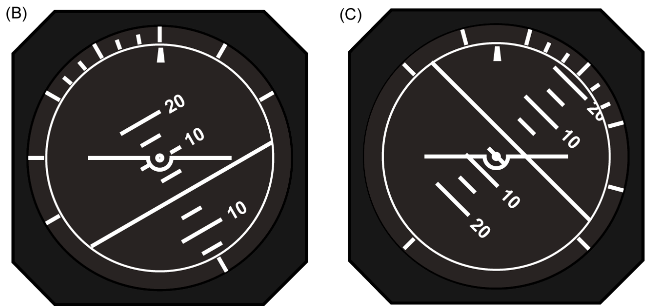


FIGURE 16-2
Modern Artificial Horizon Presentation



(A) Aircraft nose level with horizon and wings level
(B) Aircraft 10°nose high with 30° right bank
(C) Aircraft 5° nose low with 45° left bank



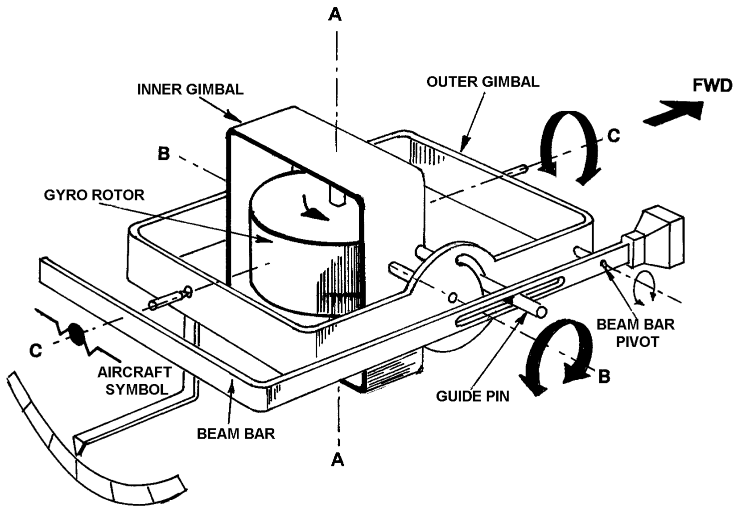
Principle of Operation

6. The artificial horizon employs an **Earth gyro**, the spin axis of which is maintained Earth vertical under the influence of gravity, regardless of whether the instrument is air driven or electrically driven.

7. Appreciate that the **beam bar** (the horizon line on the front of the instrument) is controlled, through suitable linkages, by the gyro gimbals. The aircraft symbol is attached to the instrument casing and therefore to the actual aircraft.

8. The basic construction of an artificial horizon is illustrated at Figure 16-3. The illustration could equally apply to an air driven or electrically driven instrument, the principle is the same. Note, however, that the direction of spin is shown as being anti-clockwise when viewed from above. This is always the case with air driven instruments, whereas it is normal for electrically driven artificial horizon gyros to rotate in the opposite direction.

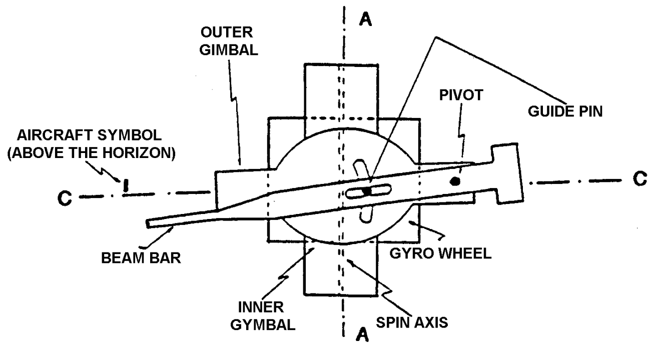
FIGURE 16-3
Artificial Horizon - Basic Construction



9. The three axes of the gyro are shown as A-A, B-B and C-C at Figure 16-3. Appreciate that axis A-A (the spin axis) will remain Earth vertical, and axis B-B will therefore remain Earth horizontal.

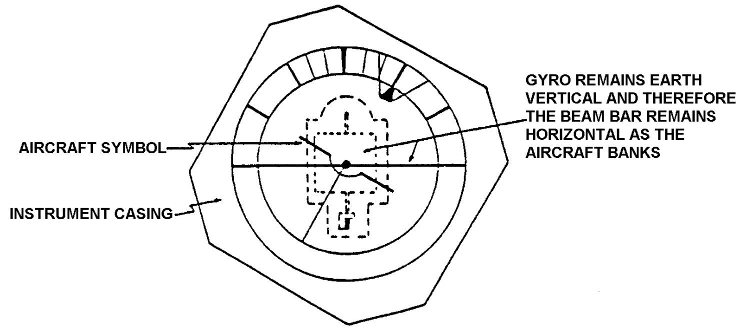
10. Should the aircraft adopt a **pitch up** attitude (nose high), the outer gimbal would be forced out of Earth horizontal, since it is attached to the instrument casing. This movement of the outer gimbal is amplified by the beam bar which now appears to the pilot to be below the aircraft symbol at the face of the instrument. This situation is illustrated at Figure 16-4.

FIGURE 16-4
Effect of Pitch Up on Beam Bar



11. Whenever the aircraft rolls, the aircraft itself, and consequently the instrument casing and the aircraft symbol, rotate about the axis C-C. The gyro spin axis remains vertical. This is illustrated at Figure 16-5.

FIGURE 16-5
Effect of Roll on Presentation



The Air Driven Artificial Horizon

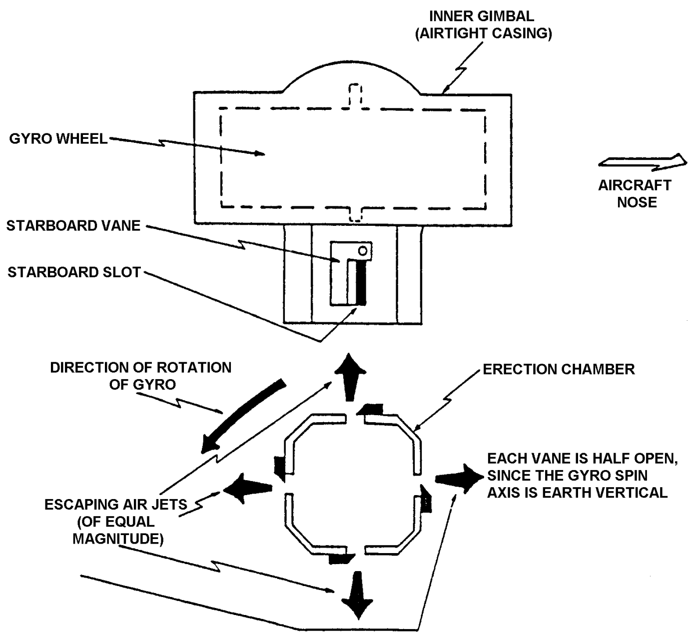
12. With the air driven instrument the air which has been used to spin the gyro is exhausted through an **erection chamber** at the base of the inner gimbal. The inner gimbal forms an otherwise airtight seal around the gyro wheel. The air is exhausted through four slots in the erection chamber, two in the aircraft fore and aft axis (one forward and one aft) and two in the athwartships axis (one port and one starboard). When the gyro is Earth horizontal and in unaccelerated flight these slots are half covered by **four pendulous vanes.**

13. The fore and aft vanes are attached to a common spindle, such that if one vane moves to close its associated slot entirely, the other will have moved to open its associated slot.

14. Likewise the athwartships vanes are attached to a common spindle such that as one slot is opened the opposite slot is closed.

15. The arrangement of slots and pendulous vanes is illustrated at Figure 16-6.

FIGURE 16-6
Erection Chamber / Pendulous Vanes in Air Driven Artificial Horizon

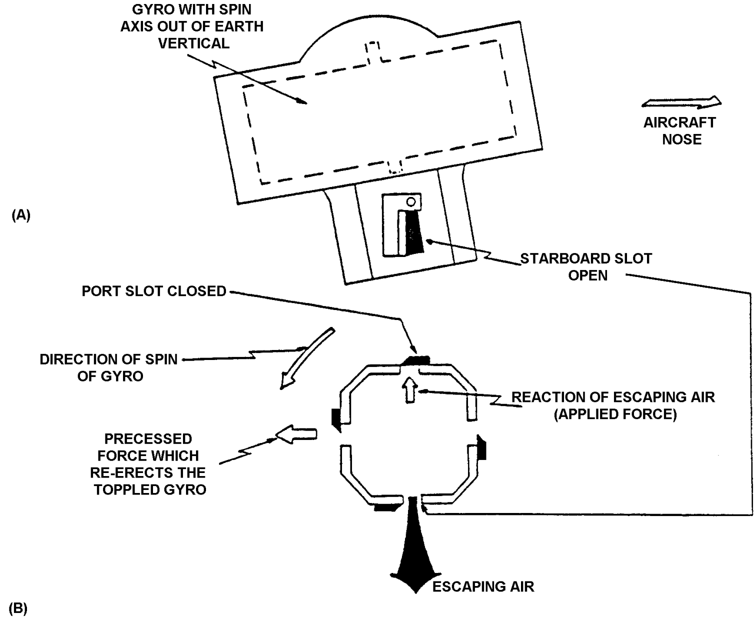


16. The distinguishing feature of an **Earth gyro** is that it is constrained by gravity to remain rigid with respect to one of the terrestrial axes. In the case of the air driven artificial horizon it is the effect of gravity on the **pendulous vanes** which retains the spin axis in the Earth vertical axis.

17. Figure 16-7(a) shows the gyro of an artificial horizon where the spin axis has departed from the vertical. The starboard pendulous vane which is visible at Figure 16-7(a) is open, and consequently the port vane will be closed. The fore and aft vanes will be unaffected by the topple illustrated in this case.

18. Figure 16-7(b) shows the erection chamber and the vanes, viewed from above. Think of the air escaping from the starboard slot as jet eflux, causing an applied reactive force to act on the gyro in the opposite direction. This applied force is precessed through 90° in the direction of spin and re-erects the toppled gyro. The same principle will apply, whatever the direction of topple.

FIGURE 16-7
Effect of Topple on Pendulous Vanes

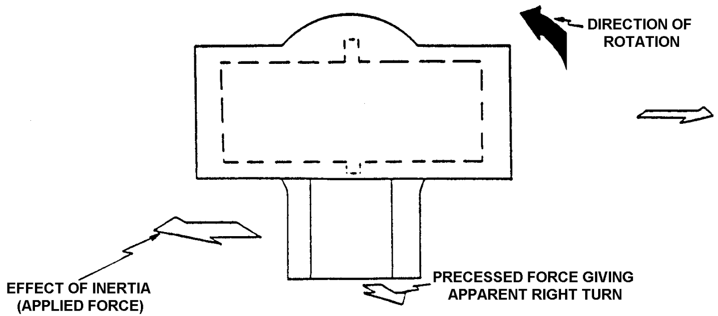


Acceleration Error

19. With an air driven artificial horizon, the instrument is likely to show an **erroneous** indication of a **climbing right turn** as the aircraft accelerates on a constant heading in level flight or along a level runway. This false indication is due to acceleration error.

20. The **apparent right turn** indicated during aircraft acceleration is caused by the effect of **inertia** upon the **erection chamber.** The chamber has a significant mass and is located at the bottom of the spin axis. As the aircraft accelerates the erection chamber wants to remain at rest or to maintain its state of uniform motion, giving the applied force shown at Figure 16-8. This force is precessed through 90° in the direction of rotation, causing the bottom of the spin axis to move towards the starboard wing, and showing an apparent right turn at the face of the instrument.

FIGURE 16-8
Effect of Inertia on Erection Chamber



21. The **apparent climb** (**apparent pitch up**) indicated during aircraft acceleration is caused by the effect of **inertia** on the **athwartships pendulous vanes.** As the aircraft accelerates the athwartships vanes lag, and this causes the starboard vane to open and the port vane to close. The applied force is now towards the port wing, but it is precessed to act towards the tail of the aircraft. This causes the bottom of the spin axis to move backwards, lowering the horizon bar and therefore indicating an apparent climb (nose high attitude).

Turning Errors

22. An acceleration error may be described as a change in the vector force acting on the instrument. During an acceleration the vector force is changing in magnitude but not in direction. During a turn, the vector force is changing in direction, and may or may not change in magnitude, depending on whether or not the TAS remains constant in the turn. The centrifugal force resulting from the change of direction during the turn will try to displace the mass of the erection chamber and will also try to move the vanes in respect to their associated slots. The effect on the indications will vary as the turn progresses, and the explanation of why this is so is complex. It is sufficient to appreciate that, with modern instruments, the errors will be insignificant at the low rates of turn associated with instrument flying.

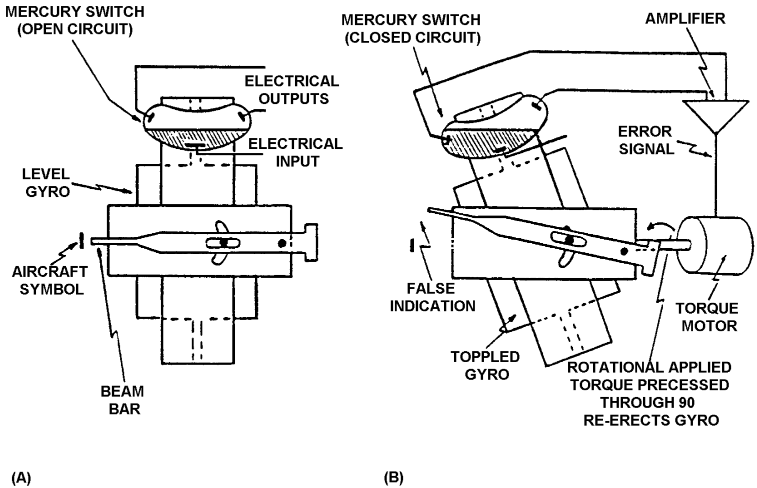
The Electrically Driven Artificial Horizon

23. The advantages of electrically-driven gyros over air driven gyros have already been discussed. The gyro in the artificial horizon needs to be as **rigid** as possible, and consequently the higher spin speed of the electrically driven instrument is significant.

24. The basic principle of the instrument is the same, regardless of the driving force, however the electrically driven artificial horizon employs different techniques to maintain the gyro spin axis Earth vertical using the influence of gravity.

25. **Mercury switches and torque motors** replace the erection chamber and pendulous vanes in the electrically driven instrument. The technique involved is illustrated at Figure 16-9.

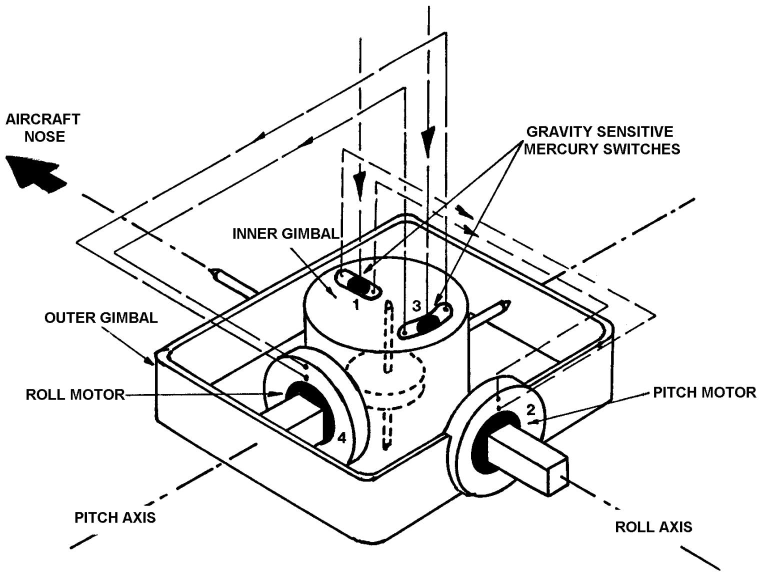
FIGURE 16-9
Electrically Driven Artificial Horizon - Erection System



26. At Figure 16-9(a) the spin axis is vertical and the mercury switch is not therefore making an electrical circuit to the torque motor. At Figure 16-9(b) the spin axis has toppled out of the vertical, the electrical circuit is completed through the mercury switch and the torque motor is energised. The force applied by the torque motor is precessed through 90° and re-erects the gyro.

27. A three dimensional diagram of an electrically driven artificial horizon is shown at Figure 16-10.

FIGURE 16-10
Electrically Driven Artificial Horizon Showing Pitch and Roll Motors



NOTE:

The mercury switch which senses topple of the spin axis in the fore and aft plane (1) is located on the inner gimbal. The torque motor associated with this mercury switch (2) is attached to the outer gimbal and applies a torque about the roll axis, which is then precessed to act about the pitch axis.

NOTE:

The mercury switch which senses topple of the spin axis in the athwartships plane (3) is again located on the inner gimbal. The torque motor associated with this mercury switch (4) applies a force about the pitch axis, which is then precessed to act about the roll axis.

Advantages of the Electrically Driven Artificial Horizon

28. The advantages of the electrically driven artificial horizon over the air driven instrument are summarised below:

(a) A faster gyro rotation gives more rigidity and therefore turning and acceleration errors take longer to take effect.

(b) Cut-out switches (mercury type inertia switches) are attached to the airframe and ensure that current flow to the torque motors is interrupted during accelerations. This is necessary since the mercury switches which control the gyro are affected by accelerations in the same way as the vanes in an air driven artificial horizon. Typically, the cut-out switches operate at 0.18 G acceleration. In any event the electrically driven gyro suffers from smaller acceleration errors than its air driven cousin, since the electrically driven gyro has no heavy erection chamber at the bottom of the instrument.

(c) It is normal to incorporate a **fast erect button** with an electrically driven artificial horizon. When this button is depressed the magnitude of the currents flowing into the two torque motors is increased significantly, and so the gyro erects rapidly. It is obviously not a good idea to use the fast erect system unless the aircraft is in straight and level flight at a constant airspeed. If the fast erect button is depressed whilst the aircraft is manoeuvring, the effect of inertia/centrifugal force on the mercury switches will cause the instrument to erect to a **false horizon**. The fast erection button increases the erection rate from the normal 5°/minute to between 120 and 180°/minute. However, because of the increased current this facility must not be used for long periods, otherwise overheating of the torque motor may occur. An operating limitation of 15 or 30 seconds is normally imposed on the continuous use of the fast erect system.

(d) It is possible to take the horizontal reference from an electrically driven artificial horizon as an output to other systems, notably the autopilot, the flight director and the weather radar aerial stabilising system. The outputs from both primary artificial horizons (the captain's and the first officer's, but not the standby instrument) can also be fed into a comparitor, which compares one with the other and alerts the pilots in the event that there is a discrepancy.

Variable Pitch Datum

29. On some artificial horizons it is possible to raise or lower the position of the miniature aircraft in the instrument window, so that it can be adjusted to lie along the horizon bar in level flight. The danger is that, should the adjusting knob be inadvertently turned in flight, the instrument will subsequently give erroneous indications of pitch attitude, and this could be confusing to the pilot in actual IMC conditions.

30. The CAA strongly recommends that in light aircraft the datum be set before flight and thereafter left well alone. The CAA further require that such movable datums be removed or otherwise rendered inoperative on aircraft having a maximum all-up weight in excess of 6,000 pounds.

Pitch and Roll Limits

31. Regardless of whether the instrument is air driven or electrically driven, artificial horizons fitted to civilian transport aircraft would typically enjoy freedom in roll through 360° and freedom in pitch through ± 75 to 85°.

The Remote Vertical Gyro

The Remote Vertical Gyro

1. The artificial horizons previously discussed have shared a common feature, namely that, in each case, the gyro has been an integral part of the indicator. The size of the gyro has therefore been limited because of the available space on the flight instrument panel. In modern aircraft, principally those incorporating a flight director into the attitude indicator, the gyro unit, **the remote vertical gyro,** is located remotely from the indicator itself, normally in the electronic equipment bay behind or beneath the flight deck. The main advantage of this arrangement is that more space is available for larger and better gyros with improved erection systems.

2. The remote vertical gyro consists essentially of an electrically driven gyro spinning about a vertical axis. The gyro has full freedom in roll and ±85° of freedom in the pitching plane. The gyro's advanced erection system ensures that the rotor is maintained within 1° of the vertical even during manoeuvres or severe turbulence. Electrical pick-offs, fixed to the gimbals, measure the displacement of the aircraft around the rotor and supply their outputs to synchros within the attitude indicator on the flight instrument panel. Reduction of turning and acceleration errors is made possible through rate switching gyros and longitudinal accelerometers which interrupt erection currents when the aircraft is subject to false gravity vectors.

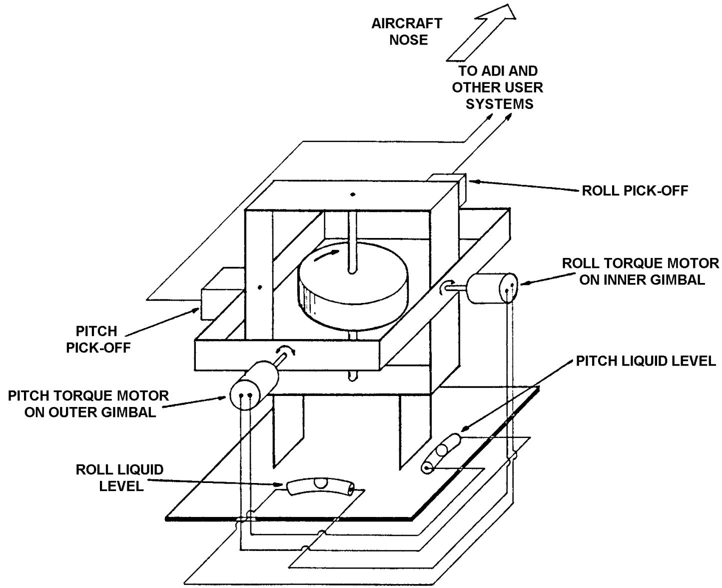
3. The remote vertical gyro is schematically illustrated at Figure 17-1. Although not shown in the diagram for the sake of clarity, the outer gimbal is suspended in an airtight case by bearings attached to the pitch torque motor and the roll pick-off retaining frames.

4. The erection system is similar to the system used in the basic electrically driven artificial horizon and consists of two torque motors and two levels (similar to spirit levels), one for pitch and one for roll. These levels contain a special electrolytic fluid. When the gyro axis is vertical, the bubbles in the levels are both centred and the electrolytic resistance in each end of the tubes is equal. Therefore no currents will flow to the torque motors. Should the axis of the rotor stray from the vertical, the bubble in one or both of the levels will be displaced, causing an unequal electrolytic resistance within one or both of the levels. As a consequence of this, currents will flow to one or both of the torque motors, as appropriate to the displacement of the gyro spin axis. The mechanical output of the torque motor is applied to the relevant gimbal, and the resultant precession will erect the gyro, centralise the bubble and cause the current flow to cease.

5. The gyro erection circuits provide fast erection for the first few minutes after power is applied to the system, and then at a slower rate to correct for any small amount of drift that may be produced by friction or Earth rotation.

6. Three remote vertical gyros are normally installed to provide for system redundancy, one each for the captain's and first officer's attitude indicators and an identical auxiliary gyro as backup for either of the primary systems.

FIGURE 17-1
Remote Vertical Gyro



7. This type of installation will normally include an attitude comparator, to warn the pilots of any discrepancy between the vertical gyro outputs. With a three gyro installation, the comparator will normally identify which of the three gyros is in error.

8. In addition to the attitude indicators, the remote vertical gyros are also used to provide pitch and roll displacement signals to the autoflight systems, flight director computers and weather radar antenna stabilising systems.

Turn and Balance Indicators

Turn and Balance Indicators

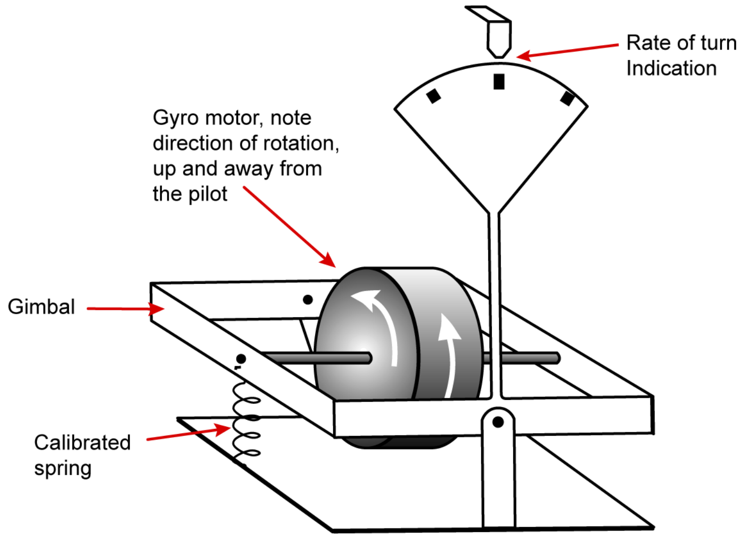
1. Two types of turn and balance indicators are considered, the traditional turn and slip (or turn and balance) indicator, and the turn coordinator.

2. The turn and slip indicator is in fact two instruments in one. The **turn indicator** shows the rate of turn of the aircraft, and utilises the properties of a **rate gyro**. The **slip indicator** enables the pilot to fly the aircraft in balance, keeping the ball in the middle, and employs a simple pendulum device.

The Turn Indicator

3. The turn indicator, more correctly known as a rate of turn indicator, employs a rate gyro which is normally electrically driven, but may also be air driven. Being a rate gyro, there is only one gimbal and consequently the gyro wheel has freedom of movement about only two axes, one of which is the spin axis. There is freedom of movement about the longitudinal (fore and aft) axis, although this movement is restricted by a **calibrated spring.** The gyro has no freedom of movement about the aircraft vertical axis, and so any torque applied about this axis, due to **yaw**, will be precessed and cause the gyro to topple against the restraining force of the spring. The construction of a turn indicator is shown schematically at Figure 18-1.

FIGURE 18-1
The Turn Indicator



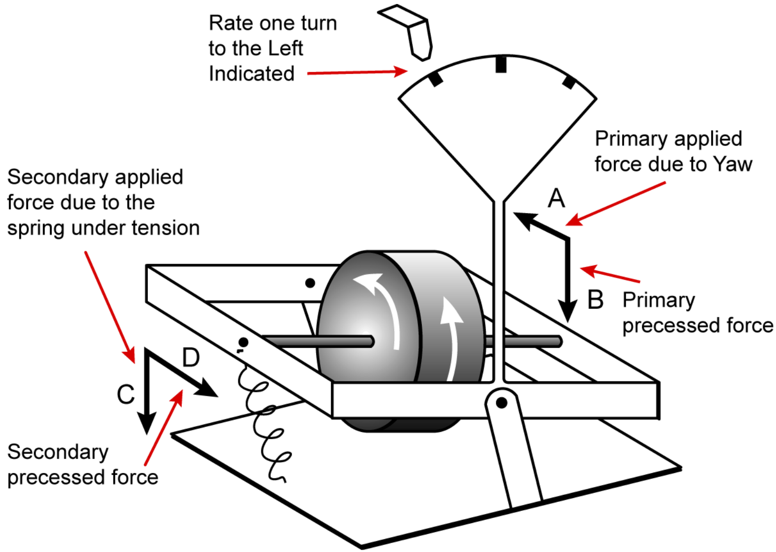
4. The gyro is mounted horizontally with the spin axis athwartships. At Figure 18-1 there is no yawing force present, and consequently the spin axis of the gyro is constrained by the spring to remain horizontal with respect to the aircraft.

5. At Figure 18-2 the aircraft is turning to the left and the yawing moment of this turn constitutes the **primary applied force A** in the diagram. This force is precessed through 90°, giving the **primary precessed force B,** which causes the gyro to topple. The amount of gyro tilt determines the rate of turn which is indicated at the instrument face.

6. As the gyro topples the spring stretches, giving the **secondary applied force C** in the diagram. This force is again precessed through 90°, giving the **secondary precessed force D.**

7. When the secondary precessed force D is equal in magnitude to the primary applied force A the secondary precession caused by the spring is proportional to the actual rate of turn. The gyro now ceases to topple, and the angle between the gyro spin axis and the aircraft yawing plane is proportional to the rate of turn.

FIGURE 18-2
The Turn Indicator - Aircraft Turning Left



8. Note that the direction of spin of the gyro is up and away from the pilot. This has the desirable effect of keeping the spin axis more or less Earth horizontal during a turn.

Turn Indicator Errors

9. The rate of turn indicated will only be correct when the aircraft is flying at the **true airspeed** for which the instrument has been calibrated. Providing that the actual TAS is within 100 knots of the calibrated TAS the error in indicated rate of turn should not exceed plus or minus five percent of the actual rate of turn.

10. During low rate turns there is little or no change of **pitch** required to maintain level flight. During steeper turns, however, it is necessary to raise the nose to maintain height during the turn. Any pitching moment which occurs whilst the spin axis of the turn indicator gyro is displaced from the aircraft yawing plane will result in additional precession, causing the instrument to **over-indicate the rate of turn**. Since the turn indicator is not normally used to monitor steep turns, this error does not seriously restrict the effectiveness of the instrument.

The Effect of Variations in Gyro Rotor Speed

11. If the rotor speed is below the design value the instrument will under-read the actual rate of turn. This is because the gyro is less rigid and therefore precesses more rapidly. The amount of secondary torque required to produce the secondary precession that matches the aircraft's rate of turn is therefore less than it should be, is achieved with less spring stretch and consequently less tilt of the spin axis. Less tilt means that a lower rate of turn is indicated.

12. Conversely, with the gyro overspeeding, the balance of forces is only achieved when the spring is stretched by more than the correct amount, because the gyro is too rigid. Therefore the indicated rate of turn is greater than it should be.

The Effect of a Weakened Spring

13. If the spring is weakened with age, in order to provide the balancing force it will need to be stretched further. This will result in a greater tilt of the spin axis, causing the instrument to **over-indicate the rate of turn.**

Bank Angle and Rate of Turn

14. The angle of bank required to achieve a given rate of turn increases as the aircraft's TAS is increased. As a rule of thumb, the angle of bank required for a rate one turn (3° per second) is calculated using the following formula:

$$\frac{\text{TAS(kt)}}{10} + 7^\circ = \text{Bank Angle}$$

15. The turn indicator is basically designed to show a rate one turn (3°/second), however you should appreciate that a rate two turn is 6°/second, a rate three turn is 9°/second, a rate four turn is 12°/second and so on.

Radius of Turn

16. The radius of a turn depends on two factors, the rate of turn and the TAS. The approximate radius of a turn (expressed in nautical miles) can be determined using the following formula:

$$\text{Radius of Turn (nm)} = \frac{\text{TAS (Kt)}}{\text{Rate of Turn (°/min)}}$$

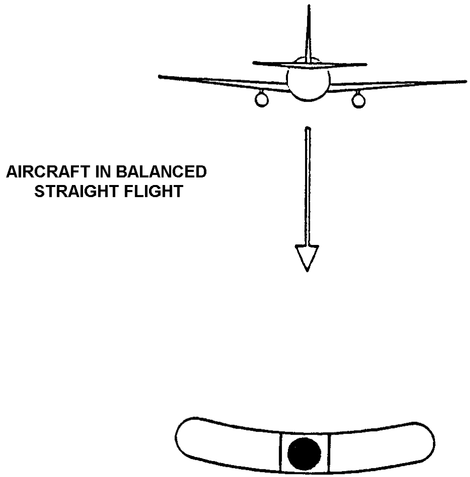
17. The radius of a rate one turn flown at a TAS of 450 kt would therefore be approximately 2½ nm.

The Slip Indicator

18. The slip (or balance) indicator works on a simple pendulum principle. In modern instruments, the system normally employed is a relatively heavy solid ball within a curved tube which contains a clear liquid to damp out unwanted oscillations. The ball is effectively the end of a pendulum, and the curvature of the tube defines the radius of arc of the pendulum arm.

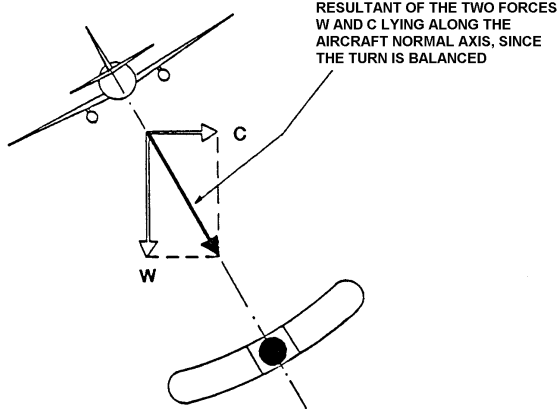
19. Consider first an aircraft in balanced straight and level flight. The only force acting on the ball is **weight** (W), that is to say mass acted on by acceleration due to gravity, and so the ball rests at the lowest point of the curved tube, as shown at Figure 18-3.

FIGURE 18-3
Aircraft in Straight and Level Flight



20. Now consider an aircraft in a balanced turn to the left. There are now two forces acting on the ball, **weight** (W), and **centrifugal force** (C). The magnitude of the centrifugal force depends on the TAS and rate of turn of the aircraft. For a turn to be balanced (no slip or skid) the resolution of the two forces must lie in the **aircraft vertical** axis, as shown at Figure 18-4.

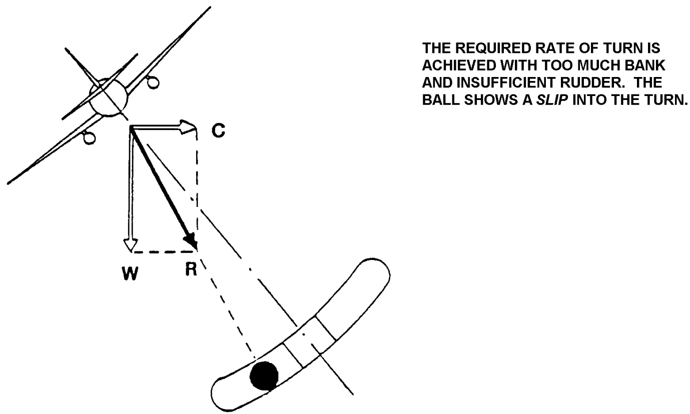
FIGURE 18-4
Aircraft in Balanced Turn to the Left



21. Finally consider an aircraft which is in an unbalanced turn to the left. Assuming that the TAS and rate of turn are as before, the weight (W) and the centrifugal force (C) will be of the same magnitude as for the balanced turn. The conclusion is that the ball shown at Figure 18-5 (slip) and at Figure 18-6 (skid) is in the same place as the ball shown at Figure 18-5 (balanced turn). **It is the tube which has changed position relative to the ball.**

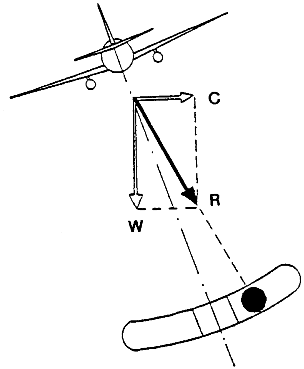
22. At Figure 18-5 the aircraft is overbanked for the TAS and required rate of turn, with too little rudder applied in the direction of turn. This results in the aircraft **slipping into the turn.** The solution is to increase the TAS or to reduce the bank angle whilst increasing the rudder pressure.

FIGURE 18-5
Aircraft Slipping into the Turn



23. At Figure 18-6 the aircraft is underbanked for the TAS and required rate of turn, and is turning at the required rate primarily because of yaw induced by over-application of rudder into the turn. The results in the **aircraft skidding out of the turn.** The solution is to decrease the TAS or to increase the bank angle whilst relaxing the rudder pressure.

FIGURE 18-6
Aircraft Skidding Out of the Turn



THE REQUIRED RATE OF TURN IS ACHIEVED WITH TOO MUCH RUDDER AND INSUFFICIENT BANK. THE BALL SHOWS A *SKID* OUT OF THE TURN.

24. An alternative to the ball-in-tube type of slip indicator is a weighted pendulum. The weighted pendulum works on precisely the same principle as the ball-in-tube, except that now the arc of movement of the visible indicator is governed by the pendulum rather than by the curvature of the tube.

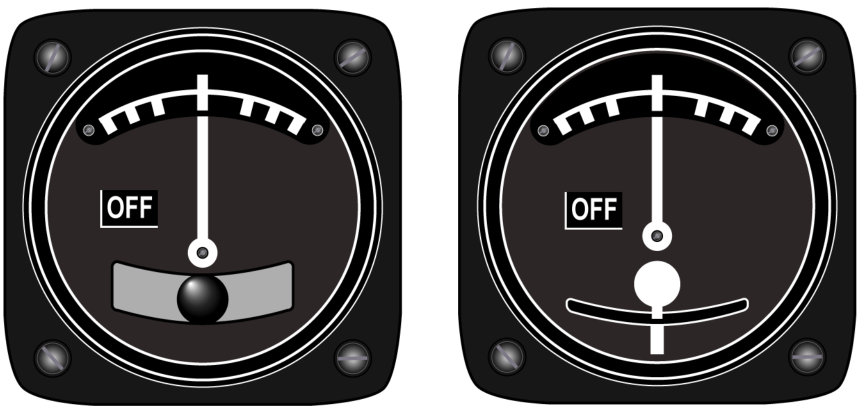
25. Two types of turn and slip indicator are shown at Figure 18-7. At Figure 18-7(a) a ball-in-tube slip indicator is shown, whilst the instrument at Figure 18-7(b) employs a weighted pendulum slip indicator. In each case the rates of turn are marked at rate one (3°/second, the inner mark), rate two (6°/second, the mid mark) and rate 3 (9°/second, the outer mark) for each direction of turn. On civil aircraft it is not unusual for the turn and slip indicator to show only rate one turns in each direction.

26. The two turn and slip indicators shown at Figure 18-7 are both electrically driven and have a power failure warning flag on the instrument face. This flag will show the word 'OFF' until the gyro rotor is operating at its proper design speed.

27. Figure 18-8 illustrates turn and slip indications which coincide with straight and balanced flight and with turns to the left and right, both in balance and with slip and with skid.

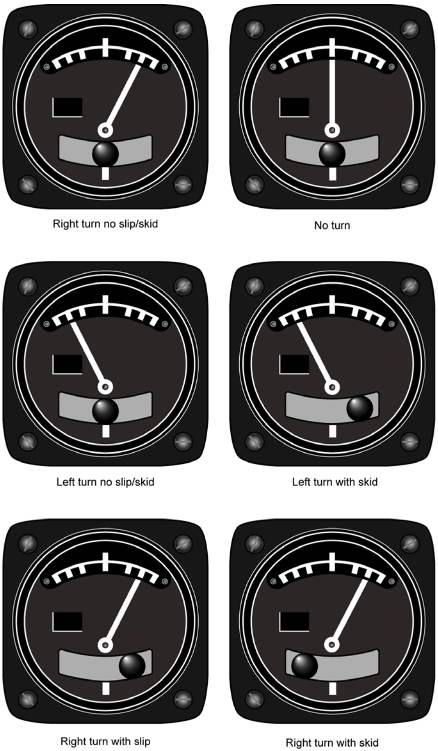
FIGURE 18-7
Different Types of Turn and Slip Indicator

(a) (b)



Turn and Balance Indicators

FIGURE 18-8
Example Turn and Slip Indications



The Turn Co-ordinator

28. The **turn coordinator** is a development of the turn and slip indicator. The slip (or balance) indicator is as previously discussed. It is the gyro arrangement and the face of the instrument showing the rate of turn which differs from the basic turn and slip indicator. A typical turn coordinator instrument face is shown at Figure 18-9.

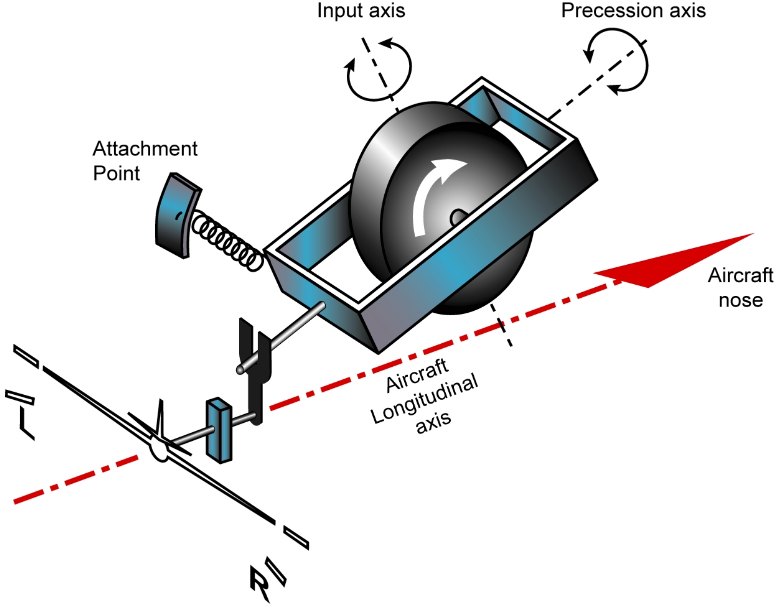
FIGURE 18-9
Turn Coordinator Presentation

29. The statement '2 min' on the face indicates that, with a balanced turn and the wing of the aeroplane on the instrument face aligned with the left or right mark as appropriate to the direction of turn, a turn through 360° will take two minutes. In other words, the instrument will be indicating a rate one turn.

30. The gyro within the instrument casing is illustrated at Figure 18-10.

31. The difference now is that the gimbal is inclined at about 30° in the vertical plane with reference to the aircraft's longitudinal axis. This means that the gyro is now sensitive to roll (or bank) as well as yaw and the initiation of a turn using bank will cause the gyro to precess. It is the ability of the instrument to respond to roll as well as to yaw which makes it react more quickly than the turn indicator to the initiation, change of rate, or completion of a turn. It is the enhanced sensitivity of the turn coordinator when compared with the turn indicator which is its principal advantage.

FIGURE 18-10
Turn Coordinator Gimbal Arrangement



32. When the gyro precesses, the movement is transmitted via a simple peg and claw arrangement, causing the aircraft symbol on the face of the instrument to show the direction of bank. A damping mechanism, normally a piston in an air tube, is usually incorporated. As in the turn and slip indicator, the spring will prevent topple when the gyro is not spinning.

33. The turn coordinator suffers the same errors as the turn and slip indicator, over-reading if the gyro speed is too high and under-reading if the gyro speed is too low. The main problem with the turn coordinator has been found to be the way in which the instrument is interpreted by the inexperienced pilot who finds himself in instrument flying conditions. The danger is that this pilot will interpret the picture as showing the bank angle of the aircraft, in other words assume that the turn coordinator is giving the same indications as the artificial horizon. To prevent this, the turn coordinator in light training aircraft may well be placarded 'No Pitch or Attitude Information'.

Inertial Navigation Systems

Inertial Navigation Systems

1. Inertial Navigation Systems (INS) are self-contained navigation systems which give continuous and accurate information regarding the position of the aircraft to which they are fitted.

2. No inertial system can operate without accelerometers. These acceleration measuring devices sense any change in the aircraft velocity (acceleration/deceleration) very accurately. This information is then **integrated** once to give velocity (speed in a given direction) and a second time to give change of position (distance travelled in a given direction).

3. Necessarily the aircraft to which the INS is fitted will itself pitch, roll and yaw, and accelerations due to these manoeuvres must not be processed by the integrators. Furthermore, the aircraft is moving across the surface of a sphere which is itself moving through space. It is therefore a complex procedure to isolate those accelerations which are relevant to the aircraft's changing horizontal position relative to the Earth.

4. Two approaches for isolating the required horizontal accelerations are available, the first of which utilises a stable platform.

Stable Platform Systems

5. Stable platforms themselves come in two forms, the **north aligned system** and the **wander angle** system.

North Aligned Platforms

6. The north aligned system involves a stable platform upon which two accelerometers are mounted at right angles to each other. One accelerometer is maintained in alignment with true north. The second accelerometer, which is set at 90° to the first, will therefore be aligned east-west. There may be a third (vertical) accelerometer, but for the time being we will ignore this.

7. As well as being stable in direction, that is to say that the north-aligned accelerometer will continue to point to true north regardless of the aircraft heading, the platform is also maintained in a state of **horizontal stability.** This means that the platform, and therefore the accelerometers, will be maintained **Earth horizontal** regardless of any pitching or rolling movements of the aircraft. The accelerometers will not therefore measure the effect of gravity and erroneously interpret this as an acceleration. Since the Earth is itself rotating, and the aircraft moving across the Earth's surface, the INS computer is required to calculate the rate at which to **topple** the platform with respect to space in order to maintain the platform in an Earth horizontal attitude.

Wander Angle Platforms

8. A development of the north aligned system described above is the wander angle system. The hardware is exactly the same as for a north aligned platform and again requires that two accelerometers are mounted at right angles to each other on a platform which is maintained Earth horizontal. With some systems the operator has the choice of operating in either the north aligned or the wander angle mode.

9. In the wander angle system no effort is made to maintain north alignment. Instead, the angle by which the platform would have to be skewed (in the horizontal plane) in order to re-align the platform with north is calculated, and used to correct the outputs of the accelerometers so that the outputs **appear** to be coming from a north aligned system.

10. The notes which follow deal principally with the north aligned system.

Accelerometers

11. Newton's laws of motion give a clue as to the principle of operation of accelerometers:

Newton's First Law. A body continues in a state of rest or uniform motion unless compelled to change that state by a force acting upon it.

Newton's Second Law. The rate of change of motion of a body is proportional to the applied force causing the change, and takes place in the direction of the applied force.

Newton's Third Law. To every applied force there is an equal and opposite reaction.

12. The most commonly used accelerometer is based on a pendulum and a closed loop feedback system and is called a force balance or force re-balance accelerometer. A schematic diagram of such an accelerometer is shown at Figure 19-1.

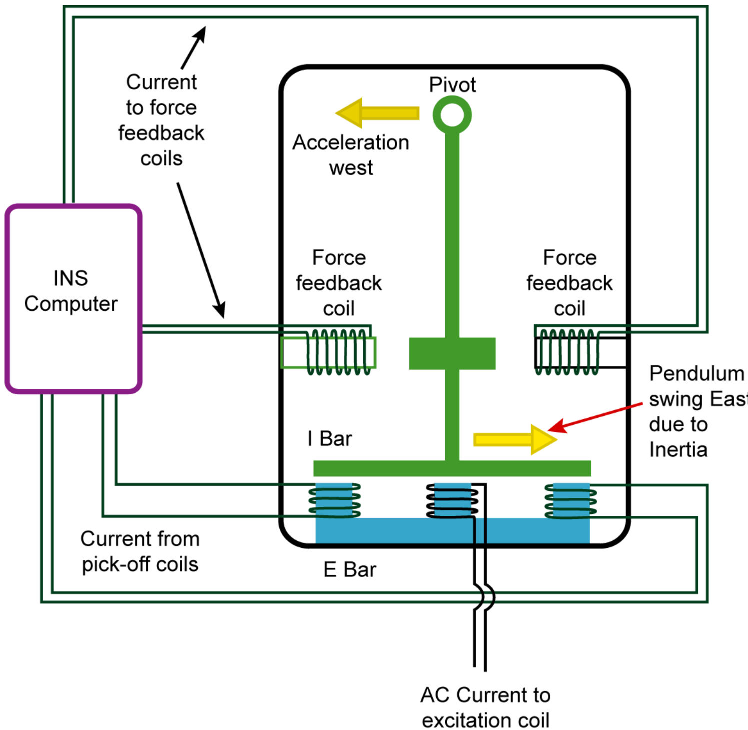
13. The pivot of the pendulum allows it to swing along only one axis, known as the **alignment axis.** In the example shown at Figure 19-1 the pendulum is aligned with an east-west axis and this accelerometer is therefore insensitive to any acceleration/deceleration which is entirely in a north-south direction.

14. The greater the rate of change of speed, the greater is the displacement of the pendulum as it lags behind the pivot. As the pendulum swings, the state of equilibrium at the **E and I bar detector** is disturbed and a greater current is induced to flow in one or other of the **pick-off coils.**

15. At Figure 19-1 the acceleration is to the west, the pendulum swings to the east (right) and a greater current flows to the left-hand pick-off coil, since the air gap between the E and the I bars is now smaller at the left hand end.

16. The difference in pick-off coil currents is sensed within the INS computer, which produces a DC current in the appropriate **force feed-back coil.** This current energises the associated electro-magnet which attracts the pendulum back to the vertical. The strength of the current required to maintain a vertical pendulum is proportional to the acceleration attempting to displace the pendulum, and it is the magnitude of this current which is used by the computer to measure the acceleration along the sensitive axis of the accelerometer.

FIGURE 19-1
Force Balance Accelerometer

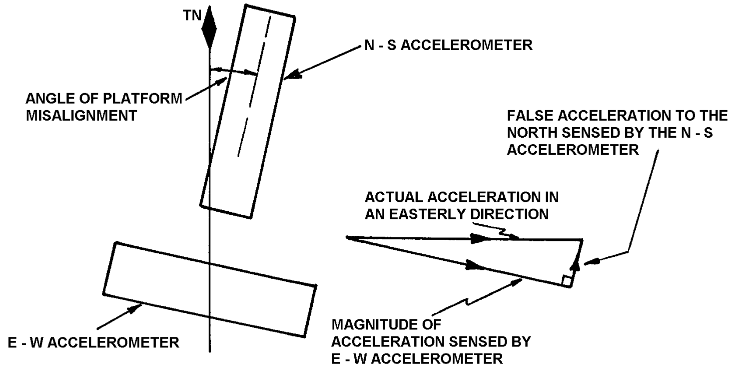


17. The accelerometer itself is extremely accurate and will measure precisely even the smallest change in aircraft velocity, subject to two very important constraints:

If the accelerometer is not **Earth horizontal** a continuous current will be required at one or other of the force-feedback coils to maintain the equal air gaps between the **I** bar and the outer arms of the **E** bar. This **bias** current is caused to flow because of gravity, but unfortunately the computer will integrate this bias current as an erroneous groundspeed. It is therefore essential that the stable platform is maintained Earth horizontal.

If the accelerometer is directionally misaligned, that is to say that the sensitive axes of the north-south and east-west accelerometers are not lying precisely in north-south and east-west planes, then the direction of acceleration will be incorrectly computed, as illustrated at Figure 19-2. It is therefore essential that the platform maintains a high degree of **directional alignment.**

FIGURE 19-2
Effect of Mis-Alignment of Accelerometers



Rate Integrating Gyros

18. The sensing gyros mounted on the platform are **rate type** gyros. Unlike the rate gyro found in the turn and slip indicator, the inertial rate gyros have no springs to inhibit the gyro wheel as it precesses. As the aircraft manoeuvres the relevant gyro (pitch, roll or azimuth) is allowed to precess, and this precession is sensed by a **pick-off coil** on the inner can pivot. The output of the pick-off coil is fed via the INS computer to the associated motor to cancel the platform movement as it attempts to follow the changing attitude of the aircraft, which would cause it to depart from an Earth horizontal and/or north aligned condition.

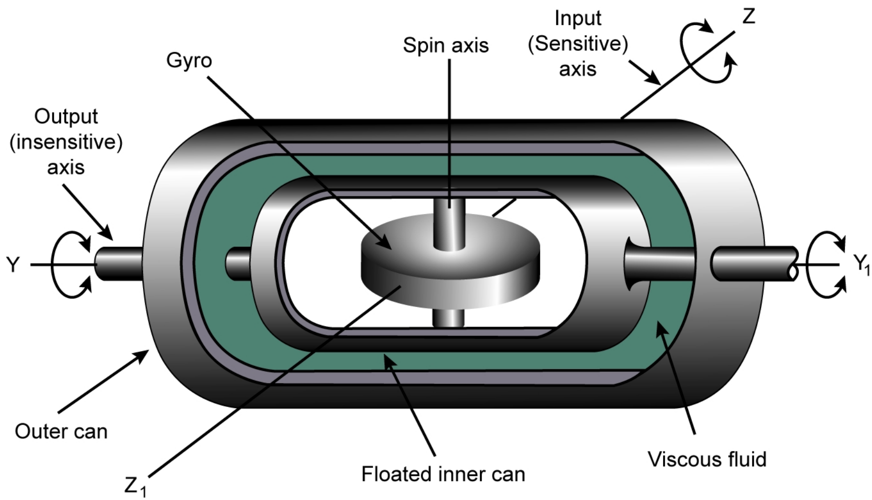
19. It is necessary that the gyros used in inertial systems are very accurate indeed. A real wander rate of 0.01°/hr is about all that is acceptable. In order to achieve this level of accuracy the gimbal and rotor assemblies of the gyro are floated in a fluid. The viscosity of the fluid controls the amount of precession which results from a given input. As viscosity varies with temperature, it is necessary that the gyros reach their operating temperature before the platform is switched into the navigation mode.

20. Like the rate gyro in the turn and slip indicator, the INS rate gyros have freedom of movement about only one axis (apart from the spin axis). Figure 19-3 shows a schematic breakdown of a typical rate integrating gyro. Note that the spin axis, the sensitive axis and the insensitive axis are all at right angles to each other.

21. The sensitive axis of the gyro at Figure 19-3 is $Z - Z^1$. Any movement about this axis results in precession of the gyro about the axis $Y - Y^1$. This rotation is picked off electrically at the pivot (the output axis $Y - Y^1$).

22. Also located at the pivot is a torque motor. This motor is fed from the INS computer with a signal which causes the motor to apply a torque to the gyro to correct for Earth rate and transport wander, in order to keep the platform Earth horizontal and north aligned.

FIGURE 19-3
INS Rate
Integrating Gyro



The Stable Platform

23. In order that the platform can remain Earth horizontal and directionally aligned as the aircraft manoeuvres, it is necessary to mount the platform within a two-gimbal system, as illustrated at Figure 19-4.

24. The platform is kept Earth horizontal and in directional alignment by means of sensing gyros, error signals from which are used to drive motors attached to the gimbal spindles, as shown at Figure 19-4. The sensing gyros and motors correct for any and all aircraft movements in the pitching, rolling and yawing planes.

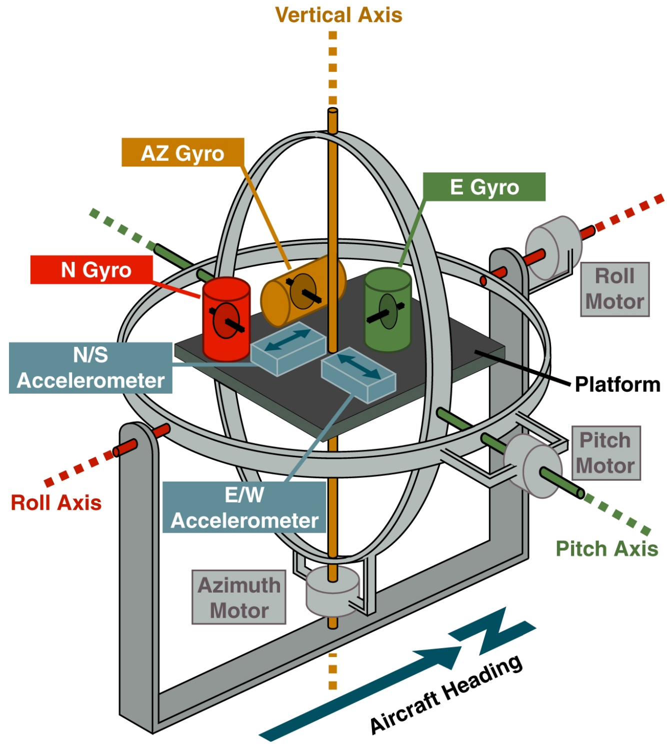
25. In the diagram at Figure 19-4 the aircraft is heading north. Any movement of the aircraft about its rolling axis will be sensed entirely by the **north gyro,** the output of which will be fed via the INS computer to the **roll motor.** On this heading any movement of the aircraft about its pitching axis will equally be entirely sensed by the **east gyro**, the output of which is used to drive the **pitch motor.**

26. Were the aircraft to be heading due east, the north gyro would sense only pitching movement and the east gyro would sense only rolling movement.

27. On all non-cardinal headings the pitch and roll motors are driven by composite outputs from both the north and east gyros, the computer takes care of the mathematics.

28. The azimuth gyro is sensitive to any yawing motion, and its output is always fed via the computer to the azimuth motor. The azimuth motor is also used to isolate the platform from aircraft heading changes.

FIGURE 19-4
The Stable Platform - Aircraft Heading True North



Corrections for Earth Rate and Transport Wander

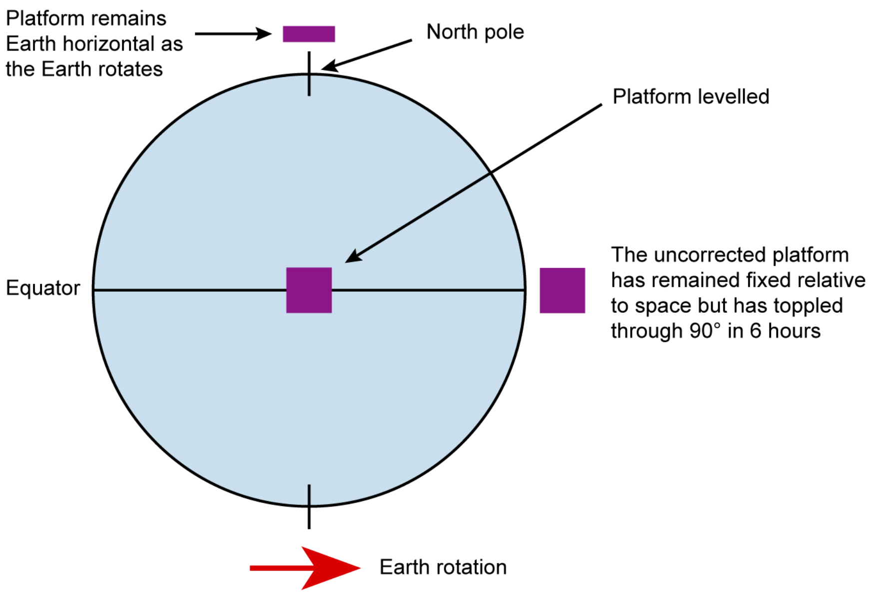
29. In order to maintain the platform Earth horizontal and north aligned, corrections are required to compensate for the rotation of the Earth, otherwise the platform will be controlled by the gyros to remain rigid in space rather than rigid with reference to the Earth. The gyro torque motors must therefore be continuously fed with Earth rate corrections. These corrections are considered in terms of topple and drift.

30. A stable platform which was levelled at either pole and remained there would not topple as a result of the rotation of the Earth. Another platform which was levelled at the equator and neither moved or corrected for Earth rotation would topple through 90° in 6 hours as a result of the Earth's rotation, as shown at Figure 19-5.

31. At the equator, the correction required to maintain the platform Earth horizontal is 15°/hour. At any other latitude the correction required is **15 x the cosine of the latitude °/hr.**

32. Conversely, a platform which was aligned with true north at the equator and neither moved or corrected for Earth rate will continue to point true north as the Earth rotates, since the meridians are parallel at the equator. At latitudes other than the equator it will be necessary to correct the alignment of the platform by applying a current to the torque motor on the azimuth gyro so that the azimuth motor can maintain north alignment. The magnitude of this correction is **15 x the sine of the latitude °/hr.**

FIGURE 19-5
Platform Topple at the Equator and Poles

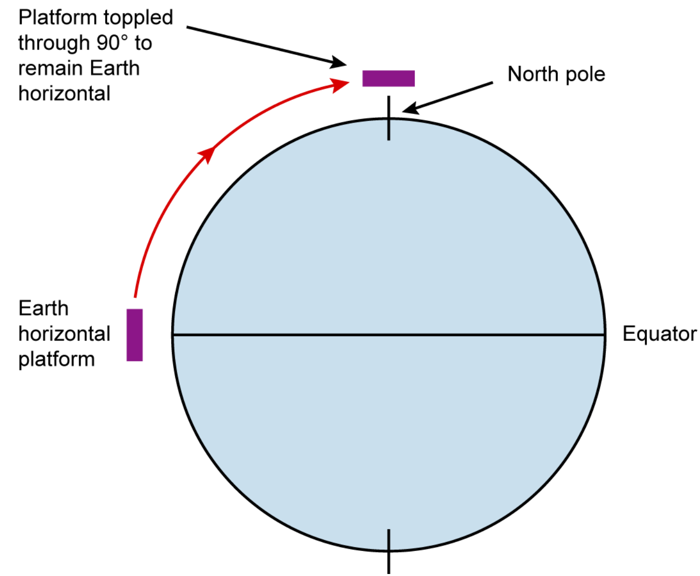


33. The problem of keeping the platform level and aligned is of course complicated by the fact that the platform is being transported across the surface of the Earth. The mathematics for transport wander as it affects the alignment of the INS platform are the same as the transport wander which we previously considered when looking at unslaved directional gyros. The computation of the correction required to maintain north alignment is based on the formula:

$$\textbf{Alignment rate} = \frac{\text{East/West component of groundspeed} \times \text{Tan latitude}}{60} \, ^\circ/\text{hr}$$

34. Transporting the platform also produces a requirement to topple the platform in order to keep it in the Earth horizontal. For example, in Figure 19-6 a platform is flown from the equator due north to the pole. In order that it remains horizontal throughout, the platform must be toppled through a total of 90°. In this case the rate of required topple is a function of the groundspeed north/south. Suppose that the aircraft's groundspeed was 600 kt. The journey time from the equator to the pole (5400 nm @ 600 kt) would be 9 hours and the required topple rate would be 10°/hr. The correction required can therefore be expressed as **groundspeed north/south ÷ 60°/hr.** Similarly, any component of the groundspeed in an east/west direction would require a correction of **groundspeed east/west ÷ 60°/hr.** On anything other than a true cardinal track, the required correction would be a composite of both of the above and the correction would be achieved by means of error signals fed to the torque motors attached to both the north and east gyros which would result in activity at both the pitch and roll motors.

FIGURE 19-6
Platform Topple - Equator to Pole Transit



Coriolis Correction

35. Accelerometers mounted on a stable platform are subject to errors caused by coriolis.

36. Coriolis effect, put simply, is the force which acts on any body which is moving across the surface of the Earth, and which tends to deflect that body to the right in the northern hemisphere and to the left in the southern hemisphere. Coriolis force is due to the Earth's rotation about its own spin axis, and is fully described in the Meteorology Theory syllabus, where its effect on moving bodies of air is considered. Any force which acts upon the accelerometers will be assumed to be a change of velocity (acceleration or deceleration), and so it is necessary for the system to 'compute out' the coriolis effect on the accelerometers. In order to do this the INS computer requires values of aircraft latitude, track and groundspeed, but of course this presents no problem, since all of these values are themselves computed by the system.

Levelling and Gyro-compassing

37. As already emphasised, the output of the INS will only be of a high standard of accuracy if the platform is maintained Earth horizontal and correctly aligned with true north. The first step is obviously to set up the platform on the ground. **This procedure can only be satisfactorily completed whilst the aircraft is stationary.**

38. Initially the platform is levelled to aircraft horizontal and if necessary coarsely aligned by manually slewing the platform so that it is aligned with the aircraft's compass heading, corrected for variation (in modern systems the coarse manual alignment is not normally required). With the platform selected to the **align mode** fine levelling is now commenced. During this process the accelerometer outputs are assumed to be due to gravity, hence the requirement that the aircraft must not be moved at this stage. The outputs of the accelerometers are used in a servo loop principle to level the platform. This process is also known as the accelerometer null technique and is completed in 1 to 1½ minutes.

39. Once the fine levelling procedure is complete, the platform automatically commences the gyro-compassing process **for which it is essential that the present position is correctly inserted into the INS.** In fact it is an incorrect latitude which will prevent successful gyro-compassing and consequent accurate north alignment, we will see why shortly.

40. During the gyro-compassing process the outputs of gyros and accelerometers are combined to achieve a precise alignment. Refer again to Figure 19-4 and let us examine the gyro-compassing process for the platform shown, which is in an aircraft which is facing north. Were the platform to be precisely aligned with true north, the east gyro (which has a north/south spin axis) would be insensitive to movement of the platform about the north/south (Earth) axis as the platform is toppled by the torque motor on the north gyro, which (in this case) is driving the roll motor to keep the platform Earth horizontal.

41. Suppose now that the platform is mis-aligned, and that therefore the spin axis of the east gyro is no longer exactly north/south oriented. The east gyro will now detect a component of the Earth rate topple, and will precess. This precession is picked off and used by the INS computer to tilt the platform. The platform is now no longer Earth horizontal and the north/south accelerometer will sense gravity. The output from the north/south accelerometer is used by the computer to feed an error signal to the azimuth motor to re-align the platform and an error signal to the pitch motor (in this case since the aircraft is facing north) to re-level the platform. This process continues until the output from the north/south accelerometer is zero. Now the east gyro is insensitive to Earth rotation, since the platform is precisely aligned. The two basic inputs which are used to achieve gyro-compassing are therefore **Earth rate** and **gravity.**

42. Since the Earth rate corrections which are fed from the computer are at a rate of 15° x the cosine of the latitude °/hr, it follows that gyro-compassing will not be achieved at high latitudes (the cosine of 90° is zero). With modern systems 70° N/S is about the limit. Furthermore, the time taken to achieve alignment will increase as the latitude at which gyro-compassing occurs increases. If the computer is using an incorrect latitude during the gyro-compassing process, the platform will not align. In the best case, if the latitude which has been entered as the start (ramp) position is wildly in error, the computer will realise that something is wrong and will invite you to check the ramp position. In the worst case, if the entered latitude is only slightly in error, a false north alignment will occur, which will give rise to an ever increasing (unbounded) error in the INS position as the flight progresses.

43. Once the levelling and alignment processes are complete the **platform ready light** will illuminate and the **navigation mode** can be selected. The aircraft can now be moved. The story of what happens next (the way in which the outputs from the platform are processed and presented to the pilots) is continued in the Inertial Data - Processing and Presentation section.

Strapped Down Systems

44. The strapped down type of inertial navigation system became feasible with the advent of high speed large capacity digital microprocessors and the introduction of the ring laser gyro.

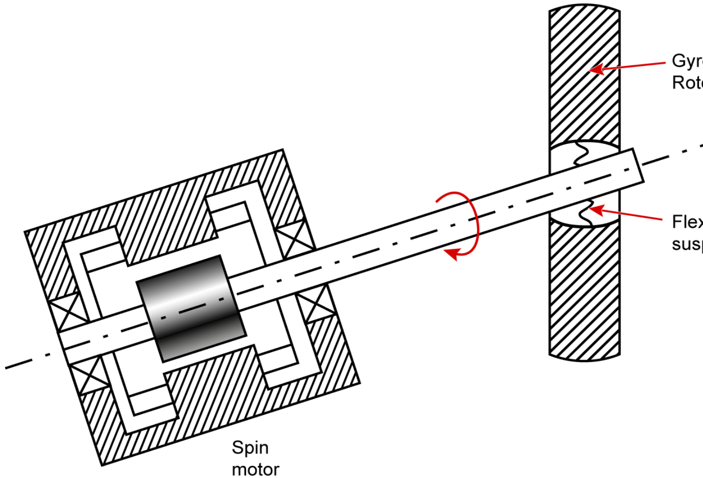
45. There is **no stabilised platform** and **three accelerometers are mounted rigidly** inside the inertial navigation unit, which is simply bolted to the aircraft structure. The accelerometers are therefore effectively fixed to the airframe, and are aligned with the aircraft's pitch, roll and yaw axes. **Three ring laser gyros** are also mounted in the same unit, and again their sensitive axes are aligned with the aircraft's pitch, roll and yaw axes.

46. Two alternative types of gyro may be used in some Strapped Down Systems - the **Tuned Rotor Gyro** or the **Fibre Optic Gyro** (FOG).

Tuned Rotor Gyro

47. The dynamically-tuned free rotor gyros, commonly called 'tuned rotor gyros' are in attempt at producing a cheaper and more robust alternative to complex and expensive conventional gyros. In their basic form they consist of a rotor which is suspended on the end of a drive shaft by a flexural suspension which allows 2 degrees of freedom at right angles to the shaft. The other end of the shaft is driven by a synchronous motor. The system is designed so that the dynamic torques caused by centrifugal forces acting on the rotor produce a positive spring rate which exactly balances the flexural spring rate at a certain speed. At this tuned condition the rotor behaves as if it were suspended by a zero-rate suspension and so behaves as a true free rotor. Figure 19-7 shows a cross sectional view of the rotor assembly of a Tuned Rotor Gyro.

FIGURE 19-7
The Tuned Rotor Gyro



Fibre Optic Gyro

48. The Fibre Optic Gyro operates in a similar manner to the ring laser gyro. It is a rotation rate sensor which uses **fibre optics** as the propogation medium for the beam of light. The big advantage of the technique is that it is relatively inexpensive when compared to the RLG as it does not require the provision of an expensive Cervit Block.

Strapped Down System Operation

49. In the strapped down system the accelerometers will clearly measure total acceleration, which will now be due to gravity (the accelerometers are not stabilised Earth horizontal), to aircraft manoeuvres and to the aircraft movement over the surface of the Earth. Of these only the third is required, and the other two output components must be isolated.

50. In order to achieve this the INS computer needs to determine the difference between the horizontal plane of the strapped down unit (aircraft horizontal) and Earth horizontal, and also the angle between the aircraft's roll (fore and aft) axis and true north during what is still referred to as the alignment (initial setting up) process. It does this by analysing the outputs of the accelerometers due to gravity. As with platform systems the aircraft must not be moved with the equipment in the align mode.

51. Once in the navigation mode the fast/high capacity processor within the INS computer uses the outputs of the RLGs in order to isolate the outputs of the accelerometers which are due solely to the movement of the aircraft relative to the Earth's surface and to relate these outputs to the Earth's north/south, east/west and vertical axes. Corrections for Earth rate and transport wander are achieved by adjusting the perceived angles between aircraft horizontal and Earth horizontal and between true north and the aircraft's roll axis.

52. The main advantage of the strapped down system over the stable platform system is that there are virtually no moving parts, making the system far more reliable, and much lighter. If a strapped down system does go unserviceable, replacement of the navigation unit is a much simpler procedure than with a stabilised platform. Initial alignment times are faster with a strapped down system since there is no platform to be aligned and the ring laser gyros achieve their stable operating temperatures much faster than their mechanical counterparts.

53. All strapped-down systems work on the same principles described above. Whereas many use the triangular ring-laser gyroscope described in the Gyro Principles section, some use a square or four-sided RLG. A further type patented by Litton Aero Products uses a four-mode laser gyroscope, with four laser paths in two planes, which together with circular light polarisation effectively combines two LRGs into one and eliminates the requirement for dither.

Inertial Data - Processing and Presentation

54. Having considered the nuts and bolts of inertial platforms and briefly discussed strapped down systems, it is now time to consider how the raw output from the system is converted into useful data, and how this is presented to the pilots.

Integration

55. Regardless of which type of inertial system is considered, they all have one thing in common. They all detect components of acceleration which are truly sensed in the Earth horizontal plane along the north/south and east/west axes (the north aligned platform). Alternatively the accelerometer outputs are corrected so that they give an output which represents that which would occur were the accelerometers so aligned (wander angle platforms and strapped down systems).

56. To produce the navigation parameters of track, groundspeed, distance and position the raw inputs of acceleration north/south and east/west must be summed (integrated) by time. Acceleration integrated with respect to time is speed, and speed integrated with respect to time is distance. These are the two necessary stages of integration which are required and which are shown at Figure 19-8.

57. Note that the distance travelled north/south in nautical miles equates to change of latitude, however in order to obtain change of longitude from the east/west distance travelled requires a further stage of computation, which is achieved within the **secant unit** of the computer.

58. Remember that east/west distance is known as departure and:

$$\textbf{Departure (nm) = Change of longitude (mins. of arc)} \times \textbf{Cosine Latitude}$$

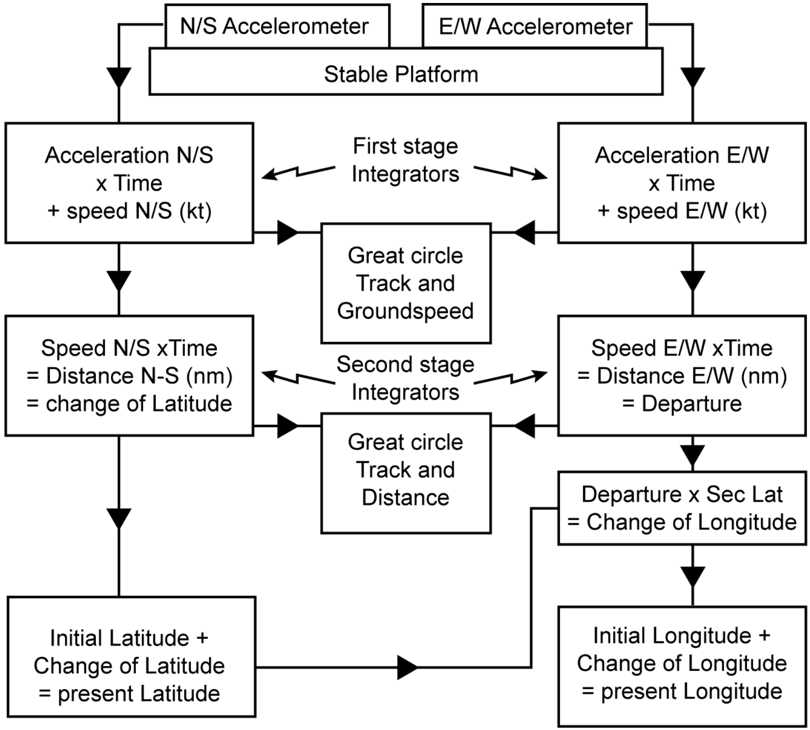
This formula must be re-arranged since in this case it is the change of longitude which is the unknown quantity, therefore:

$$\text{Change of longitude (minutes of arc)} = \frac{\text{Departure (nm)}}{\text{Cosine latitude}}$$

or:

$$\textbf{Change of longitude = Departure (nm)} \times \textbf{Secant latitude}$$

FIGURE 19-8
Stages of Integration



The Schuler Cycle

59. The key factor involved in the Schuler principle is that a stable platform which is maintained Earth horizontal behaves like a pendulum with a length equal to the radius of the Earth. The time taken for a pendulum to swing through one cycle is directly proportional to its length. Where this length is the same as the Earth's radius, the time taken for one cycle of the pendulum is 84.4 minutes.

60. Stable platforms possessing this property are said to be **Schuler tuned.**

61. In theory, once levelled the platform should remain stable and undisturbed, however vibration and turbulence shocks in flight are likely to create small disturbances. Consequently the platform is likely to be swinging continuously, hopefully by only a small amount.

62. The effect of this swing is that the outputs from the accelerometers will be in error by a maximum amount when the platform is at the extremity of its swing. The period of the swing is 84.4 minutes regardless of the magnitude of the disturbance which caused it, and so two important facts emerge:

(a) The Schuler tuned platform produces its maximum error at 21.1 and 63.3 minutes through each 84.4 minute cycle.

(b) The magnitude of the maximum error depends on the size of the disturbance which caused it, however the **mean error remains at zero** (assuming no inherent accelerometer error).

63. From the above it should be apparent that any error in the outputs of the accelerometers which is caused by Schuler tuning is **bounded,** that is to say that the error does not increase with time beyond its original maximum value.

64. Furthermore, since the output error of any accelerometer will not increase with time, the output of the first stage integrator associated with it (velocity) will also be bounded. For example, for an aircraft flying at a groundspeed of 600 kt the INS output of groundspeed might be 600 kt at minute zero, 602 kt at minute 21.1, 600 kt at minute 42.2, 598 kt at minute 63.3 and again 600 kt at minute 84.4. The mean output of groundspeed is correct at 600 kt.

65. We have discussed Schuler tuning as it applies to stable platforms which are maintained Earth horizontal. Appreciate, however, that strapped down systems are designed to deliver accelerometer outputs which are corrected (processed) to be representative of those which would be achieved were the accelerometers to be maintained Earth horizontal rather than aircraft horizontal. Strapped down systems are therefore also considered to be Schuler tuned, and to suffer similar bounded errors as a consequence.

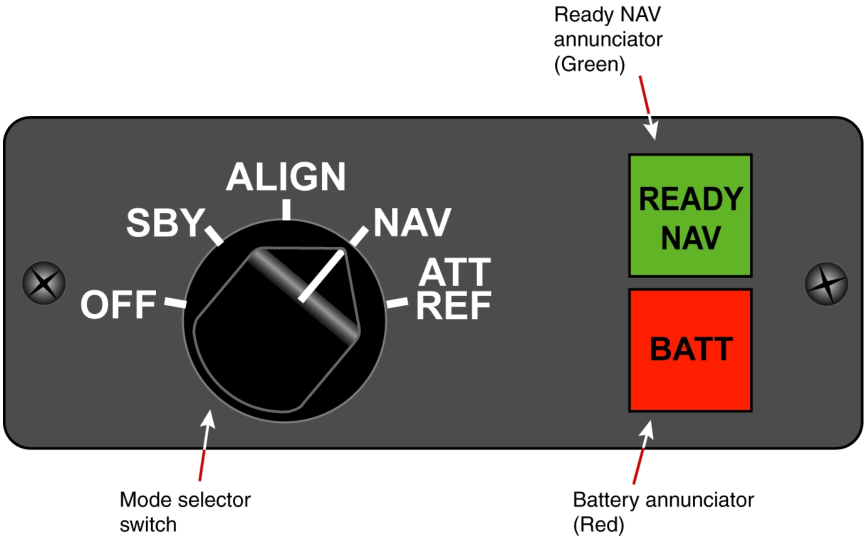
INS Control and Display Panels

66. There are many types of inertial navigation systems currently in use and the final paragraphs of this section summarise the evolution of this invaluable aid to navigation. Suffice for the moment to say that the system which is described in the following pages is an older type of INS, the operation of which is perhaps easier to understand than that of its more sophisticated successors.

The Mode Selector Panel

67. The traditional INS system comprises two control panels for control and display plus an **Inertial Reference Unit** (**IRU**) which contains the gyros, accelerometers, integration circuits etc. The simpler of the two control panels, **the mode selector unit** (**MSU**), is shown at Figure 19-9.

FIGURE 19-9
Mode Selector Unit (MSU)



68. The positions of the mode selector switch and the purpose of the two lights (Ready Nav and Batt) are discussed below:

Standby In this mode power is supplied to all parts of the system. It is normal to insert the start position (the aircraft's ramp position in latitude/longitude to the nearest tenth of a minute of arc) at this stage. Remember that an accurate latitude is essential for successful platform alignment.

Align Having inserted the start position, with the aircraft stationary and with a stable power source, the align mode can now be selected to enable levelling and gyro-compassing to commence. If necessary, the system's progress through this phase can be checked by selecting status (STS) on the control display unit. Numbers will appear in the left numerical display window, starting at 90 as the levelling procedure commences and reaching zero as gyro-compassing is completed. The power source should remain constant during this period, switching from external to internal aircraft power at this stage can persuade the INS to restart the levelling/alignment process.

Ready Nav When the levelling/alignment process is complete the green **Ready Nav light** will be illuminated, to indicate that the system is now ready for use in the navigation mode.

Nav Selection of the navigation mode means that the INS is open for business, and the aircraft can now be moved. The nav position on the control panel is detented or spring biased to prevent an inattentive first officer from de-selecting the navigation mode in flight and ruining his command prospects for a year or two.

Att Ref This position is normally for use following a computer/processing failure. All navigation computations are removed, however the platform may still be used to supply pitch, roll and heading information.

Batt In the event that the power supply to the INS is interrupted, the system will continue to operate on its own internal (battery) DC supply, for approximately 15 minutes. The red **Batt light** on the mode selector panel will illuminate when the battery supply is exhausted.

The Control and Display Unit (CDU)

69. The operation of the INS computer is controlled by **the control and display unit** (CDU): on a modern aircraft this unit may also be referred to as the **Inertial System Display Unit** (ISDU).

70. The INS computer is able to determine aircraft heading. In a north aligned platform system the true heading is simply the angle between the north/south axis of the platform and the aircraft fore and aft axis. The INS computer's memory may contain a 'look up table' of magnetic variation worldwide. Using this memory and the INS position enables the INS to display heading in degrees magnetic. For commercial transport operations it is normal for the INS to express both headings and tracks in degrees magnetic, however appreciate that the INS is referenced to true north.

71. The other primary outputs of the system are track made good, drift, groundspeed and aircraft present position expressed as latitude and longitude.

72. **With an input of TAS from the air data computer,** the INS computer can also resolve the triangle of velocities and give an output of wind velocity.

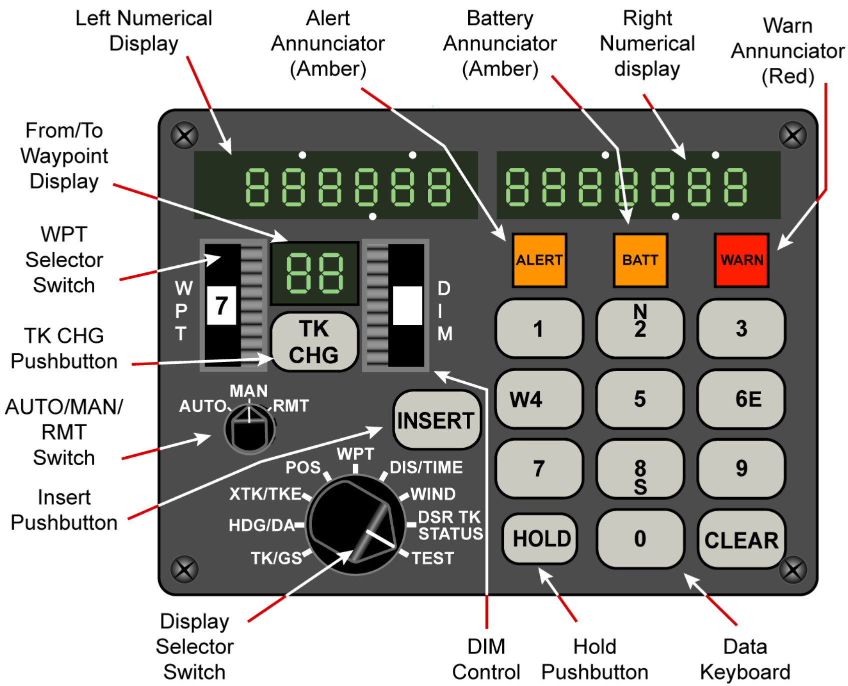
73. Providing that the INS computer is fed with details of the route, in other words it is fed with **waypoints** (the latitude and longitude of all points on the route at which the aircraft is required to change track), the computer uses spherical trigonometry logic to determine the **great circle** tracks and distances between one waypoint and the next. Should the aircraft be cleared at any time to a waypoint which is not the next waypoint, the computer will re-compute the great circle track and distance from the present position to the specified waypoint. Waypoints are identified by number (departure aerodrome as waypoint 1, the first turning point as waypoint 2 and so on) and, by convention, waypoint 0 is a floating waypoint which is the aircraft's present position. When cleared to a distant waypoint (for example waypoint number 5) the operator input would therefore request a track change 0 to 5.

74. Rather than express aircraft position in terms of latitude and longitude, the computer will normally be required to express the position of the aircraft relative to the desired track between the waypoints which define the route.

75. The CDU also provides a means of monitoring the operation of the INS by means of status displays and annunciator lights.

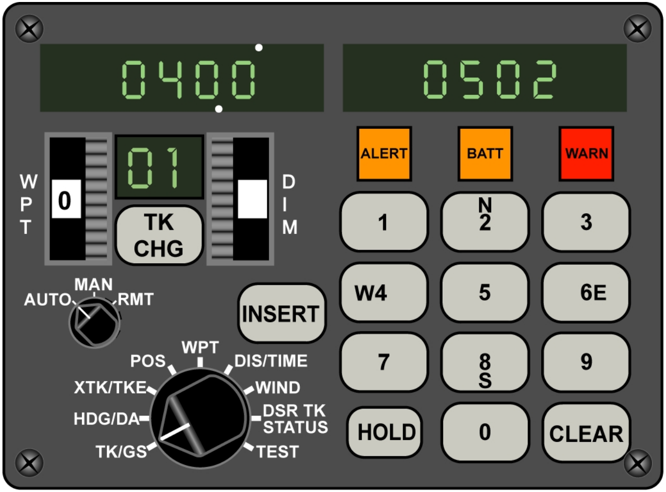
76. A typical CDU is shown at Figure 19-10.

FIGURE 19-10
Typical Control and Display Unit



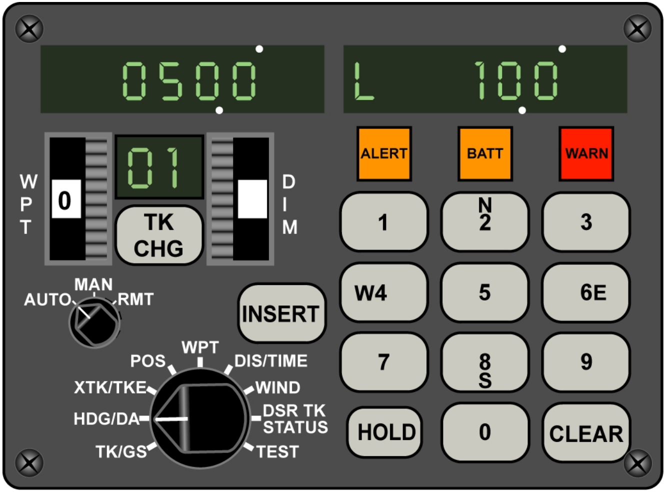
77. The functions of the display selector switch and of other controls and displays on the CDU will now be discussed in general terms.

FIGURE 19-11
Track and Groundspeed Selected



TK/GS (Track and groundspeed). The INS derived aircraft track (against a north reference specified by the operator on the bench prior to installation, normally magnetic north) is shown to the nearest tenth of a degree in the LH window. The INS derived groundspeed is shown to the nearest knot in the RH window. The track is 040.0° and the groundspeed 502 kt at Figure 19-11.

FIGURE 19-12
Heading and Drift Angle Selected

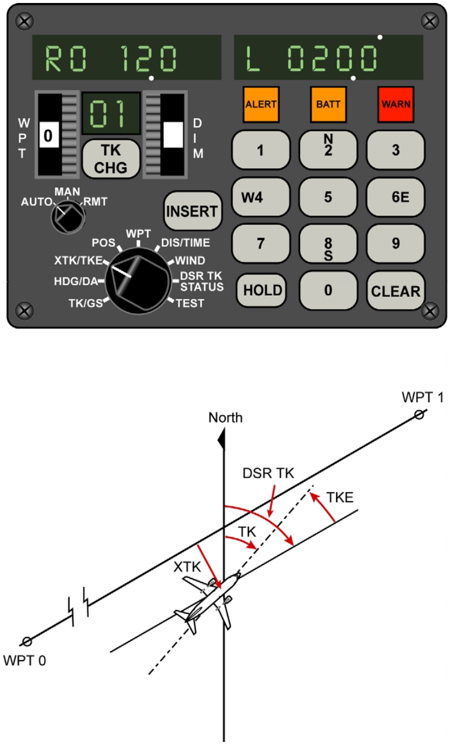


HDG/DA (Heading and drift angle). The INS derived heading is shown to the nearest tenth of a degree in the LH window. The INS derived drift angle is shown to the nearest tenth of a degree is shown in the RH window, and is preceded by an **L** (left/port drift) or an **R** (right/starboard drift). The heading is 050.0° and the drift 10° left at Figure 19-12.

XTK/TKE (Cross track distance and track error angle). The cross track distance (the displacement of the aircraft perpendicularly from the direct great circle track between the two waypoints selected) is shown to the nearest tenth of a nautical mile in the LH window. This figure is preceded by an **L** or an **R** to indicate that the aircraft is left or right of the direct track. The track angle error (the angle between the track which the aircraft would require to make good were it flying along the great circle route between the specified waypoints and the track which it is actually making good) is shown to the nearest tenth of a degree in the RH window. The **L** or **R** which precedes this value indicates that the actual track is to the left or right of the required track. The cross track error is 12.0 nm to the right and the track angle error is 20.0° to the left at Figure 19-13. The geometry of the situation is also shown at Figure 19-13.

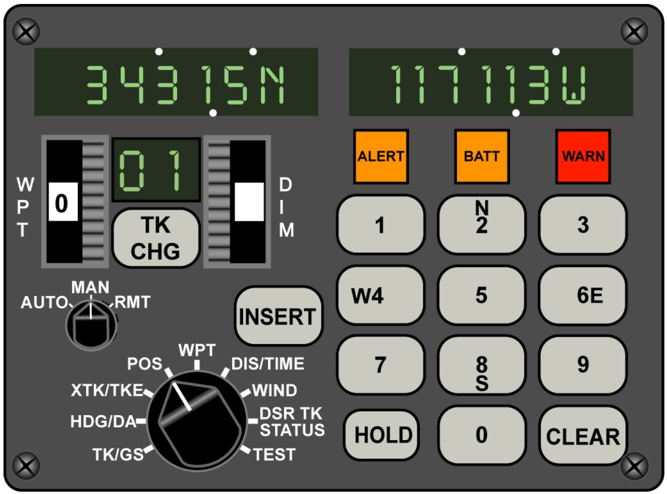
FIGURE 19-13

Cross Track Distance and Track Error Angle Selected



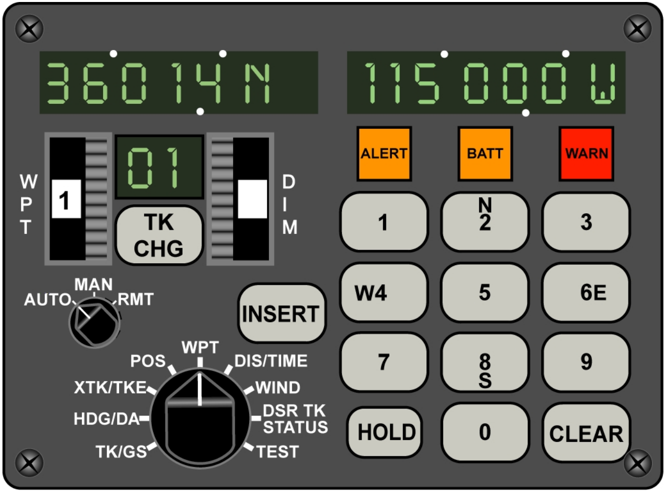
POS (Present position). The aircraft's present latitude is shown to the nearest tenth of a minute of arc, followed by an **N** (north) or **S** (south) as appropriate, in the LH window. The aircraft's present longitude is shown to the nearest tenth of a minute of arc, again followed by an **E** (east) or **W** (west) as appropriate, in the RH window. The aircraft's position is shown as 34° 31.5'N 117° 11.3'W at Figure 19-14.

FIGURE 19-14
Present Position Selected



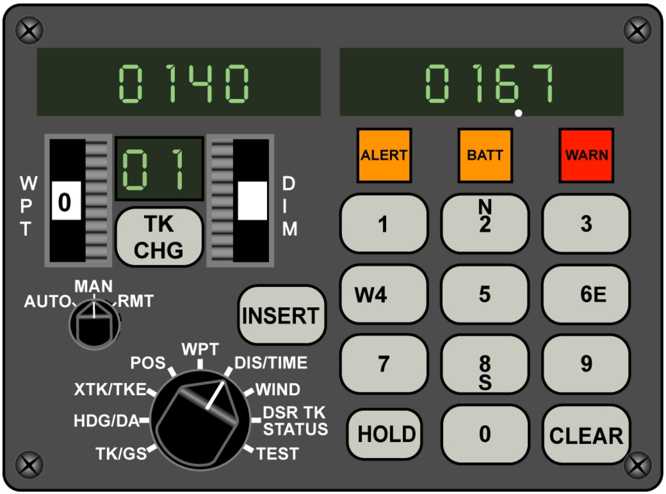
WPT (Waypoint positions). The waypoint positions are shown in latitude (LH window) and longitude (RH window) to the nearest tenth of a minute of arc. In the system which we are considering there are 10 possible waypoint selections (0 through 9). Waypoints 1 through 9 are simply selected turning points, and are normally punched into the system by the operator before the flight. Waypoint 0 represents the aircraft's position at the last time a track change from present position to a specified waypoint was selected by the operator. The position of waypoint one is shown as 36° 01.4'N 115° 00.0'W at Figure 19-15.

FIGURE 19-15
Waypoint Position Selected



DIS/TIME (Distance and time to the next waypoint). The distance to go from the aircraft's present position to the next selected waypoint is shown to the nearest nautical mile in the LH window. The lapsed time from the aircraft's present position to the next waypoint is shown to the nearest tenth of a minute in the RH window. The distance to go is shown as 140 nm and the time to go as 16.7 minutes at Figure 19-16.

FIGURE 19-16
Distance/Time Selected



WIND (Wind velocity). INS derived wind direction is shown to the nearest degree in the LH window. INS derived wind speed is shown to the nearest knot in the RH window. The W/V is shown as 155°/85 kt at Figure 19-17.

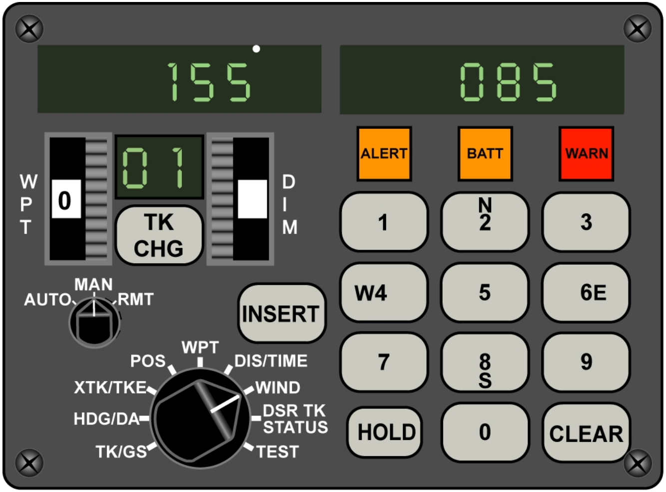
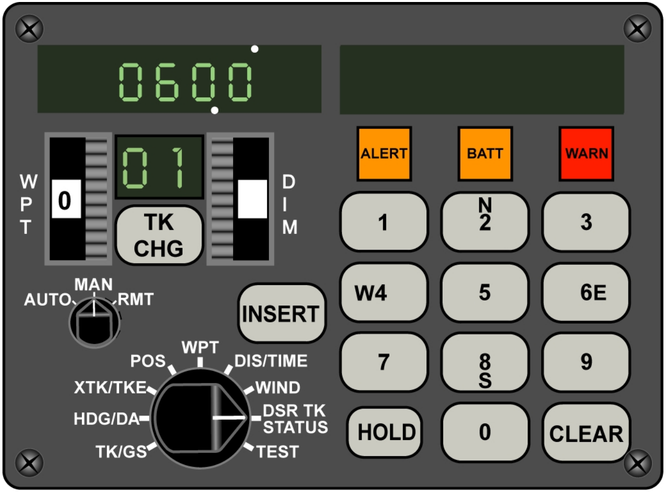


FIGURE 19-17 Wind Direction and Speed Selected

DSR TK/STS (Desired track and status). Assume for the moment that the aircraft is on the great circle track between two specified waypoints. The desired track is the track required to fly from one waypoint to the next and is shown to the nearest tenth of a degree in the LH window. The value of the desired track will change as the aircraft travels from one waypoint to the next. Remember that the INS uses great circle tracks and appreciate that therefore the desired track angle will change to account for Earth convergency. Assuming that a magnetic north reference has been specified by the operator on installation, the desired track will also change as the variation change with aircraft position is pulled from the INS computer's memory. Appreciate that the desired track always assumes that the aircraft will remain on the great circle track between specified waypoints. Should the aircraft depart from this track the desired track readout will show the track angle which is parallel to the original track as defined by the great circle between the chosen waypoints and **not** (as you might expect) the track from the present position to the next waypoint. The desired track is 060.0° at Figure 19-18. You may by now have reached the conclusion that the programme upon which the INS computer operates assumes that the INS will normally operate coupled to the flight director/autopilot, so that across track errors do not occur.

FIGURE 19-18
Desired Track and Status Selected

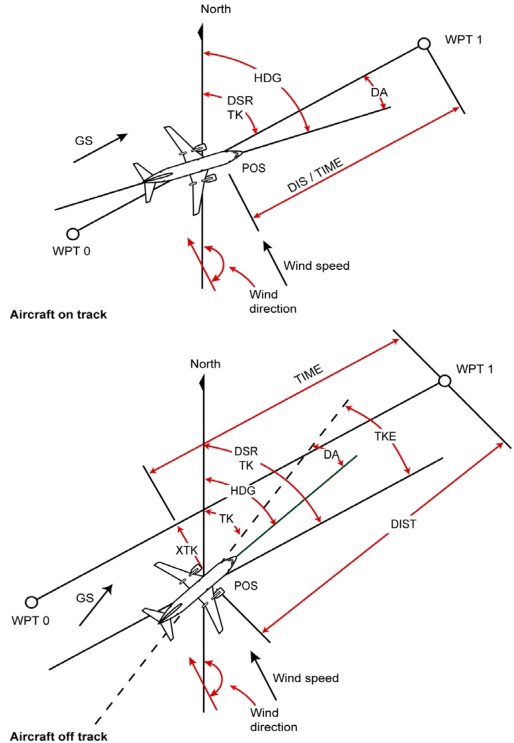


The use of the status function has already been discussed in terms of initial alignment. When airborne the status (RH) window should be blank, however, should the INS develop a fault, the status window will give a value which will identify the nature of the problem after reference to the system operating handbook.

Test (Light emitting diode test). The CDU illustrated at Figure 19-10 shows the function switch in the test position, and consequently all of the digits on the various displays are illuminated and show a number 8. This enables the operator to check that all of the LEDs are operating satisfactorily.

78. This concludes the discussion of the functions of the display selector switch. Before we continue to consider the purpose of the other buttons, selectors and annunciators on the CDU, take a look at the diagrams at Figure 19-19 and ensure that you agree with the INS navigation geometry shown there, both for the on track and the off track situations.

FIGURE 19-19
INS Navigation Geometry



79. With reference to Figure 19-10, the functions of the remaining controls are described in the following paragraphs.

80. The **waypoint selector switch** is thumbed to the appropriate waypoint number (shown in the window to the left of the thumbwheel) when loading the waypoint latitude/longitudes before flight, reloading new waypoints in flight, or checking that waypoints are correctly loaded.

81. The **from/to waypoint display** shows the two waypoints between which the INS assumes that it is flying. All digital readouts, flight director displays and autopilot commands will be based on this information, and so you can imagine the consequences of either giving the equipment the wrong to/from waypoint numbers or feeding the system with the wrong waypoint latitude/longitude to begin with.

82. The **track change push button** enables the operator to tell the system between which two waypoints the aircraft is required to fly (in the event that the system is not set up to fly sequentially through the loaded waypoints). This would normally be used to tell the INS to fly from the present position (waypoint 0) to a waypoint which is not the next waypoint, following a direct routing from ATC.

83. The **dim control** governs the brightness of the LED displays and the panel lighting.

84. The **alert annunciator** flashes to warn the operator that the aircraft is approaching the next waypoint. Typically, the alert light will come on with 2 minutes to run to the waypoint and will flash when the waypoint is crossed. If for some reason the INS is not in the automatic mode (see below), the alert annunciator will continue to flash until cleared (cancelled) by the pilot.

85. As already discussed, the red battery light on the mode selector panel illuminates when the back up power supply is exhausted. The purpose of the **battery annunciator** on the CDU is to warn the operator that the INS is operating on battery power. When the INS is initially powered up prior to levelling/alignment the internal back up power source and associated circuitry are self tested and during this test the battery annunciator is illuminated. In the event that the test is unsatisfactory the battery annunciator will remain on.

86. The **warning annunciator** illuminates when a system malfunction occurs, in this event refer to the status display, which will give a number that will identify the nature of the malfunction.

87. The **auto/manual/remote switch** determines the level of pilot intervention necessary to fly the aircraft. In the **automatic mode** the INS will automatically switch from one track to the next as each waypoint is overflown. In the **manual mode** the pilot is required to switch to the next track as a waypoint is overflown. The **remote** position enables the pilot to cross load route information (waypoints) from this INS to a second INS (if fitted) in order to avoid having to load the second INS manually. If this facility is to be used, it is even more important that the accuracy of the waypoints is checked. If a waypoint is incorrectly loaded into one INS and the data is then cross loaded the error will be duplicated. Apart from redundancy (the failure of one of the two systems is allowed for), the main advantage of having two INS is that one monitors the other. If both are loaded with erroneous route data, no discrepancy will exist between the systems (and no warning given to the pilots) despite the fact that the aircraft will at some point be heading towards an incorrect geographical location.

88. The **insert pushbutton** is used in conjunction with the **data input keyboard** to enter information into the system.

89. Finally, the **hold pushbutton** is primarily used for manually updating the INS position with a reliable fix, for example a radio fix or GPS position. The hold button is depressed as the fix position is noted, the function switch is placed in the POS (position) mode, the exact latitude/longitude of the radio fix is punched into the machine, and the hold button is then released. Appreciate that if this is done, the radial error rate assessment (discussed shortly) will be invalid, unless the position update vector is accounted for.

Checking Manually Entered Positions

90. At the initial setting up stage the start position must be fed into the INS computer with a high degree of accuracy.

91. If the initial latitude is slightly in error the platform will not remain Earth horizontal once the equipment is switched into the navigation mode, since the torque motors will be tilting the platform at an inappropriate rate, due to computer calculations based on an incorrect latitude. Likewise, and for the same reasons, the platform will not remain directionally aligned with respect to north. The same problems will apply to wander angle and strapped down systems, but for different reasons.

92. If the initial latitude setting is grossly in error the system will detect the error and warn the operator (this is one of the principal functions of the warning annunciator on the CDU whilst the equipment is in the align mode). The equipment is able to sense a gross latitude input error since the apparent drift and topple rates sensed by the rate gyros will not correspond to the corrections being applied by the torque motors.

93. An incorrect operator input of longitude will not affect the stability of the platform, but obviously the track and distance from the departure point to the first waypoint will be incorrectly computed. Furthermore, all subsequent indications of longitude will be in error by the amount of the initial input error.

94. For the reasons discussed above it is obviously essential that the ramp position is carefully entered and checked (ideally entered by one pilot and checked by the other).

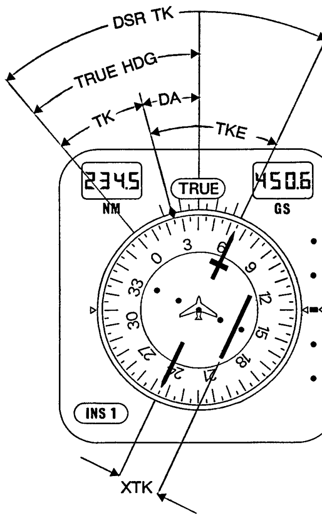
95. An incorrect input of the latitude/longitude of any of the waypoints will have serious consequences. The INS will navigate very accurately between waypoints, but it is incapable of detecting operator malfunctions. A favourite error occurs when punching in a waypoint where either the latitude or the longitude is a whole number of minutes of arc, say 50° 30'N. If you punch in 5030 (instead of 50300) your waypoint is in error by 2,727 nm! It would be unwise to assume that, because a waypoint is stored in the computer memory (data base), it is necessarily correct. A waypoint position may have been incorrectly inserted in the data loader, it may have been corrupted during the data transfer, or it may have been incorrectly entered/amended by one of your colleagues.

96. In order to check that the waypoints have been correctly inserted they should be recalled from store onto the LED display, and rechecked before flight.

97. A second check is to call up the initial great circle track and distances between consecutive waypoints, and to compare these values against those shown on the computer/manual flight plan.

Visual Presentation of INS Data

FIGURE 19-20
INS Data
Displayed on HSI



98. We have previously discussed how INS data is shown on the numerical displays of the CDU. Much of this data may be displayed on either a conventional horizontal situation indicator (HSI) or on an electronic horizontal situation indicator (EHSI) in the map mode (in a glass cockpit aeroplane). HSIs and EHSIs are described subsequently in the chapter entitled Flight Directors and Electronic Flight Information Systems, however for now we will look at a conventional HSI which is coupled to an INS output. Appreciate that the instrument (which is illustrated at Figure 19-20) is essentially a compass rose together with a track deviation indicator. The track deviation indicator can give deviations from tracks defined by VOR, ILS or (as in this case) INS. The vertical scale on the right hand side of the instrument gives glidepath deviations (fly up or fly down) and is active only when an ILS is tuned and selected to the HSI.

HDG. Aircraft heading, in this case 040° true, is displayed against the central lubber index at the top of the display.

TK. The solid diamond, in this case against 025° true, is the present track.

DA. This is the drift angle (the angle between the track diamond and the true heading lubber index line), in this case 15° left drift.

DSR TK. The desired track is indicated by the course bar or arrow, presently showing 065°T.

XTK. The horizontal deviation bar shows the distance that the aircraft is displaced from desired track (the cross track distance) with reference to the five dots. Each dot represents 3.75 nm, and therefore the display shown indicates that the aircraft is to the left of desired track by approximately 7 nm.

TKE. The track error is the angle between the track diamond and the desired track course arrow or bar. In this case the track error is 40° to the left (tracking 025°T, DSR TK 065°T).

99. Distance to go to the next waypoint is indicated in the top left window, and presently-computed groundspeed in the top right window. The window in the bottom left of the display indicates that the HSI is presently coupled to No 1 inertial navigation system, since the INS equipment is often duplicated or even triplicated.

INS Errors

100. The following may be considered, in addition to the Schuler tuning bounded errors already discussed, as likely to produce small but significant errors in the system.

Initial Levelling

101. As already discussed, if the platform is not truly Earth horizontal at the outset (when switched from the align to the nav mode) the accelerometers will sense a component of gravity which will cause a bounded error in groundspeed at the first integration stage and an unbounded error in distance travelled at the second integration stage. With a strapped down system, a similar error will exist in the event that any difference which exists between Earth horizontal and aircraft horizontal is not correctly assessed during the initial levelling/alignment procedure.

Initial Alignment

102. Again, as already discussed, if the platform is not correctly aligned with north at the outset the direction of acceleration sensed will be in error by the amount of misalignment. This will give rise to a bounded error in all track made good computations, leading to an unbounded error in indications of aircraft position. With a wander angle or strapped down system a similar error will exist in the event that the angular difference between the platform/aircraft fore and aft axis and the local meridian (true north) is incorrectly computed during the initial alignment procedure.

Real Wander of the Rate Integrating Gyros

103. The rate integrating gyros used on inertial platforms are of a very high order of accuracy, and are normally required to give real wander rates (those due to engineering imperfections) of less than 0.01°/hr. Despite this high degree of accuracy it is these real wander rates within the gyros which give rise to the most significant unbounded errors in the system. The ring laser gyros which are used with strapped down systems do not suffer from any form of mechanical precession.

Accelerometer and Integrator Errors

104. Slight imperfections in these components can give rise to errors at all stages of computation. These errors are normally much smaller than those caused by real wander of the rate gyros.

Latitude and Height Errors

105. The INS computer is programmed to correct for the varying length of the nautical mile (one minute of arc of latitude at the surface) due to the shape of the Earth. Similarly, aircraft altitude will also introduce small distance errors and, on platforms which are designed to correct for this error, it is necessary to employ a third accelerometer with its sensitive axis lying along the platform/Earth vertical axis, so that the INS computer can integrate vertical accelerations to determine altitude. All strapped down systems necessarily employ an aircraft vertical accelerometer, the output of which is integrated to give aircraft altitude.

System Accuracy

106. The minimum standards of accuracy which are specified for inertial navigation systems require a maximum circular position error rate of 2 nm/hr (on 95% of occasions) on flights of up to 10 hours duration. For flights of over 10 hours duration a cross track error of ±20 nm and an along track error of ±25 nm (on 95% of occasions) is permitted.

107. Modern systems achieve accuracies which are well within these limits and terminal errors of less than 1 nm after 10 hour flights are common, using integrated navigation systems which are discussed shortly. Large errors can occur in individual inertial systems following component failure, however INS equipped aircraft normally employ two (737) or three (747, 757 and 767) independent inertial systems, the outputs of which are continuously compared in order to identify system malfunctions.

Radial Error Rates

108. At the end of each flight the actual ramp position should be checked against the indicated ramp position shown on the INS. A **radial error rate** may then be calculated using the formula:

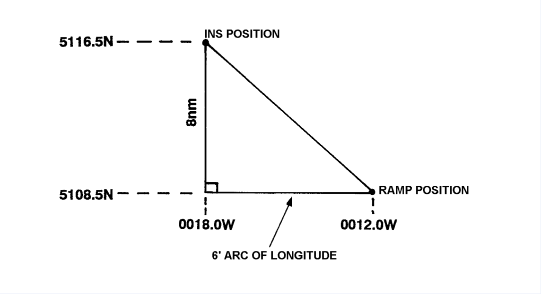
$$\textbf{Radial error rate (nm/hr)} = \frac{\textbf{Distance ramp position to INS position (nm)}}{\textbf{Time in the navigational mode (hours)}}$$

EXAMPLE

Following a flight from New York to London the INS showed a position of 51° 16.5'N 00°18.0'W when the aircraft was stationary on the ramp at London Gatwick. The ramp position was given as 51° 08.5'N 00° 12.0'W. The time in the navigation mode was 5 hours and 24 minutes. Using this information determine the INS radial error rate.

SOLUTION

Whilst the INS computer bases all calculations on spherical trigonometry, the human solution of radial rate error can be achieved to a satisfactory degree of accuracy using two-dimensional trigonometry.



The distance in longitude between the INS position and the ramp position is calculated using the departure formula:

$$\text{distance (nm)} = \text{d long (')} \times \cos\ \text{(mid) lat}$$

$$= 6 \times \cos 51°12.5'$$

Using Pythagoras:

$$\text{distance ramp to INS position}^2 = 8^2 + 3.76^2$$

$$= 78$$

$$\text{distance ramp to INS position} = \sqrt{78}$$

$$= 8.8 \text{ nm}$$

$$\text{the radial error rate} = \frac{8.8 \text{ nm}}{5.4 \text{ hrs}}$$

$$= 1.63 \text{ nm/hr}$$

If, in the examination, you are not given the value of the cosine of the precise mid latitude, use the cosine of the nearest whole degree from the table provided.

109. The examiner is also fond of asking questions of the type illustrated in the following example. Providing that you appreciate that inertial systems always navigate along great circle tracks, you should have no difficulty in answering this type of question.

EXAMPLE 19-2

EXAMPLE

An aircraft is flying from waypoint 4 at 35° 00.0'S 20° 00.0'W to waypoint 5 at 35° 00.0'S 30° 00.0'W. The INS is programmed to display tracks and headings with reference to true north. The INS is coupled to the flight control system/autopilot. Using this information answer the following questions:

1) With DSR TK/ST selected on the CDU the desired track readout which is displayed as the aircraft overflies waypoint 4 is:

(a) 270.0°

(b) 090.0°

(c) 272.9°

(d) 267.1°

2) With DIS/TIME selected on the CDU the distance to go which is displayed as the aircraft overflies waypoint 4 is:

(a) 491.5

(b) 600

(c) less than 491.5

(d) more than 491.5

3) With POS selected on the CDU and the longitude readout showing 25° 00.0'W the latitude readout will show a latitude which is:

(a) 35° 00.0'S

(b) south of 35° 00.0'S

(c) north of 35° 00.0'S

4) With XTK/TKE selected on the CDU, when the aircraft is mid way between the waypoints the cross track readout distance will show:

(a) 00.0

(b) R05.7

(c) L05.7

5) The total change in the track angle between waypoints 4 and 5 is:

(a) less than 10°, decreasing

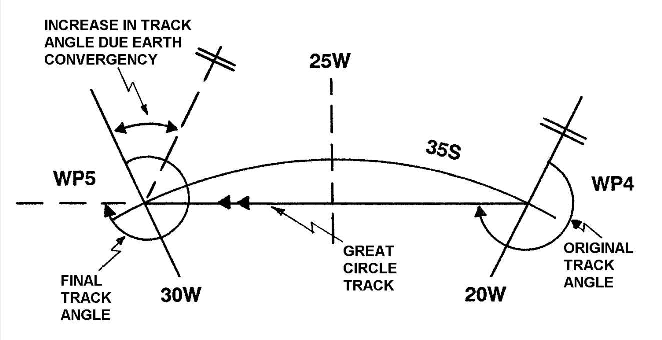
(b) more than 10°, decreasing

(c) less than 10°, increasing

(d) more than 10°, increasing

SOLUTION

1) The answer is (d). The parallel of latitude defines 270°(T), whereas the INS flies the great circle track which lies to the south of the parallel. There is no need to calculate the value of the conversion angle since only one of the options given can be correct.



2) The answer is (c). The distance of 491.5 nm, which is achieved by means of the departure formula, represents the distance between the waypoints along the parallel of latitude (the rhumb line distance). The great circle distance must therefore be less than 491.5 nm.

3) The answer is (b), as shown above.

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4) The answer is (a). With the INS coupled to the autopilot the aircraft will remain on the great circle track and there will be no cross track displacement.

5) The answer is (c). Earth convergency increases the track direction as the aircraft flies west. The total change of track direction is obviously less than 10° since the change of longitude is only 10° and Earth convergency = change of longitude x sine latitude.

The Development of Inertial Systems

110. The preceding notes have dealt with inertial systems as stand alone aids to aircraft navigation. Furthermore, for the sake of simplicity, the mode selector panel, control display unit and associated computer hardware/software which were considered were of a fairly early vintage. Over the last 20 years aircraft navigation systems have evolved very rapidly in parallel with other aircraft systems, which is why the 747-400 was designed to be operated with only two people on the flight deck.

111. What follows is a potted history of the way in which inertial navigation, as it applies to civil transport aeroplanes, has evolved.

112. The first INS systems used north aligned platforms and relatively slow processors with limited memory banks. Reliability was a problem and the limited storage capacity of the computer memory meant that the system was limited to 10 waypoints which had to be replaced in flight as they were overflown on routes which required more than 10 waypoints. It was not possible at this time to store the look up table of variation and consequently the system operated only in °(T).

113. Faster processors and larger memory banks meant that the next generation of inertial systems employed wander angle platforms and had the ability to display track and heading information in degrees magnetic. This is about where the system which we discussed fits into the story.

114. Still larger memory capacities enabled the next generation of INS to 'store' typically a thousand waypoints which could be used to produce up to something in the order of 100 routes, which could also be stored in the memory. By now the systems were becoming more reliable and duplicated INS platforms were replacing the navigator on even long oceanic sectors.

115. With the advent of strapped down systems, very high speed processors and very large memories the INS has evolved into the IRS, the inertial reference system. Now the inertial system is no longer a stand alone equipment, but rather the inertial input into an integrated system (the flight management system or FMS) which not only looks after all of the navigation requirements but also has many other functions.

116. With two or three highly reliable IRS in the system it is now possible to dispense with the gyro slaved compasses and the remote gyro compasses, since each IRS is capable of giving accurate outputs of aircraft heading and attitude, the outputs of each IRS being constantly monitored against the other(s).

117. Updating of the position determined by the inertial reference systems is now continuously achieved by the navigation module within the active flight management computer (obviously redundancy is built in here as well), providing that the aircraft is within range of suitable VOR/DME stations. The updating process is achieved using either VOR/DME or preferably DME/DME fixes which are more accurate, especially when slant range corrections are applied. Suitable VOR/DME stations are selected by the navigation module using a map of suitable stations which is automatically loaded into the active memory when the pilot types in the route number prior to departure.

118. The number of waypoints which comprise an individual route is now virtually unlimited. The basic route is entered, followed by the relevant standard instrument departure once the clearance is received. Similarly, the standard instrument approach is entered once this is known.

119. VOR/DMEs are automatically selected and the aircraft is navigated using composite data comprising the IRS outputs as updated by DME/DME or VOR/DME inputs. With a glass (EFIS) cockpit the normal mode for the screen which replaces the horizontal situation indicator (HSI) is the map mode, which gives a pictorial representation of the route as defined by the IRS waypoints; the VOR/DME stations which are stored in the navigation module (the stations currently in use are highlighted); the aircraft's present position; the output of the airborne weather radar and various other codes and symbols to indicate the operational status of the system. Navigation errors do occur, and when these are due to operator errors (principally the incorrect insertion of data) they may not be identified by the system, which cannot therefore warn the pilots. The role of the pilots as the final arbiters of the integrity of the system is absolutely paramount.

120. In addition to providing accurate navigation data, strapped down inertial systems have now replaced the remote gyro compasses and the vertical gyros and supply heading, pitch and roll information to the pilots (human and automatic). In the event an individual IRS suffers a component failure in flight which means that it is no longer capable of supplying position information (for example an integrator), or is deprived of its AC power input whilst airborne for a period which exceeds the duration of the internal battery pack, it might appear that the system will no longer be capable of supplying heading or attitude data. By selecting the attitude reference position the IRS reverts to a basic mode which isolates the navigation functions and supplies only heading and attitude data, albeit of a degraded accuracy. The important thing to appreciate is that, in the basic mode, the system is capable of achieving an acceptable level of accuracy following an airborne levelling/alignment process, but that the information supplied by the system is now limited to aircraft heading and attitude.

121. What comes next? With the advent of satellite navigation systems which are capable of determining aircraft position which is accurate to within metres anywhere in the world and for which the aircraft equipment is light, simple (reliable) and inexpensive, the future of inertial reference systems which provide the primary navigation data would appear to be limited. It is likely that, in the not too distant future, strapped down systems will be used to supply attitude and heading data (AHRS, or attitude and heading reference systems), whilst satellite systems will be used to supply primary navigation data.

The Radio Altimeter

The Radio Altimeter

1. The radio altimeter is designed to indicate to the pilot the height of the aircraft above the ground immediately beneath the aircraft. The indication is accurate and effectively instantaneous, however the radio altimeter gives **NO INDICATION OF HIGH GROUND AHEAD OF THE AIRCRAFT.**

Principle of Operation

2. The principle of operation of the equipment is that a frequency modulated carrier wave is continuously transmitted vertically downwards from the aircraft. The beam is directed downwards with a nominal width of between 20° and 40° with possibly a greater width in the athwartships plane to allow for aircraft bank. The reflected signal returning from the ground is received at a second aerial. Because the frequency of the transmitted signal is continuously changing, and because the signal will take a finite time to return via the surface to the aircraft, the received frequency will differ from the frequency of the signal being transmitted at any instant in time. The rate at which the transmitted frequency changes is a known constant. The difference in frequency between the signal being transmitted and the signal being received at any point in time can therefore be related to the time taken for the signal to travel from the aircraft to the surface and back to the aircraft.

3. If the journey time is known, and the speed of propagation is a known constant, then the distance travelled is easily calculated. In this case the distance travelled is equal to twice the height of the aircraft.

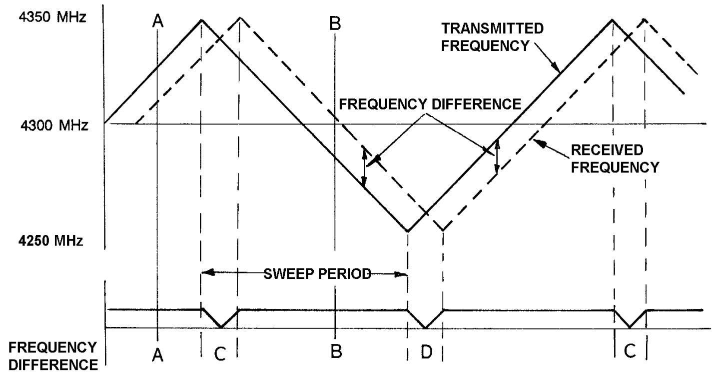
4. The frequency range used for radio altimeters is between 4200 MHz and 4400 MHz (in the SHF band). The frequency range of 1600 MHz to 1700 MHz (in the UHF band) is set aside for radio altimeters, but is not used commercially.

5. Modulation over the swept band of frequencies occurs, typically, 500 times each second. This relatively low value of sweep rate is necessary in order avoid the possibility of ambiguity of height readout at the face of the instrument.

6. Figure 20-1 shows schematically what is happening. For the reasons already discussed the received frequency will be different from the transmitted frequency at a given point in time. At time A in Figure 20-1 the transmitted frequency is increasing as time passes and therefore the received frequency will be lower than the transmitted frequency. At time B the transmitted frequency is decreasing as time passes and therefore the received frequency will be higher than the frequency being transmitted BUT THE DIFFERENCE BETWEEN TRANSMITTED AND RECEIVED FREQUENCIES HAS REMAINED CONSTANT.

7. During the time spans annotated C and D at Figure 20-1 the difference in frequency between transmitted and received signals is changing rapidly and is disregarded by the equipment.

FIGURE 20-1
Radio Altimeter
TX and RX
Frequency Profiles



8. Please appreciate that, in order to give meaning to Figure 20-1, it has been necessary to show a significant change in frequency. In fact the actual change in frequency is relatively small, being measured in kilohertz rather than megahertz, especially when the aircraft is close to the ground.

Components of the Radio Altimeter

9. A typical radio altimeter installation comprises a transmitter/receiver, separate transmit and receive aerials plus an Indicator (the Indicator may have additional repeaters at other crew positions in the aircraft).

10. A typical display is shown at Figure 20-2. Note that the maximum height displayed is 2500 ft (this is a typical value for a low level radio altimeter), and also that the scale is linear between zero and 500 ft but logarithmic between 500 and 2500 ft.

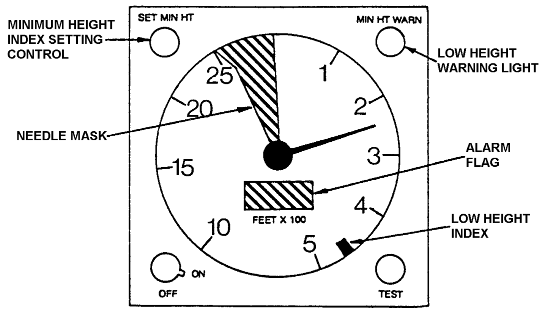


FIGURE 20-2
Typical Low Level Radio Altimeter Display

11. The needle is hidden from view behind the mask whenever:

(a) The equipment is switched off, or a power failure has occurred.

(b) A fault exists within the equipment.

(c) The aircraft is flying above 2500 ft.

The prominent alarm flag will be visible whenever:

(a) The equipment is switched off or a power failure has occurred.

(b) A fault exists within the equipment.

(c) The returning signal is subject to an unacceptable level of noise.

(d) Signals are reflected from the airframe itself.

12. Depressing the press to test button will cause the pointer to swing to a known preset reading, indicating that equipment is operating satisfactorily.

13. The low height warning light will illuminate whenever the needle is showing a height which is below that selected using the minimum height index setting control and indicated by the low height index marker (shown at 450 ft in Figure 20-2). The aircraft fit may be such that an audible warning is given in conjunction with the warning light.

Decision Height Warning Light

14. Some radio altimeter indicators are fitted with a **Decision Height (DH) selection knob.** The DH setting knob may also incorporate a press-to-test facility however its main function is to position a DH bug at the required setting. When the aircraft descends below the set decision height, a **DH warning lamp** illuminates, and in some equipments an audio tone is also generated.

Accuracy

15. The radio altimeter is deemed to have an accuracy in the order of plus or minus 3% of height or plus or minus 1 foot, whichever is the greater.

Errors of the Radio Altimeter

Transmitter to Receiver Leakage. Any leakage of RF energy between a transmitting aerial and a receiving aerial, across the surface of an airframe, may cause 'noise' or an erroneous reading on the Radio Altimeter. Installation requirements are therefore critical in systems which are used to feed radio altitude to autoland or Ground Proximity Warning Systems.

Reflections. Signal reflections from an undercarriage or other protruding surface may add to the problem described in the preceding sub-paragraph.

Multipath Signals. It is possible, occasionally, to receive a signal which has been reflected by the ground back to the aircraft, and then is reflected off the airframe and travels back to the ground for a second time before entering the receiver.

Use of Radio Altitude Information

16. Primarily radio altitude information is fed as a key input into the Ground Proximity Warning System (GPWS), Autoland System and Flight Management Computer (FMC) of large commercial airliners. Other equipments may require such information depending on the avionic fit of the aircraft.

Flight Management Systems

Flight Management Systems

1. The **Flight Management System (FMS)** is an integration of the aircraft subsystems, the purpose of which is to assist the flight crew in controlling and managing the flight path of the aircraft. The flight path is divided into **lateral and vertical profiles**, commonly known as LNAV and VNAV. The system allows the pilots to select the degree of automation required at all stages of flight and consequently the need for many routine tasks and computations is eliminated.

2. Primarily the FMS provides automatic three-dimensional navigation, fuel management and fuel monitoring together with the optimising of aircraft performance. It also provides information to the appropriate displays, including the electronic map, which is fully described in the section dealing with Flight Directors and Electronic Flight Information Systems (EFIS). FMS also provides airspeed and engine thrust cues.

3. The main components of an FMS are :

(a) **Flight Management and Guidance Computer (FMC)**

- uses both manual and automatic inputs of data to compute 3 dimensional position, performance data etc in order to fly the aircraft accurately and efficiently along a pre-defined route.

(b) **Multipurpose Control and Display Unit (MCDU)**

- the interface between the pilots and FMC.

(c) **Flight Control Unit**

- supplies the commands to control the lateral and vertical flight path of the aircraft.

(d) **Flight Management Source Selector**

- selects the sources of input to be used by the FMC.

(e) **Display System**

- any means of displaying the required data/ information to the pilots.

The Flight Management and Guidance Computer

4. A schematic diagram of the component parts of a typical flight management system is shown at Figure 21-1. The heart of the system is the **Flight Management Computer** (**FMC**) and its associated **Multipurpose Control and Display Unit** (**MCDU**)**.** A CDU of the type found in the Boeing 737 is illustrated at Figure 21-2.

FIGURE 21-1
A Typical FMS

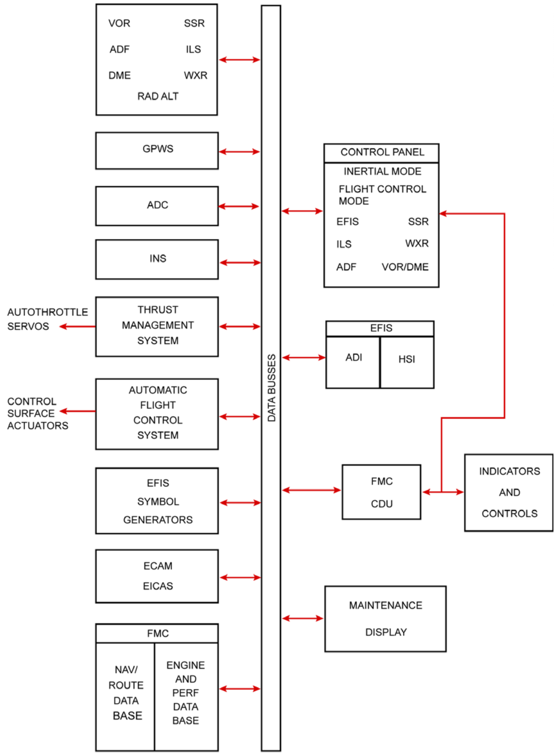
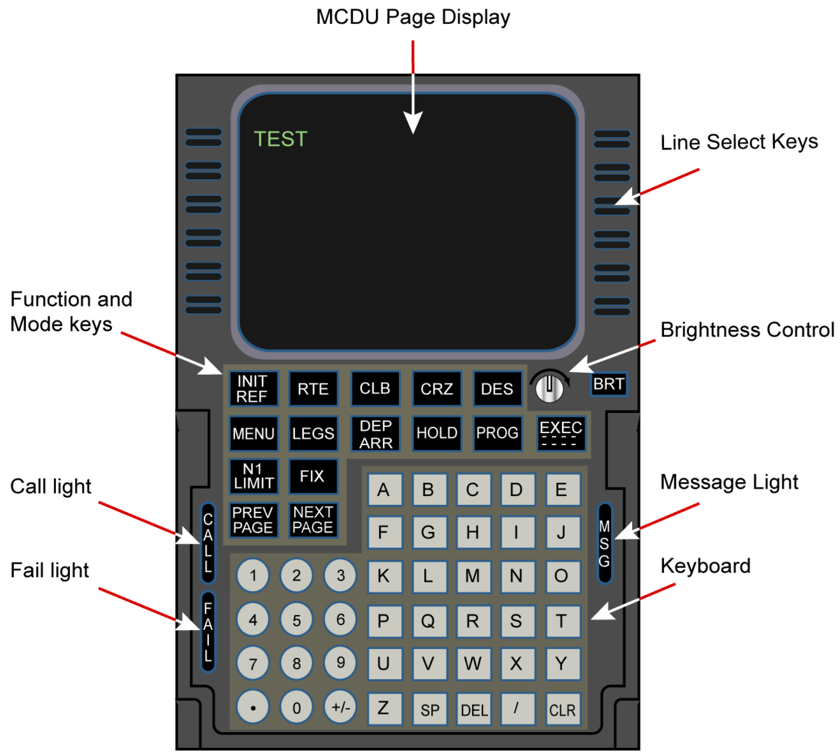


FIGURE 21-2
A Typical FMS Multipurpose Control and Display Unit (MCDU)



5. The MCDU combines flight plan information entered by the pilots with information supplied from supporting systems and information contained in memory. This enables the FMC to determine the aircraft position and to provide pitch, roll and thrust information in order to fly the profile required. Commands are sent by the FMC to the autopilot, the flight director and the autothrottle (autothrust) system. FMC navigational and performance computations are displayed on the MCDUs for reference or monitoring. Related FMC commands for lateral and vertical navigation are coupled to the AFDS and Autothrottle through the Mode Control Panel (LNAV and VNAV). The IRSs and other aircraft sensors provide additional required data. MCDUs also permit interface with the Aircraft Communications Addressing and Reporting System (ACARS). Additionally, map information is sent to the Electronic Horizontal Situation Indicator (EHSI) and displayed in the manner described in the section dealing with EFIS.

The FMC Data Base

6. The information which is stored in the FMC data base is divided into two main sections, namely **navigation information** and **aircraft performance information.**

7. The navigation data includes the location of radio navigation aids, SIDs, STARs, company routes, airports, runways, approach aids and airways structures. The data base is tailored to the needs of the individual carrier. This navigation data base is produced by a specialist agency (such as Jeppesen) and is normally updated on a **28 day cycle**. Data transfer hardware (using a magnetic tape cassette) is provided to enable the operator to load a new data base into the aircraft FMCs. In order that flight operations do not come to a grinding halt at midnight on the last day of validity of the expiring data base, the current data base together with the next effective data base are both stored in the FMCs. For the pilot then, step one when setting up the FMCs is to ensure that **the correct data base for the date of the flight is the operational one.**

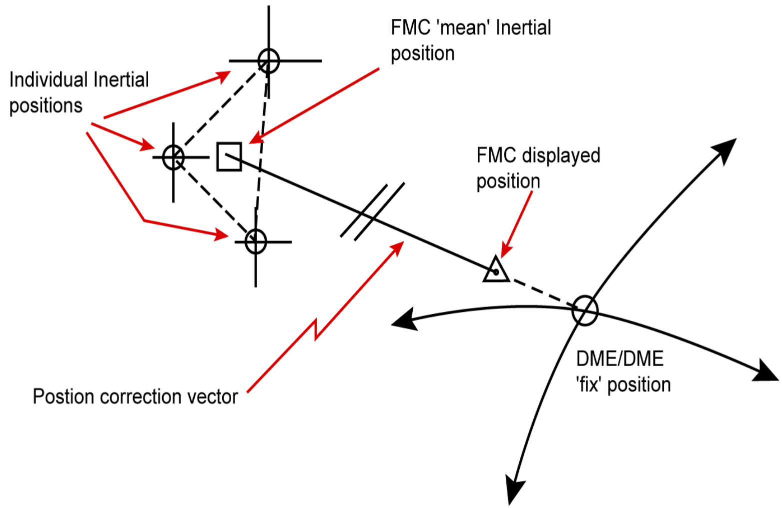
8. Within a given 28 day period it is likely that certain information contained in the navigation data base will become invalid, for example NOTAMs may inform us that a given VOR is out of service for a period of time. The pilot can access the data base and delete that VOR, but only for the duration of the flight. It is therefore impossible for the pilot to corrupt the data base itself. It is important to remember that the data base is produced by another human being and may therefore contain errors. Because of the high degree of automation involved when, basically, the FMC is driving the aeroplane, it is essential that the pilots monitor the aircraft's progress using conventional navigation techniques (raw data), and also that any errors in the data base are fed back through reporting channels so that they can be remedied.

9. During flight the FMC will search the navigation data base and automatically select the best two DME stations with which to determine the aircraft's present position. In the absence of suitable DME/DME crosscuts the system will use co-located VORs and DMEs. When DME/DME or DME/VOR fixing is not possible, for example on an oceanic leg, the aircraft's position is determined by the inertial reference systems plus a correction vector that has been developed by a **Kalman filter** over a period of time. In those systems that use GPS position as an input into the FMC, it is usually possible for the pilot to delete any satellite that has automatically been selected by the GPS receiver, in order to obtain the best fix geometry.

10. The Kalman filter uses **hybrid navigation techniques.** It takes, for example, position information from a number of sources and then statistically analyses that data (taking into account the possible errors) to produce a final solution which, in the case of position, would be the FMC position. The filter also produces the correction vector discussed in Paragraph 9.

11. Take the situation where an aircraft, equipped with say 3 inertial systems, is flying from Europe to the USA. As the aircraft crosses the UK, on its way to join the NAT track system, the FMS will be using DME/DME radio ranges to assist in determining position. Figure 21-3 gives a pictorial presentation of the computations involved.

FIGURE 21-3
Position Determination by an FMC



12. In simple terms the FMC first averages out the 3 IRS positions to determine a 'mean' inertial position. Secondly, it compares the mean inertial position with the radio aid position (in this case DME/DME ranging is used) and, taking account of the likely error in each position, it computes a final FMC position which is used to steer the aircraft along the planned track.

13. The **position correction vector** in the above example stretches between the mean inertial position and the final FMC computed position. (In an aircraft equipped with a single inertial system the vector would obviously start from that single position).

14. It will be obvious from the above explanation that, in order to develop the position correction vector over a given time, there must be a continuous supply of radio information. However, once the aircraft leaves the area of ground based radio aids the FMC can still use the 'history' of the vector to develop it further, and hence continues to provide the best possible estimate of position. As the aircraft coasts in again over the USA radio aid fixing will once again be used to 'tie down' the FMC position.

15. The accuracy of a Kalman filtering system such as the one described is dependent upon two main factors :

(a) The quality and complexity of the Kalman filter design.

(b) The error characteristics of the various 'navigation' sensors used by the system must be complementary. (i.e. any single system input which is subject to a lot of 'noise/variation', or 'drifts' in value, may cause a significant error in FMC computed position).

16. The FMCs will automatically select the VOR/DME stations which are displayed on the EHSI needles, the standby RMI needles and the DME range readouts. The system will decode the morse identifier and display letters on the screen. If a satisfactory identifier decode is not achieved, the frequency will be displayed rather than the identifier. In this event it is up to the pilot to identify ground station in the conventional manner. Similarly, providing that the FMC has been informed that the intention is to fly an ILS approach to a given runway at the destination/alternate aerodrome, the

relevant ILS will be autotuned and identified, again with the morse identifier displayed to the pilot, but this time on the Electronic Attitude Direction Indicator (EADI). Where the departure is from an ILS runway, the FMC will again autotune the ILS in order to provide centre line guidance immediately after take-off. When NDBs form part of a SID, STAR approach procedure or (unusually these days) an airways structure, these are also autotuned and identified by the FMC. The option always exists for the pilot to override the automatics by 'hard tuning' stations of his or her choice.

17. The performance data base contains all of the information normally contained within the performance manual, such as engine characteristics, the aircraft limiting speeds for the various configurations, optimum/maximum cruise altitudes and an aerodynamic model of the aeroplane. The data base may be individually tailored for an individual aeroplane within a fleet. Variables such as fuel quantity, zero fuel weight and a company cost index are entered by the flight crew. **This data is peculiar to the next sector only and is automatically dumped by the FMC following the next landing and engine shutdown.** The simplest explanation of the cost index is that it is a numerical value which tells the FMC whether the operator considers that fuel economy (with longer sector times) or minimum sector times (with a resultant higher fuel burn) is the preferred option. Please note that fixed costs, as a general rule, remain the same no matter what speed is flown. The cost index can therefore be altered on a sector by sector basis to account for the circumstances of that flight.

Modes of Operation for Dual FMC Installations

18. FMC systems are normally duplicated and each FMC has its own CDU. There are 4 modes of operation :

Dual Mode. With the system operating normally the two CDU/FMCs are interconnected and pilot entered data which is entered at one CDU is automatically transferred to the other one. In other words, one FMC provides the master function and the other the slave function. The pilots may select their own EHSI display (full or expanded VOR, full or expanded ILS, map or plan) regardless of what is displayed on the other EHSI.

Independent. The first stage of degradation of the system occurs when a disparity is sensed between the outputs of the two FMCs. Now each CDU/FMC works **independently** of the other and the pilots are left to identify the serviceable system. Each CDU will supply its own EHSI, however now the pictures on each of the EHSIs (assuming that they are in the same mode with the same range option selected) will differ.

Single. The next stage of degradation of the system when one FMC or CDU fails altogether. You are now down to a **single system operation**, however both EHSIs can be driven from the same FMC/CDU providing only that both pilots select the same mode and range setting.

Back-Up Navigation. Finally, should both FMC/CDUs fail, the pilots are left with blank EHSIs and the prospect of limited use of the FMS. Navigation is achieved by manually tuning en route and approach aids which are subsequently displayed on a conventional RMI and analogue DME readout.

19. The control of the different FMS modes is described in the Aircraft Operation Manual (AOM)

Lateral Navigation Guidance

20. The FMC calculates the great circle tracks and distances between successive waypoints in the active flight plan. These are the track lines which are shown on the EHSI map display. The active flight plan includes the SID, the STAR and any relevant holding patterns. Under normal circumstances (**managed guidance**) the FMC will command the autopilot to maintain the defined track (at a particular altitude and speed). With the aircraft flown manually (**selected guidance**) the FMC commands the human pilot to maintain a particular value of a parameter (heading, speed etc) by making selections on the Flight Control Panel (FCP). **At any time the pilot can take control of the lateral navigation** of the aircraft by going into heading mode. The FMC will automatically revert to heading mode whenever LNAV capture parameters are out of limits or when, for example, a waypoint is reached and no route is defined beyond that point.

Vertical Navigation Guidance

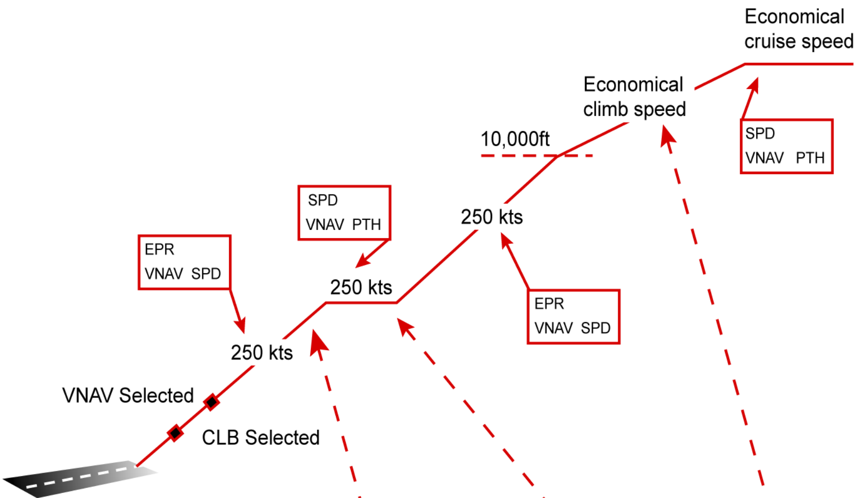
21. Providing that the pilot does not modify the climb profile, the FMC will command a climb with thrust at the airspeed limit associated with the departure airfield until above the speed limit altitude or flight level. Thereafter the climb will continue at climb thrust and **economy speed** to the demanded cruise level. Where altitude/level constraints are imposed by the SID (cross point X at/at or below/at or above a given altitude or flight level), these constraints will be shown on the EHSI map and plan displays. The aircraft will comply with these constraints providing that the FMC remains in the **fully managed mode.** In the event that ATC impose an altitude constraint, this can be entered by the pilot as a vertical revision to the waypoint to which the constraint applies. If, during the climb, the FMC senses that the aircraft will not be able to comply with the constraint due to an insufficient rate of climb, the pilot will be warned. The FMC will capture any altitude which is selected and armed on the Mode Control Panel (MCP)/Flight Control Unit (FCU).

22. During the cruise, **economy speed** will be used until the top of descent point.

23. The top of descent point is computed by the FMC, as it were, from touchdown backwards. The FMC has knowledge of the aerodrome elevation, and the QNH is manually entered by the pilots. The exact vertical distance from the cruise level to touch down is therefore known. Flight level or altitude constraints, as defined by the STAR and the approach procedure are stored in the navigational data base, and the descent profile is computed to account for these constraints. The descent will normally be computed such that, wherever possible, the engines will be at idle power (which is fuel efficient). The descent will be computed at **economy speed** down to the point where the STAR imposes a maximum speed constraint, and thereafter at speeds which will enable the slats/flaps/landing gear to be extended at the appropriate points. Wind velocities for the descent can be manually entered by the pilots in order to refine the computation.

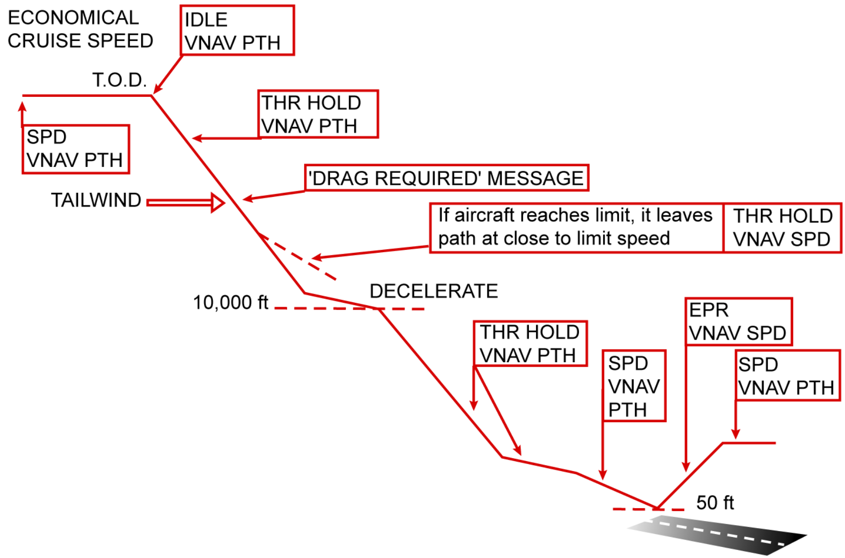
24. Typical VNAV climb, cruise and descent profiles for a B757 are illustrated at Figure 21-4 and Figure 21-5.

FIGURE 21-4
Typical VNAV Climb / Cruise Profile



waypoint constraint	Cross TMC above 4000 ft	Cross LMN at 7600 ft	Cross SYY above 20,000 ft

FIGURE 21-5
Typical VNAV Descent Profile



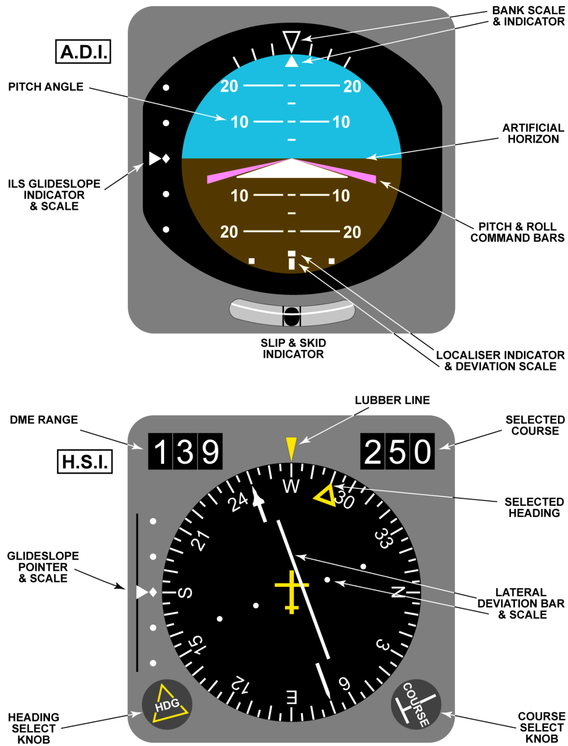
waypoint constraint	Cross RP4 above 4000 ft	Cross at FAF ALT and 170 KTS	Missed APPR ALT

Flight Director Systems

25. Flight director systems (FDS) integrate the information presented by traditional flight instruments (artificial horizon, turn and bank, gyrocompass) with the information received from external sources (VOR, DME, ILS, Radio Altimeter) to produce control commands. This achieves more accurate flight guidance and control whilst reducing the pilot's workload in terms of monitoring and co-ordinating the many individual sources of information. The FDS presentation is in the form of two displays, an **attitude direction indicator** (ADI) and a **horizontal situation indicator** (HSI). The ADI presents flight guidance commands in pitch and roll, whilst the HSI presents the navigational situation. The two displays are shown at Figure 21-6. Note the command bars on the ADI shown at Figure 21-6. An alternative presentation of the command bars are shown on the ADI illustrated at Figure 21-10.

Flight Management Systems

FIGURE 21-6
'Conventional' ADI and HSI Displays



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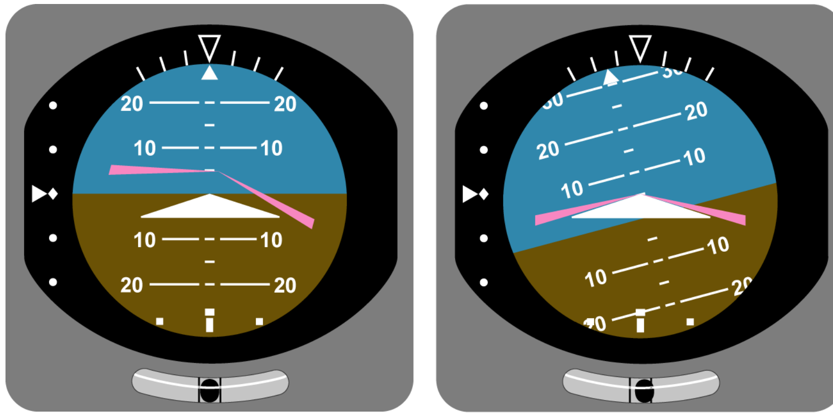
26. The student may have difficulty in visualising the display movement of the ADI in Figure 21-6, hence two diagrams are shown at Figure 21-7 which illustrate the use of the equipment. The first diagram shows commands of 5° pitch up and a right turn; the second diagram shows both commands having been satisfied and where 15° of right bank is displayed.

FIGURE 21-7



Pitch & roll command unsatisfied   Pitch & roll command satisfied

27. Flight director systems have existed for many years as analogue instruments, indeed it is an analogue system which is shown at Figure 21-6. In this chapter we will subsequently consider a system which employs modern glass cockpit technology and which therefore presents far more information to the pilots. Flight Director Systems, as such, are examined more fully in Paragraph 22.

Electronic Display Systems

28. To display all the necessary information and data concerned with in-flight management of the aircraft systems would demand a vast array of instrumentation, impossible for a typical two or three-person flight-deck crew to comprehensively monitor. Furthermore, much of the data is only relevant at certain flight phases or in particular circumstances and therefore need not be permanently displayed.

29. This has led to the development of electronic display systems (the glass cockpit) in which the data is processed and stored by large capacity computers and displayed as required on colour CRT screens in either alphanumeric form or as symbols. The following colours are being recommended in JAR 25 based on current-day common usage. Deviations may be approved with acceptable justification.

(a) Display features should be colour coded as follows:

Warnings	Red
Flight envelope and system limits	Red
Cautions, abnormal sources	Amber/Yellow
Earth	Tan/Brown
Engaged modes	Green
Sky	Cyan/Blue
ILS deviation pointer	Magenta
Flight director bar	Magenta/Green

(b) Specified display features should be allocated colours from one of the following colour sets:

	Colour Set 1	Colour Set 2
Fixed reference symbols	White	Yellow*
Current data, values	White	Green
Armed modes	White	Cyan
Selected data, values	Green	Cyan
Selected heading	Magenta**	Cyan
Active route/flight plan	Magenta	White

The extensive use of the colour yellow for other than caution/abnormal information is discouraged. In colour Set 1, magenta is intended to be associated with those analogue parameters that constitute 'fly to' or 'keep centred' type information.

(c) Precipitation and turbulence areas should be coded as follows:

Precipitation	0 - 1 mm/hr	Black
	1 - 4 "	Green
	4 - 12 "	Amber/Yellow
	12 - 50 "	Red
	Above 50 "	Magenta
Turbulence		White or Magenta

(d) Background colour: (Grey or other shade) Background colour may be used to enhance display presentation

30. The screens can be arranged to display primary information on a continuous basis, with fault or emergency information superimposed as necessary. ‘Programmes’ of secondary information can be called up and displayed when required.

31. The displayed data falls into two broad categories; navigational and aircraft systems.

32. The computer-generated electronic displays which show the navigational data are jointly known as the Electronic Flight Instrument System (EFIS). The upper screen shows the ADI whilst the lower screen shows the HSI, either in a format similar to that shown at Figure 21-6 or in one of the pilot selectable formats subsequently discussed.

33. The computer-generated electronic displays which show the aircraft systems are jointly known as either the Engine Indicating and Crew Alerting System (EICAS) or as Electronic Centralised Aircraft Monitoring (ECAM). Basically, EICAS is a Boeing term whereas ECAM is an Airbus term.

34. In the remainder of this section only the EFIS portion of the total electronic display system is considered.

Self Assessed Exercise No. 3

QUESTIONS:

QUESTION 1.

The principle of operation of the radio altimeter, and the frequency band in which it operates, are.

QUESTION 2.

In order to avoid the risk of height ambiguity in a radio altimeter a high/low/linear sweep rate is used.

QUESTION 3.

The radio altimeter receives information from the ground directly beneath the aircraft. What are the dimensions of the radio altimeter beam.

QUESTION 4.

What frequency do radio altimeters work at:

QUESTION 5.

What upper height limit does a typical low level radio altimeter work up to:

QUESTION 6.

What is the quoted accuracy of a radio altimeter.

QUESTION 7.

What is the cause of Multipath Error in a Radio Altimeter

QUESTION 8.

Name an equipment that would use radio altimeter information as an input.

QUESTION 9.

What is the purpose of the Flight Control Unit in an FMS.

QUESTION 10.

Referring in the Reference Book to FIGURE 241 of an MCDU, what are the items identified as (b), (d) and (g).

QUESTION 11.

In the absence of any other constraints, what speed will the FMS use to compute the flight profile (excluding SIDs and STARs), and why.

QUESTION 12.

How do fuel costs vary in relation to speed (ie. sector time)

QUESTION 13.

When operating in SINGLE mode, what constraints are there on the use of EHSI screens by the pilots.

QUESTION 14.

When variables such as fuel load, zero fuel weight and company cost index are input by the crew into the performance database of an FMC, how long are they valid for.

QUESTION 15.

Describe the operation of a twin FMC system when operating in INDEPENDENT mode.

QUESTION 16.

Briefly describe how the top of descent point is calculated by an FMC.

QUESTION 17.

Where would you find the instructions for controlling the different FMS modes.

QUESTION 18.

What are the general guidelines regarding FMS monitoring.

QUESTION 19.

The correction vector produced by a Kalman filter in a Flight Management Computer (FMC) can be described as follows:

QUESTION 20.

What does the term managed guidance mean, in connection with FMC operation.

ANSWERS:

ANSWER 1.

Principle of operation - Frequency Comparison

Frequency band - SHF

CH20 P20-1 Paras 2 + 4

ANSWER 2.

A low sweep rate is used

CH20 P20-1 Para 5

ANSWER 3.

The radio altimeter beam is an ellipse 20° in the fore and aft axis and 40° in the athwartships axis beneath the aircraft.

CH20 P20-1 Para 2

ANSWER 4.

4.2GHz to 4.4GHz

CH20 P20-1 Para 4

ANSWER 5.

2500ft

CH20 P20-1 Para 10

ANSWER 6.

± 3% of the height, or ± 1ft whichever is the greater.

CH20 P20-4 Para 15

ANSWER 7.

Occasionally, the radio altimeter may receive a signal which has been reflected by the ground back to the aircraft, and then is reflected off the airframe and travels back to the ground for a second time before entering the receiver.

CH20 P20-4

ANSWER 8.

GPWS, Autoland System or an FMC.

CH20 P20-4 Para 16

ANSWER 9.

The FCU supplies the commands to control the lateral and vertical flight path of the aircraft.

CH21 P21-1 Paragraph 3

ANSWER 10.

(b) Line select keys (d) Message light (g) Call light.

CH21 P21-3 Figure 21-2

ANSWER 11.

The flight profile is normally computed using economy speed as this is the most fuel efficient method of operation.

CH21 P21-8 Paragraph 21 to Paragraph 23

ANSWER 12.

An increase in speed (ie. reduction in sector time) will result in a higher fuel burn and hence increased fuel costs.

CH21 P21-6 Paragraph 17

ANSWER 13.

When operating in SINGLE mode, both EHSIs can be driven from the same FMC providing that both pilots select the same mode and range setting.

CH21 P21-7 Paragraph 18

This data is valid for the next sector only and is automatically dumped by the FMC following the next landing and engine shutdown.

CH21 P21-6 Paragraph 17

ANSWER 14.

Each CDU supplies its own EHSI however, because the FMCs are operating independently, the EHSI pictures will differ even with the same mode and range selected.

CH21 P21-7 Paragraph 18

ANSWER 15.

The FMC knows the aerodrome elevation (usually from the database) and it has manually entered QNH information; the exact vertical distance from the cruise level to touch down is therefore known. The descent is normally calculated at idle power and economy speed taking into account any altitude constraints imposed by the relevant STAR. Wind velocity may be fed in manually by the pilots to refine the descent calculations.

CH21 P21-8 Paragraph 23

ANSWER 16.

They are in the Aircraft Operating Manual (AOM)

CH21 P21-7 Paragraph 19

ANSWER 17.

It is essential that pilots monitor the aircraft's progress using raw data throughout all phases of flight, particularly the descent. Any errors in the database should be fed back through the appropriate reporting channels for subsequent correction.

CH21 P21-4 Paragraph 8

ANSWER 18.

A constantly developing vector between the mean inertial position and the FMC position

CH21 P21-5 Paragraph 11 to Paragraph 14

ANSWER 19.

Managed guidance means that the FMC will command the autopilot to maintain the defined track (at a particular altitude and speed).

CH21 P21-7 Paragraph 20

Electronic Flight Instrument Systems (EFIS)

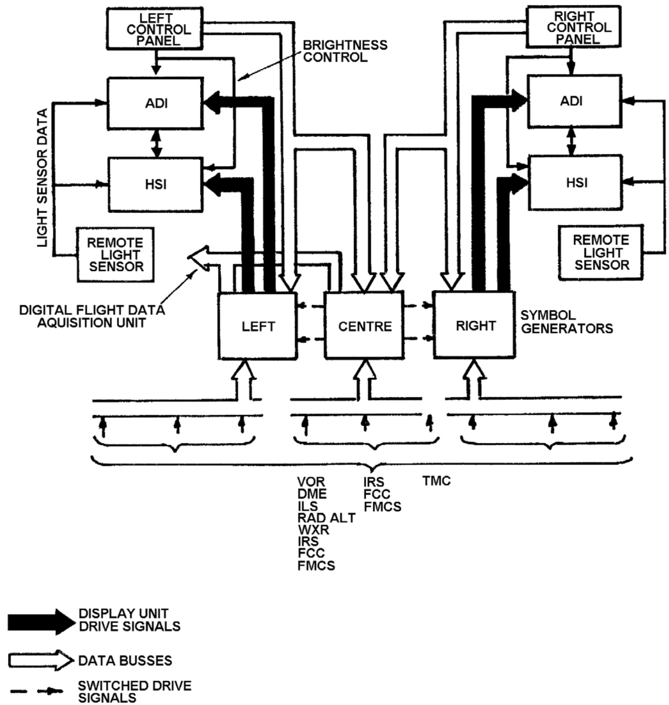
35. EFIS displays information on two, approximately 5 inch square, screens for each pilot. One screen corresponds to the ADI (attitude direction indicator) and displays mainly flight parameters whilst the other corresponds to the HSI (horizontal situation indicator), although the computer-generated displays convey far more navigation information than is possible with the conventional electro-mechanical flight director system. Figure 21-8 shows a typical interface between EFIS and signal inputs. These displays, which are generated via the Display Management Computer, are capable of presenting all of the necessary primary and secondary flight information.

36. Conventional (analogue) back-up pressure and gyro instruments are usually retained for airspeed (ASI), altitude (pressure altimeter), pitch and bank (artificial horizon) and heading (direct reading compass).

37. The symbol generators interface between the aircraft systems, the control panels and the display screens. They perform the main control functions of the EFIS, including system monitoring and generation of the digital and analogue displays on the electronic ADI (EADI) and electronic HSI (EHSI) screens.

38. Appreciate that some manufacturers refer to the **EADI** as the **Primary Flight Display** (**PFD**) and to the **EHSI** as the **Navigation Display** (**ND).**

FIGURE 21-8
EFIS Interface Diagram



The EFIS Control Panel

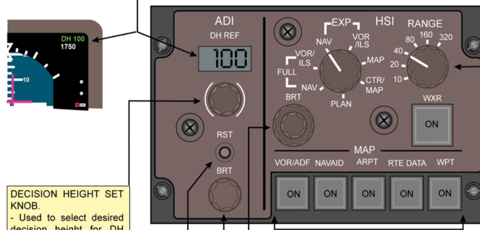
39. An example of a control panel is at Figure 21-9. Remote light sensors respond to ambient flight-deck lighting levels and adjust the CRT displays accordingly to maintain optimum display visibility. Display brightness can also be adjusted manually by brightness controls (BRT) on each half of the pilots' EFIS control panel (EADI and EHSI). The buttons at the bottom of the panel (the EHSI **Map Mode Selector Switches**) are illuminated when pressed to select on.

FIGURE 21-9

EFIS Control Panel

DECISION HEIGHT REFERENCE INDICATOR
- Displays selected decision height (also displayed on EADI).
- Display on ADI blanks when a negative decision height is selected

EHSI RANGE SWITCH.
Selects desired nautical mile range for MAP, CTR MAP, PLAN and weather radar displays



DECISION HEIGHT SET KNOB.
- Used to select desired decision height for DH alerting

DECISION HEIGHT RESET SWITCH.
PRESS - Resets a DH alert on the associated ADI.
- Changes RA display from yellow to white.

EADI DISPLAY BRIGHTNESS CONTROL.
- Used to adjust the brightness of the ADI display to the desired level.

EHSI MANUAL BRIGHTNESS CONTROL.
OUTER CONTROL - Adjusts brightness of HSI display.
INNER CONTROL - Adjusts brightness of weather radar display.

EHSI MAP MODE DISPLAY SELECTOR SWITCHES.
- Add background data/symbols to EHSI when selected.
- Multiple options can be selected simultaneously.
- Illuminated ON (white) when selected.

VOR/ADF - Displays VOR and/or ADF relative bearing radials if the VOR/ADF receiver(s) are tuned to usable stations and valid data is being received.
- Suppresses MAP mode displayed ADF Bearing Pointer.

NAV AID (Navigation Aids) - Displays FMC data base high altitude navigation aids on map scales 80, 160, 320 nm.
- Displays all FMC data base navigation aids if on map scales 10, 20, or 40 nm.

ARPT (Airports) - Displays all airports which are stored in the FMS data base and which are within viewable map area of the EHSI.

RTE DATA (Route Data) - Displays altitude constraint (if applicable) and estimated time of arrival for each active waypoint.

WPT (Waypoints) - Displays the waypoints in the FMC data base which are not in the flight plan route if the selected range is 40 nm or less.

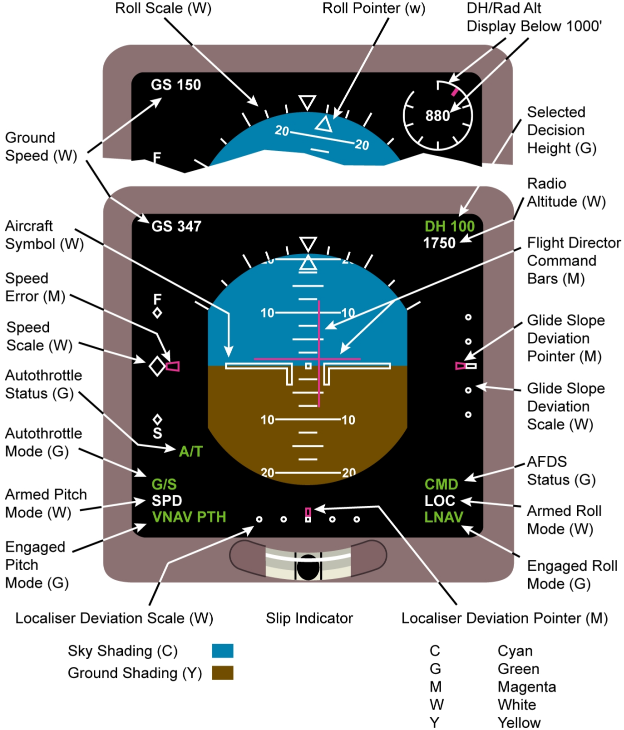
The EADI Screen

40. The upper (EADI) screen conventionally displays aircraft attitude in pitch and roll against a shaded (raster-scanned) background, the upper half of which is coloured blue (cyan) and the lower half yellow (or light brown). The source for the attitude data is the aircraft's inertial reference system(s). The EADI also displays flight director command bars for roll and pitch commands, as well as ILS localiser and glideslope deviation, selected airspeed deviation, ground speed, automatic flight system and autothrottle system operating modes, radio altitude and decision height. A typical EADI display is shown at Figure 21-10.

41. **Radio altitude** is displayed **digitally between 2500 feet and 1000 feet agl,** as shown in the top right hand corner of the EADI at Figure 21-10. **Below 1000 feet agl the display becomes analogue/ digital,** again as illustrated at Figure 21-10. There is a decision height (DH) setting knob on the EFIS control panel (Figure 21-9). At radio altitudes above 1000 feet the selected DH is displayed digitally on the EADI (Figure 21-10). Below 1000 feet radio altitude the DH is displayed as a magenta coloured marker on the circular analogue radio altimeter scale (Figure 21-10). As the aircraft descends from 1000 feet radio altitude the white circular scale segments are progressively erased in an anti-clockwise direction, so that the remaining 100 foot segments indicate the height above ground. **At 50 feet above the selected decision height an aural chime alert sounds with increasing frequency until decision height is reached.** The circular scale and marker then both change colour to amber and flash for several seconds. This alert is manually cancelled by pressing a reset button on the control panel (Figure 21-9).

FIGURE 21-10

Typical EADI Display



42. Airspeed error above (F) or below (S) a selected airspeed is indicated by a magenta pointer and scale on the left hand side of the EADI. Glideslope deviation is similarly displayed on the right hand side of the screen. Localiser deviation is indicated by a magenta pointer and scale at the bottom of the display. ILS localiser and glideslope deviations are emphasised by the appropriate pointer and scale changing colour to amber. Bank and slip are conventionally displayed on a computer-generated roll scale and 'ball-in-tube' symbol at the bottom of the screen.

43. Since data inputs from systems such as ILS and the radio altimeter are vital to both the displayed information and the automatic landing sequence, failure of these data inputs must be annunciated. In EFIS displays this annunciation frequently takes the form of **yellow flags** painted on the display screens.

The EHSI Screen

44. The lower (EHSI) screen presents a colour display of flight progress (ie. navigation data) in one of nine modes. These are selected on the EHSI section of the EFIS control panel (EHSI Mode Selector Switch) and are MAP, CTR MAP, PLAN, FULL ILS, FULL VOR, FULL NAV, EXPANDED VOR, EXPANDED ILS and EXPANDED NAV.

Map Mode

45. This is the display normally used for en-route navigation and is illustrated at Figure 21-11. It provides a moving Map display which is orientated to the aircraft's present track, with the aircraft symbol positioned at bottom centre and a 60° expanded arc of the compass rose positioned across the top of the screen. Ground features such as navaids, airports and waypoints are shown in their relative locations to a common scale (when selected on using the appropriate button(s) on the control panel (Figure 21-9). The scale of the Map picture is selected on the EHSI section of the control panel (the Range selector), which typically offers ranges of 10, 20, 40, 80, 160 and 320 nm. The weather radar picture, generated in the standard colours of green, amber and red (with magenta in some cases), can be superimposed on the display in the EXP VOR, EXP ILS, EXP NAV, CTR MAP and MAP modes, again by pressing the WXR button on the control panel.

46. Heading information is obtained from the aircraft's inertial reference system(s). When operated between the latitudes of 73°N and 65°S the compass rose is referenced to magnetic north or true north, depending upon operator preference. Above these latitudes the compass rose is referenced to true north only. Note however that the compass rose lubber line shows aircraft track and that the heading pointer is off centre in conditions other than of zero drift.

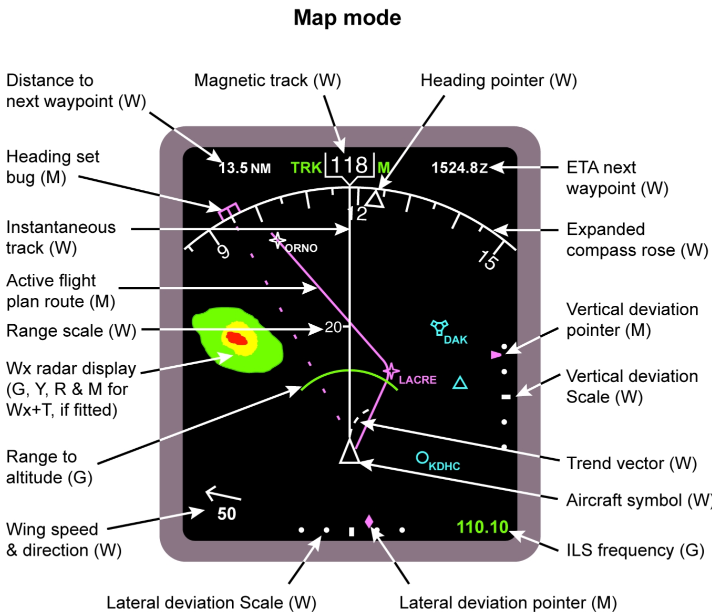
47. Wind speed is displayed digitally, with an analogue display of wind direction in the form of an arrow pointing in the appropriate direction. The wind arrow is oriented to the Map display, which is in turn oriented to the aircraft track such that the vertical axis of the display is the aircraft instantaneous track as shown at Figure 21-11. The wind velocity shown at Figure 21-11 is therefore in the order of 225°(M)/50 kt.

48. Lateral and vertical deviation from the planned flight path is indicated by pointers and scales aligned horizontally and vertically on the edges of the display. The selected range scale is overlaid vertically on the Map display, originating from the aircraft symbol. Distance and time to the next waypoint is displayed digitally.

49. A trend vector extending from the apex of the aircraft symbol shows the predicted lateral position at the end of 30, 60 and 90 second intervals based upon bank angle, groundspeed and lateral acceleration. A range to altitude arc intersects the planned track and range scale at the point where a selected target altitude will be reached at present rate of climb or descent.

50. With the EHSI in the map mode the screen is continuously displaying area navigation information. The picture is generated by the appropriate signal generator using data provided by the inertial navigation/inertial reference system and by the flight management system. The position of the aircraft as determined by the INS/IRS will be continuously monitored and updated by the FMS using fixing data received from in-range VOR/DME stations which are automatically selected by the FMS. The FMS selects stations to achieve optimum fix geometry, two DME range arcs at 90° to each other being the ideal. Obviously the automatic update aspect of the area navigation function will cease when the aircraft flies out of VOR/DME coverage. Manual update of the INS/IRS position is possible but should not normally be necessary.

FIGURE 21-11
Map Mode Display



○	Airports (C)	△	Off route waypoints (C)
	Navaids (C)	⇐	Wind direction (W)
✧	Waypoints: Active (M), Inactive (W)		

Colours
C Cyan
G Green
M Magenta
W White

CTR MAP Mode

51. Displays the same data and symbols as the MAP mode, but the aeroplane symbol is placed in the centre of the map area so that MAP information behind the aeroplane is displayed.

Plan Mode

52. Figure 21-12 shows the display generated when plan mode is selected. On the lower part of the screen the active route is displayed, but now it is oriented to **true north**. Track and heading information is on an expanded compass rose but now the lubber line shows heading with the track mark off centre in conditions other than of zero drift. Again distance and time to the next waypoint shown digitally. **Wind speed and direction is not displayed in this mode and weather radar returns cannot be superimposed.** It is a useful display mode for checking route changes as they are selected at the keyboard and before they are entered into the Flight Management System (FMS) computer.

FIGURE 21-12
Plan Mode Display

VOR and ILS Modes

53. These are illustrated at Figure 21-13 and Figure 21-14 respectively. They may be presented as a full compass rose display with conventional heading and course deviation indications, or as an expanded compass rose display upon which the weather radar picture may be superimposed on a ‘semi-map’ picture with the selected range scale displayed. In either case wind speed and direction and system source (ILS or VOR) are annunciated.

54. In expanded VOR and ILS modes (and, indeed, in Map mode) a dotted line appears from the apex of the aircraft symbol to the heading bug for a few seconds following the selection of a new set heading. The aircraft's instantaneous (current) track is displayed as a solid line extending from the apex of the aircraft symbol to the compass scale arc. Bearing of the selected radio navaid is shown by a solid line extending from the centre bar of the lateral deviation scale to the compass arc.

FIGURE 21-13
VOR Mode
Display

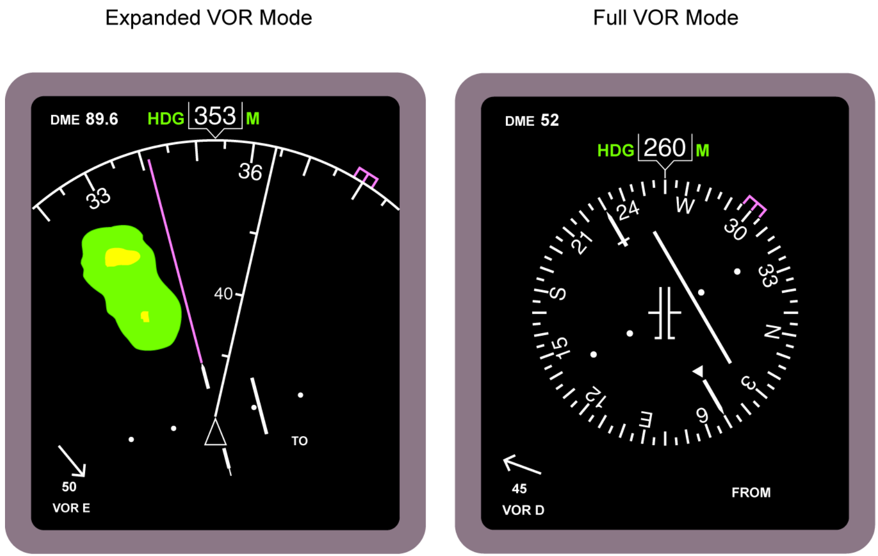
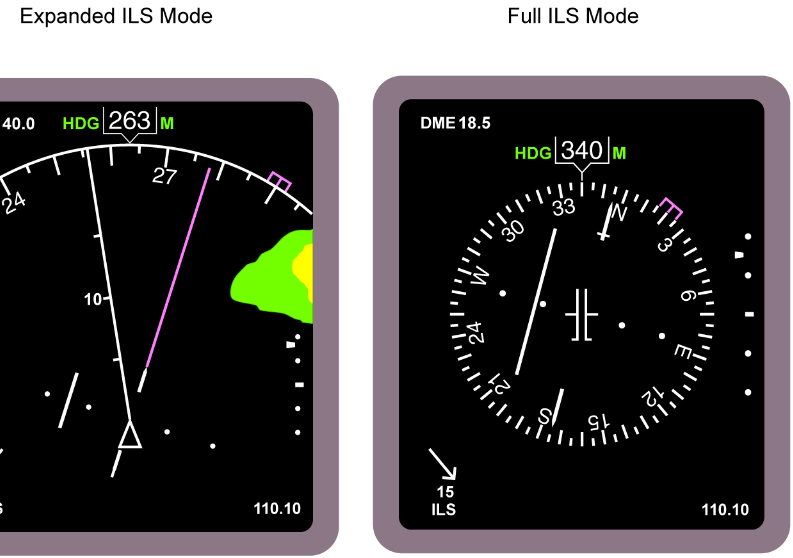


FIGURE 21-14
ILS Mode Display



Expanded Nav Mode

55. Displays lateral and vertical navigation guidance information similar to a conventional HSI. The FMC is the source of the navigation data. Weather Radar return data is displayed when the WXR Switch is On.

Full Nav Mode

56. Displays same data as expanded navigation mode with the following exceptions:

- Weather Radar displays are not available
- A full compass rose is shown in place of the expanded compass rose.

Backup Data Inputs

In the case of a display unit failure, switching to another display unit is possible. Similarly, in most systems the pilots can, independently of each other, connect their respective EADI and EHSI displays to alternate sources of input data. For example, should a symbol generator (SG) fail on the left hand side, the captain can duplicate the information shown on the right hand screens; this selection is made via the **EFI Transfer switch.**

57. The **EFI Transfer Switch** determines the SG source for the Captain's and F/O's EADI and EHSI displays. With the EFI Transfer Switch in the NORMAL position, the No. 1 SG provides display symbols for the Captain's EFIS displays, and the No. 2 SG provides display symbols for the F/O's EFIS displays. If the EFI Transfer Switch is in the BOTH ON 1 position, both sets of displays utilise symbols provided by the No. 1 SG, and the No. 2 SG is turned off. If the EFI Transfer Switch is in the BOTH ON 2 position, both sets of displays utilise symbols provided by the No. 2 SG, and the No. 1 SG is turned off.

58. The same can be done in the event of the failure of either left or right air data computers (ADC) or flight management computers (FMC). In an aircraft equipped with three inertial reference systems (IRS), number one IRS would normally supply the captain's EFIS and number two IRS the first officer's EFIS. In the event that either of these IRS were to fail, number three IRS can be selected to replace the failed system. These selections are made on a source selector switch panel.

EFIS Symbology

59. Examples and descriptions of EFIS symbols are given in the tables at Figure 21-15 to Figure 21-17.

60. The following symbols may be displayed on each EHSI depending on EFIS Control Panel selections. General colour presentation is as follows:

GREEN (G)	Indicates active or selected mode and/or dynamic conditions
WHITE (W)	Indicates present status situation and scales
MAGENTA (M) (pink)	Indications command information, pointers, symbols, and fly-to conditions, weather radar turbulence
CYAN (C) (blue)	Indicates non-active and background information
RED (R)	Indicates warning
YELLOW (Y)	Indicates cautionary information, faults, flags
BLACK (B)	Indicates blank areas, off condition

FIGURE 21-15
EFIS Symbology

SYMBOL	NAME (COLOUR)	APPLICABLE MODE(S)	REMARKS
200 nm/4.4 nm or DME 124/DME 24.6	Distance display (W)	PLAN, MAP or VOR, ILS	Distance is displayed to next FMC waypoint (nm) or tuned navaid (DME)
HDG 263 M	Heading: orientation (G) indicator (W) and reference (G)	MAP, PLAN VOR, ILS	Indicates numberically the aircraft heading
0835.4 z	ETA display (w)	MAP, PLAN	Indicates FMC calculated ETA for the active waypoint based on present groundspeed
	Selected heading marker (M)	MAP, ILS, VOR, PLAN	Indicates the set heading. A dashed line (M) extends from the marker to the aircraft symbol
24	Expanded compass rose (W)	MAP, EXP ILS, EXP VOR, PLAN	360° are available but approximately 70° are displayed
33 0 3 6 9 12 15 18 21 24 27 30	Full compass rose (W)	FULL VOR, FULL ILS	Compass data is provided by the selected IRS
80	Present track line and range scale (W)	MAP, EXP VOR, EXP ILS	Predicts ground track which will result with present heading and wind. The Displayed range mark is one half of the actual selected range
	Aircraft symbol (W)	MAP, EXP VOR, EXP ILS	Represents the aircraft, the apex of the triangle indicates the aircraft position
	Aircraft symbol (W)	FULL VOR, FULL ILS	Represents the aircraft, the centre of the symbol indicates the aircraft position
DCS	Waypoint: active (M) inactive (W)	MAP, PLAN	The active waypoint is the waypoint that the aircraft is navigating towards and the inactive waypoints are those making up the selected active route
	Altitude range arc (G)	MAP	The point at which the arc intercepts the track line predicts the point at which the reference altitude will be reached
	Trend vector (W)	MAP	Predicts aircraft directional trend at the end of 30,60 and 90 second intervals based on the bank angle and ground speed
AMBOY KILMR PARBY ARO	Active route (M). Active route modifications (W) Inactive routes (C)	MAP, PLAN	The active route is displayed with a continuous line between waypoints (M). Active route modifications are displayed with short dashes between waypoints When a change is activated the short dashes become a continuous line. Inactive routes are displayed with long dashes between waypoints (C)
	Vertical pointer (M) and deviation scale (W)	MAP	Displays vertical deviation from selected vertical profile (pointer) in map mode only. Scale indicates ± 400 feet deviation

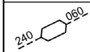
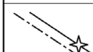
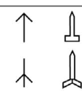
FIGURE 21-16

EFIS Symbology

SYMBOL	NAME (COLOUR)	APPLICABLE MODE(S)	REMARKS
	Glideslope pointer (M) and deviation scale (W)	ILS	Displays glideslope position and deviation in the ILS mode. The pointer is not displayed when the present track and the ILS QDM differ by more than 90º
	Drift angle pointer (W)	FULL VOR, ILS	Displays difference between FMC track angle and IRS heading
N	North pointer (G)	PLAN	Indicates map background and is oriented and referenced to true north
T/D	Vertical profile point and identifier (G)	MAP	Represents an FMC calculated point and is labelled on the flight plan path as T/C (top of climb), S/C (step climb), T/D (top of descent) and E/D (end of descent). Deceleration points have no identifier
	Radar returns (G, Y, R)	MAP, VOR, ILS	Multicoloured returns are presented when either WXR ON switch is pushed,. Most intense areas are displayed in red, lesser intensity in yellow and lowest intensity in green
	Holding pattern active (M) Modification (W) Inactive (C)	MAP, PLAN	A fixed size holding pattern appears when in the flight plan. This pattern increases to correct size when HSI range is 80 nm or less and the aircraft passes the waypoint prior to the holding pattern
	procedure turn active (M) Modification (W) Inactive (C)	MAP, PLAN	A fixed size procedure turn appears when in the flight plan. It increases to the correct size when HSI range is 40 nm or less and the aircraft passes the waypoint prior to the procedure turn
LHMZ 24L	Airport identifier and runway (W)	MAP, PLAN	Appears when selected on FMC CDU. Available when HSI range is 80 nm, 160 nm or 320 nm
22L	Airport and runway (W)	MAP, PLAN	Appears when selected on FMC CDU. Available when range is 10, 20 or 40 nm. Dashed Centrelines extend outward 14.2 nm
DTY	Selected fix circle (G) symbol and identifier (C or G)	MAP, PLAN	Presents a selected reference point (fix) via the FMC CDU FIX key. Can appear with special map symbols (VOR, VORTAC, airport or waypoint and so on) if contained in the existing database
	VOR DME/TACAN VORTAC (C)	MAP	When NAVAID switch is ON, all appropriate navaids in range appear in addition to those navaids which are standard or active. Tuned navaids are displayed regardless of the NAVAID switch and appear green. When HSI is 80, 160 or 320 nm only high altitude navaids are displayed

FIGURE 21-17

EFIS Symbology

SYMBOL	NAME (COLOUR)	APPLICABLE MODE(S)	REMARKS
 120	Wind speed and direction (W)	MAP, VOR, ILS	Indicates wind speed in knots and direction with respect to the map display orientation and compass reference
	Manually tuned navaid (G)	MAP	When a navaid is manually tuned the selected course and reciprocal are displayed
 EGSM	Airport (C)	MAP	When the ARPT switch is ON, airports within the map area are displayed. Origin and destination airports are always displayed, regardless of the ARPT switch position
 DCS 27000 1435Z	Route data (M, W)	MAP, PLAN	When the RTE DATA switch is ON altitude and ETA for route waypoints are displayed
 DVR	Off route waypoint (C)	MAP	When WPT switch is ON,data base waypoints not on the selected route are displayed when using HSI ranges 10, 20 or 40 nm
	Course indicator (M) and deviation scale (W)	VOR, ILS	Displays ILS course when ILS mode is selected and valid signals are present. VOR course is displayed when VOR MODE is selected and valid signals are present
	Offset path (M)	MAP, PLAN	Presents a dash-dot line parallel to and offset from the active route after execution on the FMC CDU
TO FROM	To / from display (W)	VOR	Display logic is the same as for non-EFIS HSIs
VOR L/R ILS	Source nav data (G)	VOR, ILS	Displays source of nav radio data based on HSI control selection
	Selected course pointer (W) and line (M)	VOR, ILS	Displays selected course as appropriately set by the VOR course selector
110.10	ILS frequency display (G)	ILS	Displayed when an ILS frequency is selected
	ADF displays (G)	ALL	Appears when valid ADF signals are being received. The narrow needle shows the left ADF and the wide needle the right ADF

Weather Radar Displays and Annunciations

61. Figure 21-11 shows weather radar returns with the EHSI selected to MAP. Figure 21-13 and Figure 21-14 show weather radar returns with the EHSI selected to expanded VOR and ILS modes.

62. The weather radar returns are colour coded **red** for the most intense returns, **amber** for lesser intensity and **green** for lowest intensity.

63. The radar has three (and possibly four) modes; **test, weather** (**WX**), (**possibly**) **weather plus turbulence** (**WX + T**) **and MAP.**

64. The test mode checks the hardware and paints a predetermined pattern on the screen to assure the operator that the various colours are being properly produced by the EFIS symbol generators.

65. The weather mode symbology is as described in the table at Figure 21-16.

66. If incorporated, the weather plus turbulence mode introduces a fourth colour (**magenta**) onto the weather paint in areas of **suspected high turbulence,** which the radar determines by identifying the areas of greatest rate of change of target intensity. This is likely to coincide with the area of greatest rate of change of vertical velocity of the air, and therefore of greatest turbulence.

67. The map mode employs a vertically broad beam to paint the land/sea surface ahead of the aircraft.

68. When the system is operating normally the radar operating mode (WX, WX + T or MAP, but not test, as this is self evident) is displayed in the top right corner of the HSI, together with the tilt angle of the scanner. These are not shown at Figure 21-11, Figure 21-13 or Figure 21-14.

69. In the event that anything goes wrong with the weather radar, the pilot is informed by means of a message which appears towards the bottom left hand corner of the HSI. These messages are typically as follows, but will vary slightly from one system to another:

WXR FAIL	Indicates weather radar has failed (no weather data displayed).
WXR WEAK	Indicates weather radar calibration fault.
WXR ATT	Indicates loss of attitude input for antenna.
WXR STAB	Indicates antenna stabilization is selected off.
WXR DSPY	Indicates loss of Display Unit cooling or an overheat condition of the HSI. Weather radar display is blanked.

Automatic Flight Control Systems

Automatic Flight Control Systems

1. This chapter is concerned with the operation of Automatic Flight Control Systems which includes the Flight Director System and Autopilot.

Flight Director Systems

2. Flight director system displays are described in Chapter 21 with conventional ADI and HSI displays illustrated by Figure 21-6 and Figure 21-7. An EADI display is illustrated at Figure 21-10.

Pitch and Roll Commands

3. Aircraft attitude is displayed by the relationship of a stationary delta shaped symbol, representing the aircraft, relating to roll and pitch commands using two bars flanking the aircraft symbol. The aircraft symbol is also related to the artificial horizon bar. On some instrument fits, the bars are replaced by parallel motion cross-pointer indicators. The command bars (or pointers) direct the pilot to fly up or down, left or right, depending upon the position and attitude of the aeroplane in relation to its predetermined flight path.

Components

4. The components of a flight director comprise:

(a) Mode select panel

(b) Computer

(c) Attitude direction indicator (ADI)

(d) Horizontal situation indicator (HSI)

(e) Nav receiver

(f) Vertical gyro

(g) Amplifier unit

(h) Magnetic compass unit.

Inputs

5. A flight director computer accepts flight path displacement information, compares it with current attitude information and presents any required attitude changes as commands or directions to a single instrument. The inputs to the computer are shown at Figure 22-1.

FIGURE 22-1
Flight Director Computer Inputs

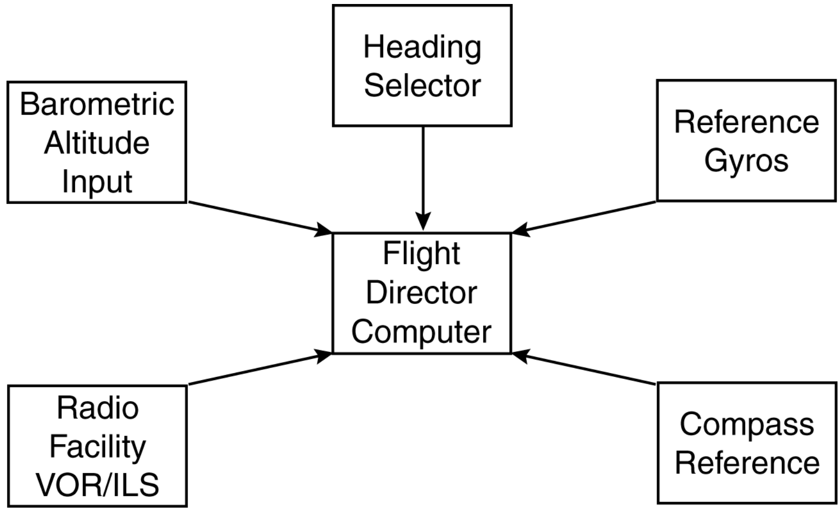


FIGURE 22-2

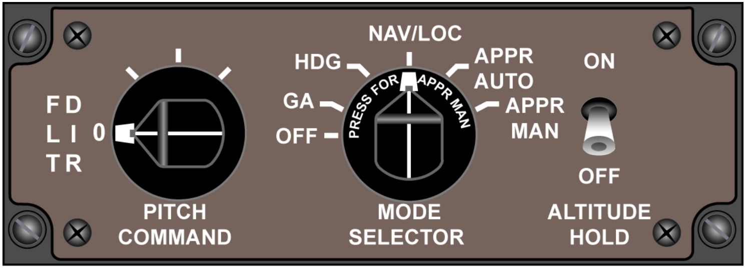
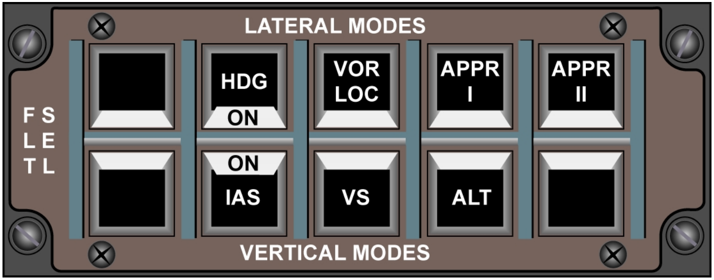


FIGURE 22-3



Command Modes (Bars)

Flight Director Computer

(Figure 22-1)

6. The operating principle of the FD computer is based on an electronic unit which will accept signals from aircraft instrumentation, radio and navigation sources, process and develop them to command position or attitude changes. If the FD system is integrated with an automatic flight control system, the signal outputs are utilised for control application. The computer signal circuits are solid state, aligned to a logic network. This ensures that all signals are scaled and adjusted to suit specific types of aircraft. All power and signal circuits are on printed circuit boards which are designed to be arranged as individual plug-in modules. The output to the command bars are displayed on the attitude director indicator (ADI) and horizontal situation indicator (HSI). This directs the pilot to fly up or down, left or right, depending on the position of the aeroplane in relation to its predetermined flight path. Signals from the ILS are fed to command bars and localiser and glidepath deviation pointers. The ADI presentation includes a glide slope vertical dot scale on the side of the instrumentation over which a glide slope pointer travels. Similarly, across the bottom of the instrument face, a localiser pointer moves over a scale to indicate displacement in relation to localiser beam. The HSI will display the navigational situation with respect to a selected heading or course. In addition the HSI will display inputs corresponding to deviations from an ILS localiser beam, VOR radial and glide path beam.

7. Reference to the table '**Mode of Operation**' indicates F/D mode selected, signals which are controlling the command bars (roll/pitch) and the F/D computer input data. As an example, consider a selection of VOR/LOC mode when intending to fly a VOR course. After tuning the VOR RX, the appropriate course must be selected on the autopilot mode select panel. The flight director selection will be to VOR/LOC. Signal inputs from compass, gyro (bank angle) and the VOR RX will be fed to the computer and once any VOR deviation, bank angle and course datum have been resolved, signals will be sent to the command bars. The pilot is requested to obey any command bar deviation to achieve the desired course.

Mode of Operation

Mode Selected	Signals Controlling Command Bars	Roll/Pitch	Input to F/D Computer
Heading (HDG)	Heading error + Bank Attitude	Roll	Compass Heading select
VOR (VOR/LOC) Radio tuned to VOR	VOR Deviation + Course Datum + Bank	Roll	Compass Gyro VOR RX
ILS localiser (VOR/LOC) Radio tuned to LOC	LOC Deviation + Bank + Course Datum	Roll	Compass Gyro LOC RX
Glideslope (G/S)	Pitch Attitude + Glide Slope Deviation	Pitch	Compass Gyro G/S RX Altitude

Flight Phase Operation

8. The flight director can be used for all phases of flight and be on for both manual or automatic flight.

Selection and Operation

(Figure 22-2 and Figure 22-3)

Take Off Flight director on with pitch bar set at target attitude. Pitch command attitude set.

Climb Select heading or NAV (VOR) as required to fly the planned departure after takeoff for lateral mode. Select vertical speed or IAS as required for vertical mode.

Cruise Select the cruise altitude required during the climb and command bars will give guidance to hold aircraft at the desired altitude. Continue with lateral navigation as required using en-route VOR navigation. Use heading where appropriate to set Intercept Heading for NAV facility capture. Where fitted, use the altitude hold facility to maintain selected level. Use the IAS or MACH (where fitted) to maintain the desired speed.

Descent Continue to use the lateral NAV facility (VOR) and select the desired cleared altitude(s) as required by ATC using the Alt input. Select vertical speed or IAS/MACH as appropriate.

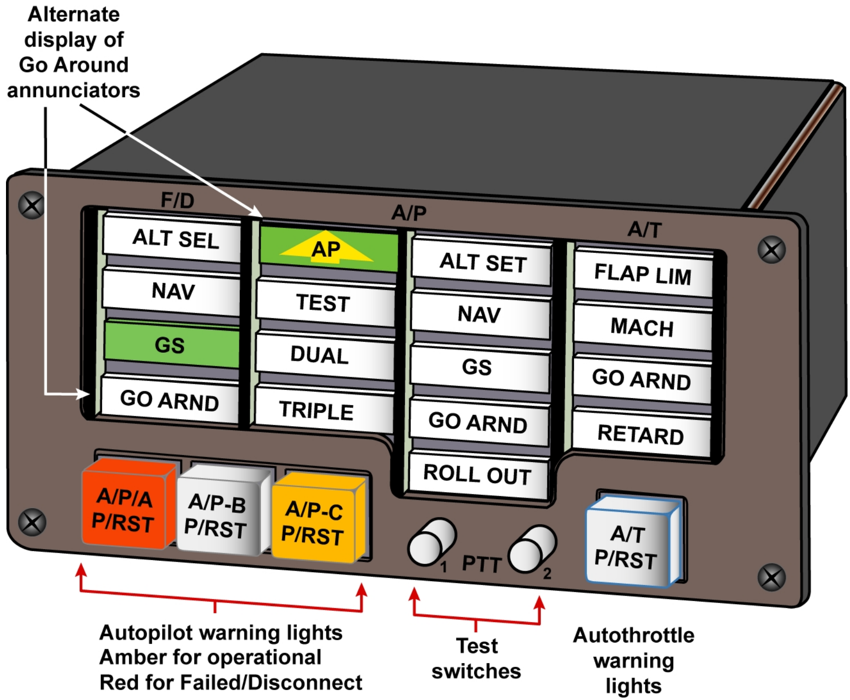
Approach Select Intercept Heading for ILS Localiser and the required altitude to intercept the ILS Glidepath. Select the appropriate approach facility for capture and tracking of glideslope and localiser beam of ILS APPR I and APPR II, select IAS as appropriate.

Land Follow the command bars down to the decision height and land as appropriate with required visual reference.

Go Around (Missed Approach) Select if go around is required. The command bars command a wings level and pitch attitude. Once go around power and IAS is achieved HDG and IAS modes may be selected.

Flight Mode Annunciation

FIGURE 22-4
Flight Mode Annunciator



9. In a more sophisticated system, indication of the selected mode of operation is displayed on a flight mode annunciator situated on the pilots panel. The FD annunciation can be combined with the autopilot and auto throttle displays in a single display.

10. The display uses light indications which have different colours to display the selected mode, i.e. Armed - white letters on black background, Capture - black letters on green background.

11. The presentation is done in various ways. A typical installation is illustrated at Figure 22-4. On an EFIS equipped aeroplane the FD annunciations are incorporated into the EFIS display. The annunciations on Figure 22-4 for the FD section are as follows:

ALT SEL	Annunciates Armed when altitude mode select switch is placed to ALT SEL and green when altitude is captured
NAV	Annunciates Armed when navigation mode is in INS VOR LOC, ILS or LAND (autoland) and green when selected course has been captured
GS	Annunciates Armed when navigation mode switch is in ILS or Land and green after slide slope capture
GO AROUND	F/D annunciates green with the arrow presentation once go around is initiated. (i.e. when No. 2 or 3 thrust lever switch is pressed after glide slope capture). Note there is no annunciation of go around Armed.

12. The flight director is switched on by pilot action with a separate FD selection (switch or button), located on the auto pilot mode select panel. When the FD is engaged it will be indicated by the illumination of an adjacent indicator light.

Limitations, Operational Restrictions

13. The commands presented on the flight director are programmed such that the structural limits of the aircraft cannot be exceeded when obeying the pitch and roll indications.

Gain Program in Approach Mode

14. During an ILS approach using the LOC and G/S roll and pitch signals, a gain programmer in the gain control section of the computer will automatically be switched in. At pre-programmed positions on the LOC and GS, following capture, the pitch and roll signals are reduced. This means that the overall response to the system is optimised, irrespective of flight conditions, and in the case of approach mode, the gain programming will reduce the deviation signals to allow for convergence of the LOC and GS beams.

Lateral and Vertical Beam Sensors

15. The task of the lateral and vertical beam sensors is to convey data input to the computer for performing the task of stabilising the attitude of the aircraft or modifying it as necessary. Disturbances about the longitudinal and lateral axis are picked up by sensors which produce a signal to be processed and presented as a command signal in roll and pitch to the F/D instrumentation. The commands of the F/D are related to the rate of change of deviation and this will therefore be reflected in the command information presented on the instrumentation.

Autopilot

JAR-OPS Requirement JAR 25.1329

16. Each automatic pilot system must be approved and must be designed so that the automatic pilot can be quickly and positively disengaged by the pilots to prevent it from interfering with their control of the aeroplane.

17. Unless there is automatic synchronisation, each system must have a means to readily indicate to the pilot the alignment of the actuating device in relation to the control system it operates.

18. Each manually operated control for the system must be readily accessible to the pilots.

19. Quick release (emergency) controls must be on both control wheels, on the side of each wheel opposite the throttles.

20. Attitude controls must operate in the plane and sense of motion specified in Jar 25.777(b) and 25.779(a) for cockpit controls. The direction of motion must be plainly indicated on, or adjacent to, each control.

21. The system must be designed and adjusted so that, within the range of adjustment available to the human pilot, it cannot produce hazardous loads on the aeroplane, or create hazardous deviations in the flight path, under any condition of flight appropriate to its use, either during normal operation, or in the event of a malfunction assuming that corrective action begins within a reasonable period of time.

22. If the automatic pilot integrates signals from auxiliary controls or furnishes signals for operation of other equipment, there must be positive interlocks and sequencing of engagement to prevent improper operation. Protection against adverse interaction of integrated components, resulting from a malfunction, is also required.

23. Means must be provided to indicate to the flight crew the current mode of operation and any modes armed by the pilot. Selector switch position is not acceptable as a means of indication.

Automatic Pilot System

24. Basic to all aircraft automatic flight control systems (AFCS) is the autopilot, which comprises closed loop systems providing control about one or more of the aircraft's primary axes (roll, pitch and yaw). The function of an autopilot is to provide a means of automatically controlling an aircraft thus relieving the pilot from the manual task of flying the aircraft for long periods. With the autopilot engaged, the pilot selects the required flight conditions and monitors the functioning of the autopilot in achieving the tasks.

25. An automatic flight control system incorporates the functions of the autopilot into the broader concept of flight management, by injecting data relevant to the planned flight path such as airspeed, altitude, heading, position relative to ground radio navaids, and so on.

26. Autopilots employ closed loop control systems which sense deviations from steady flight and apply corrections via the flying controls proportional to the rate of deviation. The functions of such a system may be summarised as follows:

(a) An attitude change about one of the principal axis is sensed by a gyroscope.

(b) The gyroscope processes and produces an output signal proportional to the rate of change of attitude.

(c) A controller compares the gyro signal with a fixed 'reference attitude' signal, computes the rate and direction of attitude change, and transmits a suitable corrective signal to a flying control servomotor.

(d) The loop is closed as the corrective action is sensed by the gyro, the output signal is removed and the servomotor returns the flying control to a 'steady-state' position. This is achieved by a 'feedback' signal of servomotor action to the controller, which is thus able to compare 'required' and 'actual' control surface movement.

27. A block diagram of an autopilot control loop for pitch control is shown at Figure 22-5, pitch and yaw control loops would be identical. The inner loop of an autopilot system serves the function of auto stabilisation about a particular axis.

FIGURE 22-5
Autopilot Inner Loop Control

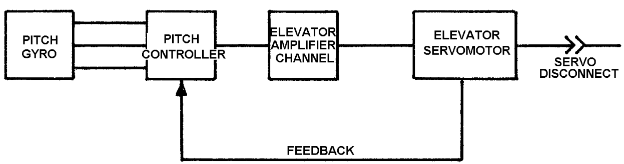


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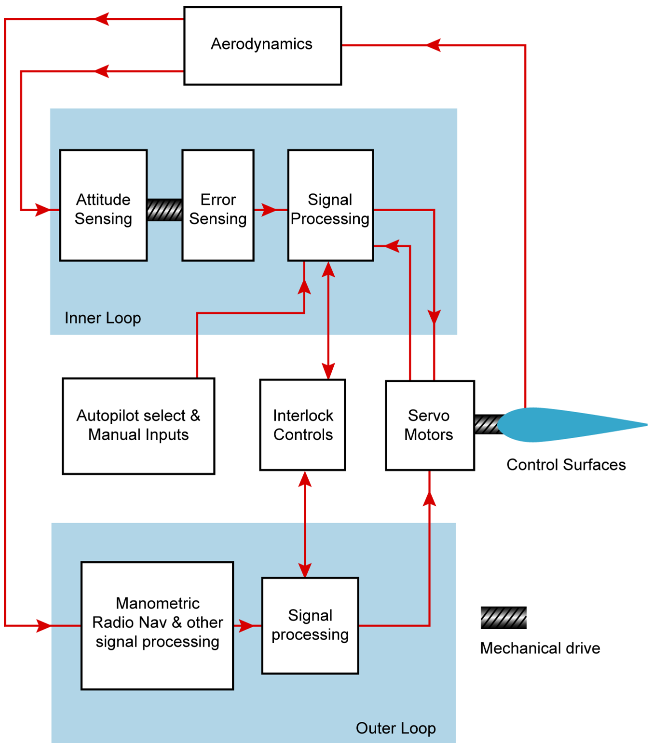
FIGURE 22-6
Autopilot Inner and Outer Loops



28. Inputs to the automatic control (inner) loop from external sources (manual, radio navigation and so on) form what is known as the outer loop control. An autopilot will have one closed (inner) loop system for each axis about which automatic flight control is to be effected. A rate gyro, sensitive to movement about one axis only, is used in each control loop. A block diagram showing inner loop stabilisation and outer loop control is at Figure 22-6.

29. Single axis control is usually about the roll axis only. Used in some light aircraft it provides lateral stabilisation. The control loop receives its input from a gyro sensitive to movement about the aircraft's longitudinal (roll) axis and whose rate of precession is therefore proportional to rate of roll. The system servomotors will activate the ailerons to correct roll and maintain ‘wings level’ flight. Outer loop control injects from the pilot (change of heading), gyrocompass (maintenance of heading) and radio navigation aids (maintenance of track) are possible. See table below.

Pitch Axis	**Roll Axis**
Manometric/Air Data	**Heading Select and Hold**
Vertical Speed	
Airspeed Select/Hold	Bank Angle hold
Mach hold	
Altitude Select/Hold	Radio Nav VOR
Pitch Hold	

<table>
<tr><td>Pitch Axis
Pitch Trim
Turbulence Mode</td><td>Roll Axis</td></tr>
<tr><td>Vertical Navigation</td><td>Lateral Navigation
INS
GPS</td></tr>
<tr><td colspan="2" align="center">ILS</td></tr>
<tr><td>Glideslope</td><td>Localiser</td></tr>
<tr><td colspan="2" align="center">Autoland</td></tr>
<tr><td>Approach
Flare</td><td>Runway align
Roll out guidance</td></tr>
<tr><td colspan="2" align="center">Control wheel steering
Touch control steering</td></tr>
</table>

30. Two axis systems normally provide attitude control about the aircraft roll and pitch axes. In addition to the ‘roll loop’ there is a second inner loop in which a rate gyro sensitive to rate of movement about the aircraft's lateral (pitch) axis provides signals to activate the elevator servomotor. Such a system will automatically maintain straight and level flight. Outer loop control injects from the pilot, altitude hold, altitude select and vertical speed functions are possible.

31. Three axis systems provide attitude control about the roll, pitch and yaw axes. Hence, a third rate gyro sensitive to rate of movement about the aircraft's normal (yaw) axis is necessary to provide signals to the rudder servomotor.

32. A full three axis autopilot will therefore have three inner control loops, plus outer control loops. Modern medium and large transport aircraft normally employ two or three independent autopilots, and each self contained autopilot is referred to as an autopilot channel.

33. Aircraft fitted with Inertial Navigation (INS) or Inertial Reference (IRS) systems normally utilise these to provide the necessary attitude references for the autopilot.

34. Some auto pilot systems incorporate an automatic trim system which is active when the auto pilot is engaged. It is generally confined to control about the pitch axis. Control of either a trim tab servo actuation or a variable incidence horizontal stabiliser is used to trim out any out of balance forces. If the auto pilot fails, the pilot will resort to manual operation of the elevator trim system.

35. **Position Trim Indicator.** Position trim indicators are incorporated into the auto pilot to indicate that signals are being supplied to the servo actuators, whether in the engaged or disengaged condition. The indicator will indicate any out of trim condition under normal operation conditions. It performs these functions by monitoring the outputs from the aileron, elevator and rudder servor amplifiers and by producing deflections of pointers from zero datum marks in response to signals supplied.

36. Most auto pilots systems will be self synchronising. Prior to engaging the auto pilot, in addition to the pre-engage requirements, the aircraft must be trimmed for the desired flight attitude and the automatic control system must be synchronised to maintain the attitude on engagement. When power is applied to the auto pilot, the attitude sensing filaments are operative and will sense aircraft attitude and supply any necessary control command signals to the servo motors. In the case of a non-self synchronising auto pilot system, switching on could cause ‘snatching’ of the aircrafts control system. In order to avoid this, the aircraft must be trimmed, by the pilot, for the desired attitude before engagement.

Automatic Flight Control

37. The inner loop control system is, on its own, a stability augmentation (or auto stabilisation) system. It will hold the aircraft on a desired flight path by detecting and correcting any deviation from that flight path. An automatic flight control system, on the other hand, is capable of manoeuvring the aircraft to comply with the requirements of a pre-planned flight path. For example, if the inner loop control system is supplied with actual magnetic heading from the gyro compass and desired magnetic heading as a manually set input it can be made to move the aileron servomotor to bank the aircraft and turn it until the actual and desired heading inputs are the same. Similarly the pitch control loop can be ‘biased’ by supplying it with a manually set desired rate of change of static pressure (rate of climb) and an actual rate of change signal from the Air Data Computer. These concepts can be expanded to achieve automatic control of the aircraft's flight path in terms of lateral and vertical navigation (LNAV and VNAV) by coupling the aircraft's navigational computer systems, Inertial navigation, Flight Management System (FMS) etc) to the inner loops of the autopilot.

38. The principal components of an automatic flight control system are as follows:

Autopilots. As already stated, large passenger transport aircraft are equipped with two or possibly three autopilots. During most flight phases one of the autopilot systems, or channels, is flying the aircraft in response to inner and/or outer loop commands. During automatic landing all the autopilot channels are engaged, in order to provide safety through redundancy.

Flight Director. A conventional flight director system senses deviation from the planned flight path in terms of deviation rate, attitude and rate of change of attitude and presents this as correction command information on the pilot's attitude direction indicator (ADI) and horizontal situation indicator (HSI). In automatic flight these computed deviation commands may be fed to the autopilot(s).

Flight Management System (FMS). A flight management system (FMS) is a complex digital system which is capable of controlling the aircraft's flight path and thrust management to achieve either least-time or least-cost operation. It usually receives data from such sources as a flight control computer, an air data computer, a thrust management computer, the engine indicating and crew alerting system (EICAS), the pilot's controls and various airframe, engine and navigational sensors. Its outputs are the ‘outer loop’ signals to the autopilots and automatic throttle controls (autothrottle). In some aircraft this system is alternatively known as the Autopilot Flight Director System (AFDS).

Central Air Data Computer (CADC). (Figure 22-7) The CADC receives what is known as manometric data from the aircraft's pitot and static sources, together with a transducer generated signal representing the true air temperature (TAT). The computer is capable of producing computed airspeed (in effect IAS, true airspeed (TAS), Mach number and pressure altitude. The CADC outputs are used for automatic speed, rate of climb/descent, altitude and Mach trim control.

Autothrottle. The automatic throttle thrust control system adjusts engine thrust to suit the flight phase conditions through take-off, climb, cruise, descent, approach and landing or go-around.

39. The outer loop data inputs, which are relevant to the aircraft's required flight path, are 'coupled' to the autopilot system to achieve conditions of 'hold', 'lock' or 'capture'. For example, a system maintaining automatic flight at a pre-selected altitude would be said to be operating in height lock (or altitude hold) mode. 'Capture' refers to the interception of radio beams from ground-located aids such as ILS or VOR. Interception of, and locking to, the ILS glideslope is known as 'glideslope capture' (and subsequently glideslope hold). Similarly, interception of, and locking to, the ILS localiser beam or VOR radial is known as 'VOR/LOC capture' (and subsequently VOR/LOC hold). Glideslope hold provides controlled flight in the vertical plane (rate of descent), whereas VOR/LOC hold provides controlled flight in the lateral plane (directional control).

40. Typical outer loop data inputs are as follows:

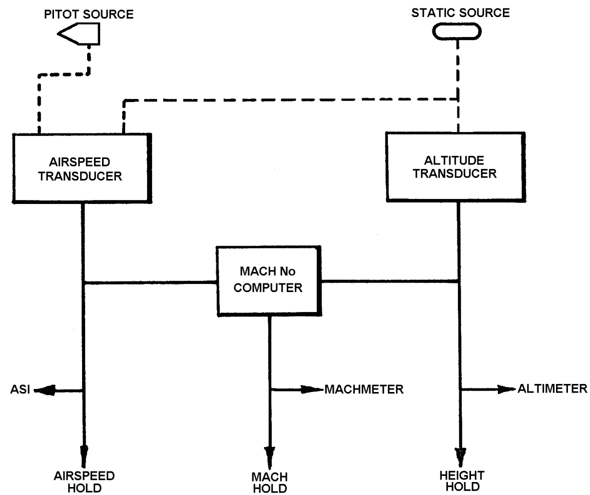
Air data (manometric data). These are data inputs which apply control about the aircraft's pitch axis:

Altitude select and hold. A manually-selected pressure altitude is fed to the pitch control inner loop in the form of an electrical signal. The inner loop controller compares this with another signal, proportional to actual static pressure, supplied from a transducer operated by a static pressure capsule in the Central Air Data Computer (CADC). The pitch control inner loop will adjust the elevator, via its servomotor to climb or descend the aircraft until the two signals agree.

Airspeed select and hold. Airspeed is controlled by sensing dynamic pressure (pitot minus static pressure). A manually selected IAS is fed to the pitch control inner loop as an electrical signal, which is compared by the loop controller with a signal from a pitot pressure-operated transducer in the CADC. In certain modes involving climb or descent (but not in the vertical speed mode) a biasing signal may be applied to the pitch control inner loop to maintain constant dynamic pressure (IAS). This would occur during a long climb at maximum permitted power, or a glide descent with the throttles at idle. In both of these cases it is necessary to control the airspeed by adjusting the pitch attitude, since variations of power are not available to achieve this end. In level flight and when climbing/descending in the vertical speed mode the airspeed is normally controlled (automatically or manually) by the throttles.

Mach hold. At high altitudes airspeed hold is usually replaced by Mach hold. By integrating the altitude and airspeed a Mach signal output can be generated, and may be used to bias the pitch control loop in certain modes involving climb or descent, as described in the preceding paragraph, or to control the autothrottles in level flight or when climbing/descending in the vertical speed mode. The combination of signals to produce a Mach hold output is accomplished by the central air data computer (CADC), as shown at Figure 22-7.

FIGURE 22-7
Central Air Data Computer (CADC) - Schematic (TAS output omitted)



Vertical speed hold. During climb or descent a constant rate of climb or descent may be required, and this can be automatically maintained by the CADC. A reference vertical speed is inserted by the pilot and the CADC compares rate of change of altitude (from the static pressure sensing) with the rate of change of altitude demanded by the pilot, biasing the pitch control loop as necessary to maintain a constant rate of change and automatically adjusting the throttles to ensure a safe IAS/Mach within pre-set maximum and minimum thrust (EPR) limitations.

41. In many systems vertical speed hold and altitude hold are selected with the same controller, which takes the form of a thumb-wheel, with a centre detent ‘altitude hold’ position, and which is rotated in the appropriate sense for climb or descent. The system can be further sophisticated to provide such features as pitch trim and vertical navigation. The transducers which convert static and pitot pressure into electrical signals are, typically, pressure capsule operated E and I transformers. The principle of the E and I transformer is covered in the altimeter section of the Instruments notes. With the autopilot engaged, the maximum pitch angle will be limited to a pre set value.

42. **Heading select and hold** obviously implies automatic maintenance of flight on a selected magnetic heading, which involves directional control. This is achieved through the ailerons, so heading hold relates to the roll control loop. Magnetic heading reference is from the aircraft's gyromagnetic compass system, also described in full in the Instruments notes. With the autopilot engaged, a limit on the maximum bank angle is pre set.

Instrument landing system (ILS). Localiser and glidepath information from the aircraft's ILS receiver comprises the outer loop inputs for this aspect of the automatic flight control system.

43. **VOR** receiver outputs may be coupled to the autopilot roll loop. The processed VOR signal, together with the manual input of the required radial, will enable the autopilot to control the flight path in relation to fixed locations on the ground. To avoid the errors which would occur within the VOR's ‘cone of confusion’, an automatic cut-off is activated on entering the cone. Whilst in the cone the roll control loop shifts into heading hold mode, until the VOR signal returns.

INS coupling. The INS, or IRS may be coupled to the autopilot pitch, roll and yaw control loops. The autopilot will then maintain the INS computed track through sequential waypoints.

44. Combination of all these data inputs will result in automatic control of the aircraft's planned flight path, in both azimuth and elevation, from take-off to touch-down. Control in azimuth is known as lateral navigation (LNAV) and control in elevation as vertical navigation (VNAV). Throughout all flight phases the automatic flight control system is normally providing both LNAV and VNAV control.

45. Most automatic flight control systems also provide the pilot with the ability to manoeuvre the aircraft in pitch and roll, without the need to disengage the autopilot. This may be in the form of pitch and turn controls on the automatic flight system control panel, or it may be by means of control wheel steering (CWS). With CWS, applying normal manoeuvring forces to the control wheel (with the autopilot engaged) achieves the desired result in that the aircraft pitches and/or rolls in response to the manual inputs but the autopilot doesn't drop off line. Once the control wheel is released the autopilot holds new attitude. In some cases, if roll angle is less than 5° at the time of release the control system will roll the aircraft wings level and hold the aircraft on a heading established one second after levelling. An alternative system, known as touch control steering (TCS), employs a thumb switch on the control yoke. When activated, the autopilot is disengaged whilst the pilot flies the aircraft to the desired attitude. Once the switch is released, the autopilot re-engages to hold the desired attitude.

Servomotors. In aircraft which are not fitted with hydraulic power-operated controls, servomotors (or servo-actuators) are connected to the aircraft's primary flight control systems to move the elevators, rudders and ailerons/roll spoilers. These servomotors are usually either electro-pneumatic, electro-mechanical or electro-hydraulic devices, depending upon whether linear (push-pull) or rotary actuation is required. They may be connected in series with the manual flight controls (that is to say between the pilot's controls and the control surface) or in parallel with them. Series-connected servomotors move the control surface without moving the associated pilot's control. Parallel-connected servomotors move both the pilot's controls and the control surfaces.

Torque Limiters. The high aerodynamic loads which can occur in flight, when large or rapid control surface movements are made, can result in excessive stress loads on the aircraft structure. In manual control the pilot can 'feel' the control surface loads and apply control movement accordingly. An automatic flight control system lacks this 'feel' so safeguards must be provided against the stresses it could impose. These safeguards are achieved by placing a torque-limiting device between the servomotor and the flying control surface. If the servomotor tries to move the control surface too rapidly or too greatly, the torque needed to achieve this movement will exceed the tolerance of the torque limiter, which will either slip, or disengage. This also protects against a servomotor 'runaway', which would otherwise drive the control surface to full deflection. Torque limiters usually take the form of spring-loaded couplings and a friction clutch. If torque becomes excessive the spring force is overcome and the friction clutch separates.

Interlocks. It is important that certain operating requirements are met before the automatic flight system is engaged with the aircraft's flight controls. For example, all system power supplies need to be operational. To ensure that automatic flight control cannot be engaged until all the necessary parameters are satisfied a system of switches and relays, known as interlocks, is incorporated in the electrical supply to the engage switch. These interlocks are connected in series, so that all must be closed to complete the engage circuit.

Adaptive Control. This is the process whereby the gain factor of a servo control loop (inner loop) is altered automatically to offset variation of handling techniques. For instance, on a system with turn co-ordination, it may be required to isolate or reduce gain at IAS's above or below certain limits.

Flight Mode Annunciator (FMA). Auto pilot, flight director and auto throttle status is presented on a flight mode annunciator (FMA) located on the flight deck. Using a series of coloured indicators, any selected mode is shown in the 'armed' or 'captured' mode. In addition to this there are aural alerts which will indicate to the pilot that auto pilot disconnection has taken place either manually or automatically. During normal flight, the modes illuminated are those in actual use. Some auto pilot systems incorporate red warning lights for signal failure in that system. On an EFIS fit aeroplane, the mode annunciation is incorporated into the EADI and EHSI display. Figure 22-4 illustrates an example of a flight mode annunciator (FMA).

Auto Pilot Phase Condition

46. **Take Off.** The auto pilot is **not used** for take off. This is a flight director only function. Flight director pitch and roll commands are displayed on the ADI and the autothrottle (if engaged) maintains take off thrust (IN or EPR) as selected.

47. **Climb.**

(a) **Pitch modes** can be selected as required and will control:

(i) Altitude hold (ALT HOLD)

(ii) Altitude select (ALT SEL)

(iii) Level change (LVL CHG)

(iv) Vertical speed (V/S)

(v) Airspeed - IAS/MACH HOLD

(vi) VNAV (with FMS)

(b) **Roll modes** can be selected as required controlling:

(i) VOR course (VOR/LOC)

(ii) Heading select (HDG/SEL)

(iii) LNAV (with FMS)

(iv) INS - Waypoint Navigation

48. **Cruise. Roll modes:**

(i) VOR Course (VOR/LOC)

(ii) Heading select (HDG/SEL)

(iii) INS - Waypoint navigation

(iv) LNAV (with FMS)

Vertical modes can be engaged to change levels

(v) **Pitch Modes** - Speed IAS/MACH

49. **Descent** - as per **Climb.**

50. **Approach.**

(a) **Pitch**. Descent, prior to final approach fix can be accomplished using VNAV, LVL CHG or V/S. Glideslope mode is engaged on intercepting the ILS glideslope.

(b) **Roll.** VOR/LOC, LVL CHG or V/S for descent or final approach.

51. **Land.** Except for auto land the auto pilot must be disconnected prior to landing.

52. **Go-around**. This mode is available when at least two auto pilot channels are engaged. It is armed automatically when flare arm is annunciated. The mode can be engaged by pressing a TO/GA (Take-off - Go-around) switch. On engagement, the auto throttle (if engaged) advances the thrust levers to go-around thrust. The auto pilot will command a pre-set nose up pitch altitude to climb the aircraft at a pre-programmed rate of climb whilst maintaining existing track.

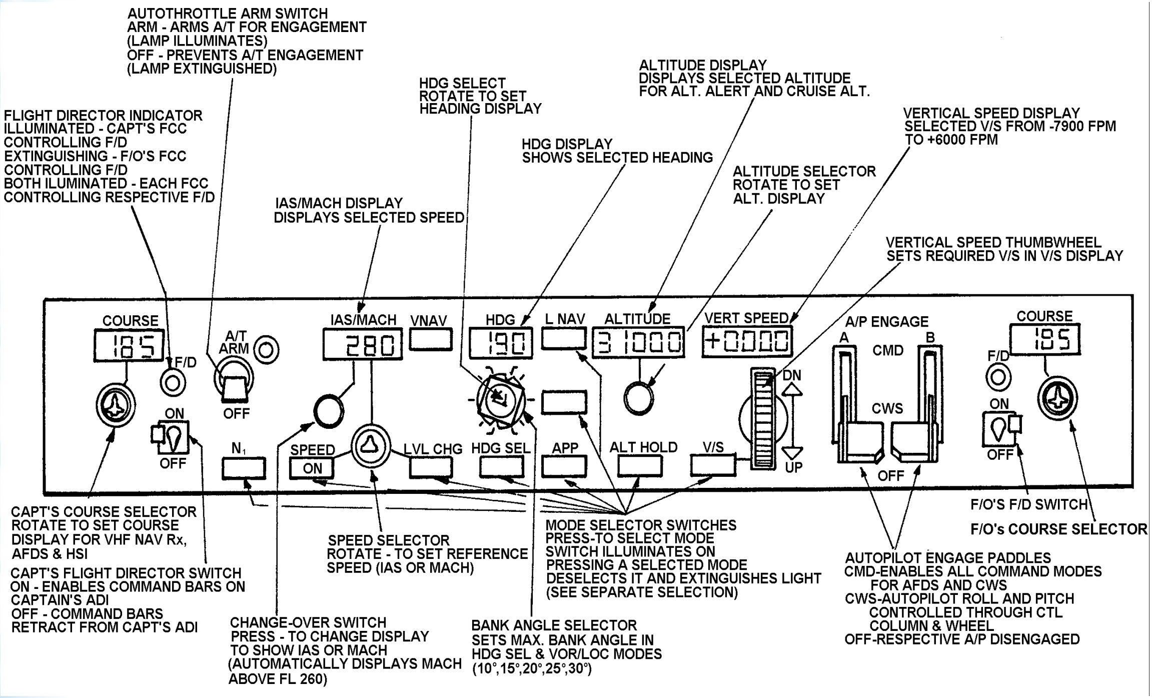
AFDS Command Modes

Boeing 737 Automatic Flight System

53. The following is a description of the fail passive automatic flight system fitted to many Boeing 737-400 series aircraft. Other aircraft systems will differ in detail, but the general principles are common to most. A fail operational system would have three, or possibly more, autopilots. It should be borne in mind that the examination syllabus is not type-related and specific data in the following descriptions need not be remembered. Autoland data for examination purposes will be based upon that given at Figure 22-8.

54. The Automatic Flight System comprises an Autopilot Flight Director System (AFDS) and an Autothrottle (A/T) System. A Flight Management Computer (FMC) provides command information to the autothrottle system to ensure that N_1 (fan) rpm does not exceed limits at high engine powers and that target values of N_1 are met. The FMC also provides command airspeeds for the A/T and AFDS. The AFDS and A/T are operated by the pilots from the AFDS Mode Control Panel, illustrated at Figure 22-8, which provides co-ordinated control of the autopilot(s), flight director, autothrottle and altitude alert functions. The status of the automatic flight system is displayed on the Flight Mode Annunciators on each pilot's EFIS ADI display

FIGURE 22-8
AFDS Mode Control Panel



Autopilot Flight Director System. This is a dual system comprising two independent Flight Control Computers (FCC), identified as A and B, with a single Mode Control Panel. The Flight Control Computers send control commands to their respective autopilot roll and pitch loops, which operate the flight controls through their own, independent, hydraulic servo systems. For Flight Director operation, each FCC positions the Flight Director command bars on the Captain's and First Officer's ADI.

55. Flight Director operation is selected by means of the Captain's and First Officer's Flight Director switch and is available in both manual and automatic flight. Command bar positioning will depend upon the command modes selected. The appropriate mode selector switches are pressed to select desired command modes. The switches illuminate ON to indicate mode selection, a mode can be deselected by pressing an illuminated switch. If engagement of a mode would conflict with the current Automatic Flight System operation, pressing that mode selection switch has no effect. All AFDS modes can be disengaged by either selecting another command mode or by disengaging the autopilot and switching off the Flight Directors.

56. Autopilot engagement/disengagement is achieved by means of the Engage Paddles. The autopilot may be engaged in either Control Wheel Steering (CWS) or Command (CMD). During single channel autopilot operation (all modes other than Approach [APP]) only one autopilot can be engaged, engaging the second autopilot in CMD or CWS disengages the first.

57. **Autopilot CWS Operation**. Moving an engage paddle to CWS engages that autopilot roll and pitch in the CWS mode, so that the autopilot manoeuvres the aircraft in response to control column/wheel pressure applied by either pilot. The control forces required are similar to those in manual operation. When the control column is released the autopilot holds the existing aircraft attitude unless aileron pressure is released with 6° or less of bank angle, in which case the autopilot rolls the aircraft wings level and holds existing heading.

58. When the autopilot is engaged in CMD the command roll and pitch modes can be manually overridden by applying control wheel/column force greater than the normal CWS/manual force. This will place the autopilot in CWS operation. Manual pitch override is inhibited when the autopilot Flight Director System is in APP mode with two A/Ps engaged. Whether A/P operation remains in CWS after an override, or returns to CMD, depends upon the AFDS modes selected and the extent of deviation during the manual override.

59. Command modes can be armed or engaged when an autopilot engage paddle switch is selected to CMD and/or one or both Flight Director switches are ON.

Altitude Acquire Mode. This is a transition manoeuvre entered automatically from a Vertical Speed (V/S), Level Change (LVL CHG) or VNAV climb or descent to a selected altitude, set by means of the Altitude Selector knob on the Mode Control Panel and displayed digitally in the Altitude Display window.

Altitude Hold (ALT HOLD) Mode. The altitude hold mode will, by means of pitch commands, either maintain the aircraft at the selected altitude or adjust the aircraft's altitude until the selected altitude is attained. Altitude hold mode, whilst not at the selected altitude, is initiated by either pressing the ALT HOLD selector switch or by selecting a new altitude whilst in ALT HOLD at the current selected altitude. Whilst changing altitudes, with ALT HOLD depressed, the selector switch will be illuminated. Upon reaching the selected altitude the ALT HOLD switch light is extinguished. When in ALT HOLD at the selected altitude LVL CHG, V/S and VNAV climb and descent modes are inhibited.

Vertical Speed (V/S) Mode. The V/S mode gives pitch commands to the autopilot to hold the selected vertical speed (ROC/ROD) and to the autothrottle to hold the selected IAS. The V/S mode has both an armed and an engaged state. Pressing the V/S selector switch engages V/S mode (unless engaged in ALT HOLD or after glideslope capture). This is annunciated and the vertical speed display changes from blank to show present V/S. Desired V/S can now be selected with the V/S thumbwheel.

60. V/S mode becomes armed if, whilst in ALT HOLD at the selected altitude, a new desired altitude is selected which differs by more than 100 ft from that previously selected. V/S armed is annunciated and V/S mode is engaged by moving the thumbwheel.

Level Change (LVL CHG) Mode. The LVL CHG mode co-ordinates pitch and thrust commands to make automatic climbs and descents to pre-selected altitudes at selected airspeeds. A LVL CHG climb or descent is initiated by selecting a new altitude and pressing the LVL CHG mode selector switch to engage the mode.

Vertical Navigation (VNAV) Mode. Pressing the VNAV mode selector engages VNAV mode and the Flight Management Computer commands AFDS pitch and autothrottle to fly the pre-selected vertical flight profile held in the FMC. This profile includes pre-programmed climbs, cruise altitudes, speeds, descents and height constraints at specified waypoints. The vertical profile usually ends with an ILS approach to the destination airport. The EFIS ADI displays with VNAV engaged are VNAV PTH or VNAV SPD for the pitch engaged mode and SPD, N_1, RETARD or ARM for the autothrottle engaged mode.

Lateral Navigation (LNAV) Mode. With LNAV engaged the Flight Management Computer applies roll commands to the AFDS to intercept and track the pre-programmed active route, including terminal approach procedures such as STARs and ILS approach. For LNAV to engage there must be an active route programmed into the FMC. LNAV mode will automatically disengage if the active route is not captured within certain criteria or if overridden by selecting HDG SEL.

61. During automatic flight along a pre-programmed vertical and lateral flight path, VNAV and LNAV are the usual pitch and roll engaged modes of AFDS operation.

Heading Select (HDG SEL) Mode. Pressing the HDG SEL switch sends roll commands to the AFDS to turn the aircraft onto, and maintain, the heading set in the heading display. The bank angle during the turning manoeuvre is limited by the Bank Angle Selector.

VOR/LOC Mode. Pressing the VOR/LOC switch sends roll commands to the AFDS to turn the aircraft onto, and maintain, a selected VOR course if a VOR frequency is tuned, or the localiser inbound front course if a localiser frequency is tuned.

Approach (APP) Mode (Dual Autopilot Operation). When APP is selected the AFDS is armed to capture and hold the ILS localiser and glideslope. For a fully automatic landing both autopilots must be engaged for dual operation. This provides fail passive control throughout the landing flare and touchdown, or an automatic go-around. During fail passive operation the flight controls respond to the autopilot demanding the lesser control movement, thus providing protection against servomotor runaway. The approach and landing sequence and profile is similar to that described at Figure 22-9 and the associated text, with minor differences in mode change radio altitudes.

Automatic Go-Around. The autopilot go-around mode requires dual autopilot operation and arms automatically when FLARE ARMED is annunciated. Go-around mode is engaged by pressing a TO/GA (take-off/go-around) switch. Upon engagement the autothrottle advances the thrust levers for go-around N_1 rpm.

62. The autopilot initially commands a 15° nose-up pitch attitude, to climb at a programmed rate and to maintain existing track. Once a radio altitude of 400 ft is attained other pitch and roll modes may be selected, below 400 ft RA the autopilots must be disengaged to change pitch or roll modes from Go-Around.

63. **The two Flight Directors** operate in the same command modes as the autopilot, but drive the command bars on the Captain's and First Officer's ADIs. Exceptions to this are:

64. **Take-Off**, which is a Flight Director only mode (see below).

Flare. There is no flare capability during a normal (rather than an automatic) landing approach. At approximately 50 ft RA on an ILS approach the command bars retract from view.

65. In take-off mode the FD initially commands 10° nose-down pitch and wings level. At 60 kt on the take-off roll the pitch command changes to 15° nose-up, wings level, and remains there until a sufficient rate of climb is acquired. Thereafter it commands pitch to maintain selected IAS plus 20 kt. Above 400 ft RA an autopilot can be engaged in CMD.

66. **The Autothrottle** moves the thrust levers by means of a separate servomotor for each lever. It is engaged by moving the autothrottle arm switch to ARM. The autothrottle can be selected to operate in a number of modes, described in the following text.

Take-Off Mode. This is engaged by pressing either TO/GA switch with the aircraft on the ground, the autothrottle armed and desired take-off N_1 rpm selected on the Flight Management Computer (FMC) controls. The autothrottle then advances the thrust levers to take-off thrust. At 64 kt IAS thrust hold engages to prevent the autothrottle changing thrust lever positions until 400 ft RA is attained and approximately 18 seconds after lift-off have elapsed. Reduction of thrust to climb thrust can then be made by pressing the N_1 selector switch.

N_1 Mode. In this mode the autothrottle positions the thrust levers to maintain the limiting N_1 set on the FMC.

Speed Mode. This is available throughout the flight, once the take-off phase is completed. The autothrottle positions the thrust levers to maintain the selected target speed, but will not allow the N_1 limit to be exceeded. The autothrottle attempts to equalise N_1 on both engines, but will not permit greater than 8° of thrust lever disagreement.

Retard Mode. During a LVL CHG or VNAV descent the autothrottle retards the thrust levers, if necessary until they reach the aft stop. During landing retard mode engages 22 seconds after FLARE is engaged, or at approximately 27 ft RA, whichever occurs first.

Go-Around Mode. The autothrottle GA mode arms when descending below 2000 ft RA and may be engaged at any time until 2 seconds after touchdown, by pressing either TO/GA switch. Once engaged, the autothrottle advances the thrust levers to the reduced GA setting for a 1000 to 2000 ft per minute climb rate.

Automatic Landing (Autoland)

67. An automatic flight control system incorporating autoland facilities must embody safety features that ensure that:

(a) There is sufficiently rapid response to prevent deviation from the flight path in the event of disturbances (windshear, turbulence).

(b) The effect of a servomotor 'runaway' must be limited, so that safe recovery action can be taken by the pilot.

(c) 'Passive' control system failures (those which do not cause an immediate deviation from the flight path) are indicated to the pilot.

68. Consequently, to meet the requirement of a safe approach and landing an autoland system must:

(a) Not deviate from the flight path as the result of an active (runaway) malfunction.

(b) Have sufficient control authority for accurate maintenance of the flight path.

(c) Incorporate warning of passive failure.

(d) Not prevent completion of the intended manoeuvre, following an active or passive failure.

69. The criteria are met by the adoption of the system-redundancy concept, whereby duplication or triplication of systems ensures that a single failure within a system has an insignificant effect on overall performance during approach and landing.

70. There are two types of autoland systems in use, the first is called a fail passive (fail soft) system and the second a fail operational (fail active) system.

71. An automatic landing system is fail passive if, in the event of a failure, there is no significant out-of-trim condition or deviation of flight path or altitude but the landing is not completed automatically. Essentially this means that, should the system fail, it will not present the aeroplane to the pilot in a seriously out of trim condition or a point where the aircraft has already deviated seriously from the localiser, the glideslope or the target airspeed. For a fail passive automatic landing system the pilot assumes control of the aircraft after a failure.

72. To achieve these requirements a fail passive system incorporates two, independent, autopilot channels, both of which must be engaged for the approach, and a monitoring system which ensures that there are no 'out of limits' discrepancies between each of the autopilot channels.

73. Should one of the autopilot channels (or indeed the monitoring system) fail during the approach to an automatic landing, the approach will be continued on a single autopilot channel. An autoland is no longer possible and the crew should revert to category 1 minima, either landing manually or going around, depending on what is (or isn't) seen at the category 1 decision height. Ample warnings are given to the pilots, however, should they ignore them, the remaining autopilot channel will disengage itself, typically at 350 ft (radio altimeter). It will do this because, due to the malfunction, it has been unable to 'arm' the autoland functions (flare, throttle retard and roll-out).

Fail operational. An automatic landing system is fail operational if, in the event of a failure below alert height, the approach, flare and landing, can be completed by the remaining part of the automatic system. In the event of a failure, the automatic landing system will operate as a fail passive system. Such a system implies a considerable degree of both redundancy and self-monitoring. This is achieved by incorporating three or more independent autopilot channels, all of which are engaged for an automatic approach and landing. Thus, if one channel suffers a failure it can be disengaged without compromising the ability of the automatic flight control system to complete a fully automatic landing. The number of engaged autopilot channels is shown on the Electronic Flight Information System (EFIS) display as a caption. LAND 3 indicates three channels engaged and a fail operational system. LAND 2 indicates two channels engaged and a fail passive system. LAND 1 indicates only a single channel engaged which, during automatic approach means the system has suffered a passive failure and will not complete an automatic landing. During all other flight phases (climb, cruise, descent) only one autopilot channel is engaged.

74. **Alert height** is a specified radio height, based on the characteristics of the aeroplane and its fail operational landing system. In operational use, if a failure occurred above the alert height in one of the required redundant operational systems in the aeroplane, (including, where appropriate, ground roll guidance and the reversionary mode in a hybrid system) the approach would be discontinued and a go-around executed unless reversion to a higher decision height is possible. If a failure in one of the required redundant operational systems occurred below alert height, it would be ignored and the approach continued.

75. **A multiplex control system**, is one containing two or more independent systems and embodies monitoring devices which compare system functions against a datum, selecting the most accurate and discarding those outside pre-set limits. Multiplex systems may be duplex, triplex or even quadruplex, depending upon the number of independent flight controls systems incorporated.

76. In the approach/land mode, gain, flare and decrab programmes are utilised when the aircraft is coupled to the instrument landing system in the localiser and glidslope modes. The purpose of the gain programme is to reduce the gain of the beam deviation signals which will allow for convergence of the LOC and CS beams. The decrab programme allows for the effects of crosswinds. Without this the aircraft would turn up at a position downwind of the localiser beam centre at a particular drift or 'crab' angle. The flare programme mode will take over from the glideslope and will generate a pitch command signal to bring the aircraft on to a pre-set rate of descent (typically 2 feet/sec) for the final part of the approach to the runway. Just prior to touchdown the flare mode will be automatically disengaged.

Low Visibility Landings (Autoland)

77. The following paragraphs are extracted from the JAR-AWO (All Weather Operations) dealing with low visibility landings. It deals with the relationship between the type of autoland system employed and the decision height for Category 2 and 3 Autolandings.

Category 2 Operations

78. An ILS precision approach with a decision height below 60m (200ft) down to 30m (100ft).

Jar Requirements

General

Safety Level. The safety level for precision approaches with decision heights below 60m (200ft) down to 30m (100ft) must not be less than the average safety level achieved in precision approaches with decision heights of 60m (200ft) and above. Hence, in showing compliance with the performance and failure requirements, the probabilities of performance or failure effects may not be factored by the proportion of approaches which are made with the decision height below 60m (200ft).

Go-around rate. The proportion of approaches terminating in a go-around below 150m (500ft) due to the approach system performance or reliability may not be greater than 5%.

Control of flight path. The approach system must either:

(a) Provide information of sufficient quality to the flight crew to permit the manual control of the aeroplane along the flight path within the prescribed limits; or

(b) Automatically control the aeroplane along the flight path within the prescribed limits.

Control of speed. Automatic throttle control must be provided unless it is demonstrated in flight that speed can be controlled manually by the crew within acceptable limits and without excessive workload. When making an approach using an automatic throttle system the approach speed may be selected manually or automatically.

Manual control.

(a) In the absence of a failure, the approach down to the decision height must not require a change in the means of control (eg. a change from the automatic flight control system to flight director).

(b) The use of a manual mode or the transition from an automatic mode to manual control must not require exceptional piloting skill, alertness or strength.

Oscillations and deviations. The approach system must cause no sustained nuisance oscillations or undue attitude changes or control activity as a result of configuration or power changes or any other disturbance to be expected in normal operation.

Decision height recognition. Decision height recognition must be by means of height measured by a radio altimeter.

Go-around. The go-around may not require exceptional piloting skill, alertness or strength to maintain the desired flight path.

Equipment

Installed equipment. The approach guidance system must include:

(a) Two ILS glidepath and localiser receivers with indication at each pilot's station;

(b) An automatic approach coupler or a flight director system with display at each pilot's station (or an alternative giving equivalent performance and safety);

(c) A radio altimeter with displays at each pilot's station of:

(1) radio altitude, and

(2) the selected decision height (eg. an index on an analogue scale or a digital indication);

(d) Clear visual indication at each pilot's station (eg. an alert light) when the aeroplane reaches the pre-selected decision height appropriate to the approach;

(e) Automatic or flight director go-around system or acceptable attitude indicators;

(f) Audible warning of automatic pilot failure (for automatic approach);

(g) An automatic throttle system where necessary;

(h) An appropriate equipment failure warning system; and

(i) Excess-ILS-deviation alert at each pilot's station (eg. amber flashing light).

Minimum equipment. The minimum equipment which must be serviceable at the beginning of an approach, for compliance with the general requirements of this Subpart 2 and those relating to performance and failure conditions, must be established, For example, where justified by a system safety assessment in accordance with 25.1309 one ILS receiver may be unserviceable.

Performance

Flight path and speed control. The performance of the aeroplane and its systems must be demonstrated by flight tests supported where necessary by analysis and simulator tests. Flight testing must include a sufficient number of approaches conducted in conditions which are reasonably representative of actual operating conditions and must cover the range of parameters affecting the behaviour of the aeroplane (eg. wind, speed, ILS characteristics, aeroplane configurations, weight, centre of gravity etc).

Decision height. The decision height must not be less than 1.25 times the minimum permissible height for the use of the approach systems.

Localizer and glide path receivers. The localizer and glide path receivers must comply with the minimum performance standards of EUROCAE ED-46 and ED-47, or RTCA DO-131A and DO-132A respectively as follows:

Localizer Class D

Glide Path Class D

Radio altimeters. The radio altimeter must comply with the minimum performance standards of EUROCAE ED-30 or RTCA DO-155.

Excess-deviation alerts.

(a) Excess-deviations alerts must operate when the deviation from the ILS glide path or localizer centre line exceeds a value from which a safe landing can be made from offset positions equivalent to the excess-deviation alert, without exceptional piloting skill and with the visual references available in these conditions.

(b) They must be set to operate with a delay of not more than 1 second from the time that the values determined in the sub paragraph are exceeded.

(c) They must be active at least from 90m (300ft) to the decision height but the glide path alert should not be active below 30m (100ft)

Go-around climb gradient. The aeroplane Flight Manual must contain either a WAT (Weight, Altitude, Temperature) limit corresponding to a gross climb gradient of 2.5% with the critical engine failed and with the speed and configuration used for a go-around, or the information necessary to construct a go-around gross flight path with an engine failure at the start of the go-around from the decision height.

Controls, Indicators and Warnings

Mode selection and switching.

(a) A positive and continuous indication must be provided of the modes actually in operation. In addition, where engagement of a mode is automatic (eg. localizer and glide path acquisition) clear indication must be given when the mode has been armed by a member of the flight crew.

(b) Where reliance is placed on the pilot to detect non-engagement of go-around mode when it is selected, an appropriate indication or warning must be given.

(c) The system must be designed so that no selection or changes of switch settings (other than system disengagement) need to be made manually below a height of 150m (500ft) in the absence of a failure.

Presentation of Information to the crew.

(a) The display of information to the crew, including that required to monitor the flight path, must be compatible with the procedures specified in the aeroplane Flight Manual. All indications must be designed to prevent crew errors.

(b) Essential information and warnings necessary to the crew in the use of the approach system must be so located and designed as to permit both their accurate use in normal operation and the rapid recognition of malfunctions, in all expected lighting conditions.

Audible warning of automatic pilot disengagement.

(a) Where the approach flight path is controlled automatically, an audible warning must be given following disengagement of the automatic pilot or loss of the automatic approach mode.

(b) For aeroplanes with automatic landing systems, the same warning must be used for automatic approach as is used for automatic landing.

Failure Conditions

Automatic pilot. The automatic pilot must comply with Jar Requirements.

Flight director system.

(a) The flight director system, or alternative form of information display, must be so designed that the probability of display of incorrect guidance commands to the pilot is remote when credit is taken for an excess-deviation alert.

(b) The deviation profile method must be used in assessing failures of flight director systems.

(c) Wherever practicable, a fault must cause the immediate removal from view of the guidance information but, where a warning is given instead, it must be such that the pilot cannot fail to observe it whilst using the information.

Radio altimeter. The radio altimeter must be such that the probability of the provision of false height information leading to a hazardous situation is Extremely Remote. The warning must be given by the removal or obscuration of displayed information, at least in the height band from 30m (100ft) downwards.

Excess-deviation alerts. The excess deviation alerts must be such that the probability of failure to operate when requires is not Frequent.

Aeroplane Flight Manual

General. The aeroplane Flight Manual must state:

(a) Limitations, including the minimum decision height to which the aeroplane is certificated;

(b) Normal and abnormal procedures;

(c) Changes to the performance information, if necessary (eg, approach speed, landing distance, go-around climb); and

(d) Minimum required equipment, including flight instruments.

(e) The maximum head, tail and cross wind components in which the performance of the aeroplane has been demonstrated.

Category 3

The criteria for Category 3 operations are divided into the following types of operation:

- CAT 3 A - Decision heights below 30m (100ft) but not less than 15m (50ft).
- CAT 3 B - Decision heights below 15m (50ft)
- No Decision Height.

The minimum RVR (Runway Visual Range) provides visibility at and below the decision height so that, if either the automatic landing system or the ILS fails when the aeroplane is below the decision height, the pilot can carry out a manual landing with an acceptable safety level. The ground guidance system is a CAT III or a CAT II ILS.

Decision height below 15m (50ft). Aeroplanes with a fail-operational landing system are certified to auto land with a decision height below 15m (50ft).

In this type of operation, the RVR will enable the pilot to make a decision at decision height and in addition be able to control the aeroplane during the ground roll. The RVR limit is usually set in the range 200m to 150m.

No decision height. Aeroplanes with fail-operational landing systems with automatic ground roll control (or ground guidance) are certified to auto land without a decision height. If the ground roll system is fail operation, no RVR limit is necessary. If it is not fail operational, then the RVR limit is applied to enable the pilot to control the aeroplane safely on the ground in all likely conditions.

In these visibility conditions, hard braking may be required on the ground roll and therefore the anti-skid system must be operational.

Requirements

General

Safety level. The safety level for precision approaches with decision heights below 30m (100ft) or no decision height may not be less than the average safety level achieved in precision approaches with decision heights of 60m (200ft) and above. Hence, in showing compliance with the performance and failure requirements, the probabilities of performance or failure effects may not be factored by the proportion of approaches which are made with the decision height below 30m (100ft).

Go-around rate. The go-around rate below 150m (**500ft**) attributable to the landing system performance or reliability may not be greater than 5%. Additionally, for decision heights below 15m (50ft) and no decision height, the probability of go-around below the alert height attributable to the landing system performance and reliability must be such that compliance is achieved (Previous paragraph - Safety level).

Minimum flight crew. The workload associated with use of the minimum decision height must be considered.

Control of flight path and ground roll. The landing system must control the aeroplane within the prescribed limits along the flight path to touchdown and along the runway when appropriate and specifically:

(a) The primary mode of controlling the aeroplane must be automatic until the main wheels touch the ground and for operation with no decision height, control must be automatic until the nose wheels touch down;

(b) For decision heights below 15m (50ft) a fail-operational landing system (automatic or hybrid) must be provided which, when appropriate, includes provision for control of the aeroplane along the runway during the ground roll down to a safe speed for taxi.

Ground roll control.

(a) Limitations on visibility conditions must be established in accordance with the following:

(1) If the aeroplane has neither automatic ground roll control nor a ground roll guidance display, operation will be limited to visibilities in which the pilot can always control the ground roll by visual reference;

Note. Precise RVR values to be used are subject to operational regulation and will vary from one State to another, but values in the region of 200 to 150m have been used.

(2) If either automatic ground roll control or a ground roll guidance display is fitted, but is fail-operational, the aeroplane may land in visibilities which are lower than in sub-paragraph (a)(1) of this paragraph but still are normally sufficient to allow the pilot to complete the ground roll safely in the event of failure of the system;

(3) Fail-operational ground roll control or guidance will permit operation with no visibility limitations.

(b) Where certification is requested for operation with no visibility limitations (see sub-paragraph (a)(3) of this paragraph) operation of automatic pilot quick-release control on the control wheel may not disengage the automatic ground roll function unless alternative head-up guidance remains available.

Control of speed. Automatic throttle control must be provided unless:

(a) The decision height is 15m (50ft) or greater; and

(b) It is demonstrated in flight that speed can be controlled manually by the crew within acceptable limits and without excessive work-load.

Manual control. The transition from an automatic mode to manual mode or the use of a manual mode may not require exceptional piloting skill, alertness or strength.

Oscillations and deviations. The landing system may cause no sustained nuisance or undue attitude changes or control activity as a result of configuration or power changes or any other disturbance to be expected in normal operation.

Alert height. For a fail operational system with a decision height below 15m (50ft) or with no decision height, an alert height must be established and must be at least 30m (100ft).

Note. It may be operationally useful for the alert height to be somewhat higher than 30m (100ft) since this would permit reversion to a higher decision height in the event of system failure. A maximum value should be established during certification and it should not normally be above 90m (300ft).

Minimum break-off height. The minimum approach break-off height (MABH) must be established such that:

(a) A catastrophic effect is extremely improbable if a go-around is carried out without external visual references according to the standard procedure; and

(b) With all engines operation, the probability that the aeroplane will touch the ground during this procedures is less than 10^4.

Decision height. The decision height may not be lower than the MABH. When the decision height is during the landing flare, it must be below the height at which the major attitude changes associated with this manoeuvre take place.

Decision height recognition. Decision height recognition must be by means of height measured by a radio altimeter.

Go-around.

(a) The go-around from any point on the approach to touchdown may not require exceptional piloting skill, alertness or strength and must ensure that the aeroplane remains within the obstacle limitation surface specified in ICAO Annex 14, for a precision approach runway Category II or III.

(b) For decision heights below 15m (50ft) automatic go-around must be provided.

(c) When automatic go-around is provided, it must be available down to touchdown.

(d) When automatic go-around is engaged, subsequent ground contact should not cause its disengagement.

Equipment

Installed equipment. The following items of equipment must be installed for certification to the decision heights specified unless it is shown that the intended level of safety is achieved with alternative equipment, or the deletion of some items:

Note. This list is based on experience with conventional medium and large jet transports and it is recognised that changes may be appropriate in significantly difference applications.

(a) All decision heights below 30m (100ft) or no decision height:

(1) Two ILS glide path and localizer receivers with the first pilot's station receiving information from one, and the second pilot's station receiving information from the other;

(2) One radio altimeter with display at each pilot's station;

(3) Clear visual indication at each pilot's station (eg. an alert light) when the aeroplane reaches the pre-selected decision height appropriate to the approach;

(4) An appropriate equipment failure warning system; and

(5) An excess-ILS-deviation alert at each pilot's station (eg. amber flashing light).

(6) In case of aeroplanes having a minimum flight crew of two pilots, an automatic voice system, which calls when the aeroplane is approaching the decision height (or when approaching the ground during a no decision height approach) and when it reaches decision height).

Note. The number of ILS receivers and radio altimeters may need to be increased in order to provide fail-operational capability where required.

(b) Decision height 15m (50ft) or greater:

(1) Fail-passive automatic landing system or

(2) Fail-passive automatic approach system without automatic landing, provided that:

(i) It is demonstrated that manual landings can be made without excessive workload in the visibility conditions; and

(ii) The aeroplane has a low approach speed, and is manoeuvrable and the height of the pilot's eyes above the wheels is small;

(3) Automatic throttle control, unless it can be shown that speed control does not add excessively to the crew work-load; and

(4) Automatic or flight director go-around system or suitable attitude indicators.

(c) Decision height below 15m (50ft)

(1) Fail-operational automatic landing system or fail-operational hybrid landing system;

(2) Fail-passive automatic go-around;

(3) Automatic throttle control; and

(4) Automatic ground roll control or head-up ground roll guidance as appropriate to the limitations on visibility conditions or RVR.

(d) No decision height:

(1) Fail-operational automatic landing system;

(2) Fail-passive automatic go-around;

(3) Automatic throttle control;

(4) Fail-operational or fail-passive automatic ground roll control or head-up ground roll guidance (see JAR-AWO 304 and 305); and

(5) Anti-skid braking system.

Minimum equipment. The minimum equipment which must be serviceable at the beginning of an approach for compliance with the general criteria of this Subpart 3 and those relating to performance and failure conditions must be established.

Performance

Performance demonstration.

(a) Flight path and speed control must comply with the provisions of JAR-AWO.

(b) Touchdown performance of automatic landing systems must comply with the provisions listed in JAR-AWO. For operation with no decision height, compliance with the lateral touchdown performance criteria must be demonstrated at main wheel and nose wheel touchdown.

(c) The automatic throttle system must comply with the provisions of JAR-AWO.

(d) Compliance with JAR-AWO may be demonstrated primarily by flight test.

Compliance with sub-paragraphs (a) and (b) of this paragraph and JAR-AWO must be demonstrated by analysis and simulator tests supported by flight tests. Flight testing and any associated analysis must include a sufficient number of approaches and landing conducted in conditions which are reasonably representative of actual operation conditions and must cover the range of parameters affecting the behaviour of the aeroplane (eg. wind conditions, runway and ILS characteristics, aeroplane configurations, weight, centre of gravity).

Localizer and glide path receivers. The localizer and glide path receivers must comply with the minimum performance standards of EUROCAE ED-46 and ED-47 or RTCA DO-131A and DO-132A respectively as follows:

Localizer Class E

Glide Path Class D

The achieved centering accuracy must be taken into account in the analysis to show compliance with JAR-AWO.

Radio altimeters. The radio altimeter must comply with the minimum performance standards of EUROCAE ED-30 or RTCA DO-155.

Head-up display. Where a head-up display is fitted as part of a hybrid system, its performance need not meet the same criteria as the primary system provided that:

(a) It meets the overall performance requirements, taking into account the probability that it will be used; and

(b) It is sufficiently compatible with the primary system so as to retain pilot confidence.

Automatic ground roll control.

(a) When automatic ground roll control or head-up ground roll guidance is being used, the probability must be less than 5% than the point on the aeroplane centreline between the main wheel will deviate more than 8.2m (27ft) from the runway centreline on any landing.

(b) Additionally, when the operation is predicted on the provision of fail-operational ground roll control, the probability must be less than 10^{-6} that the outboard landing gear will deviate to a point more than 21.3m (70ft) from the runway centreline while the speed is greater than 40 knots.

Landing distance. If there is any feature of the system or the associated procedures which would result in an increase to the landing distance required the appropriate increment must be established and scheduled in the aeroplane Flight Manual.

Controls, Indicators and Warnings

Mode selection and switching.

(a) A positive and continuous indication must be provided of the modes actually in operation. In addition, where engagement of a mode is automatic (eg. localizer and glide path acquisition) clear indication must be given when the mode has been armed by a member of the flight crew.

(b) Where reliance is placed on the pilot to detect non-engagement of go-around mode when it is selected, an appropriate indication of warning must be given.

(c) The system must be designed so that no manual selections or changes of switch settings need to be made below a height of 150m (500ft) in normal operation, other than system disengagement or selection of automatic go-around as necessary.

Indications and warnings.

(a) The display of information to the crew, including that required to monitor the approach, flare and ground roll must be compatible with the procedures specified in the aeroplane Flight Manual and normal crew tasks. All indications must be designed to minimise crew errors.

(b) Essential information and warnings necessary to the crew in the use of the landing system must be so located and designed as to permit both their accurate use in normal operation and the rapid recognition of malfunctions in all expected lighting conditions.

(c) Any malfunction of the landing system or of the ILS glide path or localizer ground station which requires a missed approach must be annunciated positively and unambiguously to each pilot, so that pilot action may be initiated promptly without further interpretation.

(d) Notwithstanding sub-paragraphs (a), (b) and (c), of this paragraph, for fail-operational systems, failure warnings may be inhibited below alert height if:

(1) The failure does not preclude continuation of an automatic landing; and

(2) The failure requires no specific action of the flight crew; and

(3) Information on the occurrence of any failure warnings so inhibited is subsequently available to flight and maintenance crews.

(e) Where the capability of the aeroplane is dependent on equipment serviceability and modes selected, means must be provided whereby the pilot can readily determine the capability at alert height (eg. fail-operational status, ground roll availability).

Failure Conditions

(a) The automatic landing system must comply with the provisions of JAR-AWO.

(b) The radio altimeter, and excess-deviation alerts must comply with the provisions of JAR-AWO.

Fail-passive automatic landing system.

(a) For a fail-passive automatic landing system, failure conditions resulting in the loss of automatic landing height control capability below decision height may not be Frequent.

(b) For a fail-passive automatic landing system, any failure condition which is not Extremely Remote must be automatically detected and neutralised before it has a significant effect on the trim, flight path or attitude.

Fail-operational landing system (Automatic or Hybrid).

(a) For a fail-operational landing system, the probability of total loss of the landing system below the alert height must be Extremely Remote. Demonstration of compliance must be by means of a suitable analysis programme supported, where necessary, by a simulation and flight test programme. Special precautions must be taken to ensure that redundant sub-systems are not vulnerable to simultaneous disengagement or failure warning.

(b) A fail-operational landing system must operate as a fail-passive system following a first failure which leads to a loss of fail-operational capability.

(c) A fail-operational automatic throttle system must be provided unless the effect of loss of automatic throttle control is Minor.

Head-up display (or other form of guidance display). Where a head-up display or other form of guidance display is fitted for use in the event of automatic landing system failure, the combination of the two systems must comply with JAR-AWO. In addition, the failure modes of the display may not be such as might lead a pilot to disengage a satisfactorily functioning autopilot and obey the malfunctioning display.

Nose-wheel steering. In showing that the nose-wheel steering system complies with JAR, account must be taken of the effect of the visibility conditions on the ability of the pilot to detect steering faults and to take over control.

Automatic go-around. Total failure (shutdown) of the ILS glide path or localizer ground station may not result in loss of automatic go-around capability.

Aeroplane Flight Manual

General. The aeroplane Flight Manual must state:

(a) Limitations, including the minimum crew, alert height, the decision heights for which the aeroplane is certificated etc, (see JAR-AWO on visibility conditions).

(b) Permitted configurations (eg. flap setting, number of engines operating).

(c) Normal and abnormal procedures.

(d) Changes to the performance information, if necessary (eg, the approach speed, landing distance required, go-around climb); and

(e) Minimum required equipment including flight instrumentation.

Certification Documentation

Documentation required. Documentation providing the following information is required for certification:

(a) A specification of the aeroplane and the airborne equipment.

(b) Evidence that the equipment and its installation comply with the applicable standards.

(c) A failure analysis and an assessment of system safety.

(d) A performance analysis demonstrating compliance with the performance criteria of JAR-AWO.

(e) Flight test results including validation of any simulation.

(f) Limitations on the use of the system and crew procedures to be incorporated in the aeroplane Flight Manual.

(g) Evidence that the crew work-load complies with JAR.

(h) Inspection and maintenance procedures shown to be necessary by the system safety assessment.

Automatic Landing Sequence

79. Figure 22-9 illustrates the profile of an automatic landing sequence for a fail operational (and subsequently fail passive) system with triple redundancy. During cruise and the initial stages of the landing approach the system operates on one channel. Multichannel operation is required for automatic landing and the two disengaged channels are armed by pressing the APPR switch on the flight control panel at a specified stage of the approach. This also arms the ILS localiser and glideslope modes. The two ‘off-line’ autopilot channels are continuously supplied with all relevant outer loop control signals and they operate on a comparative monitoring basis.

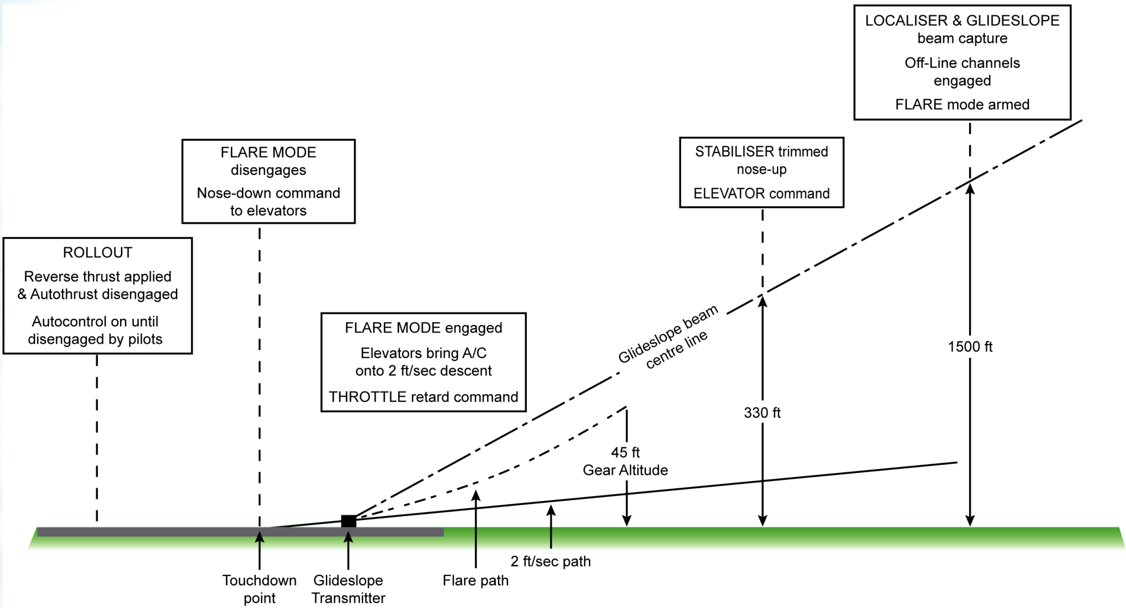
80. The radio altimeter becomes operative at, typically, 2500 ft. At 1500 ft radio altitude the localiser and glideslope beams have been captured and the armed off-line channels automatically engage to provide triple channel operation and the autoland status annunciator displays LAND 3 (LAND 2 if one channel fails to engage, for any reason). 'Flare' mode is armed and roll and pitch control is by the localiser and glideslope beam signals.

81. At 330 ft radio altitude the aircraft's horizontal stabiliser is automatically positioned to trim the aircraft into a nose-up attitude, with the elevators providing pitch control. When a ‘gear altitude’ (height of landing gear above the ground) of 45 ft is reached, flare mode is automatically engaged. Flare mode takes over pitch control from glide slope signal and brings the aircraft onto a controlled 2 ft per second descent path. Simultaneously, the auto throttle system begins retarding the throttle to reduce thrust to match the flare requirements.

82. At about 5 ft gear altitude, just prior to touchdown, flare mode disengages and touchdown and roll-out mode engages. At about 1 ft gear altitude the pitch attitude is decreased to 2°. On touchdown the elevators are automatically repositioned to bring the nose wheels into contact with the ground and hold them there during roll-out.

83. Autothrottle is disengaged when reverse thrust is selected, the automatic flight control system remains on until dis-engaged by the pilots.

FIGURE 22-9
Typical Automatic Landing Sequence - Fail Operational System



Self Assessed Exercise No. 4

QUESTIONS:

QUESTION 1.

The colour used to display cautionary information (e.g. alert messages) on an EFIS screen is.

QUESTION 2.

The EADI and EHSI EFIS displays are also known as.

QUESTION 3.

What range scales are typically available, when in Map mode on the EHSI.

QUESTION 4.

What is the purpose of the trend vector as displayed on an EHSI display.

QUESTION 5.

What guidance information is being given, with regard to the ILS, on the EHSI display shown at FIGURE 242 in the Reference Book.

QUESTION 6.

On the EHSI display shown at FIGURE 243 in the Reference Book, the symbol ‘RBT’ is a :

QUESTION 7.

What is the purpose of the altitude/range arc (VNAV performance line) as used in an EHSI display.

QUESTION 8.

On the EHSI display shown at FIGURE 244 in the Reference Book, the marked line is called:

QUESTION 9.

When using the colour weather radar test facility together with an EFIS display, what form does the test normally take.

QUESTION 10.

If the colour weather radar is operating normally (not on TEST), what information will be displayed on the EHSI screen in addition to the picture itself.

QUESTION 11.

Describe how the flight director information is displayed to the pilots.

QUESTION 12.

What is the source and nature of the information fed to the flight director computer.

QUESTION 13.

How are the armed and engaged modes annunciated in the EFIS equipped aircraft.

QUESTION 14.

Regarding the flight director, what selection and operation is actioned by the pilot for the climb mode.

QUESTION 15.

Explain the operating principle of the flight director computer.

QUESTION 16.

What is the purpose of the flight director flight mode annunciator (FMA).

QUESTION 17.

How is exceedance of the structural limits of the aircraft prevented on the flight director presentation.

QUESTION 18.

Describe the task of the gain program in the FD approach mode.

QUESTION 19.

State how the commands of the flight director are affected by the rate of change of deviation.

QUESTION 20.

List the components of a flight director system.

ANSWERS:

ANSWER 1.

Yellow

CH21 P21-21 Para57

ANSWER 2.

The EADI and EHSI displays are also known as the Primary display and Navigation display respectively.

CH21 P21-12 Para38

ANSWER 3.

Typical range scales are 10, 20, 40, 80, 160, 320 nm

CH21 P21-17 Para45

ANSWER 4.

The trend vector indicates the predicted lateral position of the aircraft at the end of 30, 60 and 90 second intervals based upon bank angle, groundspeed and lateral acceleration.

CH21 P21-22 Fig 21-15

ANSWER 5.

The indications are fly right and fly down.

CH21 P21-23/24 Fig 21-16/17

ANSWER 6.

The symbol 'RBT' is a VOR

CH21 P21-23 Fig 21-16

ANSWER 7.

It predicts where, in front of the aircraft, the reference altitude (in a climb or descent) will be reached.

CH21 P21-22 Fig 21-15

ANSWER 8.

The marked symbol is called the course line.

CH21

ANSWER 9.

The test mode checks the hardware, and then paints a predetermined pattern on the EFIS screen to assure the operator that the various colours are being properly produced by the EFIS symbol generators.

CH21 P21-25 Para61

ANSWER 10.

The radar operating mode (WX, WX + T, or MAP) is displayed in the top right corner of the EHSI, together with the tilt angle of the scanner.

CH21 P21-25 Para65

ANSWER 11.

The flight director presentation is in the form of two displays, an attitude direction indicator (ADI) and a horizontal situation indicator (HSI). The ADI presents flight guidance commands in pitch and roll and the HSI presents the navigation situation.

Page Ref 022-21-09/10

ANSWER 12.

The flight director computer is fed with flight path displacement information. The inputs to the computer are from radio facilities (VOR/ILS), barometric altitude, heading selector, compass reference and reference gyro.

Page Ref 022-22-2

ANSWER 13.

In an EFIS equipped aircraft the flight director annunciations are incorporated into the EFIS display and appear on the EADI. (See Figure 21-10 on Page 21-22).

ANSWER 14.

Select heading, or NAV (VOR) as required to fly the planned departure after take-off for lateral navigation. Select vertical speed or IAS as appropriate for vertical mode.

ANSWER 15.

The operating principle is based on an electric unit accepting signals from aircraft instrumentation, radio and navigation sources, processing and developing them into command position or attitude changes.

Page Ref 022-22-3

ANSWER 16.

The flight mode annunciator will indicate to the pilot the selected mode of the flight director eg. HDG,GA.

Page Ref 022-22-5

ANSWER 17.

The commands presented on the flight director are programmed such that the structural limits of the aircraft cannot be exceeded when obeying the pitch and roll commands.

Page Ref 022-22-6

ANSWER 18.

At pre-programmed positions on the ILS, LOC and GS the pitch and roll commands are reduced. The gain programming will reduce the deviation signals to allow for convergence of the LOC and GS beams.

Page Ref 022-22-6

ANSWER 19.

Disturbances about the longitudinal and lateral axis are picked up by sensors thus producing a signal which, after processing, is presented as a command signal in roll and pitch. The commands are related to the rate of change of deviation which is reflected in the command information presented on the instrumentation.

Page Ref 022-22-6

ANSWER 20.

The components of a flight director are:

Computer

Mode Select Panel (MSP)

NAV receiver

Attitude Direction Indicator (ADI)

Horizontal Situation Indicator (HSI)

Vertical gyro

Amplifier unit

Magnetic compass unit

Page Ref 022-22-1

Flight Envelope Protection

84. The introduction of Fly-By-Wire (FBW) control of jet transport aircraft has a number of advantages as follows:

Artificial stability
Load alleviation function
Flight envelope protection
Damping of structural loads

Improved handling
Weight saving

85. One of the advantages, Flight Envelope Protection, is highlighted in the following text:

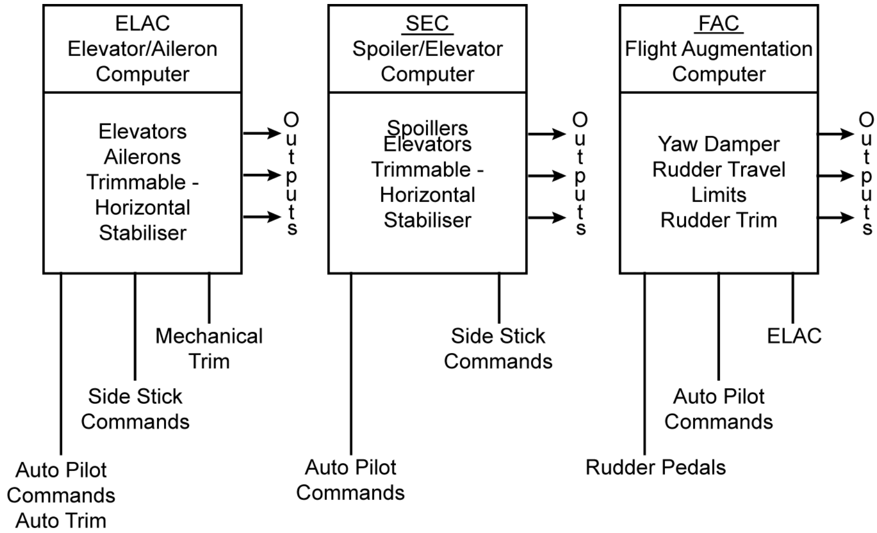
Flight Envelope Protection

Function. In most conventionally controlled aircraft it would be possible to initiate a manoeuvre which could go beyond the flight envelope limits. With FBW controlled aircraft, computers are used to interpret pilot commands and control the deflection of the requisite control surface. The computers are programmed such that the command signals generated are modified to match the flight criteria for that aircraft. In this way, the computers limit the response of the controls, thereby ensuring that the flight envelope limits of pitch, bank, yaw, speed, angle of attack and 'g' forces are not exceeded.

Input and Output Data

86. It is normal for a FBW active control system to have several different computers. Each computer will have a specific function but no single computer will be permitted to have control without its commands being monitored by at least one other computer. Eg. the Airbus A320 has seven flight control computers which process pilot and autopilot inputs according to normal, alternate or direct flight control laws. The computers are identified by their function, ie. ELAC - Elevator/Aileron Computer, SEC - Spoiler/Elevator Computer and FAC - Flight Augmentation Computer inputs and outputs are shown in Figure 22-10.

FIGURE 22-10
Input and Output Commands FBW Computers



Cockpit Controls

Each pilot has a sidestick controller with which to exercise manual control of pitch and roll. These are on their respective lateral consoles. The two sidestick controllers are not coupled mechanically, and they send separate sets of signals to the flight control computers.

Two pairs of pedals, which are rigidly interconnected, give the pilot mechanical control of the rudder.

The pilots control speed brakes with a lever on the centre pedestal.

The pilots use mechanically interconnected handwheels on each side of the centre pedestal to control the trimmable horizontal stabilizer.

The pilots use a single switch on the centre pedestal to set the rudder trim.

There is no manual switch for trimming the ailerons.

Yaw Damper

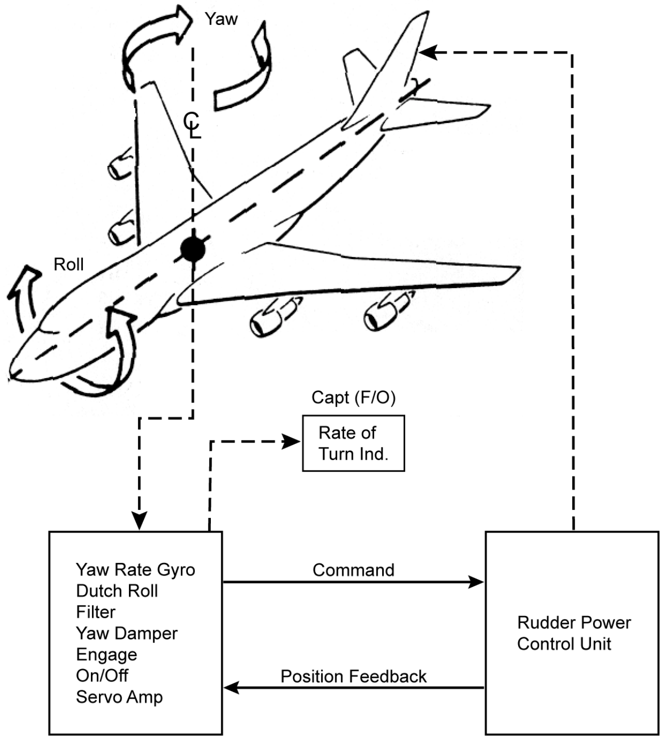
87. The installation of yaw damper systems was introduced to counter the occurrence of Dutch Roll which is a yawing-rolling motion experienced by aircraft, especially those with a swept wing configuration. Dutch Roll is explained in detail in Chapter 8, Principles of Flight.

88. The natural damping of some aircraft depends on the size of the rudder, vertical stabiliser and the speed of the aircraft, damping being more responsive at higher speeds. The requirement to control Dutch Roll would be a function of the pilot, which would be very tedious. Such an action requires displacement of the rudder to assist the vertical stabiliser in its stabilising function. Therefore, to relieve the pilot of the requirement to control Dutch Roll, yaw dampers were introduced to do it automatically.

89. The yaw damper system will function with either the automatic control system i.e. autopilot engaged or disengaged. If the automatic control system utilises a two axis auto pilot (roll and pitch) the third axis will be provided by a sub system known as a Yaw Damper system.

90. The operation of a yaw damper system is shown schematically by Figure 22-11. Central to the system operation is an 115 A/C driven yaw rate gyroscope housed in the yaw damper computer which also contains the engage switching and servo amplifier.

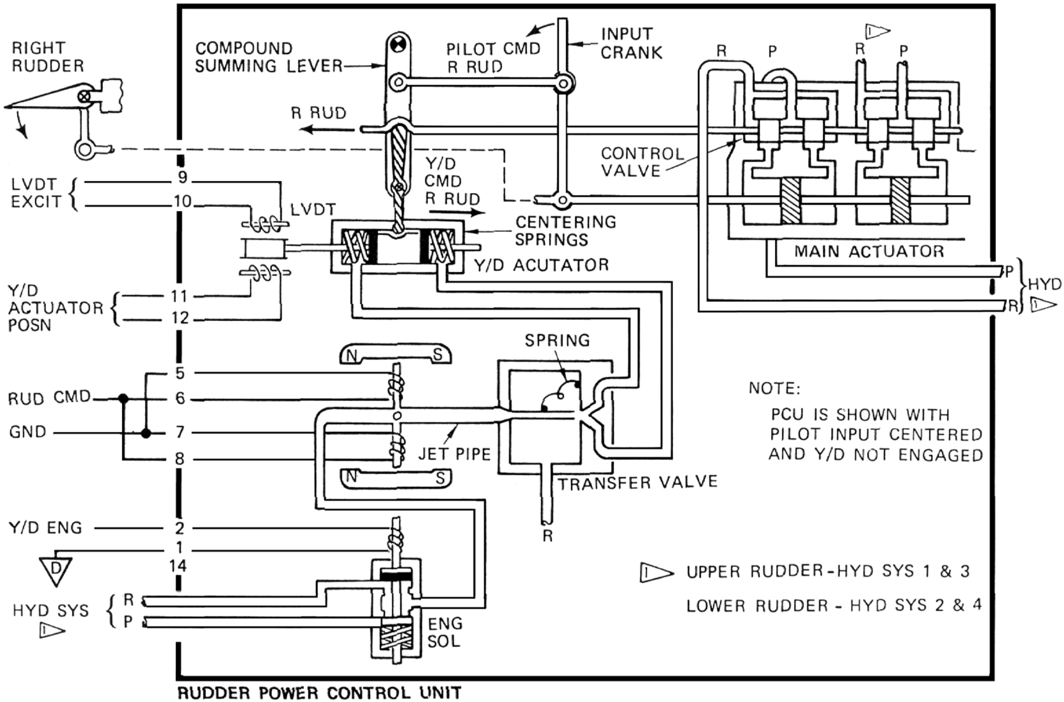
FIGURE 22-11
Yaw Damper System



91. The yaw rate gyro in each computer senses yaw rate. The signal is filtered to pass only yaw signals in the Dutch Roll frequency band. The filter discriminates between frequencies associated with flexing of the aircrafts fuselage and with normal turns, so that only those frequencies associated with Dutch Roll are allowed to pass. A yaw damper control section on the power control unit (PCU) accepts an engage and command signal from the computer. Pilot action is required to initially switch the yaw damper(s) on. Switching on will activate a solenoid valve in the PCU allowing hydraulic fluid pressure to the yaw damper control section. A transfer valve accepts command signals from the computer to port fluid to the yaw damper actuator ram to move it in the required direction. The yaw damper actuator is connected by linkage to move the control valve of the PCU ram in the required direction. See Figure 22-12.

92. Note. The PCU lineage differs from the auto pilots PCU's. The yaw damper is a series system, ie. it adds to or subtracts from the pilots rudder\pedal inputs.

FIGURE 22-12
Rudder Power Control Unit



93. There is no feedback into the rudder control system so yaw damper operation will not be seen as rudder pedal movement.

94. Indication of yaw damper operation can be seen on a Rate of Turn Indicator (ROT) (if installed) on the Attitude Direction Indicator (ADI). The ROT indicator takes a signal almost directly from the yaw rate gyro output. It is not filtered, so any time the yaw rate gyro precesses due to yaw, the ROT indicator will move away from neutral. Movement of the rudder is indicated on a Control Position Indicator (CPI).

Rudder Power Control Unit Simplified

95. Inputs to the rudder power control unit are either manual (pilot commands) or by the yaw damper actuator located in the PCU. A rudder pedal input rotates the summing lever left or right. The control valve moves left or right supplying hydraulic sources to the rudder actuator. Moving the main actuator repositions the summing lever and control valve to stop mechanical motion. When the yaw damper is engaged the solenoid valve opens supplying hydraulic pressure to the transfer valve. Yaw damper commands from the transfer valve move the yaw damper actuator. The summing lever pivots about the centre of the summing lever, moves the control valve and the rudder actuators move. The yaw damper LVDT signal is proportional to the position of the yaw damper actuator and is used in the system for signal follow-up. See Figure 22-12.

96. The transfer valve, referred to is used in the power flying control unit. The transfer valve is electrically operated by an input signal from the autopilot. This in turn will activate a spool valve and via an A/P select valve, direct hydraulic fluid pressure to the main actuator thus moving the control surface in the required direction.

Note. LVDT is a Linear Voltage Differential (Displacement) Transformer (Transmitter).

Automatic Stabiliser Trim

Introduction

97. The human pilot corrects a ‘pitch out of trim’ condition by applying elevator, then centres the elevator by trimming with the horizontal stabiliser

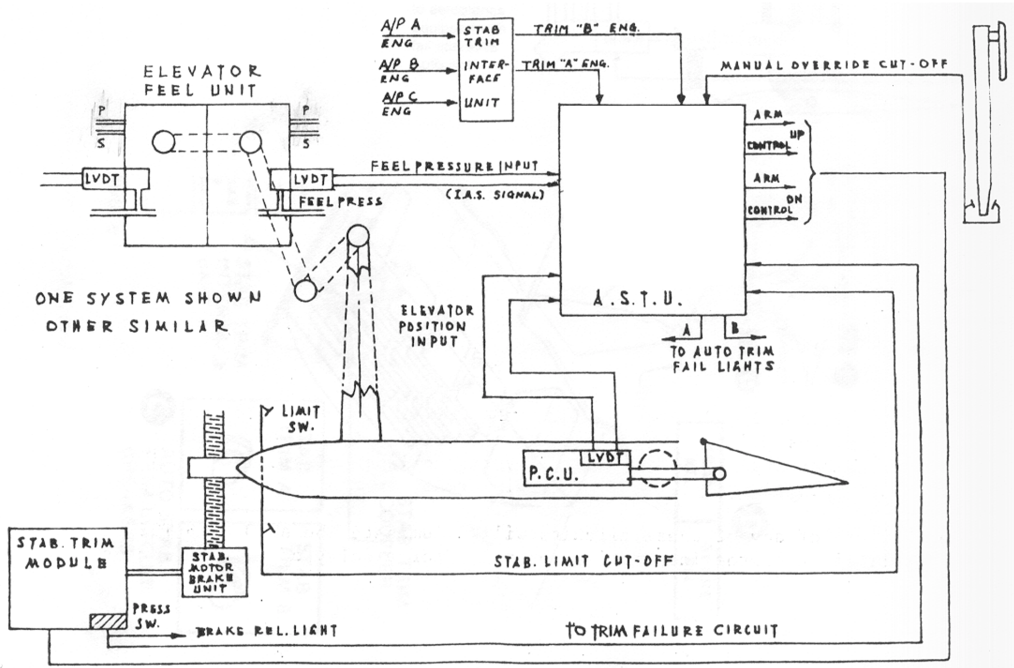
98. Correction a ‘pitch out of trim’ condition purely with elevator has two disadvantages (1) DRAG (2) Restricted elevator movement. i.e. The elevator trim increment must be taken from the elevator movement required for manoeuvre.

99. The auto pilot only controls the elevator (pitch channel), so a ‘pitch out of trim’ condition would be corrected by the auto pilot by a sustained elevator displacement.

100. The separate AUTO STAB TRIM system then runs the stabiliser to return the elevator to its natural position, i.e. aligned with the stabiliser.

FIGURE 22-13

Stabiliser Trim System



The AUTO STAB TRIM system is switched ON by engaging an auto pilot. See Figure 22-13.

An elevator position signal is taken from the ELEVATOR P.C.U. ram position transmitter.

101. Naturally the amount of elevator movement is dependent on I.A.S., and an I.A.S. signal is provided by a transmitter in the ELEVATOR FEEL UNIT sensing FELL UNIT OIL PRESSURE. The Feel unit uses PITOT and STATIC inputs to adjust the feel pressure, so the feel pressure is a function of airspeed. All rather involved, but it seems to work. Feel pressure is also modified by stabiliser feedback (mechanical linkage).

The two signals, ELEVATOR POSITION and I.A.S. are fred to the A.S.T.U. (AUTO STAB TRIM UNIT).

Trim Threshold

102. When the elevator reaches a certain angular displacement, the stabiliser is driven to neutralise the elevator position. There is a 5 second delay to prevent transient response to cransients.

LOW SPEED threshold - 2.5° ELEVATOR movement reducing progressively with increasing airspeed till:-
HIGH SPEED threshold - 0.25° ELEVATOR movement.

FIGURE 22-14

Stabiliser Trim System Control Inputs

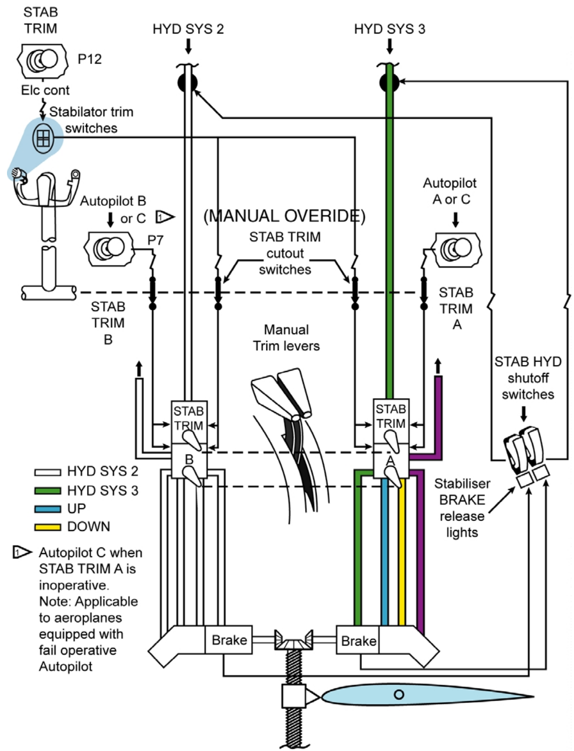
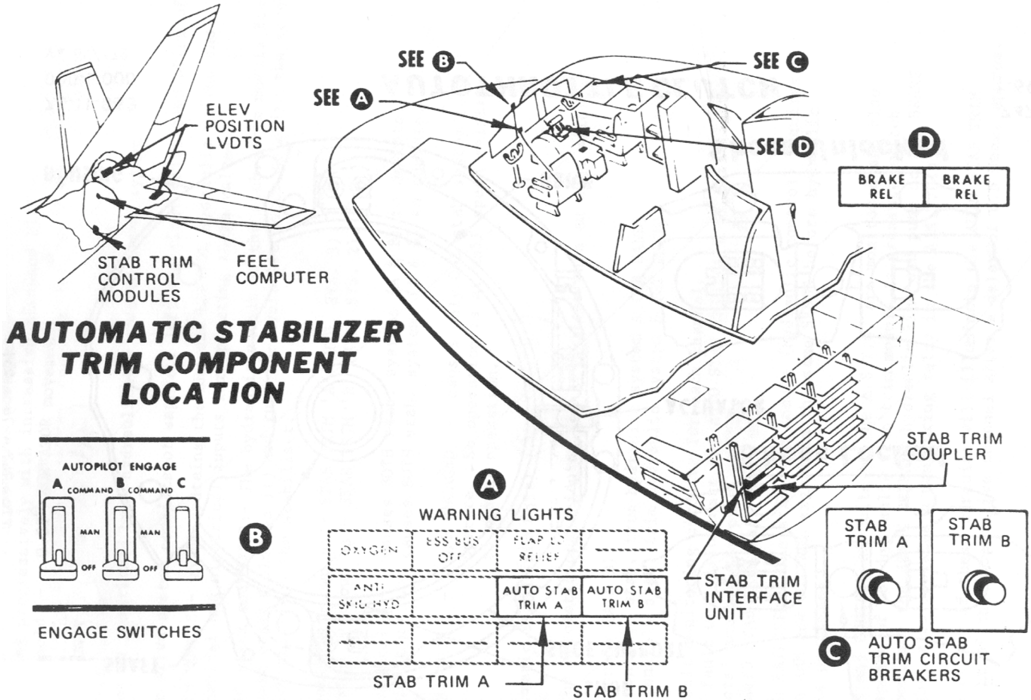


FIGURE 22-15
Component Location



Hydraulic Drive Unit

The stabiliser is positioned by a hydraulically powered drive system.

The hydraulic drive system consists of separate left and right units consisting of:

(1) STAB TRIM MODULE - which contains the ARM and CONTROL hydraulic valves and the mechanical and electrical inputs for operating these valves.

(2) HYDRAULIC BRAKE and MOTOR - The hydraulic output of the stab. trim module feeds a hydraulic brake (releases it) and runs the hydraulic motor to drive the screw jack which positions the stabiliser.

R.H. drive system is known as - STAB TRIM 'A' (HYD. SYS. 3).
L.H. drive system is known as - STAB TRIM 'B' (HYD. SYS. 2.).

MANUAL LEVER control - causes both stab. systems (A & B) to operate.
MANUAL ELECTRIC TRIM - causes both stab. systems (A & B) to operate

AUTOMATIC STAB. TRIM - A/P ENGAGED - causes only ONE stab. system to operate. So operation is at half speed relative to manual operation.

A.S.T.U. (Automatic Stab. Trim Unit)

1. Controls the automatic operation of the stabiliser when an auto pilot is engaged.
Has two channels (B & B)
Channel A controls stab. trim hydraulic system A (RT HAND)
Channel B controls stab. trim hydraulic system B (LT HAND)

2. Contains fault monitoring circuits for each channel.
A detached fault will illuminate the AUTO STAB TRIM (A & B) fail light on the pilots' annunciator panel (See Figure 22-15, Part A).
FAULTS:- AUTO STAB TRIM fail light ON.

(a) CONTROL FAILURE:- The stab trim module (hydraulic control) requires both an ARM and CONTROL signal in the same SENSE before hydraulic drive is permitted.

The ARM and CONTROL circuits of the A.S.T.U. are monitored and also the response of the hydraulic stab trim module (a pressure switch which monitors the BRAKE RELEASE hydraulic line and puts on the BRAKE REL light, also operates a monitoring relay in the A.S.T.U).

Failure of any of these circuits will illuminate the AUTO STAB TRIM fail light of the inoperative channel after a delay of 8.5 secs. approx. In this case the stab trim drive motor fails to operate because:

(i) the A.S.T.U. is faulty or

(ii) the stab trim hydraulic module is faulty.

In either case an
(i) AUTO STAB TRIM fail light is ON (Figure 22-15 (A))
(ii) BRAKE REL LIGHT is OUT (Figure 22-15 (D))

AUTO STAB TRIM fail light remains ON till A/P disconnected.

(b) OUT OF TRIM:- Stabiliser trim drive stops when the stabiliser and elevator are aligned.

If the stabiliser is not trimmed out after 12 secs. of a trim demand being made, the AUTO STAB TRIM fail light of the inoperative channel will come ON.

In this case the control system is good but the drive system is suspect, either:-

(i) Drive has failed due to a faulty drive motor or drive mechanism. There is no trimming action at all. The fault can be identified as a drive problem as the BRAKE REL light should be ON, as the system is being correctly commanded electrically and hydraulically to run.

OR

(ii) Trimming not complete. System is running, but has not completed trim operation in 12 secs. AUTO STAB TRIM fail light will go out when trimming complete. This can be due to:

(1) Slow operating drive mechanism, or
(2) NORMAL operation, when a large trim demand is made and the system has just not completed its trim action in 12 secs.

(c) POWER LOSS:- If STAB TRIM D.C. power is lost at 'any time', the AUTO STAB TRIM fail light will come ON.

STAB TRIM 'A' D.C. C/B - P.7. ESS. BUS.
STAB TRIM 'B' D.C. C/B - P.7. NO. 2. FLT. INST. BUS
See Figure 22-15, Part C.

Single Channel Auto Pilot Operation

(1) A or C auto pilots engaged - A.S.T.U. Channel A engaged and operates stab trim hydraulic system A (RT HAND).

(2) B auto pilots engaged - A.S.T.U. Channel B engaged and operates stab trim hydraulic system B (LT HAND).

See Figure 22-15, Part B.

Multi Channel Auto Pilot Operation

103. Auto changeover - after G/S CAPTURE.

- After LAND is selected on the NAV MODE SELECT switch the first A/P whose engage switch is at COMMAND will be in control of the A/C. (FIRST IN CONTROL - see A/P LAND SECTION). It will also engage its associated trim channel in the A.S.T.U.
- After G/S CAPTURE all the selected A/Ps become PITCH ENGAGED.
- If a CONTROL FAILURE occurs, after G/S CAPTURE, in the STAB TRIM CHANNEL first engaged, the AUTO STAB TRIM fail light will come ON and control will be automatically

switched to the other A.S.T.U. channel.

The fail light is locked on until the associated AP is disconnected.

'C' Auto Pilot - A.S.T.U. engagement in MULTI AUTO PILOT operation.

If 'C' A/P is engaged in a multi approach, after G/S Capture it is always put into a STANDBY condition, relinquishing any A.S.T.U. engagement function it may have had to auto pilots A or B. In the STANDBY condition it can now engage either A or B A.S.T.U. channel, if failure conditions require it.

Auto Stab Trim - Deactivation

104. The movement of the stabiliser under AUTO STAB TRIM control is stopped when:-

- AUTO PILOT DISCONNECTED - Completely disconnects the A.S.T.U. output (ARM & CONTROL) to the Stab Trim Hydraulic module.
- TURB MODE selected - Completely disconnects the A.S.T.U. output (ARM & CONTROL) to the Stab Trim Hydraulic module. The AUTO STAB TRIM fail lights are still active and will illuminate if an OUT OF TRIM CONDITION occurs during TURB MODE operation.
- MANUAL OVERRIDE SWITCHES - Physically operated by the control column. A 2° - 3° manual movement of the control column, either way, stops AUTO STAB TRIM in the opposite direction by cutting the CONTROL output from the A.S.T.U. to the Stab Trim hydraulic module. See Figure 22-15.
- STABILISER LIMIT SWITCHES - Physically operated by the stabiliser. They limit total stabiliser movement under AUTO STAB TRIM control to 2½° N.D. to 13° N.U. by cutting the ARM signal from the A.S.T.U. to the Stab Trim Hydraulic module.

Unscheduled Stab Trim Operation

105. Apply the EMERGENCY DRILL, i.e. CUT OFF the hydraulics on the Stabiliser Hydraulic Cut Off switches on the pedestal, and then sort it out from there. But ALWAYS shut off the HYDRAULICS as an automatic action. See Figure 22-15.

Trim Servo Actuator

106. Another example of an automatic pitch trim system is where a trim servo actuator is used to move a trim tab on the elevator. This is an automatic device which will trim out any elevator load when the autopilot is engaged. Assuming that the load is associated with an UP elevator displacement, then a directional signal is applied to the down trim sensor. The trim tab is displaced downward at a particular rate given by time sensors and time delay circuitry. With the autopilot disengaged, a set of relay contacts change over, thus transferring manual control to the trim tab servo actuator. The pilot will now trim out any elevator load in the normal manner.

Thrust Computation

Thrust Computation

Function

1. A fully automatic computation of required thrust, for various conditions of flight, is presented on the flight deck by an EPR (Engine Pressure Ratio) indicator. The computations are derived from measuring T.A.T (Total Air Temperature) and pressure altitude. The indications on the E.PR. computer gauge are presented for various phases of flight i.e. take off, climb, maximum continuous, cruise and go-around thrust.
EPR is the ratio between turbine exhaust pressure and low pressure (LP) compressor inlet pressure

Electronic Engine Control

2. Many modern gas turbine engines incorporate electronic controls that monitor engine performance and operate the engine controls to maintain certain parameters within pre-set operating limits. These parameters are, typically, engine spool speeds, exhaust gas temperature and engine pressure ratio (EPR). The system may act simply as a limiter, preventing pre-set parameters from being exceeded, or it may be a supervisory control system, which maintains the thrust condition set by the pilot, regardless of changing atmospheric conditions.

3. Currently the ultimate extension of this concept is full authority digital engine control (FADEC) which replaces most of the hydro-mechanical and pneumatic functions of the fuel control system and virtually takes over all steady state and transient control of the engine conditions. The fuel system retains only sufficient controls (throttle and HP shut-off cock) for safe operation in the event of a major electronic failure.

Components

E.P.R Indicator

4. This is a gauge on the flight deck which indicates T.A.T in degrees centigrade, the selected mode (i.e. take off) and the computed thrust for that mode. See Figure 23-1

Thrust Computation

FIGURE 23-1
TAT / EPRL Indicator

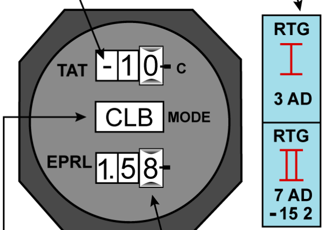
TOTAL AIR TEMPERATURE Indicator

Failure warming flag appears over indicator with electrical power loss or system failure. Any TAT failure will also cause failure flag to appear over EPR indicator

RATING SELECTOR LIGHTS (Blue)

Computer driven repeater light for rating selector on EPRL mode selector panel. Illumination of light indicates that selected alternate rating computer is operational.

If light does not illuminate when corresponding rate is selected full rating will be applied



SELECTED FLIGHT MODE

Displays mode selected on EPRL mode selector panel.

EPR LIMIT SELECTED MODE

Failure warning flag appears over indicator with electrical power loss, system failure or concurrently with failure warning flag on TAT indicator. EPR limit is automatically corrected for wing and/or nacelle anti-ice appliction.

Selector

5. The selector is used to choose a setting of required flight mode from take off, climb, max continuous cruise to go around. Some installations have an additional feature whereby a thrust rating can be inputted to utilise pre set values. The indicated thrust readings will be the maximum for conditions of TAT and flight level. A slew switch can be used to reduce these values where maximum thrust is not required.

6. If the EPR system is linked to the auto throttle system, provision is made for selection of auto throttle mode i.e. EPR, Mach Lock or Speed Lock.

7. A test facility is provided to check the serviceability of the system prior to departure.

8. System engagement, in whatever mode, is presented to the pilot by illuminated captions. See Figure 23-2.

FIGURE 23-2

TAT / EPRL Control Panel and Indicators

RATING SELECT SWITCHES

When switch is depressed, upper portion of switch illuminates green. I will provide -3A Dry performance for mode selected. II will provide -7A Dry -15% performance. Push switch second time to cancel selection.

Rating light on pilots' center panel should also illuminate.

NOTE: With CON or CRZ mode selected full rating will be displayed — Rating I or II will not be provided.

EPR DECREASE DISPLAY

Displays derate increments of manual derate switch.

MANUAL EPR DECREASE SWITCH

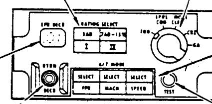
When switch is positioned to DECR the EPR Decrease Display will indicate decrease in increments of .01 EPR up to a maximum of .10 EPR. The decrease will also be indicated in the position of the "bugs" on the EPR indicator if the knob is in the automatic (in) mode. RTRN position will cancel previous decrease input by .01 increments.

Displays "88" when master dim and test switch is placed in TEST position.

EPRL MODE SELECT SWITCH

Selects phase of flight for EPR limit. Will automatically position to GA when A/P or F/D is engaged and glide slope is captured or when flaps move to 30.

NOTE: May position to GA when electrical power source is changed.



Refer to Chap. 7 for A/T annunciators.

TEST SWITCH

When the switch is depressed the following values will appear in the TAT and EPRL windows of the digital indicator as the modes are selected:

TAT — +10° (±.5)
EPRL — (all values ±.005)

	7A/SP	3A DRY	7A DRY -15%
	FULL RATING	RATING I	RATING II
TOD	1.511	1.494	1.404
CON	1.586	1.586	1.586
CLB	1.586	1.352	1.446
CRZ	1.520	1.520	1.520
GA	1.500	1.463	1.382

EPR Computer

9. The EPR computer takes inputs from TAT system and altitude sensing and processes this to transmit a required thrust parameter depending on mode selected.

Input Data Signals

10. Inputs of T.A.T altitude and mode are required. The data comes from CADC (Central Air Data Computer) and the mode comes from the pilot selected input on the E.P.R selector.

Output Signals

11. Output from the EPR computer to the indicator display EPR required plus the pilot select mode (i.e. take off) and an indication of T.A.T.

System Monitoring

12. Initially the serviceability will be checked by the crew prior to departure by using the test facility. When the test button is pressed, EPR values will appear in the indicator which are checked for accuracy against set values in the aircraft operations manual.

13. Any selected mode is indicated and must be verified by the crew member.

14. The indicated value for any selected mode can be verified by using charts in the aircraft operations manual.

15. Any failure of the outputs is usually indicated by flags which drop to obscure the indicated readings.

Auto-Thrust

FFRATS Approach Procedures

Automatic Go-Around (FFRATS)

Manual Go Around (FFRATS)

Speed Reversion (FFRATS)

Fast – Slow Indicator (FFRATS)

Signal Interfacing to Throttle Lever Mechanism

System Monitoring

Limitations, Operation Restrictions

Auto-Thrust

Function and Application

1. An auto thrust (autothrottle) system can provide all or part of the following functions:

- Automatic thrust setting for take off, climb and go-around
- Automatic speed control in cruise, approach and landing phases of flight.

2. The autothrottle system can be used independently or coupled to the auto flight system (i.e. auto coupled approach and landing)

3. Figure 24-1 indicates a typical autothrottle system illustrating components and their interaction.

4. The elements are shown schematically and are typical of a larger multi engined jet transport system.

5. In conjunction with the TAT/EPR (Total Air Temperature/Engine Pressure Ratio System) the complete auto throttle system provides three primary modes of control:

- EPR Control
- MACH hold
- Speed Control

6. In EPR control, the A/T commands the engine thrust levers so that the engine with the highest EPR reaches and maintains the EPR limit mode selected (minus any incremental decrease selected)

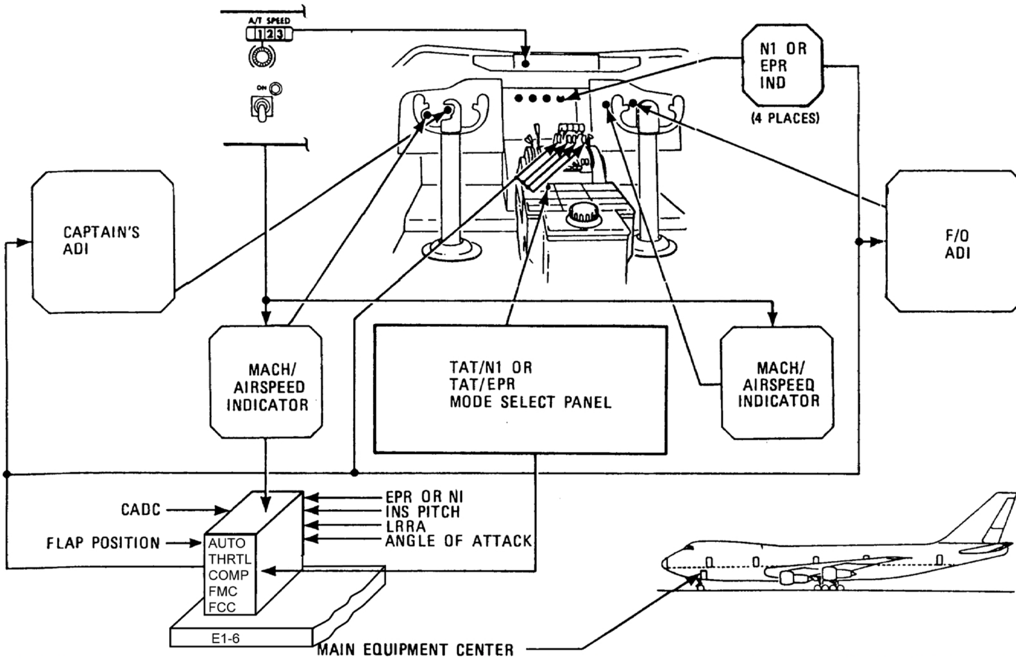


FIGURE 24-1
Typical Autothrottle System

7. The computer continuously compares all engine EPRS and selects the one with the highest value as the controlling unit. The EPR mode is used during take off, climb max continuous thrust and cruise flight. The mach hold function is used during cruise flight. The speed function is used to acquire and maintain a selected airspeed for descent holding approach and land. The speed is selected on a 'thumbwheel' selector on the mode select panel which rotates a bug on the captains and first officers air speed indicators. The mach mode is 'switched' in when the desired mach is indicated, usually in the captain's machmeter.

FIGURE 24-2

TAT/EPRL Control Panel

RATING SELECT SWITCHES
When switch is depressed, upper portion of switch illuminates green. I will provide .3A Dry performance for mode selected. II will provide .7A Dry - 15% performance. Push switch second time to cancel selection.

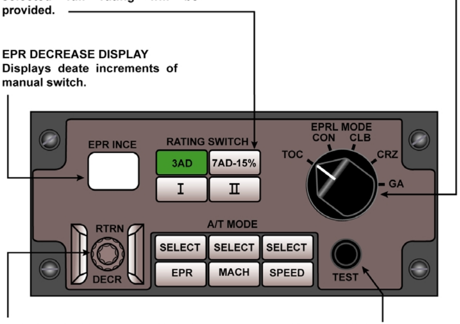
Rating light on pilots centre panel should also illuminate.

Note: With CON or CRZ mode selected full rating will be provided.

EPR DECREASE DISPLAY
Displays deate increments of manual switch.

EPRL MODE SELECT SWITCH
Select phase of flight for ERP limit. Will automatically position to GA when A/P or F/D is engaged and glide slope is caotured or when flaps move to 30.

Note: May position to GA when electrical power source is changed.



MANUAL EPR DECREASE SWITCH
When switch is positioned to DECR the EPR Decrease Displat will indicate decrease in increments of .01 EPR up to a maximun of .10 EPR. The decrease will also be indicated in the position of the "bugs" on the EPR indicator if the knob is in the automatic (in) mode. RTRN position will cancel previous decrease input by .01 increments.

Display 88 when master dim and test switch is placed in TEST position.

TEST SWITCH
When the switch is depressed the following values will appear in the TAT and EPRL windows of the digital indicator as the modes are selected

TAT _ + 10 (±.5)
EPRL _ (all values ±.005)

Mode of Operation

Auto throttle mode of operation is selected by the A/T Total Air Temperature (TAT) engine pressure ratio (EPR) selection panel Figure 24-2. (Mode Select Switch).

EPR mode is used for take off, climb and go around

MACH mode is used for cruise mach control.

SPEED Mode is used for cruise, descent approach and landing.

NOTE:

An automatic go around (GA) will automatically shift the select switch to the GA position.

8. The speed mode utilizes an airspeed error signal derived from the captains airspeed indicator. The error signal is obtained from the difference between the selected (desired) airspeed and the actual airspeed. The desired airspeed is adjusted by a thumbwheel on the autothrottle speed selector/ indicator. The EPR MODE uses the EPRL selected/indicated from the TAT/EPRL system mode, modified by any derate. The A/T computer controls the EPR command counter and index on the four engine EPR indicators.

The INS pitch attitude and an accelerometer within the A/T computer are used to detect changes in airspeed to adjust the throttles.

MACH MODE The mach mode uses the mach mode signal sensed by the central air data computer (CADC) on selection. And holds it at that value. The A/T computer also controls the fast/slow indicators on the ADI (Attitude Director Indicator).

Full Flight Regime Autothrottle System (FFRATS)

9. FFRATS offers automatic thrust control for all phases of flight. It provides minimum airspeed protection (ALPHA), flap placard protection (FLAP LIM) and engine overboost protection. Selective engine ratings are available and each rating may be further reduced with a variable derate control. The system monitors pack valves, APU bleed air valve, wing anti-ice and nacelle anti-ice switches and will make all required adjustments to the appropriate EPR limits. See Figure 24-3.

10. Airspeed Select (SPEED) and Mach Hold (MACH) are provided for low and high altitude cruise. FFRATS does not provide minimum N1 requirements for wing and/or engine/nacelle anti-ice protection.

Before Takeoff (FFRATS)

11. Select the TOD mode on the A/T EPRL Mode Selector Switch. For reduced thrust, Rating Select Switches I or II may be activated and a further reduction may be made by using the Manual EPR Derate Switch.

Takeoff (FFRATS)

12. Advance thrust levers to the vertical position, momentarily stabilizing at approximately 1.1 EPR. The initial setting value (1.1 EPR) is not as important as obtaining symmetrical thrust. Place the A/T engage switch to ON and allow the A/T to advance the thrust to the selected setting. The engine with the highest EPR is the controlling engine and all thrust levers will stop advancing when the controlling engine reaches the selected rating limit.

NOTE:

For normal operations, the thrust should not be adjusted while the A/T is advancing the thrust levers to the takeoff setting. If another thrust lever is adjusted to a higher thrust than the controlling engine, the A/T system will select that engine for control. Random changes in control can prevent the system from obtaining limit EPR by 80 KIAS (THR HLD).

---

13. Make the final trim of the thrust when THR HLD annunciation is displayed on the A/T FMA. Adjustments are made by advancing low thrust engines to the desired setting as required.

Setting Full Thrust During Reduced Thrust Takeoff

14. If full thrust is desired during a reduced thrust takeoff using Rating Select I or II or Manual derate, press either Rating Select Switch. This will cause the TAT/EPR limit mode to revert to the full thrust rating. Any variable derate will zero.

Rejected Takeoff (FFRATS)

15. Disengage the A/T while manually retarding the thrust levers to idle.

NOTE:

If a takeoff is rejected below 80 KIAS (before THR HLD), the A/T will attempt to advance the thrust levers to the limit EPR if it is not disengaged and/or physically restrained. Once the reverse thrust levers are moved to the interlock position, the A/T system will automatically disengage.

Climb (no Altitude Restriction) FFRATS

16. Initiate flap retraction on flap/speed schedule. At V2 + 40 select CLB on the A/T EPRL Mode Selector Switch. Any alternate rating or variable derate used for takeoff will not revert to the full thrust rating when the EPRL Mode Selector Switch is changed to CLB. The TAT/EPRL Indicator will show CLB and the limit EPR for the rating in use. The EPR Indicators will show the limit EPR minus the manual derate reduction. Any reduced thrust setting may be cancelled by pressing the illuminated Rating Select Switch, or by holding the Manual EPR Derate Switch to RTRN. Manual Derate automatically cancels with any change in Rating Selection.

17. If reduced climb thrust is desired, activate Rating Select Switch I or II and/or use the Manual Derate switch for the desired thrust reduction. Engage an A/P in CMD mode when desired and position the Altitude Mode Switch to ALT SEL and the A/P Speed Mode Switch to VS. Adjust the Vertical Speed Control for level flight or a slight rate of climb. Continue to retract the flaps on the flap/speed schedule. Use the Vertical Speed Control as required to prevent the airspeed from approaching the flap placard speeds.

CAUTION: IF THE AIRSPEED IS INADVERTENTLY ALLOWED TO EXCEED THE FLAP PLACARD SPEED, THE A/T WILL RAPIDLY RETARD THE THRUST LEVERS AND FLAP LIM WILL BE DISPLAYED ON THE A/T FMA.

NOTE:

Some flight crews have misinterpreted the above thrust reduction as a malfunction of the system. The FLAP LIM mode was designed as a safety backup in case the flight crew is distracted from normal duties.

18. Modify the rate of climb to stabilize the airspeed below the flap 5 placard speed until the leading edge flaps have retracted to the green indication at flaps 1, then stabilize below the flap 1 placard speed. Flap retraction speed schedule or higher airspeed must be maintained during flap retraction. At takeoff weights exceeding 800,000 lbs. (363,000 kgs.), flap limit and flap retraction speeds may differ by less than 10 knots until LED's are fully retracted. Select V2, + 100 on the A/T Speed Selector and use the A/P VS to establish the speed.

19. When on speed, select the A/P Speed Mode Switch to IAS. When able, select the desired climb speed schedule on the A/T Speed Selector and use the A/P VS to establish the speed. A small amount of asymmetric thrust during climb has little effect on trip fuel (20 pounds). Both engines on one side can be up to .05 EPR higher relative to the other side with no significant drag penalty.

20. The reduced amount of thrust caused by the EPR difference would result in increased time and distance for climb to initial cruise altitude which could affect the ability to comply with ATC departure requirements. There could be as much as 2 minutes and 15 miles increase in time and distance to cruise altitude.

Intermediate Level Off (FFRATS)

21. Set the Altitude Selector to level off altitude and the Altitude Mode Switch to ALT SEL. At Altitude Select Capture, position the A/T Speed Selector to the desired level off airspeed and activate the A/T SPEED Mode. The A/T FMA will display SPEED.

NOTE:

Any variable derate will reset to zero.

Resume Climb (FFRATS)

22. Select the desired clearance altitude on the Altitude Selector and ALT SEL on the Altitude Mode Switch. Select EPR Mode on the A/T control panel and verify EPR mode display on the A/T FMA. Use the autoflight controls as required to establish the desired climb speed schedule. Position the A/P Speed Mode Switch to Mach (if installed) at the airspeed-Mach crossover during the climb.

Cruise (FFRATS)

23. The MACH Mode should be used for all high altitude cruise. The MACH Mode is an engine referenced system while the SPEED mode is an inertial referenced system. Therefore, the MACH Mode is less sensitive to atmospheric disturbances. The MACH Mode has low gain and limited authority for initial corrections from a stabilized Mach cruise. At 35,000 feet it is limited to adding EPR to the reference EPR up to the limit of the selected rating or reducing the EPR below the reference EPR by .20 initially.

Step Climb

24. Under ideal conditions, the flight crew initiates cruise 2,000 feet above optimum altitude and maintains that flight level until the weight has been reduced to the 2,000 feet below optimum altitude weight, then, step climbs 4000 feet.

Descent (FFRATS)

25. At the top of descent set the Altitude Selector to clearance altitude and the Mode Switch to ALT SEL. Disengage the A/T. Set the A/P Speed Mode Switch to VS and set the Vertical Speed Control to approximately 3,000 fpm or as required to obtain the desired descent schedule. Reduce thrust to idle when on desired descent schedule. Select Mach (if installed) on the A/P Speed Mode Switch or maintain descent Mach with the Vertical Speed Control. Select IAS on the A/P Speed Mode Switch when intercepting the constant IAS descent schedule. Set the A/T Speed Selector to the desired level off airspeed and select SPEED on the A/T control panel. At Altitude Select Capture, engage the A/T. The A/T FMA will display SPEED.

NOTE:

The descent may be made with the A/T engaged in the SPEED Mode with the A/P Vertical Speed Control used to control the descent speed schedule.

26. Do not use the A/T for descent through icing conditions. The A/T does not provide minimum thrust protection for wing and/or engine/nacelle anti-ice operation.

NOTE:

Disengage the A/T in severe turbulence.

27. During cruise a slightly asymmetrical thrust condition may develop due to the small adjustments being made by the A/T through the cable 'dead zone'. It is not necessary for the flight crew to constantly balance the thrust if the EPR settings are within a reasonable tolerance. For example at Mach.84 both engines on one side may in some cases be as much as .05 EPR higher than the other side with no significant drag penalty.

NOTE:

When the margin between the EPR Required and the EPR Limit is less than 0.5. The allowable EPR disparity may have to be reduced in order to maintain the target Mach number.

FFRATS Approach Procedures

28. The A/T FMA will annunciate GO ARND during an A/P – F/D ILS approach at GS capture or when landing flaps are set when the A/P and F/D are not engaged. Extend flaps on flap/speed schedule.

NOTE:

Selecting the next flap setting approximately 10 knots above the minimum maneuvering speed for the present configuration will avoid nuisance A/T activity due to Alpha mode.

29. Coordinate A/T Speed Selector changes with flap extension, i.e.; call for flaps 1 and select VREF + 60 as the flaps start to move from the up position. Set the missed approach altitude on the Altitude Selector and the runway heading on the Heading Selector. The A/T Speed Selector will normally be set for the desired target airspeed. The setting for final approach and landing will be bug + 5 Kts.

NOTE:

The A/T is optimized to handle normal variations in pressure, temperature and wind encountered during the final approach and landing. During turbulent conditions, the controlled airspeed may be 8 to 10 knots above the command airspeed bug for a normally operating system. The Fast-Slow Indicator will not reflect this additional speed.

30. The A/T FMA will indicate RETARD at 30 feet radio altitude during an autoland and the thrust levers will be retarded to the idle stop.

NOTE:

For manual approaches for some airplanes, the thrust levers must be manually retarded during the flare. The F/E should follow the thrust lever movement during the flare and ensure that they reach the idle stop and remain there during the initiation of reverse thrust. The A/T will disengage during reverse thrust operation.

Automatic Go-Around (FFRATS)

31. Depress a GO-AROUND SWITCH (thrust lever 2 or 3). The A/T FMA will indicate EPR. Retract the flaps to 20 and retract the gear after a positive rate of climb. The F/E will monitor the engines and equalize go-around trust, if required. Select BUG + 40 on the A/T Speed Selector.

32. Disengage all A/P's except one and select HDG on the Navigation Mode Switch at BUG + 40. Check that the Altitude Selector is set for the desired altitude and the Altitude Mode Switch is set to ALT SEL. Retract the flaps to 5 on schedule at a safe altitude and select CLB on the A/T EPRL Mode Selector Switch. At ALT SEL Capture, activate the A/T SPEED mode.

BE CAREFUL NOT TO INADVERTENTLY ACTIVATE THE GO-AROUND SWITCHES WHILE REDUCING THRUST DURING THE LANDING IN AIRPLANES WITH FINGER ACTIVATED SWITCHES ON THE THRUST LEVERS.

Manual Go Around (FFRATS)

33. Check the A/T EPRL Mode Selector Switch is on GA, activation of the A/T will advance the thrust levers to obtain go-around thrust. Retract the flaps to 20 and retract the gear at a positive rate of climb. At a safe altitude retract the flaps on schedule and set the A/T Speed Selector to the desired speed. At desired altitude, activate the A/T SPEED Mode. The A/T FMA will annunciate SPEED.

Speed Reversion (FFRATS)

34. The A/T can be operated in SPEED REVERSION mode when the TAT/EPRL system is inoperative. FLAP LIM and ALPHA protection are available and ENGINE OVERBOOST is limited by forward thrust lever restrictions. SPEED will be annunciated on the A/T FMA. The airplane airspeed is controlled by the A/T Speed Selector. If the A/T TAT EPR limits system fails while the A/T is engaged, the A/T will disengage and the A/T light will flash red. The system may be re-engaged in the SPEED REVERSION Mode by resetting the A/T light (if required), setting the desired airspeed with the A/T SPEED SELECTOR and activating the A/T ENGAGE switch.

Fast – Slow Indicator (FFRATS)

35. The Fast-Slow Indicator on the ADI's is driven by the A/T and will be in view with the A/T engaged in SPEED or MACH Mode or when the A/T is not engaged. It is driven out of view when the A/T is engaged in EPR mode. When the A/T is not engaged, or engaged in the SPEED mode, the Fast-Slow Indicator responds to longitudinal acceleration and airspeed error. The airspeed error is the difference between the actual airspeed and one of the following:

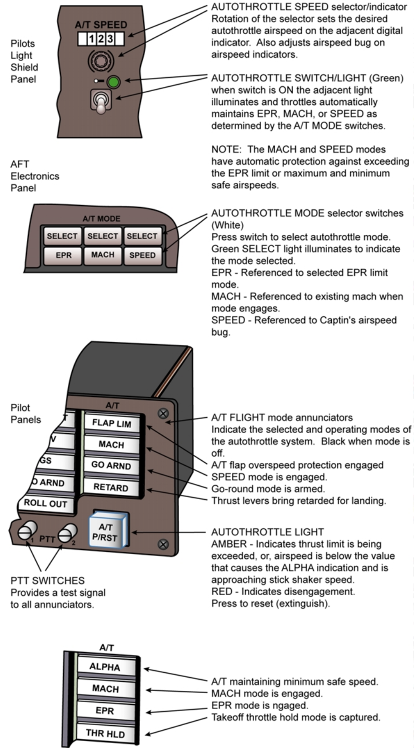
- The Command Airspeed Bug on the Captain's airspeed indicator.
- The FLAP LIM speed.

- The ALPHA REFERENCE speed.

36. When the A/T is engaged in the MACH Mode, the Fast-Slow Indicator shows Mach error. The Fast-Slow Indicator full scale denotes approximately + 10 knots in the SPEED Mode and + .025 Mach in the MACH Mode.

FIGURE 24-3

Full Flight Autothrottle System (FFRATS)



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Signal Interfacing to Throttle Lever Mechanism

37. A typical autothrottle interface system is illustrated at Figure 24-4. This is representative of a typical servo link for the control of airspeed.

38. Airspeed information is provided by the air data computer (ADC) via a control transformer link. The selected speed is compared to actual airspeed and the difference produces an error signal fed to a servo amplifier and thence to the throttle servomotor. The servomotor will be activated to cause the throttle to move by an amount proportional to the error signal amplitude. Assuming that an error signal is produced as a result of decreased airspeed, the servomotors will be activated to cause the throttles to open by an amount proportional to the error signal. As the airspeed increases the error signal is reduced until the throttles move towards the close position and balance is restored. The throttles will not return to the original position because a higher amount of thrust will be required to obviate the drag resulting from the higher airspeed. Error signals are balanced by feedback signals which are relevant to throttle servo motor position and rate of position change. This results in the total output to the servo amp being reduced to zero and throttle actuation ceases. Any change of airspeed error signal is passed through an integrator circuit.

39. Changes in airspeed due to pitch are anticipated by rate of pitch attitude change signal which is applied to the servo amp by pitch rate gyro. The throttle servomotor transmits drive to the throttles via a clutch mechanism.

System Monitoring

40. The autothrottle system performance should be closely monitored at all time during engagement. Mode selection must be cross-referenced to indications and/or warnings. Any discrepancies or warnings must be actioned in accordance with the operations manual.

41. When the autothrottle is used in the approach phase in the IAS mode, the approach gain system is in use. Approach gain of the autothrottle system is determined either by glideslope capture or by radio altitude and flap position. Approach gain will provide high gain setting for more precise control of IAS. When flap selection are made, the gain control will reduce throttle movement accordingly. If turbulence is experienced during approach, the gain will keep the system high on speed.

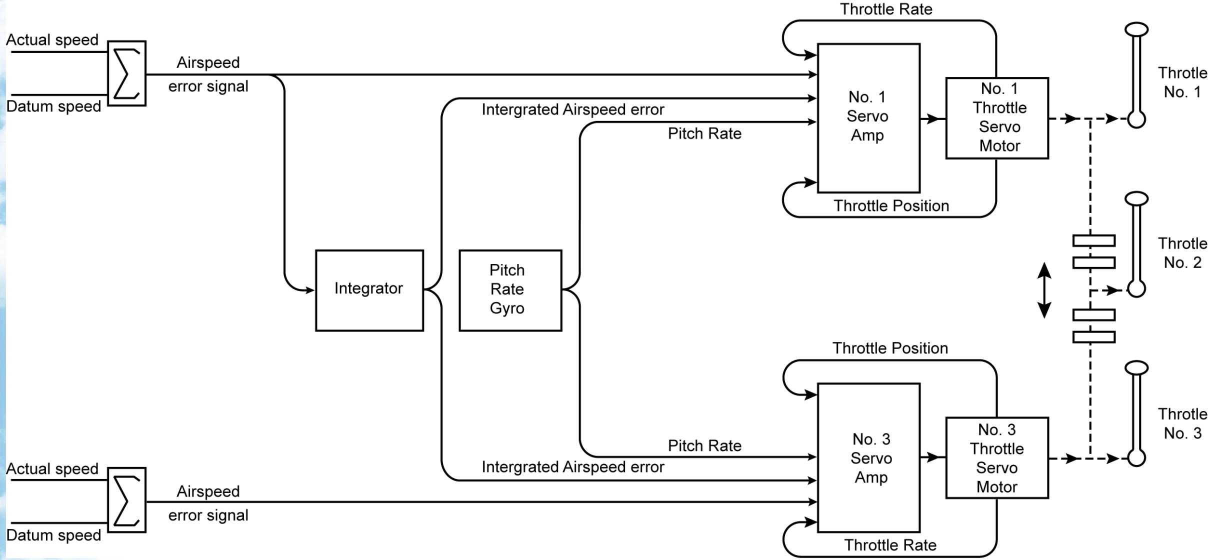
Limitations, Operation Restrictions

42. Protection is provided to avoid aircraft/engine limitation exceedance. Minimum speed protection is provided by angle of attack sensors. Engine overboost is protected by engine indicating limit sensors. Flap limit speed is protected by sensors monitoring flap position and airspeed.

43. It is generally recommended that the auto throttle be turned off when flying the aircraft manually.

44. Since the autothrottle system does not provide minimum RPM's during wing or engine nacelle anti-icing it should not be used in these circumstances.

FIGURE 24-4
Autothrottle
Interface System



Warning and Recording Equipment

Warnings General

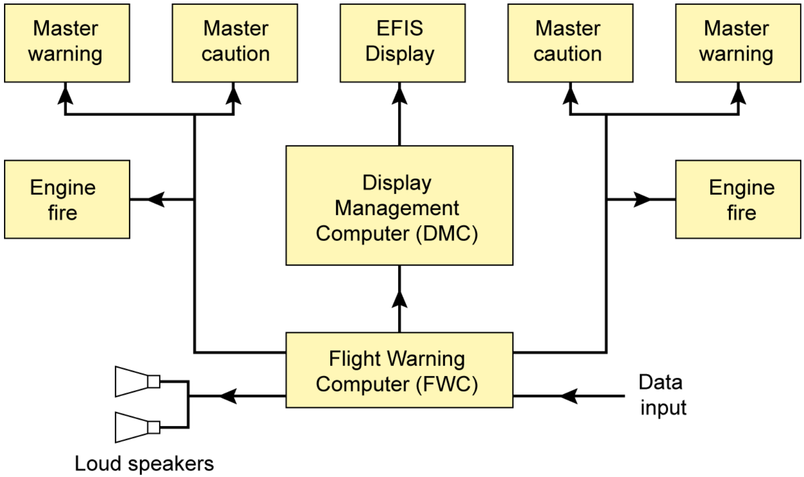
Warning and Recording Equipment

Warnings General

Flight Warning System

1. The function of the flight warning system is to identify system failures or conditions requiring action by the pilots. The type of warning depends on the degree of urgency or the type of hazard involved. Warnings may be visual, aural or tactile or in combination.

Warning and Recording Equipment



2. Warnings are classified into those requiring immediate attention (red light), those requiring urgent attention (amber light) and minor problems that are recorded but may not be displayed to the pilots.

3. The master warning light (red) is a general alert and may be accompanied by an aural warning such as a continuous repetitive chime. The EFIS system displays the dedicated alert.

4. The master caution light (amber) is a general alert and may be accompanied by an aural warning such as a single stroke chime. The EFIS system displays the dedicated alert.

5. On aircraft types without EFIS systems the general alert system of master warning, master caution and aural warnings remains the same, but the dedicated alert consists of an annunciator panel with sections for red light warnings and amber light warnings.

6. Included in the annunciator panel there may be blue, white or green lights. Blue lights indicate electrical power availability, valve position or system status. White lights indicate systems selected OFF. Green lights indicate a fully extended condition, for example leading edge devices or thrust reverser.

7. Some electro-mechanical analogue gauges may include a warning light which illuminates when the limiting parameter is exceeded, for example exhaust gas temperature. Failure of an electromechanical instrument is indicated by a flag appearing.

Altitude Alert System

JAR Requirements

8. An operator shall not operate a turbine propeller powered aeroplane with a maximum certificated take-off mass in excess of 5700 kg or having a maximum approved passenger seating configuration of more than 9 seats or a turbojet powered aeroplane unless it is equipped with an altitude alerting system capable of:

(a) Alerting the flight crew upon approaching preselected altitude in either ascent or descent; and

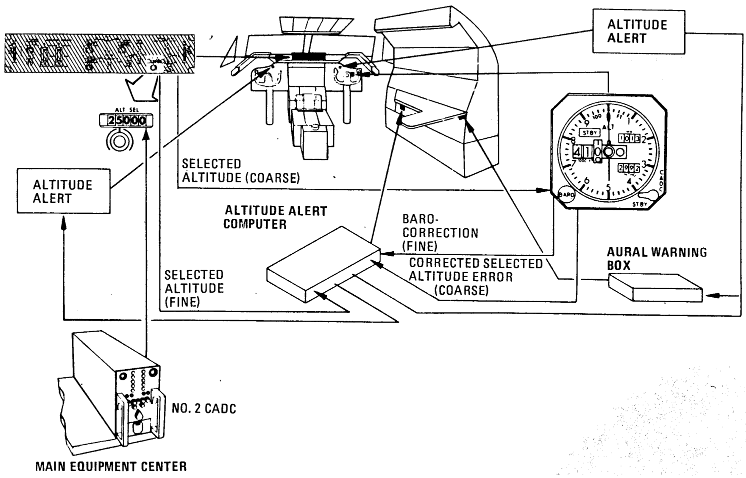
(b) Alerting the flight crew by at least an aural signal, when deviating above or below a preselected altitude, except for aeroplanes with a maximum certificated take-off mass of 5700 kg or less having a maximum approved passenger seating configuration of more than 9 and first issued with an individual certificate of airworthiness in a JAA Member State or elsewhere before 1 April 1972 and already registered in a JAA member State on 1 April 1995.

Function

9. The altitude alert system provides a visual and aural alert to the pilots when the aircraft is approaching, or deviating from a selected altitude. The altitude is manually selected using an altitude select control usually located on an auto pilot flight director mode select panel. The relevant inputs are directed through a computer in association with the central air data computer.

10. Block diagram components. See Figure 25-2.

FIGURE 25-2
Altitude Alert



Operation and System Monitoring

(See Figure 25-3).

11. The No 2 CADC (Figure 25-2) supplies actual altitude to the A/P – F/D mode select panel from where the resultant altitude error is routed through the first officer's altimeter for barometric correction then routed to the altitude alert computer.

12. The pilot selects the required altitude on the ALT SEL selector which is displayed in the ALT SEL window.

13. As the aeroplane approaches to within 900 ft of the selected altitude, the two altitude alert lights (amber) will illuminate steady and A C – chord will sound for 2 seconds. As the aircraft continues towards the selected altitude the alert lights will extinguish when within 300 ft. See Figure 25-3.

14. If the aircraft then deviates from the selected altitude, the alert lights will illuminate flashing amber and the tone will sound for 2 seconds when the aircraft is 300 ft from the selected altitude the light will extinguish when reaching 900 ft deviation or returning to the selected altitude.

15. The deviation mode is inhibited when the landing gear is down. Pilots will enter assigned altitude or flight level in the ALT SEL and will be cross-checked by another crew member. Any of the indications or alerts must be verified and where necessary actioned in the event of a deviation of altitude or flight level.

FIGURE 25-3
System Operation and Indication

ALTITUDE ALERT

PILOTS' PANELS

ALTITUDE ALERT LIGHT (Amber)

- Illuminates steady (and 2 second aural tone sounds) when approaching 900 feet above or below selected altitude; remains illuminated until 300 feet above or below altitude.
- Extinguishes when 300 feet above or below selected altitude and remains extinguished while within that 300 feet above or below range.
- Illuminates flashing (and 2 second aural tone sounds) when deviating 300 feet above or below selected altitude; light continues to flash until 900 feet above or below, at which time the light extinguishes and the system is automatically reset for subsequent altitude alerting.
- "Deviation from altitude" alerting is inoperative when landing gear is extended.

Ground Proximity Warning System (GPWS)

Ground Proximity Warning System (GPWS)

JAR Requirements

1. An operator shall not operate a turbine powered aeroplane:

   (a) Having a maximum certificated take-off mass in excess of 15 000 kg or having a maximum approved passenger seating configuration of more than 30; or,

   (b) Having a maximum certificated take-off mass in excess of 5700 kg or a maximum approved passenger seating configuration of more than 9 after 1 January 1999, unless it is equipped with a ground proximity warning system.

2. The ground proximity warning system required by JAR-OPS must automatically provide, by means of aural signals, which may be supplemented by visual signals, timely and distinctive warning to the flight crew of sink rate, ground proximity, altitude loss after take-off or go-around, incorrect landing configuration and downward glideslope deviation.

General

3. Over the last twenty years or so it is estimated that over 50% of transport aircraft losses have been caused by 'controlled' flights into the ground, for such reasons as inattention, confusion, vertigo, distraction, instrument reading error, poor visibility and navigation error. The GPWS is designed to prevent this sort of accident, by giving the flight deck crew advanced warning, both audibly and visually, of an unsafe flight condition close to the ground.

4. There are now three types of GPWS in use; basic GPWS, advanced GPWS, and enhanced GPWS.

Basic GPWS

5. The system is designed to alert the pilot in any of the following circumstances:

(i) Mode 1-If an excessive descent rate is observed.

(ii) Mode 2-If an excessive terrain closure rate is observed.

(iii) Mode 3-If a loss of altitude occurs after take-off or following a go-around.

(iv) Mode 4-If unsafe terrain clearance occurs, and the aircraft is not in the landing configuration.

(v) Mode 5-If the aircraft departs too far below the glidepath during an ILS approach.

(a) There is a subtle difference between modes 1 and 2 above. A rate of descent which is acceptable at 2000 ft might not be so acceptable at 200 ft. Additionally, a mode 2 warning may occur with the aircraft in level flight over steadily rising ground.

(b) The GPWS will not provide a warning if the aircraft is flying towards, for example, a vertical cliff. This is because terrain clearance is determined by the radio altimeter, which determines the height of the aircraft relative to the surface immediately beneath the aircraft. In other words, the cessation of a mode 2 warning means only that the aircraft has cleared the terrain beneath it. There may still be rising terrain ahead of the aircraft.

(c) The GPWS should not be considered to be a windshear detector, however the system may be activated by the effect of marked windshear where this results in the loss of climb capability (mode 3) or an increased rate of descent (modes 1, 2, 4 and 5).

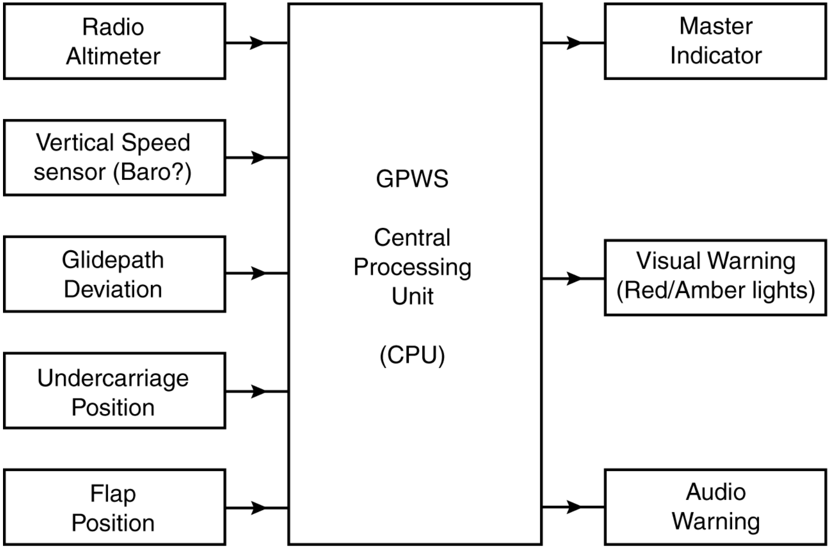
6. The GPWS receives inputs from the **radio altimeter**, the **ILS glidepath receiver, a vertical speed sensor** (which may very well be a barometric altimeter, the output of which is integrated by the GPWS to derive vertical speed), a switch which is activated by the **landing gear** and a further switch which is activated when the **flaps** are selected to the landing position.

7. Candidates who are not yet familiar with the radio altimeter should bear in mind that its function is to provide an accurate reading of the vertical distance between the aircraft and the ground immediately beneath the aircraft, hence note (b) above.

8. The GPWS is normally activated between 2450 ft and 50 ft above the surface, this height being determined by the radio altimeter. The GPWS must never be de-activated (i.e. by pulling the circuit breakers), except when using approved procedures at those airfields where GPWS inhibition is specifically required. A block schematic diagram of GPWS is shown in Figure 26-1.

Ground Proximity Warning System (GPWS)

FIGURE 26-1
GPWS Block Schematic



Aural and Visual Warnings and Alerts

9. The CAA define a warning as a command generated by the GPWS equipment, which requires an immediate response to initiate a maximum gradient climb to a safe altitude. This would normally be achieved with the aircraft 'wings level', unless the terrain avoidance situation requires that the aircraft follows a curved track.

10. The CAA define an alert as a caution generated by the GPWS equipment, which requires immediate action by the crew to correct the aircraft flight path or configuration.

11. The response to all GPWS alerts or warnings must be both positive and immediate. Establishing the cause of the GPWS activation is of secondary importance.

12. As far as basic GPWS is concerned, modes 1 to 4 will generate only warnings, whilst mode 5 will generate only an alert.

13. Warnings for modes 1 to 4 consist of both a red PULL UP light (see Figure 26-2) and an audible warning WHOOP WHOOP followed by the spoken words PULL UP.

14. An alert for mode 5 consists of both a steady amber GLIDESLOPE light (Figure 26-2) and a spoken audible GLIDESLOPE warning, which is repeated more and more frequently as deviation below the glidepath increases.

15. The mode 1 to 4 audible warning has priority over the mode 5 audible alert, however it is possible to have both the (red) warning light and the (amber) alert light illuminated at the same time.

16. It is normally impossible to mute/suppress either the audible or the visual warnings for modes 1 to 4, these will continue until the unsafe condition(s) are remedied.

17. All GPWS warnings are automatically suppressed whenever the aircraft stall warning (stick shaker) or stall identification system (stick pusher) is operating.

18. The CAA would prefer that the GPWS warn the pilot of terrain hazards at least 20 seconds before ground collision would occur if no corrective action were taken. However, as far as possible, the system should be designed so that it does not give nuisance warnings, that is to say unwanted and unnecessary warnings when the aircraft is in fact operating safely within permitted limitations and procedures. In order to eliminate the majority of these nuisance warnings it may be necessary to limit the period of time between the initiation of the terrain warning and the time that collision with the ground would occur if no corrective action were taken to 10 seconds, and to 5 seconds during the final approach phase, where pilot response might be expected to be more rapid.

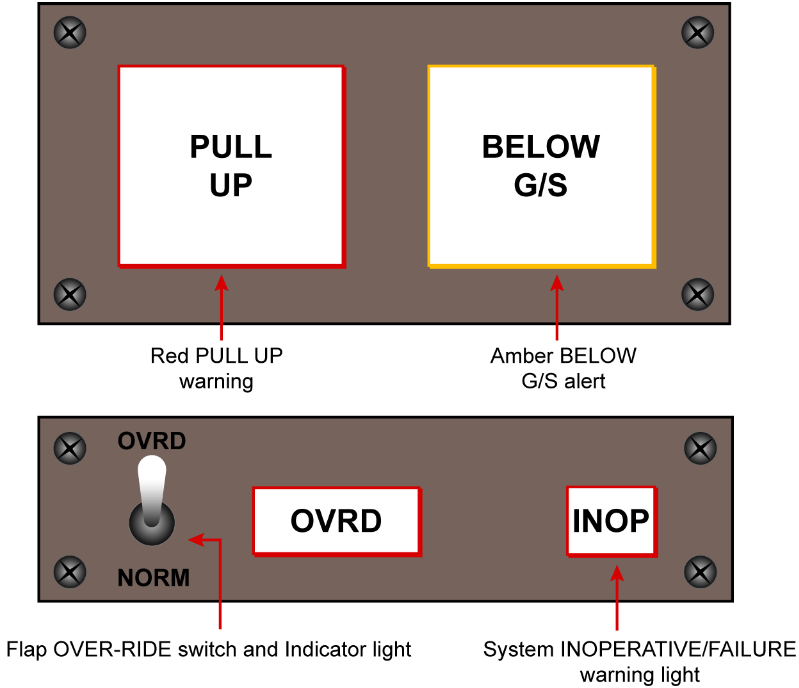
19. A genuine warning is defined as a warning provided by the equipment in accordance with its technical specifications (in other words, there is a very real risk of ground impact if no corrective action is taken). A nuisance warning has already been defined in the preceding paragraph. A false warning is one which occurs as a result of a failure or fault of or within the equipment (the GPWS itself, or any of its inputs) which causes it to operate outside of its technical specification.

Control and Display Units

20. Typical control and display units are illustrated at Figure 26-2. Appreciate that the red PULL UP (warning) and the amber BELOW GLIDESLOPE (alert) lights are mounted so as to be predominantly within the pilot's field of vision.

Ground Proximity Warning System (GPWS)

FIGURE 26-2
Typical GPWS Control and Display Unit



21. As with many modern equipments, GPWS normally includes built in test equipment (BITE). On the ground prior to engine start the check list will normally require the flight crew to run a BITE test on the GPWS. A crew initiated BITE test will normally illuminate the PULL UP, BELOW G/S and INOP lights as well as activating the audible warnings, assuming of course that the equipment has passed the test.

22. Crew initiation of the BITE system is inhibited in flight, however the system will continue to self test throughout the trip. The flight crew will have no indication that this is happening, unless the GPWS fails the test, in which case the INOP light will be illuminated. The INOP light will also illuminate if the power to the GPWS is interrupted, or if any of the inputs to the GPWS are lost.

23. Pressing the PULL UP light will test the system integrity and will illuminate the PULL UP, BELOW G/S and INOP lights as well as activating the audible warnings.

24. The INOP light will illuminate if the GPWS itself fails, if the power to the GPWS is interrupted, or if any of the inputs to the GPWS are lost.

25. If an approach is to be made with an unusual flap configuration (for example, an asymmetric approach would not normally be made with full flap), it is necessary to warn the GPWS, and this is done by the FLAP NORMAL/OVER-RIDE SWITCH. With the switch in the over-ride position, the OVRD light will be illuminated.

System Operation

26. The following notes and diagrams give brief details of the operating parameters of the equipment in each of the five modes.

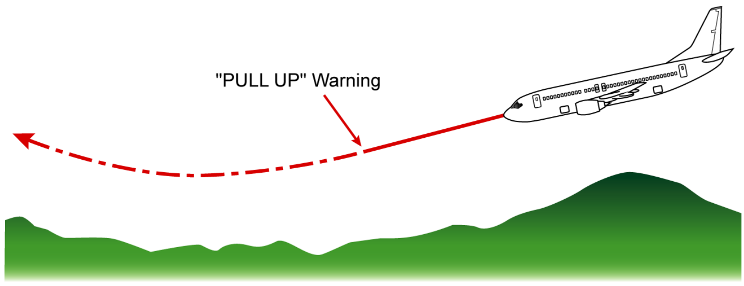
27. The operating parameters subsequently described are typical rather than rigid. The CAA permit that alternate envelopes may be approved where it can be shown that they are more suited to a particular operation.

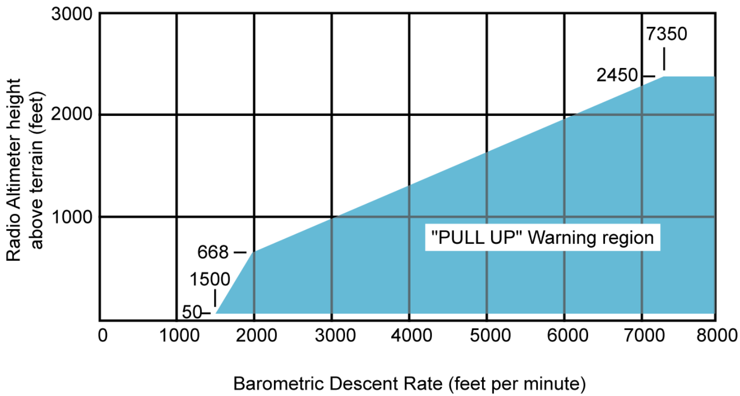
Mode One - Excessive Descent Rate

28. Mode 1 is activated when the barometric descent rate is excessive with respect to the aircraft height above the terrain, as determined by the radio altimeter. The warning envelope for mode 1 has an upper limit of 2450 ft above the ground, and at this height a warning will be given if the barometric rate of descent exceeds 7350 ft per minute. At the lower limit of the envelope, which is 50 ft above the ground, a barometric descent rate of 1500 ft per minute or more will cause mode 1 activation. The full operating parameters for mode 1 are shown at Figure 26-3.

FIGURE 26-3

Mode 1 -
Excessive Descent
Rate

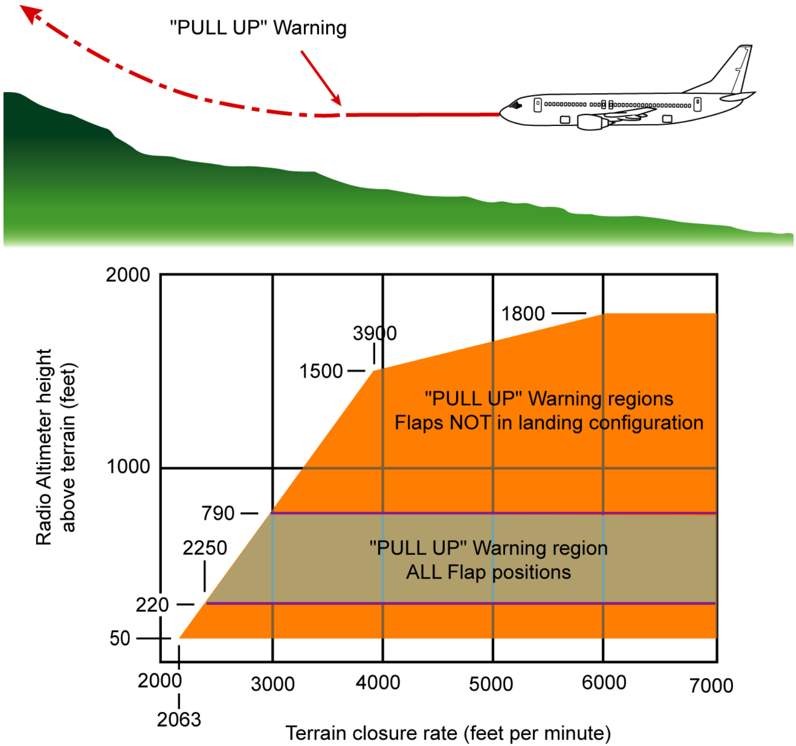




Mode Two - Excessive Terrain Closure Rate

29. Mode 2 activation occurs when the aircraft is flying into rising terrain. Activation is achieved by measuring the terrain closure rate as determined by the radio altimeter. With the aircraft at 1800 ft above the terrain, and with the flaps not in the landing configuration, mode 2A will be activated if the terrain closure rate is equal to or in excess of 6000 ft per minute. The lower limit of mode 2A operation is 50 ft above the ground, and at this height a warning will be given if the terrain closure rate exceeds 2063 ft per minute, with the flaps not in the landing configuration. With the flaps in the landing configuration, mode 2B will be activated, at the upper parameter of 790 ft agl, if the terrain closure rate is equal to or exceeds 3000 ft per minute, or at the lower parameter of 220 ft agl, if the terrain closure rate is equal to or exceeds 2250 ft per minute. The full operating parameters for mode 2 are shown at Figure 26-4.

FIGURE 26-4
Mode 2 -
Excessive Terrain
Closure Rate



Mode Three - A Loss of Altitude After Take-off or Go-Around

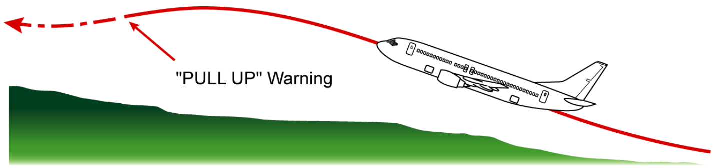
30. Mode 3 is activated if an excessive height loss is experienced after take-off or during a go-around procedure, when the aircraft is between 50 ft and 700 ft above the ground, as determined by the radio altimeter. With the aircraft at 700 ft agl, mode 3 will be activated if the accumulated barometric height loss exceeds 70 ft. With the aircraft at 50 ft agl, mode 3 will be activated if the accumulated barometric height loss exceeds 10 ft. Mode 3 is inactive when the landing gear and flaps are both in the full landing configuration. The full operating parameters for mode 3 are shown at Figure 26-5.

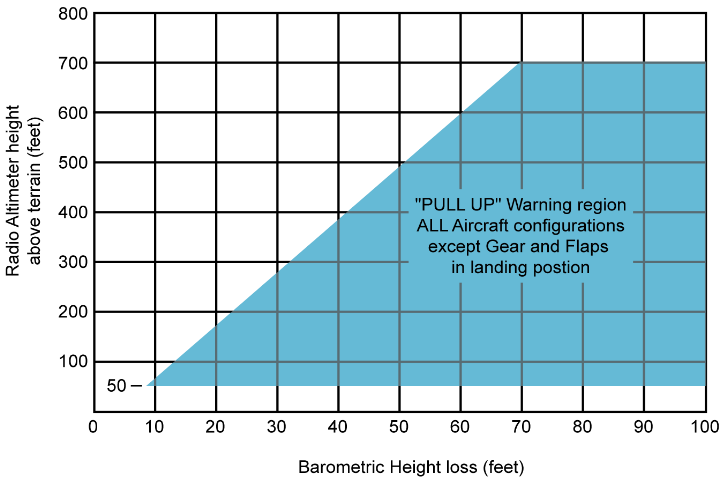
31. Mode 3 is armed automatically after take-off or go-around, when either gear or flaps are retracted.

Ground Proximity Warning System (GPWS)

FIGURE 26-5

Mode 3 - Loss of Altitude After Take-Off or Go-around



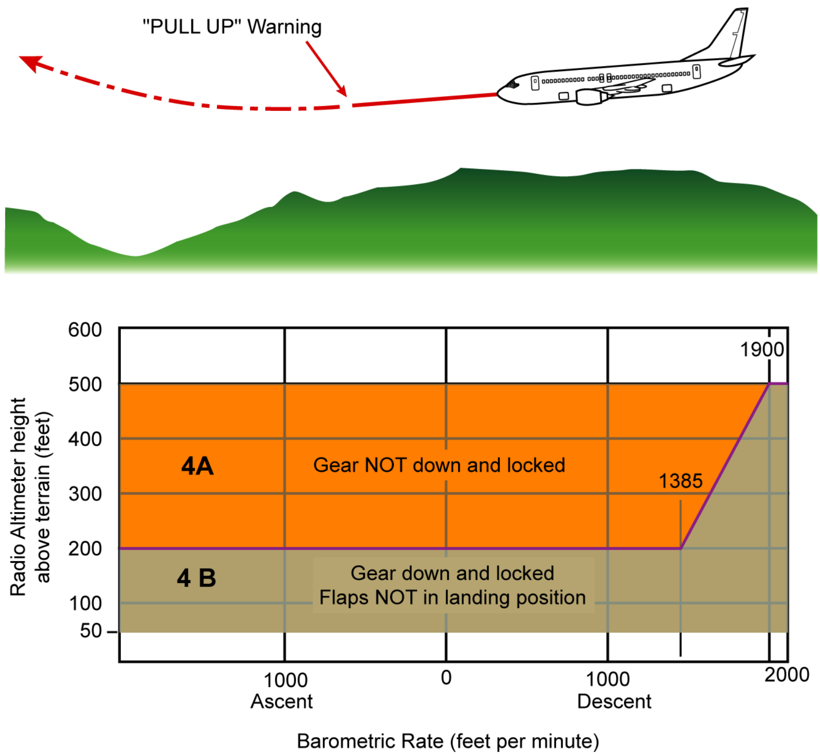


Mode Four - Unsafe Terrain Clearance

32. Mode 4 is activated when an unsafe terrain clearance situation is experienced, with the aircraft not in the landing configuration. Regardless of the barometric rate, mode 4A will be activated when the terrain clearance reduces to 500 ft agl, unless the landing gear is fully down. At 500 ft agl, as determined by the radio altimeter, and a barometric rate of descent of 1900 ft per minute or more, mode 4B will be activated unless the flaps are also in the landing position. The lower operating parameter for mode 4B is, 50 ft agl, and the full operating parameters for mode 4 are shown at Figure 26-6.

33. Mode 4 is armed automatically after take-off or go-around when climbing through a radio altimeter height of 700 ft.

FIGURE 26-6
Mode 4 - Unsafe Terrain Clearance



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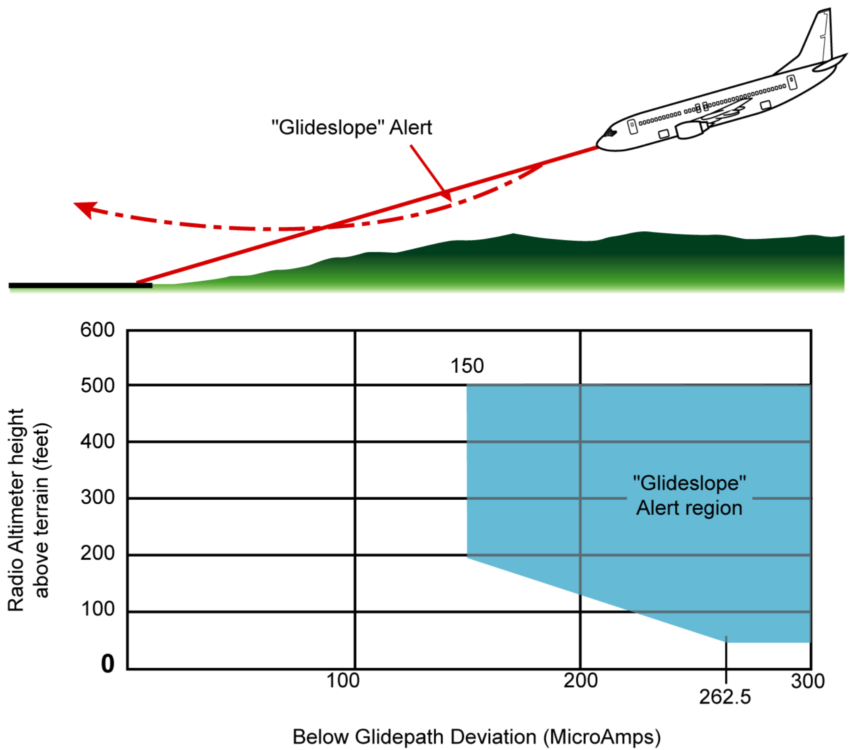
Mode Five - Aircraft Below the ILS Glidepath

34. Mode 5 is activated when the aircraft is significantly below the ILS glidepath, with the aircraft between 500 ft and 50 ft agl as determined by the radio altimeter, and with the landing gear down. The glideslope visual alert may occur at the same time as the visual and aural pull-up warnings on the occasions when the pull-up alert is due to an active mode 1, 2 or 4, but not mode 3. If mode 3 is activated, mode 5 is automatically inhibited. The full operating parameters for mode 5 are shown at Figure 26-7.

FIGURE 26-7
Mode 5 - Descent Below the ILS Glidepath



35. A mode 5 alert may be inhibited (both the visual and the audible indications) by pushing the amber light cover. The alert will then be automatically re-armed when the aircraft climbs above 1000 ft agl or descends below the lower inhibition height of, normally, 50 ft.

Advanced GPWS

36. Advanced GPWS systems were introduced in order to overcome the principle disadvantage of basic GPWS, namely that the nature of the hazard is not identified in the basic system when a warning is given.

37. With a basic system, when a warning is given, the pilot is unaware of whether the problem is one of excessive descent rate (mode 1), of excessive terrain closure rate (mode 2), of a loss of barometric altitude after take-off or following a go-around (mode 3) or of unsafe terrain clearance with the aircraft not in the landing configuration (mode 4).

38. In the advanced GPWS this problem is overcome by introducing an alert (as well as a warning) for modes 1 to 4. Under normal circumstances the alert will precede the warning, and it is the alert which will give an audible identification of the nature of the hazard. Appreciate that, with advanced GPWS, the amber alert light (which in basic GPWS was activated only in mode 5) will now illuminate for modes 1 to 5 inclusive. It is for this reason that the amber alert light is now inscribed with the words Ground Proximity (rather than the word Glideslope, as in the basic system).

39. With advanced GPWS a mode 1 alert will elicit an audible statement of SINK RATE, a mode 2 alert TERRAIN TERRAIN, a mode 3 alert DONT SINK, a mode 4A alert TOO LOW GEAR, a mode 4B alert TOO LOW FLAPS and finally a mode 5 alert (as before) GLIDESLOPE. All of these audible alerts are of course accompanied by an illuminated amber light.

40. With advanced GPWS modes 2 and 4 also operate terrain closure or proximity alerts and warnings at certain critical airspeeds. Thus a mode 4B alert and then warning could be given even with the undercarriage down if the airspeed is excessive. The upper limits of the activation envelopes may also be increased, but not above 2500 ft (radio altimeter).

41. As with basic GPWS, a mode 5 alert will not become a warning in the event that no corrective action is taken. With modes 1, 2 and 4A, if no corrective action is taken, the alert will rapidly be replaced by a warning comprising a red light and an audible WHOOP WHOOP PULL UP message. For mode 4B the audible message when the alert becomes a warning is TOO LOW TERRAIN, but see the note at the bottom of the table at Figure 26-6, which summarises the above.

42. Another difference between basic and advance GPWS is that, with the advanced system, a mode 6 alert is included. Mode 6 isn't really in the same category as modes 1 to 5, in that it is a statement that the aircraft has arrived at a certain point in the approach (the decision height on a category 2 or 3 ILS autoland approach, as determined by the radio altimeter). The action which needs to be taken by the operating pilot at this point is either to continue and land visually (if the specified visual references are present) or to execute a missed approach procedure.

43. Figure 26-8 lists the audible warnings which occur in both basic and advanced GPWS.

Enhanced Ground Proximity Warning System (EGPWS)

44. The current generation of GPWS equipments suffer from two main problems :

(a) Short warning time to potential impact with terrain

(b) Minimal advice generated as to the best method of avoiding the hazard.

The development of EGPWS is an attempt to overcome these problems.

45. EGPWS operation depends upon a worldwide terrain database which is held in the EGPWS computer memory. The system uses data provided by GPS in order to display surrounding terrain below, at or above the aircraft's altitude. In brief, the system sounds an audible warning if the aircraft's projected flight path takes it too close to terrain.

46. With a normal GPWS equipment pilots have no visual display to confirm terrain details, merely the alert and warning aural message/lights (typically 30 seconds warning) which identify that a hazard exists. The EGPWS system displays the surrounding terrain up to 320 miles away, from its database, and provides up to a 60 second warning of a potential impact.

47. The visual display can present terrain details on the aircraft's weather radar display or EFIS screen in one of three colours, depending on proximity. Green terrain is below the aircraft, yellow is above, and red is well above. Screen resolution also gets denser as the height of the terrain increases. If the system issues an alert, the terrain that poses a threat is shown as a solid block of yellow or red.

FIGURE 26-8
Basic and Advanced GPWS Audible Warnings

GPWS MODE			BASIC EQUIPMENT Alert	BASIC EQUIPMENT Warning	ADVANCED EQUIPMENT Alert	ADVANCED EQUIPMENT Warning
1	Excessive descent rate		–	"Whoop Whoop Pull Up"	"Sink Rate"	"Whoop Whoop Pull Up"
2	Excessive terrain closure rate		–	"Whoop Whoop Pull Up"	"Terrain, Terrain"	"Whoop Whoop Pull Up"
3	Altitude loss after take–off or go–around		–	"Whoop Whoop Pull Up"	"Don't Sink"	–
4	Unsafe terrain clearance while not in the landing configuration	4A Proximity to terrain – Gear not locked down	–	"Whoop Whoop Pull Up"	"Too Low Gear"	"Whoop Whoop Pull Up"
		4B Proximity to terrain – Flaps not in landing configuration	–	"Whoop Whoop Pull Up"	"Too Low Flaps"	"Too Low Terrain" (see note below)
5	Descent below ILS glideslope		"Glideslope"	–	"Glideslope"	–
6	Descent below "minimums"		–	–	"Minimums"	–

NOTE: Some GPWS equipment manufacturers refer in their product literature to "Too Low Terrain" as an alert. The view of the CAA is that the response should be as for a warning.

Discretionary Response to Warnings (UK Procedures)

48. The following is an extract from UK CAP 516 (GPWS Guidance Material).

49. All alerts and warnings must be responded to immediately. However, in the case of a warning, the response may be limited to that required for an alert, but only when the following conditions apply:

(a) The aircraft is operating by day in meteorological conditions which will enable the aircraft to remain 1 nautical mile horizontally and 1000 feet vertically away from cloud and in a flight visibility of at least 5 nautical miles, and

(b) It is immediately obvious to the commander that the aircraft is not in a dangerous situation with regard to terrain, aircraft configuration or the present manoeuvre of the aircraft.

Reporting

50. The following is a quote from a CAA Airworthiness specification concerning GPWS:

51. The effectiveness of GPWS will be greatly enhanced if there is rapid detection and correction of operational procedures or equipment faults which lead to nuisance warnings. It is therefore strongly recommended that installations should include provision for automatically logging the occurrence of a warning and, if possible, relating this directly to flight path parameters recorded on a flight data recorder, even though there is not a CAA requirement to this effect.

52. There are two levels of reporting in respect of GPWS activation.

Company Reports. The CAA require that all GPWS warnings, be they real, nuisance or false, should be reported by the pilots to the operator, thereby ensuring that appropriate analysis or remedial action can be taken. These reports form the basis upon which GPWS performance is monitored. The legend of Peter and the Wolf applies to GPWS. Repeated unwanted alerts and warnings may reduce crew confidence in the system to such an extent that response urgency becomes degraded. Consequently, all alerts and warnings should be reported so that they can be investigated and the causes of spurious activation eliminated.

Reportable Occurrences. The CAA considers that GPWS warnings should be regarded as 'Reportable Occurrences', which are sent to the CAA Safety and Data Analysis Unit (SDAU) at Gatwick (using form CA 1718), when:

(a) Aircraft separation from the ground was less than that planned or anticipated.

(b) The requisite crew response to a warning leads to a hazardous situation (such as confliction with other traffic).

(c) Repetitive false or nuisance warnings occur at a particular location or in a particular aircraft or equipment.

(d) The warning is experienced in IMC or at night and is established as having been triggered by a high rate of descent (mode 1).

(e) The occurrence involves failure to select gear or land flap by the appropriate point on the approach (mode 4).

53. Alerts are not normally regarded as reportable occurrences unless:

(i) Any difficulty or hazard arises as a result of the alert.

(ii) There is undue repetition in a particular equipment or location.

54. Further information concerning GPWS reportable occurrences will be found in the Civil Aviation Publication (CAP) 516.

Traffic Collision Avoidance Systems

Traffic Collision Avoidance Systems

1. The purpose of Traffic Collision Avoidance Systems (TCAS) is to alert pilots to the presence of SSR transponding aircraft in their vicinity and to assist in the detection and resolution of potential conflicts. The equipment is designed to operate independently of other (ground based) systems which are used by air traffic services for the prevention of collisions.

2. The equipment which is currently available for aircraft may be either TCAS I, which provides information on other traffic in the vicinity which may become a threat, or the more recently developed TCAS II, which extends this capability by presenting avoidance manoeuvre advice but only **in the vertical plane.**

3. Passenger carrying aircraft with more than thirty passenger seats intending to operate in the USA, whatever their country of registration, are now required by the FAA to carry and operate TCAS II.

4. The UK CAA position on TCAS II is, currently, to permit its use in UK airspace but not to make its use mandatory.

5. The following notes are related to TCAS II which is the current ‘state of the art’ equipment.

Definitions

6. The following definitions may be used in TCAS terminology:

Traffic Advisory (TA). Indications showing approximate positions relative to own aircraft in the horizontal plane (azimuth), or in both the horizontal and vertical planes, of transponding aircraft in the vicinity which may become a threat.

Resolution Advisory (RA). An aural and visual recommendation of manoeuvres or manoeuvre restrictions in the vertical plane to resolve conflicts with aircraft transponding SSR mode C (altitude).

Corrective Advisory. A resolution advisory that advises the pilot to deviate from the current rate of climb or descent.

Preventive Advisory. A resolution advisory that advises the pilot which rates of climb or descent need to be avoided.

Intruder. An aircraft operating SSR mode A, mode C or mode S that is predicted to enter the TCAS aircraft's collision area.

Collision Area. A volume of airspace defined by the TCAS computer that varies in size according to the closure rate.

Caution Area. A volume of airspace which begins 35 to 45 seconds before an intruder aircraft is predicted to enter the TCAS aircraft's collision area.

Warning Area. A volume of airspace which begins 20 to 30 seconds before an intruder aircraft is predicted to enter the TCAS aircraft's collision area.

Proximate Traffic. Any transponder replying within a 6 nm radius and ± 1200 ft vertically of the TCAS aircraft.

Other Traffic. Any transponder replying traffic not classified as proximate or intruder and within ± 2700 ft vertically and in display range. A typical system has three range settings, 16 nm, 8 nm and 4 nm.

Principle of Operation

7. The TCAS system continuously interrogates the SSR transponder of other aircraft in the vicinity. The bearings of the transponder replies are determined automatically using directional aerials. The range of the other aircraft is determined (as in DME) by measuring the time interval between the transmission of the interrogation and the reception of the response. The TCAS II computer is therefore able to continuously resolve the three dimensional position of other aircraft in the vicinity, **providing that they are SSR equipped.** Flight paths (based on these positions) are predicted by the TCAS II computer and, where it is predicted that they will penetrate the collision area of the TCAS aircraft, a TA or RA is produced depending on the closure rate and therefore the time to close.

8. TCAS II is designed to be able to cope with a traffic density of 24 aircraft within 5 nm range of an individual aircraft.

Equipment

9. The basic TCAS II equipment consists of:

Directional Antennae. One or two transmit/receive directional aerials are fitted almost flush with the fuselage. These are typically approximately 9 inches in diameter.

Computer Unit. The function of the TCAS II computer unit has already been discussed.

SSR Transponder. The standard SSR frequencies are employed by TCAS (transmission 1030 MHz and reception 1090 MHz), and the SSR transponder is used to produce the TCAS interrogation pulses and to receive and demodulate the response pulses.

Displays. TCAS information can be displayed on an electronic VSI or IVSI or on a dedicated TCAS display. Alternatively TCAS information can be super-imposed on the EFIS display.

Control Panel. Provides the system control function.

Symbol Codes and Colours

10. Industry standard colour coding is used to give visual cues as to the level of the threat which is posed:

11. **Red** represents an immediate threat and prompt action is required to avoid the intruder. Red is only used in conjunction with a **resolution advisory** (RA) and in this case the intruder aircraft is shown as a **solid red square.**

12. **Amber** represents a moderate threat and in this case, flight conditions permitting, a visual search is recommended in order to prepare for avoidance of the other aircraft. Amber is only used in conjunction with a **traffic advisory** (TA) and in this case the intruder aircraft is shown as a **solid amber circle.**

13. **Cyan** (light blue) represents proximate traffic and other traffic. A **solid cyan diamond** is used to denote **proximate traffic** and a **hollow diamond** to indicate **other traffic.**

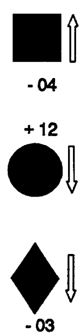
14. Any associated numerals or symbols are displayed in the same colour as the primary symbol.

15. In addition to the traffic colour codes discussed above, during a resolution advisory the rate of climb/descent to be avoided can be displayed on the electronic VSI/IVSI as a red arc with the recommended rate of climb or descent (when appropriate) shown as a green arc. Those portions of the VSI/IVSI scale which are neither red or green arced are considered to be safe rates of climb/descent.

16. Examples of TCAS symbology are shown at Figure 27-1.

FIGURE 27-1
TCAS Symbology

The arrows which are adjacent to the symbols appear only when the intruder aircraft are climbing or descending at 500 ft/min or more.



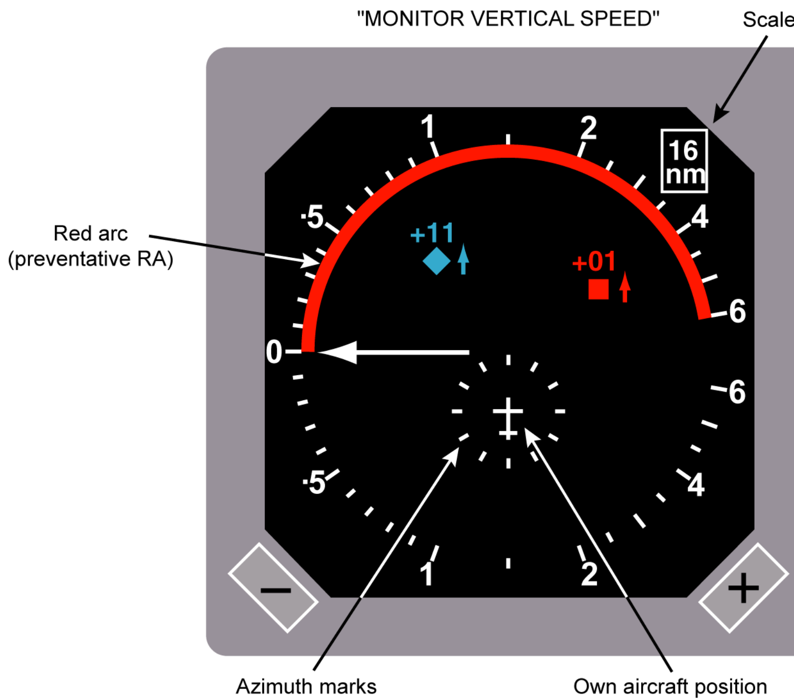
Resolution Advisory (Red).
The intruder is 400 ft below own aircraft and is climbing.

Traffic Advisory (Amber).
The intruder is 1200 ft above own aircraft and is descending.

Proximate Traffic (Cyan).
The traffic is 300 ft below own aircraft and is descending.

17. Figure 27-2 shows how these symbols might be displayed on an electronic VSI/TCAS combined display unit. TCAS information can also be superimposed on the EFIS display

FIGURE 27-2
Typical VSI/TCAS Display



Warnings

18. TCAS I was designed to inform the pilot of traffic in the vicinity so that visual acquisition and avoidance could be achieved. However, with an increased sophistication and bearing accuracy TCAS II extends the warning information (TA) into a resolution advisory (RA). The aural advice accompanying an RA may be made up of a single word or a message repeated two or three times.

19. Some examples of aural advisories are:

Preventive RA - **Monitor Vertical Speed** (meaning check the VSI/IVSI for colour coded arcs)

Corrective RA - **Climb Climb Climb** (meaning climb at the rate indicated or greater).

Descend Descend Descend (meaning descend at the rate indicated or greater).

Reduce Climb Reduce Climb (meaning reduce vertical speed to that indicated).

Reduce Descent Reduce Descent (meaning reduce vertical speed to that indicated).

Clear of Conflict (meaning separation is increasing).

20. This list is not exhaustive and other advisories may be given, particularly if the intruder aircraft changes its flight path. The required crew response to any resolution advisory is to follow promptly (within 5 seconds) and smoothly the advice given. In the case of a corrective advisory which requires an increase or decrease of vertical speed the response time must be shorter (2.5 seconds).

Coordinated Resolution Advisories

21. When the TCAS II aircraft and the intruder aircraft both have TCAS II and mode S capability the TCAS computers coordinate, via the mode S data link, the manoeuvre advice given to each of the two flight deck crews. The purpose of this is to avoid both flight deck crews being given resolution advice which would lead them into greater conflict.

22. It is for this reason that manoeuvres should **never** be made in a direction opposite to that given in an RA. This is because the sense may have been determined following an exchange of data with the established threat.

23. **RAs may be disregarded only when pilots visually identify the potentially conflicting traffic and decide that no deviation from the current flight path is needed.**

24. If pilots receive simultaneously an instruction to manoeuvre from ATC and an RA, and both conflict, **the advice given by TCAS should be followed.**

Operating Constraints

25. RAs which recommend a descent are inhibited close to the ground (700 - 1000 ft as determined by the radio altimeter) for obvious reasons. Similarly RAs which recommend an increase in descent are inhibited below a radio altitude (ground clearance) of 1450 - 1800 ft. All RAs are inhibited below a radio altitude of 500 ft and all TA aural alerts below a radio altitude of 400 ft.

26. If tracking is lost during a resolution advisory the RA is terminated without the usual Clear of Conflict aural message.

27. The priority of TCAS advisories may need to be coordinated with other alerts, such as windshear and GPWS.

28. The priorities are, typically:

(a) **windshear**

(b) **GPWS**

(c) **TCAS**

29. When TCAS is inhibited by windshear or GPWS it reverts to the traffic alert only mode and all TCAS aural alerts are inhibited.

Flight Rules and TCAS/ATC Coordination

30. The UK CAA position on TCAS is described in Aeronautical Information Circulars and the interface with ATC responsibilities for collision avoidance is clarified therein. The reference document for crew reaction to TCAS II indications is CAP 579.

31. Two extracts from current AICs dealing with TCAS are reproduced below:

RA Occurrences in UK Airspace Leading to Departure from an ATC Clearance - Legal Aspects

32. When flying in class A, B or C airspace, or in Class D or E airspace under IFR or at night, an aircraft initiating a manoeuvre in accordance with a Resolution Advisory (RA) is likely to depart from its air traffic control clearance. This would be a prima facie breach of Rule 31(3)(a) of the Rules of the Air Regulations, 1991 (as amended). However, departure from a clearance given by an ATC unit for the purpose of avoiding immediate danger is sanctioned by Article 69(3)(a) of the Air Navigation Order 1989 (as amended). In these circumstances Rule 31(4) of the Rules of the Air

Regulations also permits the commander of the aircraft to depart from an ATC clearance. Therefore, to the extent that the action in response to an RA is taken for the purpose of avoiding immediate danger and provided that the TCAS II equipment and its installation are certificated by the state of registry and that its operation by flight crew is in accordance with instructions for the use of this equipment specified in their Company's operations manual, acting in accordance with an RA will be lawful.

33. Pilots are reminded that, in UK Airspace, there are two reporting requirements imposed upon the commander of an aircraft which departs from an air traffic control clearance:

(a) Rule 31(4) of the Rules of the Air Regulations, 1991 (as amended), requires that the commander should as soon as possible inform the appropriate Air Traffic Control Unit of the deviation and give particulars of the departure in accordance with Article 69(4) of the ANO.

(b) Article 69(4) of the Air Navigation Order, 1989 (as amended), requires that the commander shall cause written particulars of the departure and of the circumstances giving rise to it, to be given within 10 days to the UK Civil Aviation Authority. For the Evaluation period only, in the event that an aircraft has departed from an air traffic control clearance in compliance with an RA, the legal obligation under Article 69(4) will be satisfied if the commander submits a TCAS II Evaluation Pilot Report Form to the UK Civil Aviation Authority.

ATC Interface

34. The operation of TCAS II equipment will affect ATC operations to some degree, irrespective of the type of airspace, ATC will expect flight crew to react to TCAS II indications in accordance with the guidance provided in CAP 579. In particular:

(a) Traffic Advisory (TA). ATC is aware that pilots are not expected to take avoiding action solely on the basis of TA information. Requests for traffic information should not be made unless the other aircraft cannot be seen and the pilots believe their aircraft is about to be endangered;

(b) Resolution Advisory (RA). ATC is aware that pilots are expected to respond immediately to an RA. If required, avoiding action should be the minimum necessary for conflict resolution. ATC should be informed as soon as possible of any deviation from an ATC clearance. Pilots should be aware that any deviation from an ATC clearance has the potential to disrupt the controller's tactical plan and may result in temporary reduction of standard separation against aircraft other than those which originally caused the RA. It is vital that flight crew maintain a good look out and return to their original flight path as soon as it is safe and practical to do so.

Future Developments

35. The third stage of airborne collision avoidance systems (ACAS) defined by ICAO requires the accurate tracking of an intruder to enable avoidance to be carried out in the horizontal plane as well as the vertical plane. This TCAS III capability has proven to be more difficult to develop because of the tracking accuracy required and all plans for TCAS III have now been discontinued.

Self Assessed Exercise No. 5

QUESTIONS:

QUESTION 1.

Describe the function of the Flight Warning System.

QUESTION 2.

The Flight Warning System comprises a Flight Warning Computer and a Display Management Computer. These provide outputs to ....

QUESTION 3.

List the three classifications of warning and the display colour associated with them, where appropriate.

QUESTION 4.

Describe how warnings and cautions are typically displayed on EFIS equipped aircraft.

QUESTION 5.

Describe the function of the Altitude Alert System.

QUESTION 6.

The Altitude Alert Computer receives inputs of .... and provides outputs to ....

QUESTION 7.

Suppose a pilot has selected a required altitude of 38,000 feet, with the aircraft at 35,000 feet and climbing. Describe the sequence of events as the aircraft approaches the selected altitude.

QUESTION 8.

Describe the sequence of events when the aircraft deviates from a selected altitude.

QUESTION 9.

GPWS uses inputs from ………………..

QUESTION 10.

A GPWS Alert is defined as ……………………

QUESTION 11.

In advanced GPWS the audible alert “Terrain Terrain” would be triggered by which mode.

QUESTION 12.

In a GWPS system, mode 5 is inhibited when ……………………

QUESTION 13.

Referring to GPWS, mode 1 gives warning of ……………………

QUESTION 14.

GPWS Mode 4 is activated ……………………...

QUESTION 15.

GPWS mode 3 will operate if altitude loss occurs before the aircraft has reached ………………..

QUESTION 16.

Providing an intruder is Mode "C" equipped the information presented on a TCAS display will comprise.

QUESTION 17.

Name the 2 types of Resolution Advisory that may be generated by TCAS.

QUESTION 18.

What is the definition of Proximate Traffic as used in TCAS.

QUESTION 19.

What coloured symbol is used on a TCAS screen to indicate a Traffic Advisory.

QUESTION 20.

What is the advantage that mode S capability gives to TCAS II equipped aircraft.

ANSWERS:

ANSWER 1.

The function of the Flight Warning System is to identify system failures or conditions requiring action by the pilots.

022-25-1

ANSWER 2.

The display Management Computer outputs to the EFIS Display. The Flight Warning Computer provides outputs to the Captain's and First Officer's Master Warning, Master Caution and Engine Fire warning displays and to loud speakers.

022-25-1

ANSWER 3.

Warnings are classified into those requiring immediate attention (red light), those requiring urgent attention (amber light) and minor problems that are recorded but may not be displayed to the pilots.

022-25-1

ANSWER 4.

The master warning light (red) is a general alert and may be accompanied by an aural warning such as a continuous repetitive chime. The master caution light (amber) is a general alert and may be accompanied by an aural warning such as a single stroke chime. In both cases the EFIS system displays the dedicated alert.

022-25-1

ANSWER 5.

The altitude alert system provides a visual and aural alert to the pilots when the aircraft is approaching, or deviating from a selected altitude. The altitude is manually selected using an altitude select control usually located on an autopilot flight director mode select panel. The relevant inputs are directed through a computer in association with the central air data computer.

022 25-2

ANSWER 6.

The Altitude Alert Computer receives inputs of Selected Altitude, Barometric Altitude and Selected Altitude Error. It provides outputs to the Captain's and First Officer's visual alerts and to the aural warning box.

022-25-3

ANSWER 7.

As the aeroplane approaches to within 900 feet of the selected altitude, the two alert lights will illuminate steady amber and a C chime will sound for two seconds. As the aircraft continues towards the selected altitude the alert lights will extinguish when within 300 feet.

022-25-3

ANSWER 8.

If the aeroplane deviates from the selected altitude, the alert lights will illuminate flashing amber and the C chime will sound for 2 seconds when the aircraft is 300 feet from the selected altitude. The lights will extinguish when the aircraft reaches 900 feet deviation from the selected altitude, or returns to the selected altitude.

022-25-3

ANSWER 9.

Rad alt, glideslope receiver, static pressure monitor, landing gear position monitor and flap position monitor.

CH26 P26-3 Fig 26-1

ANSWER 10.

A GPWS Alert is a caution requiring an immediate response to correct the flight path and/or configuration of the aircraft.

CH26 P26-3 Para 10

ANSWER 11.

Mode 2

CH26 P26-13 Fig 26-8

ANSWER 12.

Mode 5 is inhibited whenever Mode 3 is active.

CH26 P26-10 Para 34

ANSWER 13.

Excessive barometric descent rate

CH26 P26-6 Para28

ANSWER 14.

Mode 4 is activated when descending below a height of 200 ft measured by the radio altimeter, with the landing gear down and the flaps not in the landing configuration.

CH26 P26-9 Fig 26-6

ANSWER 15.

700 ft terrain clearance.

CH26 P26-8 Fig 26-5

ANSWER 16.

Range, bearing and relative height.

CH27 P27-2 paragraph 7

ANSWER 17.

Corrective Advisory and Preventive Advisory.

CH27 P27-1 paragraph 6

ANSWER 18.

Proximate Traffic is any transponding aircraft replying with a 6nm radius and ±1200 ft vertically of the TCAS aircraft.

CH27 P27-2 paragraph 6

ANSWER 19.

A solid amber circle.

CH27 P27-3 paragraph 12

ANSWER 20.

If a TCAS II aircraft and an intruder both have TCAS II and mode S capability, the TCAS computers co-ordinate, via the mode S data link, the manoeuvre advice given to each of the two flight deck crews (thus preventing them from getting into a position of greater conflict).

CH27 P27-5 paragraph 21

Overspeed Warning

Angle of Attack Sources

Stall Warning System

Overspeed Warning

1. The overspeed warning system provides a distinct aural warning any time the maximum operating speed of V_{MO}/M_{MO} is exceeded. The warning can be silenced only by reducing speed below V_{MO}/M_{MO}. The aural warning may be a clacker sound.

2. On an electronic attitude direction indicator (EADI) the airspeed indicator tape displays red warning bands at maximum and minimum airspeeds. Amber bands indicate maximum and minimum manoeuvring speeds under current aircraft configurations of slat and flap deployment.

On an electro-mechanical airspeed indicator a ‘barber’s pole’ displays maximum speed. ‘Bugs’ may be set on the bezel to display manoeuvring speeds.

3. Input data for the mach/overspeed warning system is obtained from the central air data computer (CADC), the flight management guidance computer (FMGC) for the aircraft weight and centre of gravity, and the position of the slats and flaps.

4. A test of the V_{MO}/M_{MO} overspeed warning system will cause the clacker to sound to give indication of a serviceable system.

5. In the event of failure of the overspeed warning system the clacker will not sound if the aircraft exceeds V_{MO}/M_{MO}. The airspeed indicator will however display the overspeed condition.

Angle of Attack Sources

6. Angle of attack is defined as the angle between the relative airflow and the chord line of the wing.

7. Some sort of direct angle of attack indication would be of great value to the pilot for safe and efficient control of the aircraft. This applies particularly at low airspeeds where the rate of change of angle of attack with changing airspeed is greatest and where the aircraft may be approaching a stall condition.

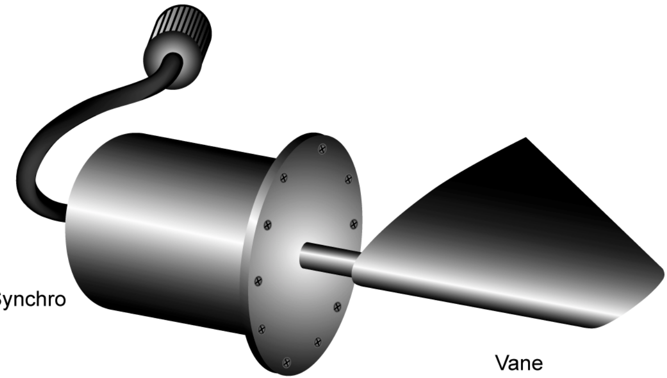
8. Practical angle of attack sensor/indicator systems normally consist of some form of measuring vane, a means of transmission of the vane angle and an indication/display system.

Measuring Vanes

9. The measuring vane is normally situated on the fuselage near the nose of the aircraft, or alternatively on a wing leading edge.

10. A fuselage mounted vane consists of a small balanced vane which is pivoted at the point of static balance. The vane is therefore free to align itself with the relative airflow and the position of the vane and any movement of the vane is converted to electrical signals by means of a synchro and fed either to a flight deck indicator or the flight management system (FMS). In order to prevent the vane from icing it normally incorporates an electrical heating element and additionally some form of damping device in order to prevent random and/or erratic outputs in turbulent air. A typical fuselage-mounted angle of attack vane is shown at Figure 28-1.

FIGURE 28-1
Angle of Attack Vane



11. A wing mounted vane is pivotally mounted on the leading edge of the wing, is spring centred and can move either upwards or downwards depending on the airflow. Again the output is fed either to a flight deck indicator or the FMS.

Indicated Angle of Attack

12. Because of the complex airflow over an aircraft the angle measured by either type of vane will not normally be the true angle of attack, but rather an indicated angle of attack. Where the angle of attack is used either by the pilot or by the FMS simply as a reference in controlling the aircraft it makes no difference whether it is indicated or corrected (true) values that are used. Provided that a datum is established, the changing values of either the indicated or the true values of angle of attack about that datum will be the same. However, where angle of attack is used to determine or control the actual flight path of the aircraft, the corrected (true) value must be used. At speeds in excess of about Mach 0.5 the relationship between true and indicated angle of attack is a function of the Mach number.

Angle of Attack Indicators

13. A separate indicator on the pilot's display panel may be used to present angle of attack, to enable the pilot to adjust speed and attitude for optimum aircraft performance. In previous generations of aircraft which employed an angle of attack indicator, the device was frequently mounted on the top of the instrument panel as an early version of a head up display. By aligning two moving datums, the angle of attack appropriate to the airspeed and configuration could be maintained during those portions of the flight which involved primarily 'visual flying'.

14. Modern displays take various forms, ranging from the single pointer-and-gauge type illustrated at Figure 28-2 to the type found in glass cockpits, where the angle of attack may be electronically displayed against a vertical scale. The display may be coloured green, yellow, and red, with the stick-shaker normally operating at the line between the green and yellow segments.

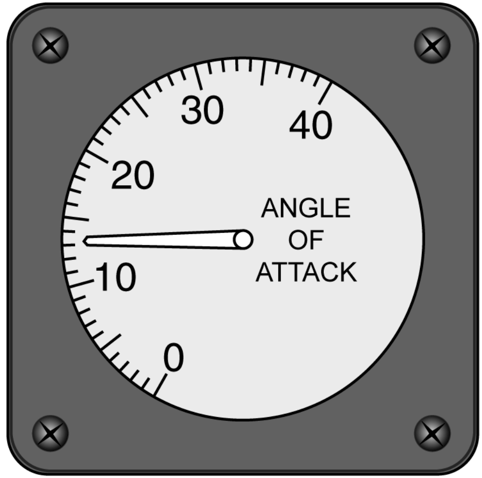


FIGURE 28-2
Angle of Attack Indicator

Other Uses of Angle of Attack Data

15. The true angle of attack may be used by the flight control and management system as one of the inputs required to compute optimum performance.

16. The central air data computer (CADC) can use the true angle of attack (together with an input of slat/flap/landing gear configuration) in order to correct for position or pressure errors at the static vents.

17. The true angle of attack can be used too by the autothrottle computer in order to provide the power settings required in order to maintain a defined climb or descent profile.

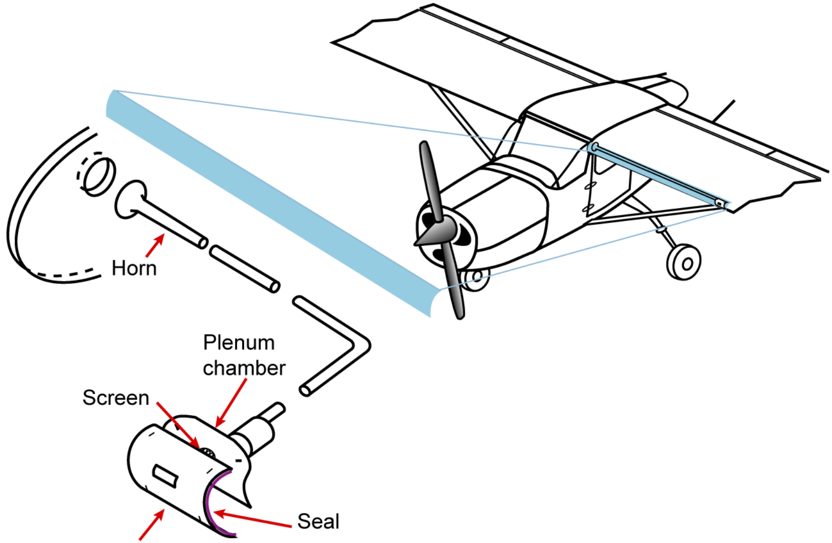
Stall Warning System

18. Many aircraft do not experience aerodynamic buffet when approaching a stall condition and require a stall warning system to warn the pilot of an impending stall. A stall warning strip may be attached to the wing leading edge to induce aerodynamic buffet to warn the pilot of high angle of attack, before the wing stalling angle is reached. A simple stall warning system may consist of a horn, more sophisticated systems may include a stick shaker and stick push. The regulatory margin between stall warning and stall is for the warning to occur 5% or 5Kt CAS avove the stall speed, whichever is greater.

Simple Pneumatic Stall Warning System

19. A plenum chamber in the leading edge is connected to a reed-operated horn in the cockpit. The slot in the adjustable plate aligns with the stagnation point in normal flight. At high angles of attack the slot enters a low pressure area sufficient to draw air through the reed/horn assembly and emit a noise to alert the pilot to an impending stall.

FIGURE 28-3
Pneumatic Stall Warning System

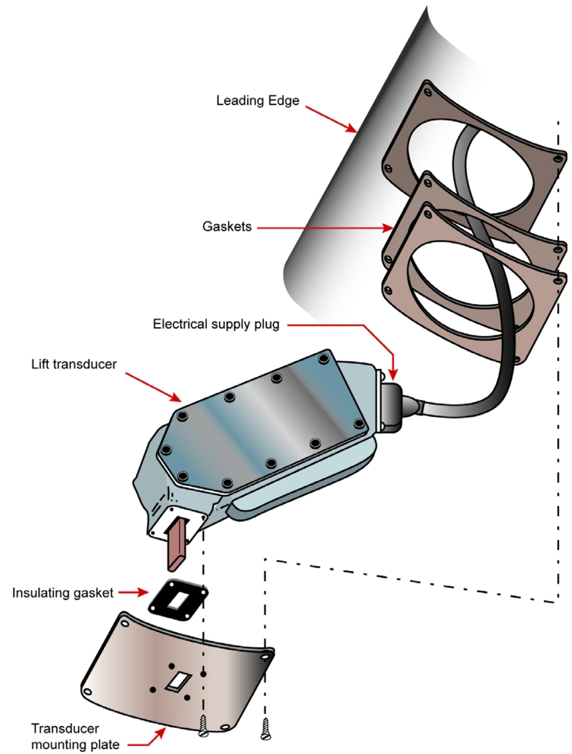


Simple Electric Stall Warning System

20. In this system, a lightly spring loaded vane is positioned in the leading edge just below the stagnation point. In normal flight airflow maintains the vane against its rear stop. As the aircraft angle of attack increases, the stagnation point moves towards the lower surface of the leading edge and the localised flow moves the vane forward to operate the lift transducer microswitch, completing the circuit to illuminate a warning light and sound a warning horn.

FIGURE 28-4

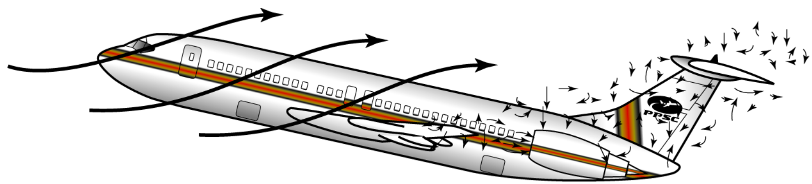
Stall Warning Vane



Stall Protection System

21. The full stall protection system is designed for aircraft with rear fuselage mounted engines and a T-tail stabiliser configuration. Flight testing of aircraft of this configuration demonstrated that at high angles of attack the increasingly turbulent boundary layer wake from the wing could cause engine flame-out and loss of elevator effectiveness. In addition, the high induced drag causes the airspeed to decay and the aircraft starts to descend, maintaining the same fuselage angle to the horizontal. This increases the angle of attack of the relative airflow onto the wing which further exacerbates the problem. Eventually, the aircraft may have a very low forward speed and a very high vertical rate of descent. The aircraft is now in a ‘deep stall’ or ‘super stall’ condition and recovery may not be possible.

FIGURE 28-5
Aircraft Approaching Deep Stall



22. The stall protection system works in a series of stages to prevent the aircraft entering a deep stall.

These stages are:

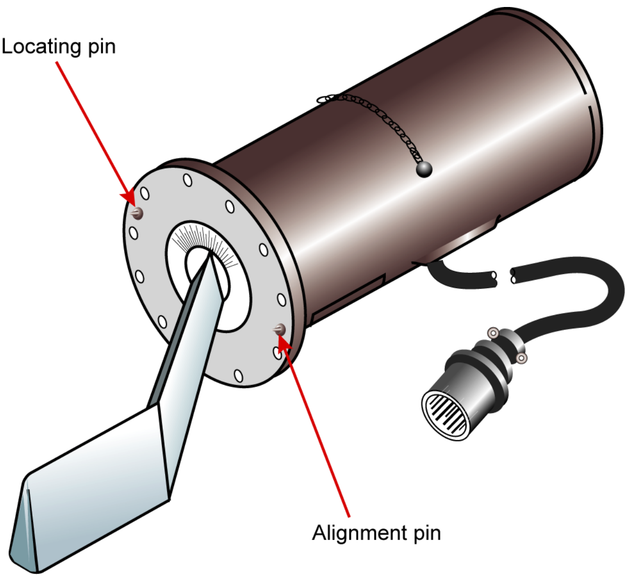
(i) 1st Stage - Auto ignition

(ii) 2nd Stage - Stall warning (stick shake)

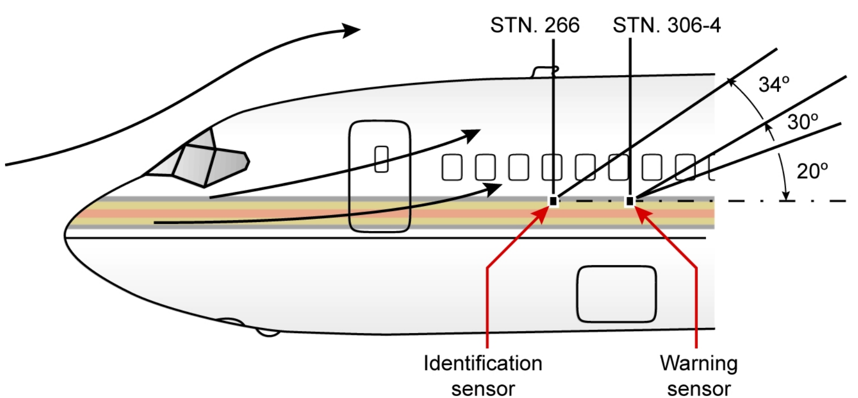
(iii) 3rd Stage - Stall identification (stick nudge or push)

The three stages are triggered by angle of attack sensors on each side of the fuselage.

FIGURE 28-6
Angle of Attack Sensor



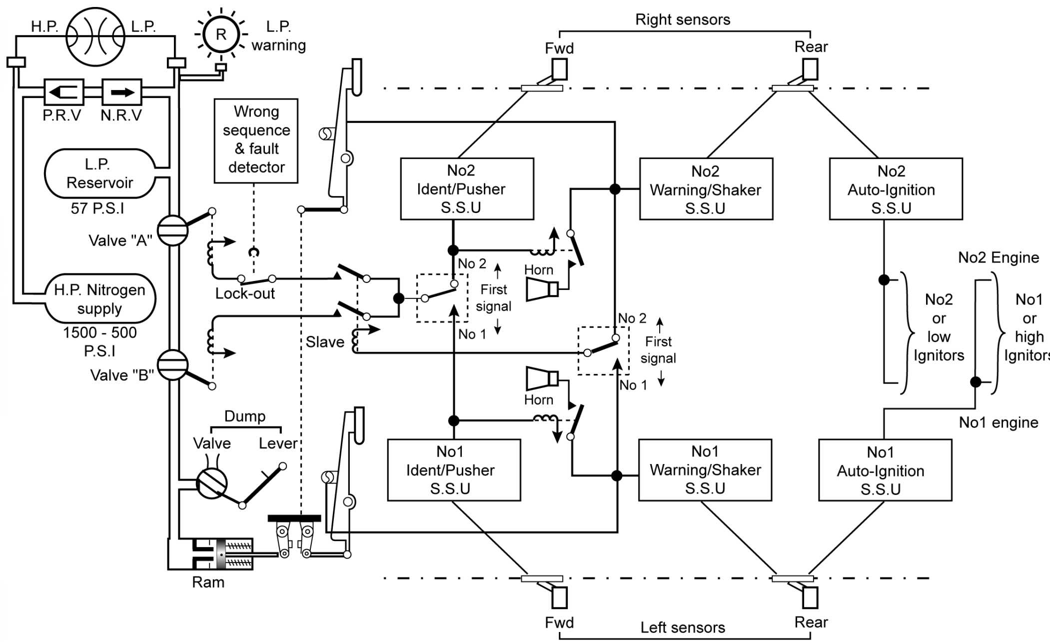
Overspeed Warning



23. The warning sensor signals auto-ignition and stick shake. The identification sensor sounds the horn and activates the stick push system.

Stall Protection System Schematic

FIGURE 28-8
Stall Protection - Simplified Schematic



Overspeed Warning

Stage 1. At 20° alpha angle automatic ignition occurs to prevent engine flame-out. Ignitor ON lights in the cockpit illuminate.

Stage 2. At 30° alpha angle stick shaker motors vibrate the control columns to simulate buffet and the stall identification system is armed.

Stage 3. At 34° alpha angle the stall identification system operates. Horns sound, amber valve A and B lights illuminate, nitrogen is introduced into the stick push pneumatic ram and red lights advise the pilots that the system has been activated.

24. The effects of the three stages are additive. The stall protection system cancels automatically when the alpha angle has been reduced.

A manual dump control is provided to permit the pilot to depressurise the ram in the event of system malfunction and inadvertent operation.

Flight Data Recorders (FDR)

Flight Data Recorders (FDR)

JAR Requirements

JAR-OPS 1.715 Flight Data Recorders-1

1. An operator shall not operate any aeroplane first issued with an individual Certificate of Airworthiness either in a JAA Member State or elsewhere on or after 1 April 1998 which:

2. Is multi-engine turbine powered and has a maximum approved passenger seating configuration of more than nine (9); or

3. Has a maximum certificated take-off mass over 5700 kg, unless it is equipped with a flight data recorder that uses a digital method or recording and storing data and a method or readily retrieving that data from the storage medium is available.

4. The flight data recorder shall be capable of retaining the data recorded during at least the last 25 hours of its operation except that, for those aeroplanes with a maximum certificated take-off mass of 5700 kg or less, this period may be reduced to 10 hours.

The Flight Data Recorder Must, With Reference to a Timescale, Record (See AMC OPS 1.715 (c):

5. The parameters necessary to determine altitude airspeed, heading, acceleration, pitch and roll attitude, radio transmission keying, thrust or power on each engine, configuration of lift and drag devices, air temperature, use of automatic flight control systems and angle of attack;

6. For those aeroplanes with a maximum certificated take-off mass over 27 000 kg, the additional parameters necessary to determine positions or primary flight controls and pitch trim, radio altitude and primary navigation information displayed to the flight crew, cockpit warnings and landing gear position; and

7. For aeroplanes specified in (a) above, the flight data recorder must record any dedicated parameters relating to novel or unique design or operational characteristics of the aeroplane.

8. Data must be obtained from aircraft sources which enable accurate correlation with information displayed to the flight crew.

9. The flight data recorder must start automatically to record the data prior to the aeroplane being capable of moving under its own power and must stop automatically after the aeroplane is incapable of moving under its own power.

10. The flight data recorder must have a device to assist in locating that recorder in water.

11. Aeroplanes with a maximum certificated take-off mass of 5700 kg or less may have the flight data recorder combined with the cockpit voice recorder.

An aeroplane may be despatched with the flight data recorder required by this section inoperative provided that (see IEM OPS 1.715 (h):

- It is not reasonably practicable to repair or replace the flight data recorder before the commencement of the flight;
- The aeroplane does not exceed 8 further consecutive flights with the flight data recorder unserviceable;
- Not more than 72 hours have elapsed since the flight data recorder was found to be unserviceable; and
- Any cockpit voice recorder required to be carried is operative, unless it is combined with the flight data recorder.

JAR-OPS 1.720 Flight Data Recorders-2

12. An operator shall not operate any aeroplane first issued with an individual certificate of airworthiness, either in a JAA Member State or elsewhere on or after 1 January 1989 up to and including 31 March 1998 which has a maximum certificated take-off mass over 5700 kg unless it is equipped with a flight data recorder that uses a digital method of recording and storing data and a method of readily retrieving that data from the storage medium is available.

13. The flight data recorder shall be capable of retaining the data recorded during at least the last 25 hours of its operation.

The flight data recorder must, with reference to a timescale, record:

14. The parameters necessary to determine altitude, airspeed, heading, acceleration, pitch and roll attitude, radio transmission keying unless an alternative means is provided to enable the recordings of the flight data recorder and the cockpit voice recorder to be synchronised, thrust or power on each engine, configuration of lift and drag devices air temperature, use of automatic flight control systems and angle of attack; and

15. For those aeroplanes with a maximum certificated take-off mass over 27,000 kg, the additional parameters necessary to determine positions of primary flight controls and pitch trim, radio altitude and primary navigation information displayed to the flight crew, cockpit warnings and landing gear position. (See AMC OPS 1.720 (c)/1.725(c).)

Data must be obtained from aeroplane sources which enable accurate correlation with information displayed to the flight crew

16. The flight data recorder must start to record the data prior to the aeroplane being capable of moving under its own power and must stop after the aeroplane is incapable of moving under its own power.

17. The flight data recorder must have a device to assist in locating that recorder in water.

18. An aeroplane may be dispatched with the flight data recorder required by this section inoperative provided that:

19. It is not reasonably practical to repair or replace the flight data recorder before the commencement of the flight;

20. The aeroplane does not exceed 8 further consecutive flights with the flight data recorder unserviceable;

21. Not more than 72 hours have elapsed since the flight data recorder was found to be unserviceable; and

22. Any cockpit voice recorder required to be carried is operative, unless it is combined with the flight data recorder. (See IEM OPS 1.720 (g)/1.725 (g).)

JAR-OPS 1.725 Flight Data Recorders-3

23. An operator shall not operate any turbine-engined aeroplane to which JAR-OPS 1.715 or JAR-OPS 1.720 is not applicable and which has a maximum certificated take-off mass over 5700 kg unless it is equipped with a flight data recorder that uses a digital method of recording and storing data and a method or readily retrieving that data from the storage medium is available, except that for aeroplanes registered in a JAA Member State on 1 April 1995 and first issued with an individual certificate of airworthiness in a JAA Member State or elsewhere before 1 April 1975, the continued use of non-digital recorders is acceptable until 1 April 2000.

24. The flight data recorder shall be capable of retaining the data recorded during at least the last 25 hours of its operation.

The flight data recorder must, with reference to a timescale, record:

25. For aeroplane first issued with an individual Certificate of Airworthiness, either in a JAA Member State of elsewhere before 1 January 1987:

26. The parameters necessary to determine altitude, airspeed, heading and normal acceleration; and

27. For those aeroplanes with a maximum certificated take-off mass over 27 000 kg that are of a type first type certificated in a JAA Member State or elsewhere after 30 September 1969, the additional parameters necessary to determine:

- Radio transmission keying unless an alternative means is provided to enable the recordings of the flight data recorder and the cockpit voice recorder to be synchronised;
- The attitude of the aeroplane in achieving its flight path; and

- The basic forces acting upon the aeroplane resulting in the achieved flight path and the origin of such forces.

28. For aeroplanes first issued with an individual certificate of airworthiness either in a JAA Member State or elsewhere on or after 1 January 1987 but before 1 January 1989:

- The parameters necessary to determine altitude, airspeed, heading and normal acceleration; and

29. For those aeroplanes with a maximum certificated take-off mass over 27,000 kg that are of a type first type certificated in a JAA Member State or elsewhere after 30 September 1969, the additional parameters necessary to determine:

- Radio transmission keying unless an alternative means is provided to enable the recordings of the flight data recorder and the cockpit voice recorder to be synchronised; and
- Pitch and roll attitude, thrust or power on each engine, configuration of lift and drag devices, air temperature, use of automatic flight control systems position, of primary flight controls and pitch trim, radio altitude and primary navigation information displayed to the flight crew, cockpit warnings and landing gear position. (See AMC OPS 1.720 (c) /1.725 (c).)

30. Data must be obtained from aircraft sources which enable accurate correlation with information displayed to the flight crew.

31. The flight data recorder must start to record the data prior to the aeroplane being capable of moving under its own power and must stop after the aeroplane is incapable of moving under its own power.

32. The flight data recorder must have a device to assist in locating that recorder in water.

33. Any aeroplane may be dispatched with the flight data recorder required by this section inoperative provided that:

- It is not reasonably practicable to repair or replace the flight data recorder before the commencement of the flight;
- The aeroplane does not exceed 8 further consecutive flights with the flight data recorder unserviceable;
- Not more than 72 hours have elapsed since the flight data recorder was found to be unserviceable; and
- Any cockpit voice recorder required to be carried is operative, unless it is combined with the flight data recorder. (See IEM OPS 1.720 (g)/1.725 (g).)

Flight Data Recorder Types

34. Flight data recorders are classified as Type 1, Type II and Type IIA depending upon the number of parameters to be recorded and the duration required for retention of the recorded information.

General Requirements

35. The recorder is to record continuously during flight time.

36. The recorder container is to:

- be painted a distinctive orange or yellow colour;
- carry reflective material to facilitate its location; and

- have securely attached an automatically activated underwater locating device.
- be shock, temperature and fire proofed.
- The recorder is to be installed so that:
- the probability of damage to the recording is minimized. To meet this requirement it should be located as far aft as practicable. In the case of pressurized aeroplanes it should be located in the vicinity of the rear pressure bulkhead;
- it receives its electrical power from a bus that provides the maximum reliability for operation of the recorder without jeopardizing service to essential or emergency loads; and
- there is an aural or visual means for pre-flight checking that the recorder is operating properly.

Type I Flight Data Recorder

37. This recorder will be capable of recording, as appropriate to the aeroplane, at least the 32 parameters in Figure 29-1. However, other parameters may be substituted with due regard to the aeroplane type and the characteristics of the recording equipment.

Types II and IIA Flight Data Recorders

38. These recorders will be capable of recording, as appropriate to the aeroplane, at least the first 15 parameters in Figure 29-1. However, other parameters may be substituted with due regard to the aeroplane type and the characteristics of the recording equipment.

Additional Information

39. A type IIA recorder, in addition to a 30-minute recording duration, is to retain sufficient information from the proceeding take-off for calibration purposes.

40. The measurement range, recording interval and accuracy of parameters on installed equipment is usually verified by methods approved by the appropriate certificating authority.

41. The manufacturer usually provides the national certificating authority with the following information in respect of the flight data recorder:

- manufacturer's operating instructions, equipment limitations and installation procedures;
- parameter origin or source and equations which relate counts to units of measurement; and
- manufacturer's test reports.

42. The operator usually supplies position error curves for pilot-static parameters, at various angles of attack and side slip, for the calibration and read-out of the recordings.

Flight Data Recorders (FDR)

FIGURE 29-1
List of FDR Parameters

Serial number	Parameter
1	Time (UTC when available, otherwise elapsed time)
2	Pressure-altitude
3	Indicated airspeed
4	Heading
5	Normal acceleration
6	Pitch altitude
7	Roll altitude
8	Radio transmission keying
9	Power on each engine
10	Trailing edge flap or cockpit control selection
11	Leading edge flap or cockpit control selection
12	Thrust reverser position
13	Ground spoiler/speed brake selection
14	Outside air temperature
15	Autopilot/autothrottle/AFCS mode and engagement status

16	Longitudinal acceleration
17	Lateral acceleration
18	Pilot input and/or control surface position-primary controls (pitch, roll, yaw)
19	Pitch trim position
20	Radio altitude
21	Glide path deviation
22	Localiser deviation
23	Marker beacon passage
24	Master warning
25	NAV 1 and 2 frequency
26	DME 1 and 2 distance
27	Landing gear squat switch status
28	GPWS (ground proximity warning system)
29	Angle of attack
30	Hydraulics, each system (low pressure)
31	Navigation data (latitude/longitude, groundspeed and drift angle)
32	Landing gear or gear selector position

NOTE:

NOTES The preceeding 32 parameters satisfy the requirements for a Type I FDR. Vso stalling speed or minimum steady flight speed in the landing configuration.
VD design diving speed.

If signals readily available:

43. If further recording capacity is available, recording of the following additional information should be considered:

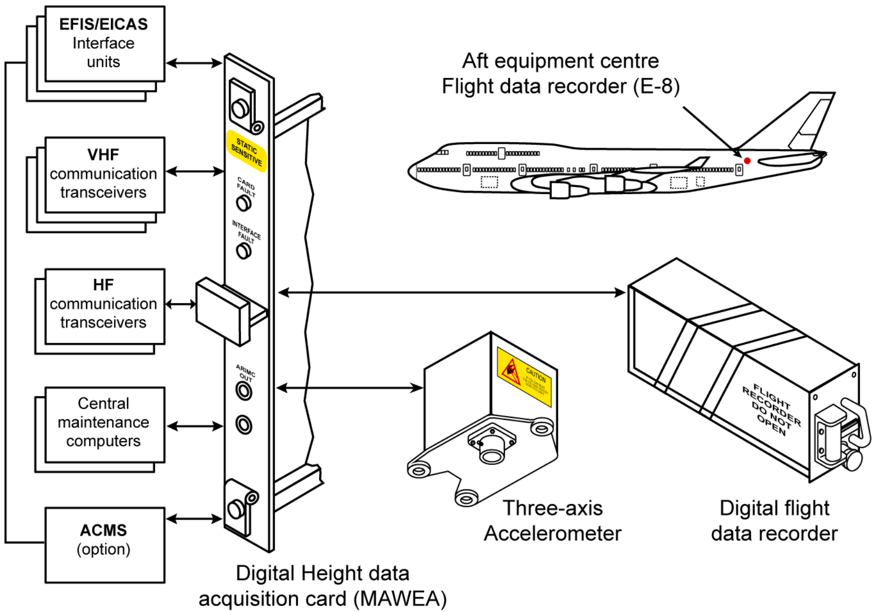
44. operational information from electronic displays systems, such as electronic flight instrument systems (EFIS), electronic centralized aircraft monitor (ECAM) and engine indication and crew alerting system (EICAS). Use the following order of priority:

- parameters selected by the flight crew relating to the desired flight path, e.g. barometric pressure setting, selected altitude, selected airspeed, decision height and autoflight system engagement and mode indications if not recorded from another source;
- display system selection/status, e.g. SECTOR, PLAN, ROSE, NAV, WXR, COMPOSITE, COPY, etc.;
- warnings and alerts;
- the identity of displayed pages for emergency procedures and checklists;

- retardation information including brake application for use in the investigation of landing overruns and rejected take-offs; and
- additional engine parameters (EPR, N1, EGT, fuel flow, etc.)

45. Figure 29-2 illustrates the components of a typical FDR System.

FIGURE 29-2
Boeing 747 Flight Data Recorder System



AIDS

Mandatory and Optional Recording Status

46. Flight data recorders are mandatory whereas the Aircraft Integrated Data System (AIDS) is optional.

47. AIDS provides the capability of gathering data for aircraft performance, engine condition and ADU health monitoring. The information can be displayed on a MCDU or printed out as real time parameters to give a ‘snap shot’ of current conditions or stored and down-loaded later.

48. A further option available under AIDS is the Aircraft Communication Addressing and Reporting System (ACARS). This system permits fault messages to be sent to the ground by VHF radio, and permits the operator’s engineering base to interrogate the aircraft in flight.

CFDS

49. A centralised fault display system (CFDS) includes a centralised fault data interface unit (CFDU) which memorises faults occurring on a set number of previous fights (e.g. 63) or a set number of failures (e.g. 200) which ever comes.

Flight Data Recorder Power Supply

50. The flight data recorder receives its electrical power supply from a bus that provides the maximum reliability for operation of the recorder without jeopardising service to essential or emergency loads.

Flight Data Recorder Monitoring

51. Prior to the first flight of the day, the built-in test features of the cockpit voice recorder and the flight data recorder should be monitored. An annual inspection should be carried out of the CVR and FDR to ensure that the recorders operate correctly for the nominal duration of the recording. The FDR system should be re-calibrated every five years. When parameters of altitude and airspeed are provided by sensors that are dedicated to the FDR, there should be re-calibration at least every two years.

Cockpit Voice Recorders (CVR)

Cockpit Voice Recorders (CVR)

JAR Requirements

JAR-OPS 1.700 Cockpit Voice Recorders-1

1. An operator shall not operate an aeroplane first issued with an individual Certificate of Airworthiness, either in a JAA Member State or elsewhere, on or after 1 April 1998, which:

Is multi-engine turbine powered and has a maximum approved passenger seating configuration of more than 9; or

Has a maximum certificated take-off mass over 5700 kg, unless it is equipped with a cockpit voice recorder which, with reference to a time scale records:

- Voice communications transmitted from or received on the flight deck by radio;
- The aural environment of the flight deck, including without interruption, the audio signals received from each boom and mask microphone in use;
- Voice communications of flight crew members on the flight deck using the aeroplane's interphone system;
- Voice of audio signals identifying navigation or approach aids introduced into a headset or speaker; and

- Voice communications of flight crew members on the flight deck using the public address system, if installed.

2. The cockpit voice recorder shall be capable of retaining information recorded during at least the last 2 hours of its operation except that, for those aeroplanes with a maximum certificated take-off mass of 5700 kg or less, this period may be reduced to 30 minutes.

3. The cockpit voice recorder must start automatically to record prior to the aeroplane moving under its own power and continue to record until the termination of the flight when the aeroplane is no longer capable of moving under its own power. In addition, depending on the availability of electrical power, the cockpit voice recorder must start to record as early as possible during the cockpit checks prior to engine start at the beginning of the flight until the cockpit checks immediately following engine shutdown at the end of the flight.

4. The cockpit voice recorder must have a device to assist in locating that recorder in water.

5. In complying with this section, aeroplanes with a maximum certificated take-off mass of 5700 kg or less may have the cockpit voice recorder combined with the flight data recorder.

6. Any aeroplane may be despatched with the cockpit voice recorder required by this section inoperative provided that:

- It is not reasonably practicable to repair or replace the cockpit voice recorder before the commencement of the flight;
- The aeroplane does not exceed 8 further consecutive flights with the cockpit voice recorder unserviceable;
- Note more than 72 hours have elapsed since the cockpit voice recorder was found to be unserviceable; and

- Any flight data recorder required to be carried is operative unless it is combined with a cockpit voice recorder.

JAR-OPS 1.705 Cockpit Voice Recorders-2

7. After 1 April 2000 an operator shall not operate any multi-engined turbine aeroplane first issued with an individual certificate of airworthiness, either in a JAA Member State or elsewhere on or after 1 January 1990 up to and including 31 March 1988 which has a maximum certificated take-off mass of 5700 kg or less and a maximum approved passenger seating configuration of more than 9, unless it is equipped with a cockpit voice recorder which records:

- Voice communications transmitted from or received on the flight deck by radio;
- The aural environment of the flight deck, including where practicable, without interruption, the audio signals received from each boom and mask microphone in use;
- Voice communication of flight crew members on the flight deck using the aeroplane's interphone system;
- Voice or audio signals identifying navigation or approach aids introduced into a headset or speaker; and
- Voice communications of flight crew members on the flight deck using the public address system, if installed.

8. The cockpit voice recorder shall be capable of retaining information recorded during at least the last 30 minutes of its operation.

9. The cockpit voice recorder must start to record prior to the aeroplane moving under its own power and continue to record until the termination of the flight when the aeroplane is no longer capable of moving under its own power. In addition, depending on the availability of electrical power, the cockpit voice recorder must start to record as early as possible during the cockpit checks, prior to the flight until the cockpit checks immediately following engine shutdown at the end of the flight.

10. The cockpit voice recorder must have a device to assist in locating that recorder in water.

11. An aeroplane may be dispatched with the cockpit voice recorder required by this section inoperative provided that:

- It is not reasonably practicable to repair or replace the cockpit voice recorder before the commencement of the flight;
- The aeroplane does not exceed 8 further consecutive flights with the cockpit voice recorder unserviceable;
- Not more than 72 hours have elapsed since the cockpit voice recorder was found to be unserviceable; and
- Any flight data recorder required to be carried is operative unless it is combined with a cockpit voice recorder.

JAR-OPS 1.710 Cockpit Voice Recorders-3

12. An operator shall not operate any aeroplane with a maximum certificated take-off mass over 5700 kg first issued with an individual certificate of airworthiness, either in a JAA Member State or elsewhere, before 1 April 1998 unless it is equipped with a cockpit voice recorder which records:

- Voice communications transmitted from or received on the flight deck by radio;
- The aural environment of the flight deck;
- Voice communications of flight crew members on the flight deck using the aeroplane's interphone system;
- Voice or audio signals identifying navigation or approach aids introduced into a headset or speaker; and
- Voice communications of flight crew members on the flight deck using the public address system, if installed.

13. The cockpit voice recorder shall be capable of retaining information recorded during at least the last 30 minutes of its operation.

14. The cockpit voice recorder must start to record prior to the aeroplane moving under its own power and continue to record until the termination of the flight when the aeroplane is no longer capable of moving under its own power.

15. The cockpit voice recorder must have a device to assist in locating that recorder in water.

16. An aeroplane may be dispatched with the cockpit voice recorder required by this section inoperative provided that:

- It is not reasonably practicable to repair or replace the cockpit voice recorder before the commencement of the flight;
- The aeroplane does not exceed 8 further consecutive flights with the cockpit voice recorder unserviceable;

- Not more than 72 hours have elapsed since the cockpit voice recorder was recorder was found to be unserviceable; and
- Any flight data recorder required to be carried is operative.

General Requirements

17. The recorder is to be designed so that it will record at least the following:

- voice communication transmitted from or received in the aeroplane by radio;
- aural environment on the flight deck;
- voice communication of flight crew members on the flight deck using the aeroplane's interphone system;
- voice or audio signals identifying navigation or approach aids introduced in the headset or speaker; and
- voice communication of flight crew members using the passenger address system, if installed.
- The recorder container is to:
- be painted a distinctive orange or yellow colour;
- carry reflective material to facilitate its location; and
- have securely attached an automatically activated underwater locating device.
- be shock, temperature and fire-proofed.

18. To aid in voice and sound discrimination, microphones in the cockpit are to be located in the best position for recording voice communications originating at the point and co-pilot stations and voice communications of other crew members on the flight deck when directed to those stations. This can best be achieved by wiring suitable boom microphones to record continuously on separate channels.

19. The recorder is to be installed so that:

- the probability of damage to the recording is minimized. To meet this requirement it should be located as far aft as practicable. In the case of pressurized aeroplanes it should be located in the vicinity of the rear pressure bulkhead;
- it receives its electrical power from a bus that provides the maximum reliability for operation of the recorder without jeopardizing service to essential or emergency loads;
- there is an aural or visual means for pre-flight checking of the recorder for proper operation; and

Performance Requirements

20. The recorder will be capable or recording on at least four tracks simultaneously. To ensure accurate time correlation between tracks, the recorder is to record in an in-line format. If a bi-directional configuration is used, the in-line format and track allocation should be retained in both directions.

21. The preferred track allocation is as follows:

- Track 1 – co-pilot headphones and live boom microphone

- Track 2 – pilot headphones and live boom microphone
- Track 3 – area microphone
- Track 4 – time reference plus the third and fourth crew members' headphone and live microphone, if applicable.

NOTE:

Track 1 is located closest to the base of the recording head.

22. The recorder, when tested by methods approved by the appropriate certificating authority, will be demonstrated to be suitable for the environmental extremes over which it is designed to operate.

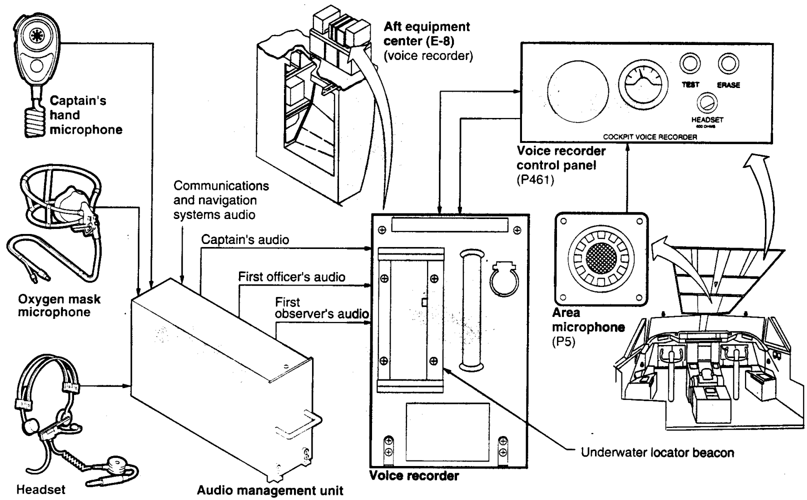
23. Means will be provided for an accurate time correction between the flight data recorder and cockpit voice recorder. One method of achieving this is by superimposing the FDR time signal on Track 4 of the CVR.

Additional Information

24. The manufacturer usually provides the national certificating authority with the following information in respect of the cockpit voice recorder:

- manufacturer's operating instructions, equipment limitations and installation procedures; and manufacturer's test reports.

FIGURE 30-1
Boeing 747 Cockpit Voice Recorder



25. The multiple-channel voice recorder with cockpit area microphone records the last 30 minutes of flight crew communications and is automatically erased so that only the last 30 minutes remain recorded.

26. Input to the voice recorder is from the area microphone, from the captain's, first officer's, and observer's received audio, and from hot microphone input to the Audio Management Unit.

27. The cockpit voice recorder control panel, which has test and erase buttons, is located on the aft over head panel. The area microphone in the cockpit is on the forward overhead panel.

28. The recorder unit is located in the aft equipment center and is accessible for replacement. It includes an underwater locator beacon (ULB) with replaceable battery.

29. The complete tape can be erased when the airplane is on the ground and the parking brake is set. Figure 30-1 illustrates the components of the Boeing 747 CVR system.

Powerplant and System Monitoring Instruments

Powerplant and System Monitoring Instruments

1. In order to monitor the operation of the aircraft engines the pilot must be provided with adequate instrumentation displaying the various parameters that indicate performance of the engines and their associated systems. As aircraft engines have become more complex the extent of instrumentation required has grown. By the end of the 1970's the complexity of a large transport aircraft instrument panel had virtually reached the practical limits of cockpit space and required two pilots and a flight engineer to monitor the array of gauges.

2. This and the demand by the airline operators for reduction in flight crew numbers led to the introduction of electronic 'glass cockpit' instrumentation with compact presentations, displaying information on an as-required basis only.

3. This section looks at the operating principles and displays of conventional gauges and meters and concludes with an overview of electronic engine and system information displays.

Pressure Gauges

4. Pressure instruments used in connection with aircraft systems are either of the direct reading type or the remote reading type.

5. A direct reading pressure gauge is one in which the fluid pressure to be sensed is fed directly to an elastic element within the gauge. Deformation of the element due to pressure is used to operate a mechanism that drives a pointer across the face of the gauge.

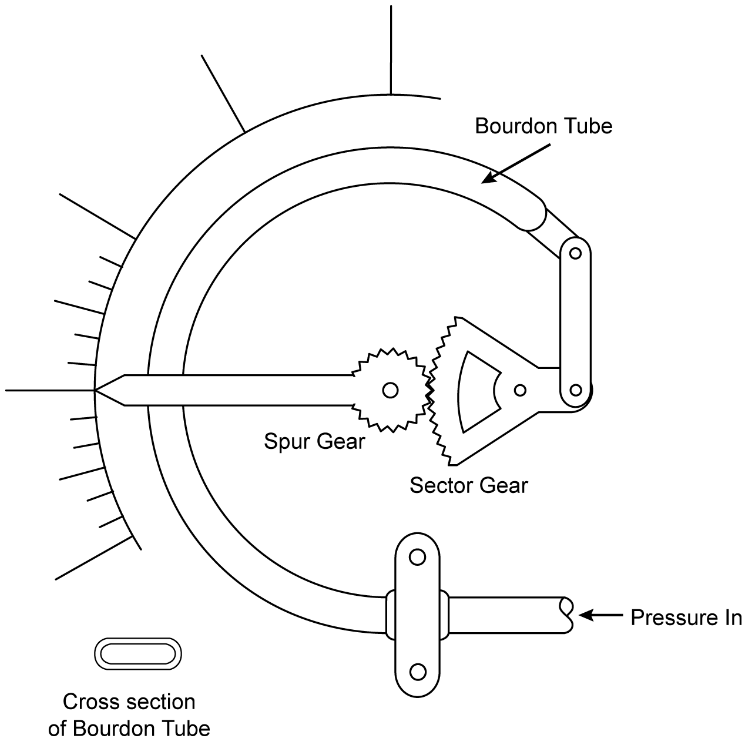
6. A remote reading pressure gauge is one in which a transmitter senses the fluid pressure at source, and transmits a proportionate signal to a remote indicator. Remote reading pressure gauges have the advantage that lengthy pipes carrying pressurised fluid, with the accompanying potential for leakage, are not required.

7. Pressures are defined in one of two ranges. Absolute pressure is pressure above absolute zero, that is the difference between the pressure being measured and the pressure in a complete vacuum. Gauge pressure is the difference between the pressure being measured and ambient atmospheric pressure.

8. Direct reading pressure gauges used to measure high pressures invariably operate on the Bourdon tube principle. The fluid under pressure is led to the inside of a flattened tube formed into an almost complete circle and closed at the end. As pressure increases the tube tends to straighten, but the material from which the tube is made, usually bronze or copper alloy opposes this tendency. The partial straightening motion is mechanically transmitted through linkages and gearing to a pointer, which moves against a calibrated scale on the face of the pressure gauge.

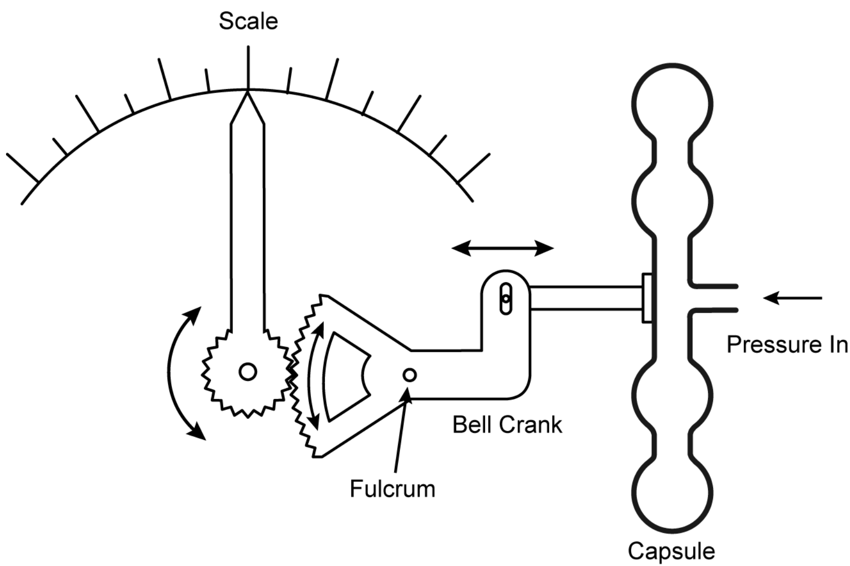
9. The Bourdon tube principle is illustrated at Figure 31-1. This type of gauge is typically used for the measurement of engine lubricating oil pressure.

FIGURE 31-1
Bourdon Tube Gauge



10. Direct reading pressure gauges that are used to measure lower pressures are usually of the capsule type. In these the fluid under pressure is led to the inside of a capsule formed of two corrugated metal discs joined around their periphery. As fluid pressure increases the capsule expands and the expansion is transmitted to a moving pointer by means of linkages and gearing. This principle is illustrated at Figure 31-2.

FIGURE 31-2
Capsule Operated Pressure Gauge



11. A typical use of the capsule type of pressure gauge in engine instrumentation is the MAP gauge. Since MAP stands for manifold absolute pressure, the MAP gauge contains two diaphragms. One of these is an aneroid capsule, sealed and evacuated so that it is sensitive to atmospheric pressure. The inside of the other capsule is connected to the engine inlet manifold and is sensitive to manifold pressure between throttle valve and the cylinder inlet valves. The two capsules are mechanically linked so that the combined transmitted movement to the gauge pointer is proportional to the difference between manifold pressure and absolute zero.

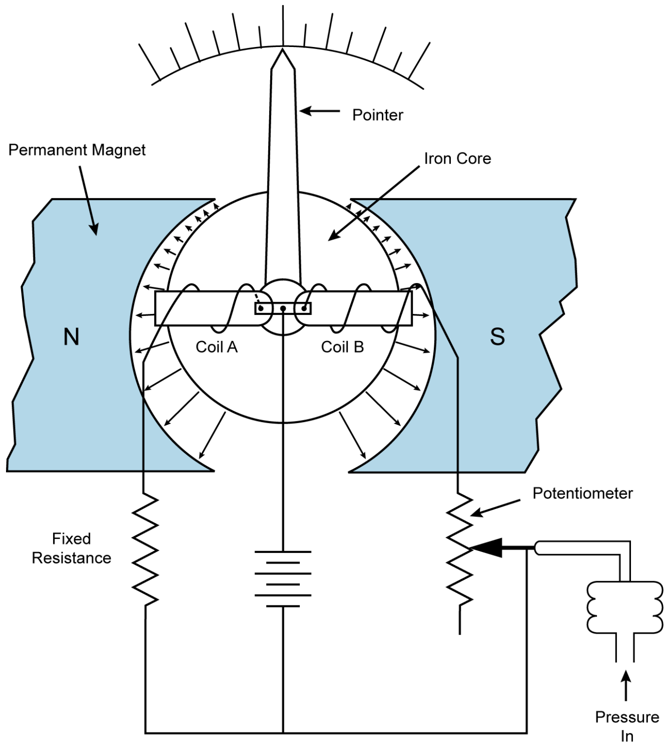
12. Remote reading pressure gauges in aircraft systems usually employ a transmitter in which the fluid pressure is led to the inside of a bellows capsule. Mechanical movement due to expansion of the bellows unit is converted into an electrical signal of voltage and current proportional to the pressure being sensed, and transmitted to a remote indicator in the cockpit. A ratiometer or a synchronous motor then moves the indicator pointer against a calibrated scale on the face of the pressure gauge.

13. The electrical circuit for a simple ratiometer is shown at Figure 31-3. The instrument consists of an iron core, mounted on a spindle and situated eccentrically between the poles of a permanent magnet so that the gap between the poles and the core varies in width. Two coils, A and B, are mounted on the iron core and supplied with current from a common source. The current to coil A passes through a fixed resistance, whilst the current to coil B passes through a resistance that is varied by movement of the pressure-sensing bellows in the transmitter.

14. When the current flowing through each coil is identical, electromagnetic forces are produced that exactly oppose each other and the core (and gauge pointer) is held stationary. Let us suppose that expansion of the pressure-sensing bellows decreases the value of the bellows-operated resistance. The current flowing through coil B will now be greater than that flowing through coil A.

15. The electromagnetic field of coil B is now greater than that of coil A and coil B will move towards the larger gap where the permanent magnetic field flux density is lower. This will rotate the iron core, moving the gauge pointer across the calibrated scale. As it does so, the electromagnetic field produced by coil A is moving into a narrowing gap of increasing permanent flux density, arresting the rotation of the core. The gauge pointer now indicates the new, increased pressure.

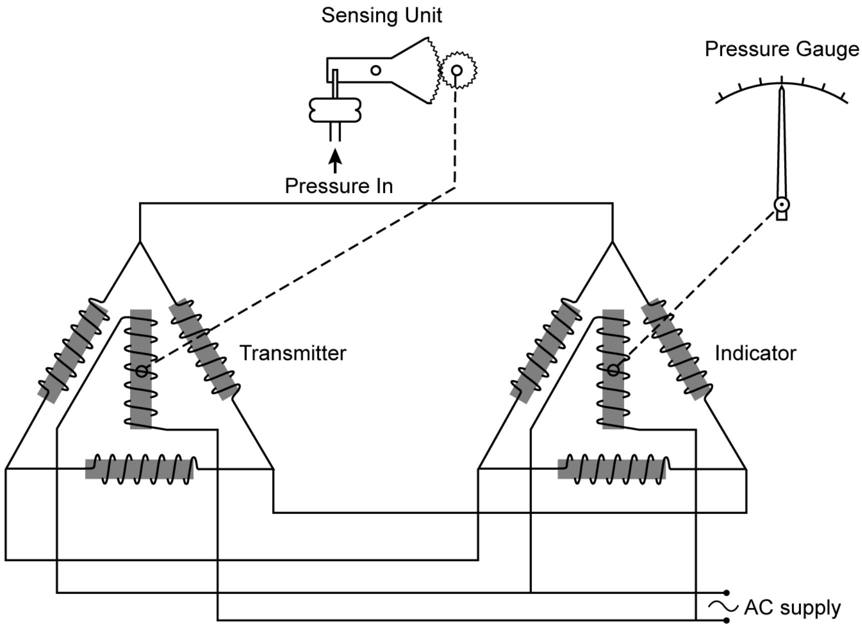
FIGURE 31-3
Principle of Ratiometer Instrument



16. A synchronous, or Desynn remote reading system is shown at Figure 31-4. This type of transmitter was very common in older aircraft, but is rarely found nowadays. Movement of pressure-sensing bellows or a Bourdon tube rotates the transmitter rotor an amount proportional to the change in pressure. Current flowing through a coil wound around the rotor creates an electromagnetic field, which induces a discrete current strength in each of the three coils in the transmitter stator.

17. The transmitter stator coils are connected to three identically located stator coils in the remote reading indicator, where the discrete current flows produce an electromagnetic field identical to that at the transmitter. The indicator rotor takes up a position dictated by the stator field, where the interaction of the rotor and stator fields is identical to that at the transmitter. The indicator, or gauge pointer is attached to the indicator rotor and it moves around the calibrated face of the gauge.

FIGURE 31-4
Desynn Synchro System

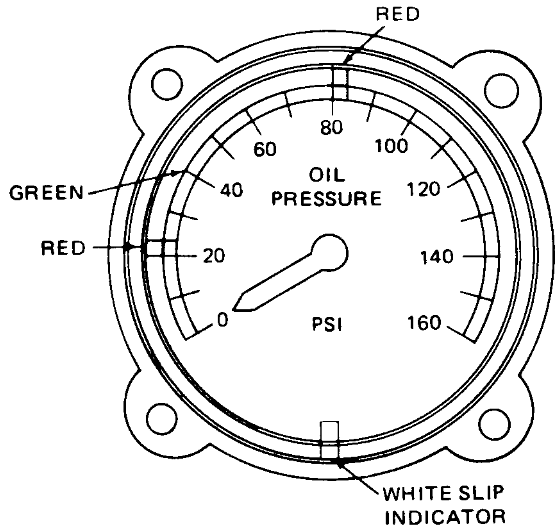


Pressure Indicators

18. The conventional pressure gauge display is typically a circular scale calibrated in units of pressure around which a pointer, pivoted at the centre of the scale, rotates. Occasionally linear displays are used in which the scale is positioned vertically and a pointer moves up the scale as pressure increases. The latter type of display is sometimes favoured where comparative measurements are required, in which case a number of such displays are mounted side-by-side.

19. Engine pressure gauges are colour coded to define safe operating limits. A circular-scale piston engine oil pressure gauge is shown at Figure 31-5. Red lines indicate the minimum and maximum permissible operating pressures and a green arc indicates the normal operating pressure range. If these coloured markings appear on the glass cover of the instrument a white slippage mark is painted on the glass and extended to the instrument case. Thus, if the cover glass should move the slippage mark will be broken and the two lines offset.

FIGURE 31-5
Engine Oil Pressure Gauge with Range Markings



20. MAP gauges are also colour coded, with a blue arc defining the cruise power region in which lean mixture may be used and a green arc showing the range in which rich mixture is required. A red line indicates the maximum allowable MAP to be used for take-off power.

21. In piston engines with fuel injection systems the fuel pressure is directly proportional to engine power and therefore to fuel consumption. Fuel pressure gauges are consequently often calibrated in terms of engine power or fuel flow rate in addition to the fuel pressure scale. In such cases the coloured arcs on the face of the gauge are similar to those on the MAP gauge, a blue arc for cruise power and lean mixtures and a green arc for higher power and rich mixtures.

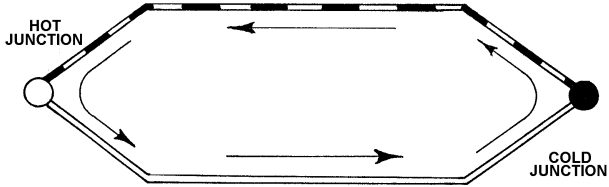
Temperature Gauges

22. The principal temperature gauges associated with piston engine instrumentation are for the measurement of oil temperature and cylinder head temperature (CHT). In addition, carburettor air temperature (CAT) and exhaust gas temperature are sensed and indicated in many piston engines, the former to indicate icing hazard and the latter to indicate combustion efficiency.

23. In gas turbine engines oil temperature and either turbine inlet or exhaust gas temperature gauges are essential requirements and air inlet temperature is usually indicated.

24. High temperatures, such as cylinder head and exhaust gas temperatures, are usually measured by thermocouples. Thermocouples convert heat energy into electrical energy at the source of heat. They rely upon what is known as the Seebeck effect where two wires of dissimilar metal are laid parallel and joined at both ends, as shown in Figure 31-6. An emf is produced when there is a temperature difference between the two junctions, the greater the temperature difference, the greater the emf.

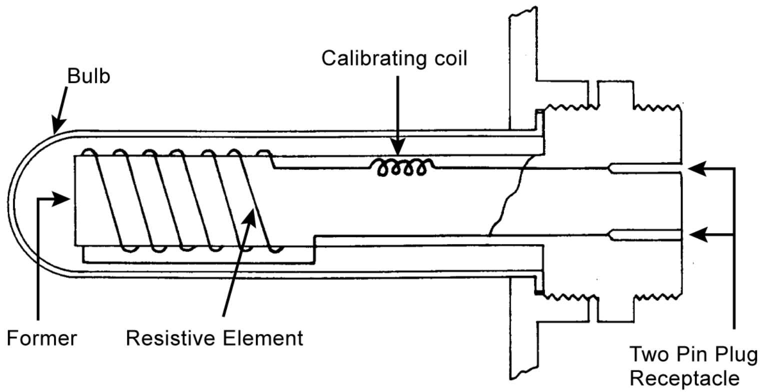
FIGURE 31-6
Thermo-Couple



25. The hot junction represents the thermocouple probe, placed in the hot gas flow (immersion type) or connected to a point on the cylinder head (surface contact type). The cold junction is formed at the temperature indicator, which measures current flow. The greater the temperature difference, the greater the emf produced and therefore the greater the current flow.

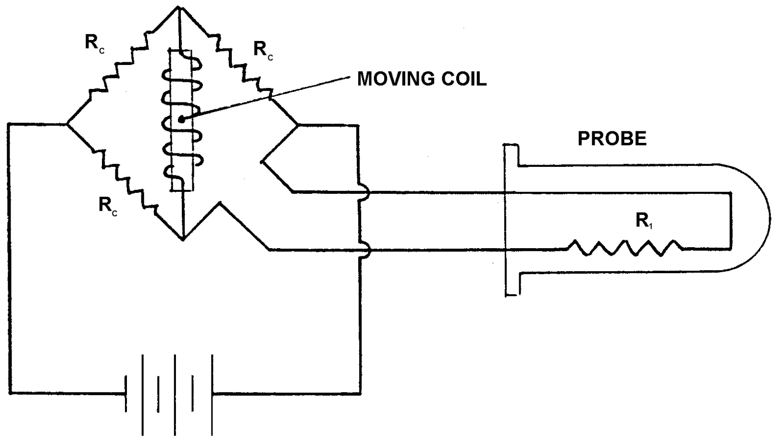
26. Lower temperature ranges, such as oil temperature, fuel temperature, carburettor air temperature or gas turbine inlet air temperature are sensed using a resistive probe connected to an electrical circuit such as the ratiometer described above or a Wheatstone bridge circuit. A temperature-sensing probe is illustrated at Figure 31-7.

FIGURE 31-7
Temperature Sensing Probe



27. The temperature probe, containing a resistance of known value, is placed in the medium whose temperature is to be measured. This resistor (R_1) forms one of the four resistors in a Wheatstone Bridge circuit as shown at Figure 31-8. A moving coil is connected across the bridge circuit as shown, and low voltage is applied to the circuit. If all four resistors are of equal value there will be no potential difference across the coil that bridges the resistors, and therefore no current flow through it.

FIGURE 31-8
Temperature Probe



28. The resistance of a conductor varies directly with its temperature. Hence, if the temperature of the medium surrounding the probe increases, the resistance of the probe will increase. This causes a reduction in current flow on that side of the bridge, which results in a current flow through the moving coil. The greater the temperature change, the greater the current flow through the coil.

29. The moving coil lies in a permanent magnetic field. Current flow through the coil creates a second magnetic field, concentrated in the soft-iron core of the coil. Magnetic attraction will cause the coil to rotate, operating the pointer of a temperature indicator. Coil movement is controlled by a linear spring and the amount of magnetic force in the coil will depend upon the current flow through it. The greater the potential difference (the greater the sensed temperature change at the probe), the greater the current flow.

30. Air temperature measurement is of great significance when determining engine performance, particularly in the case of gas turbine engines. Ideally, the air temperature measured should be the static air temperature (SAT) at the flight level of the aircraft. In most cases however, measurement of SAT is not possible due to adiabatic compression of the air as a result of flight speed or friction heating of the moving air in contact with the surface of the aircraft.

31. The increase in measured air temperature above SAT is known as the ram rise and the temperature measured as a result is known as ram air temperature (RAT). Ram rise can be calculated, since it is a function of airspeed, and aircraft operating manuals contain charts or graphs from which the air temperature gauge reading can be corrected to give SAT for any given Mach number, by subtracting the tabulated ram rise. Air data computers make this correction automatically.

32. Temperature sensors are not usually capable of sensing, or recovering, the full temperature increase due to ram rise and the extent to which they do is known as the recovery factor of the sensor. For example, a sensor which measures SAT plus 85% of the ram rise is said to have a recovery factor of 0.85. The tables in the operating manuals take the recovery factor of each sensor into account.

33. As with pressure gauges, the scales of engine temperature gauges are often coloured to indicate performance limitations.

34. The face of the oil temperature gauge is marked with a green sector to indicate the safe normal operating temperature range. Red lines on the gauge indicate the minimum and maximum safe operating temperatures. Generally the minimum safe temperature is 40°C. The normal operating range is typically 60°C to 70°C and is indicated by a green arc. The maximum permissible temperature is usually about 100°C.

35. The carburettor air temperature gauge is typically calibrated from -50°C to +150°C. There are three coloured arcs on the face of the gauge. Yellow, from -10°C to +15°C, indicates the icing hazard range. Green, from +15°C to +40°C, is the normal operating range. Red, above +40°C indicates detonation hazard.

36. EGT gauges for gas turbine engines may be marked with a red line or arc to indicate maximum permissible limits, but do not usually have other coloured arcs.

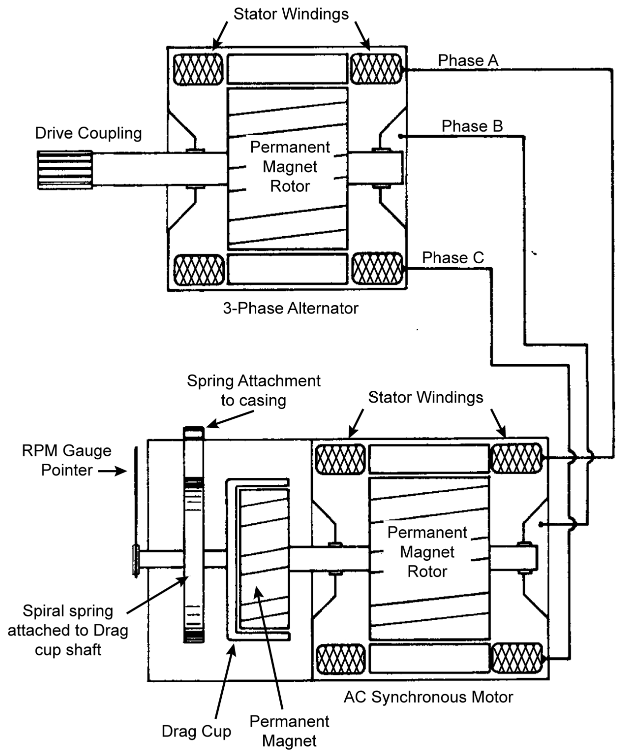
RPM Indicators

37. An instrument that measures and indicates rotary speed is called a tachometer.

38. Mechanical tachometers, commonly used with smaller piston engined aircraft, are connected to the engine drive by means of a flexible shaft, which rotates a magnet within the indicator casing. The resultant rotating magnetic field causes a spring-restrained rotor to attempt to follow the rotating field. The greater the speed of rotation, the more the rotor moves against the restraining spring force and the limited arc of movement produced is used to drive the tachometer pointer a controlled distance around the face of the rpm gauge. This device is known as a drag cup.

39. RPM measurement in larger piston engines is often by means of an electric tachometer of the type illustrated at Figure 31-9. A 3-phase alternator is driven by the engine accessories drive. The output of the alternator is fed to the three phase windings in the stator of a synchronous motor, where a rotating magnetic field is set up by the alternating current. The speed of rotation of the generator determines the frequency of the output alternating current, which in turn determines the rotary speed of the stator field in the motor

FIGURE 31-9
RPM Indicator



40. The rotor of the synchronous motor is a permanent magnet and so it is constrained to follow the rotating magnetic field of the stator, turning at the same speed as the alternator. The motor drives a drag cup similar to that described above in connection with the mechanical tachometer. Piston engine rpm gauges are calibrated in revolutions per minute.

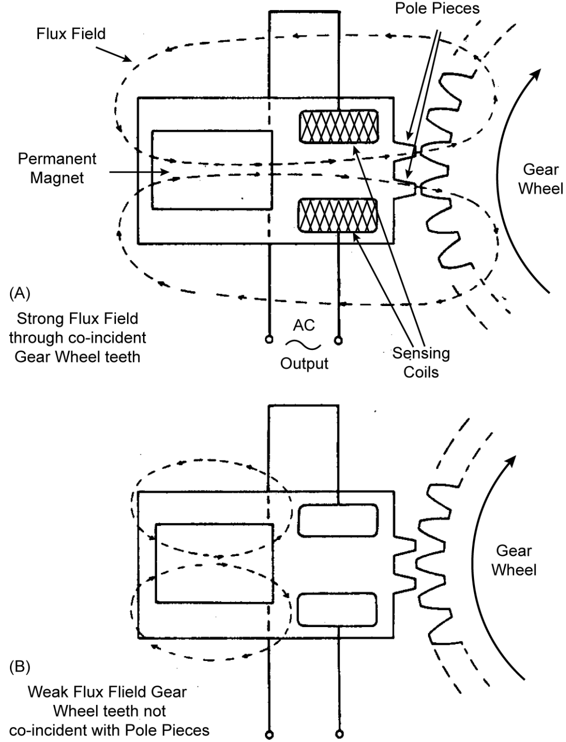
41. An alternative to the above, commonly used with gas turbine engines, is a servo-operated system in which the output of the engine-driven alternator is first converted to square waveform and thence to a series of DC pulses at twice the frequency of the alternator AC output. The pulsed DC powers a torquemotor that drives the rpm gauge pointer through a potentiometer.

42. Turbo-fan engine rotary speed is usually displayed in terms of percentage rpm with the scale calibrated from 0% - 100% in 10% increments, where 100% represents optimum turbine speed. Rotary speed is often measured and indicated by means of a tacho-probe indicator.

43. Tacho probe/indicator systems are made up of a flux switch, in the form of a magnetic probe, the principle of which is illustrated at Figure 31-10.

FIGURE 31-10

Tachometer Probe / Indicator



44. The flux density of the field of a permanent magnet in the probe is greatest when the two "prongs" of the probe are coincident with two teeth of a gear wheel, which is rotating at the same speed as the fan shaft. Thus as the gearwheel rotates the flux density in the probe fluctuates at a rate directly proportional to the speed of rotation. The fluctuating flux density induces an emf in sensing coils mounted within the probe, which are connected to a circuit in which the probe output is converted to DC to drive the rpm gauge pointer through a DC torquemotor and potentiometer.

45. RPM gauge displays usually incorporate coloured arcs and limit markings, with a green arc indicating the normal operating range of engine speeds and a red line indicating maximum permissible rpm.

Consumption Gauges

46. There are two principal methods of measuring fuel flow in aircraft, independent and integrated systems.

47. Independent systems use a metering device fitted in the fuel supply line that converts mechanical movement of a spring-loaded vane into a proportionate electrical output, either by means of a potentiometer or an a.c. synchro of the type described above in connection with pressure indicators and illustrated at Figure 31-4. The electrical output operates an indicator, which is simply a milliammeter whose scale is calibrated in rate of fuel flow (gals/hr, kg/hr and so on).

48. The moving vane flowmeter normally incorporates a spring-loaded bypass valve, which will open in the event of the vane sticking and blocking fuel flow to the engine. Opening of the valve may activate a warning indicator on the flow gauge.

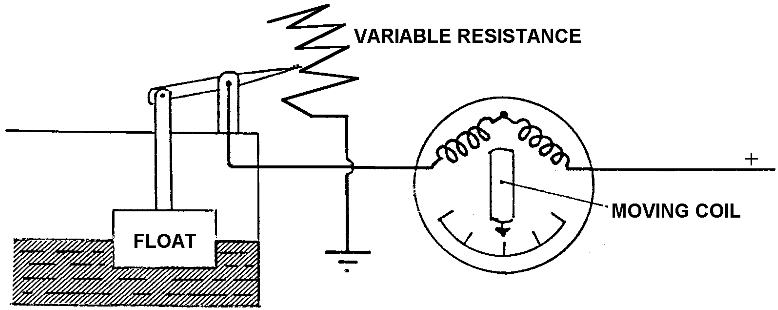
49. Integrated systems display both fuel flow (kg/hr) and total fuel consumed (kg). To achieve this the system includes an integrator, which computes fuel flow against time to give fuel used. Fuel flow metering and transmission is a little more complex than in the independent system. In essence it consists of a turbine impeller in the fuel supply line to the engine whose speed of rotation depends upon rate of fuel flow. Speed of rotation is converted into an electrical signal by electro-magnetic induction and fed to a computer. The computer operates the fuel flow and consumption gauges.

50. In large turbine engine fuel systems the flowmeter is usually situated in the high-pressure fuel line between the fuel control unit (FCU) and the burner nozzles. The associated gauge typically indicates both flow rate and fuel used and may incorporate a low fuel flow warning.

Fuel Quantity Gauges

51. In light aircraft, fuel tank quantity is usually measured by means of a float-operated variable resistance. The float, inside the tank, adjusts a variable resistance, or potentiometer to produce a current flow proportional to float position (fuel level). A Wheatstone bridge circuit or a ratiometer operates a moving coil in the fuel gauge, moving a pointer over a scale that is usually calibrated in volumetric quantity (gallons or litres). A schematic diagram of this form of measurement is at Figure 31-11.

FIGURE 31-11
Resistive Quantity Indicator



52. The resistive form of tank contents measurement illustrated at Figure 31-11 suffers from two significant disadvantages. In the first place the float-operated transmitter is highly susceptible to fuel movement within the tank and will give a false reading if aircraft attitude changes. Secondly, since it is purely liquid level within the tank that is being sensed, if the fuel expands due to increased temperature then the gauge will again give a false indication of tank contents.

53. In the relatively small tanks of a light aircraft the errors due to these failings are not of great significance, but in a large aircraft fuel tank the changes in fuel level due to movement or expansion may well be sufficient to cause appreciable errors, especially in an integrated system.

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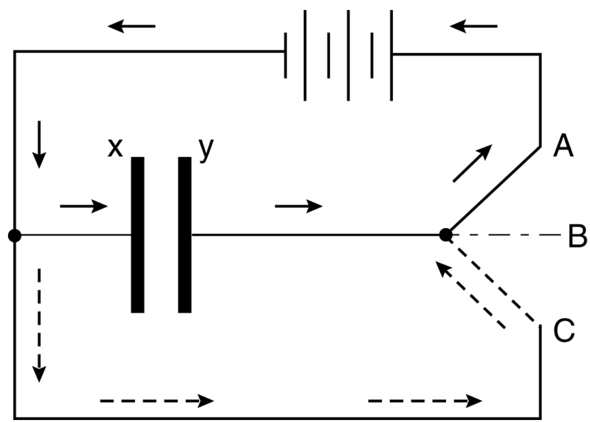
54. The inherent errors of the resistive system of tank contents measurement can be overcome by using a capacitive system of measurement. The principle of a capacitor is shown at Figure 31-12. If a potential difference exists between two conductors separated by an insulator (called a dielectric) an electrical charge will be stored in the conductors. If the switch in Figure 31-12 is placed in position A, electrons will flow from plate y, through the battery, to plate x until the potential difference between x and y equals the potential difference of the battery

FIGURE 31-12
Capacitor Circuit

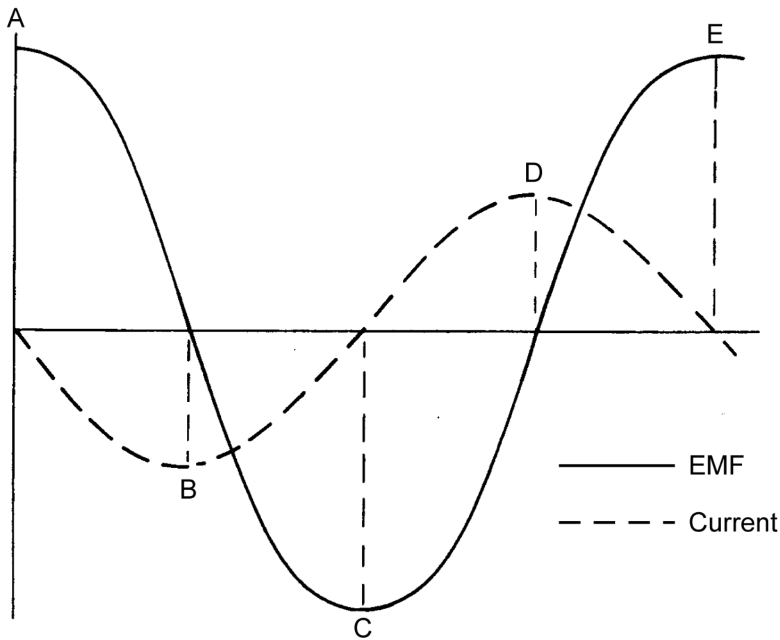


55. If the switch is moved to position B, plates x and y will remain negatively and positively charged respectively.

56. When the switch is moved to position C the plates are short-circuited and there is a flow of electrons from x to y until the potential difference between the plates is zero.

57. When a capacitor is supplied with alternating current the rising and falling voltage has a somewhat similar effect to the switch in the circuit at Figure 31-12. Figure 31-13 illustrates the effect

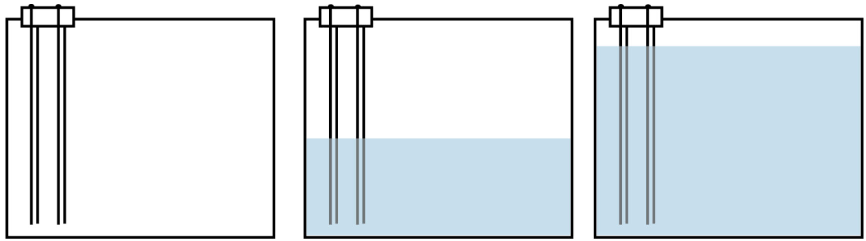
FIGURE 31-13
Effect of Capacitor in AC Circuit



58. At point A the voltage is at peak value and is virtually constant with respect to time for a finite period and the capacitor is fully charged, with no current flowing into or out of it. As the circuit voltage falls between points A and B the voltage stored in the capacitor is greater than circuit voltage and so there is a current flow out of the capacitor. In other words, the capacitor is discharging its stored charge, the rate of discharge current flow being greatest at point B where the rate of change of circuit voltage is greatest. Between points B and C circuit voltage is rising and current flow is now into the capacitor. The capacitor is now being charged and at point C it reaches maximum charge once more, but of opposite polarity. From point C to point D the capacitor is discharging as circuit voltage falls to zero. Thus there is a charge and discharge every half cycle of the AC supply, with the current leading the voltage by 90°.

59. Measurement of fuel tank contents using capacitance is achieved by making the "plates" of the capacitor in the form of two concentric tubes (i.e. one inside the other) extending from the top of the tank almost to the bottom. Fuel has a dielectric strength more than twice that of air so the capacitance, or charge-storing capability, of the capacitor varies in proportion to the fuel level in the tank. The principle is illustrated at Figure 31-14.

FIGURE 31-14
Changes in Capacitance with Changing Fuel Level



60. Let us assume that the capacitance of the air in the empty fuel tank is 100 picofarads (pF) and that the fuel to be used has a capacitance exactly twice that of air (remember that in reality it is more than twice). When the tank is half full, as in the second diagram of Figure 31-14, the capacitance of the fuel in the lower half of the tank is 200 pF and that of the air in the upper half is 100 pF. Half the tank is filled with fluid of double capacitance so the total capacitance will be 100 + 50 = 150 pF. If the tank were a quarter full the total capacity would be 125 pF, at three-quarters full it would be 175 pF and when completely full it would be 200 pF.

61. A capacitance fuel tank contents circuit is shown at Figure 31-15. It comprises two loops, one of which contains the capacitor probe in the fuel tank whilst the other is a balancing loop.

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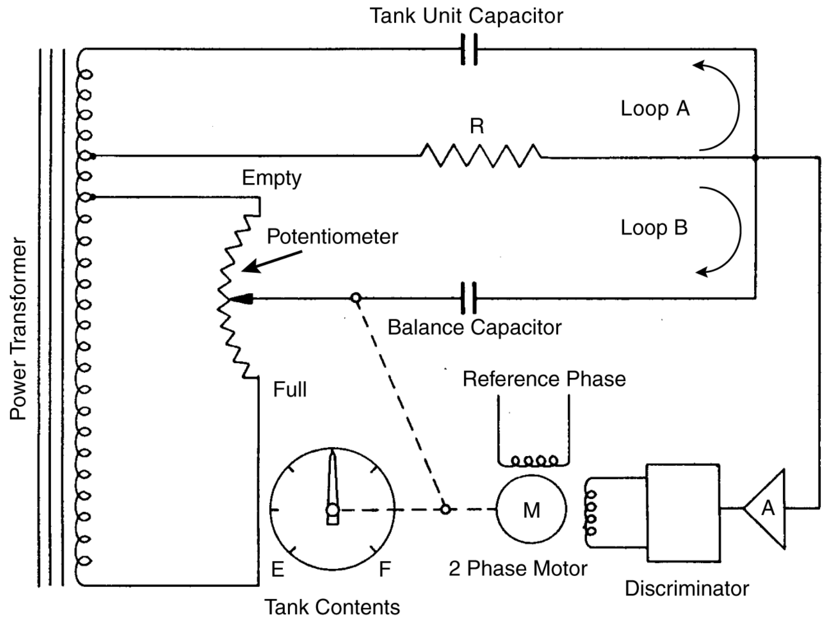
FIGURE 31-15
Capacitance Tank Contents Gauging Circuit



62. The loops are supplied with AC from a power transformer. The voltage to loop A, containing the tank capacitor unit, is maintained at a constant value and the current flow in the loop will depend upon the level of fuel in the tank, as we have already seen.

63. Loop B contains a reference capacitor of fixed value and is connected to the power transformer through a balance potentiometer, or variable resistance, so the voltage in loop B is variable. The balance potentiometer is adjusted by a two-phase motor, which also drives the pointer of the tank contents gauge. The two phase windings of the motor are known as the reference phase and the control phase, the reference phase is continuously energised from the power transformer whilst the control phase is only energised when there is an imbalance of current flow in the two loops, A and B. Both phases must be energised for the motor to rotate.

64. When the fuel level in the tank is steady the balance potentiometer will have taken up a position such that the current flow in loop B is the same as that in loop A. There is therefore no voltage difference across resistance R and the control phase of the two-phase motor is not energised.

65. Let us now assume that the level in the tank begins to fall. The capacitance of the tank unit falls and therefore the current flow in loop A falls below that in loop B. The imbalance between the two loops results in a voltage difference across the resistance R and a signal voltage is applied to the control phase of the motor, via an amplifier. Because the current in the balancing loop B is greater than that in the sensing loop A, a discriminator ensures that the control phase current is lagging behind that in the reference phase. This determines that the direction of rotation of the two-phase motor drives the gauge pointer and potentiometer wiper arm toward empty.

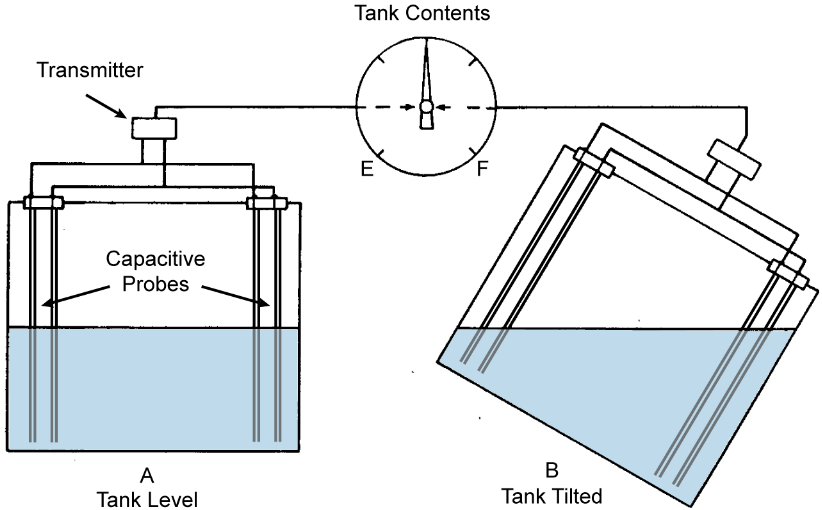
66. The movement of the potentiometer wiper arm causes the current flow in the balancing loop B to fall. When it is equal to the reduced current flow in the sensing loop A, the two loops are once again in balance. There is now no voltage difference across the resistance R, so signal voltage to the control phase falls to zero and the motor stops, with the gauge pointer indicating the new lower tank level.

67. The "empty" datum setting is manually set during calibration of the system by inserting a potentiometer between the tank unit and power transformer. Placing a potentiometer between the balance potentiometer and the power transformer allows the “full” datum to be set.

68. When the temperature of the fuel in the tank increases its volume increases and its density decreases. The decrease in density is almost exactly matched by a decrease in permittivity, or dielectric constant (K), which determines the capacitance of the fuel. The result is that, although this is a volumetric system of tank contents measurement (i.e. it measures tank level), any change in volume due to temperature change is compensated by the changed dielectric constant of the fuel and the gauge reading remains essentially unaffected. This is extremely useful because it means that the fuel tank gauges can be calibrated in terms of weight rather than volume, and the energy available in fuel is dependent upon weight, not volume.

69. In the large fuel tanks of a typical transport aircraft a number of tank capacitor units are inserted into each tank and connected in parallel with the balance loop of the tank contents indicator. By this means the gauge indication will remain constant regardless of the aircraft attitude or fuel surge within the tank. The principle is illustrated at Figure 31-16.

FIGURE 31-16
Capacitance Fuel Gauging and Aircraft Attitude



Torque Meter

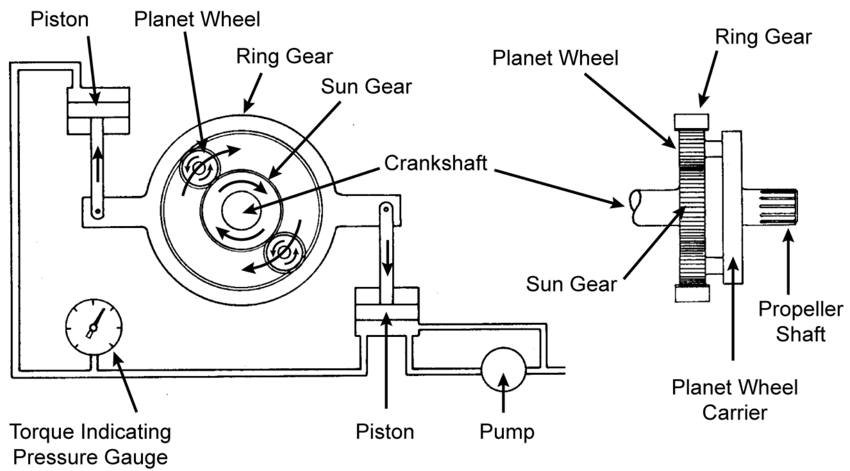
70. The power developed by an engine driving a propeller is measured and indicated by a torquemeter. Some large supercharged piston engines are fitted with these instruments, but in general they are more commonly associated with turbo-propeller powerplants. Torque is the turning moment transmitted through the propeller reduction gearing and is principally considered in terms of positive torque developed by the engine when it is driving the propeller. When the propeller is windmilling it develops negative torque and drives the engine.

71. The power output of the engine is the product of rpm and torque, so clearly if both are known the power can be calculated, either manually or by a simple computer. Since exceeding the torque limitations of the engine and its reduction gearing can lead to serious damage, it is essential that the pilot of a turbo-prop aircraft be provided with a torque meter.

72. Aircraft torque meters are usually either hydro-mechanical or electronic devices.

73. The hydro-mechanical type operate on the principle of pistons that are moved in their cylinders an amount proportional to the turning moment. The cylinders are connected to a closed hydraulic system and so the system pressure, due to the movement of the pistons, is directly proportional to torque and is used to operate a pressure gauge calibrated to indicate torque in ft. lb or N m. The principle is illustrated at Figure 31-17.

FIGURE 31-17
Torque Meter Priciple



74. Electronic torque meters are typically actuated by strain gauges placed in a suitable location in the drive train between engine and propeller. The small electrical output of the strain gauges is amplified and fed to the torque meter in the cockpit.

75. Torque indicators are sometimes colour-coded in a similar manner to other engine instruments, with red lines indicating permissible torque limits and a green arc to indicate the normal operating range. In some cases a yellow arc in the lower range cautions against the onset of negative torque.

Flight Hour Meter

76. Piston engine tachometers often incorporate an hour meter to record the total flight hours of the aircraft or engine. With these the hour meter is usually calibrated to record one hour of flight based upon the cruise rpm of the engine. For example, if the engine is designed to cruise at 2200 rpm, the meter will record one hour of flight for every 132,000 revolutions of the crankshaft (2200 x 60).

77. An alternative type of hour meter utilises an electric clock powered only when the battery master switch is closed, when the aircraft wheels leave the ground or when the engine is running. The "wheels off the ground" recorder is actuated by the aircraft squat switch and truly records flight time, the "engine running" recorder is usually activated by a pressure switch operated by the engine lubricating oil pressure.

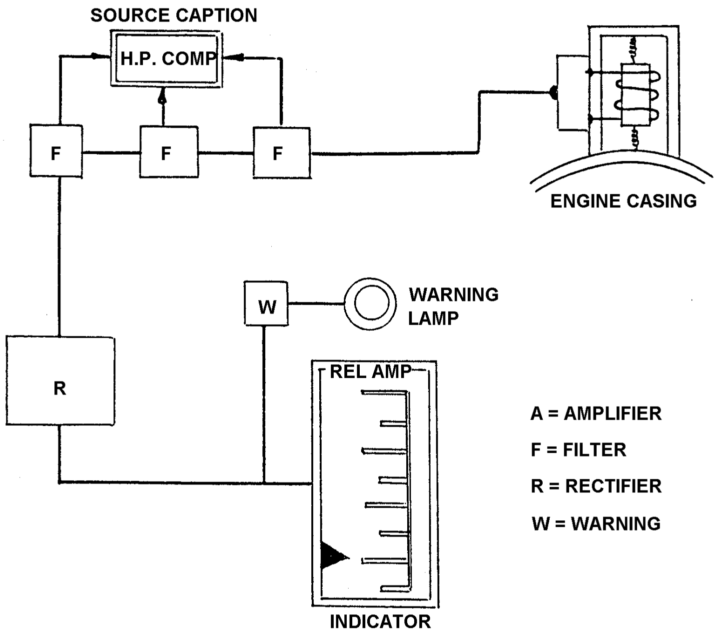
78. Indication of flight hours is usually displayed in digital form, with hours and tenths of hours recorded. The display is cumulative and is very similar to the odometer (total distance) display associated with a car speedometer.

Vibration Monitoring

79. Vibration in a gas turbine is relatively limited since the engine is finely balanced both statically and dynamically. Excessive levels of vibration are an indication of loss of dynamic balance, due perhaps to compressor or turbine blade damage. Accessory drive and reduction gearboxes produce vibration due to the meshing gearwheel teeth, but this is of predictable frequency and amplitude under normal operating conditions.

80. Consequently, comparison of measured vibration levels against reference criteria provides an extremely useful indication of engine health. An example of a simple vibration monitoring system is shown at Figure 31-18.

FIGURE 31-18
Vibration Monitoring System



81. The vibration pick-up consists of a spring-suspended permanent magnet surrounded by a coil of wire. The pick-up is mounted on the engine or gearbox casing in the appropriate plane (usually radially). With vibration the fixed coil vibrates linearly relative to the suspended magnet, which tends to remain relatively stationary due to inertia. The relative movement of the coil in a permanent magnetic field induces a small emf in the coil. This is amplified and fed through integrating filters and a rectifier to drive the indicator.

82. The indicator displays the amplitude of the sensed vibration, relative to a fixed datum. A warning light is incorporated in the system and this illuminates when the signal to the indicator reaches a pre-set level. The filters in the electrical circuit enable comparison between the frequency of the measured vibration and known frequency ranges for the various rotating components of the engine. This assists in isolating the source of vibration (HP compressor, fan, accessory gearbox, HP turbine and so on). This feature is particularly useful in multi-spool engines.

83. In more recently developed systems the electro-magnetic vibration pick-up is replaced by a piezoelectric accelerometer. This comprises a piezoelectric crystal clamped between a mass and the engine casing. Vibration of the casing causes the crystal to be alternately compressed and relaxed between mass and casing, generating a small charge in the crystal proportionate to engine vibration. The output signal from the piezoelectric crystal is amplified and fed to the cockpit display, which is calibrated in units of relative amplitude of vibration.

84. In turbo-fan engines the unit most susceptible to damage is the fan. Any loss of fan blade material due to foreign object ingestion damage upsets the dynamic balance of the fan and causes vibration. Consequently, fan vibration is always monitored on modern high bypass engines. The sensing pick-up is mounted on the fan casing such that the axis of the suspended magnet, or piezoelectric crystal, is coincident with the fan radius.

Self Assessed Exercise No. 6

QUESTIONS:

QUESTION 1.

Describe the function of the Overspeed Warning System.

QUESTION 2.

List the input data for the Overspeed Warning System.

QUESTION 3.

The regulatory margin between stall warning and stall is ……………………….. .

QUESTION 4.

Describe the function of a simple Pneumatic Stall Warning System.

QUESTION 5.

List the sequential stages of operation triggered by the angle of attack sensors of a Stall Protection System.

QUESTION 6.

What is the minimum period of flight time for which a Flight Data Recorder must be capable of retaining recorded data.

QUESTION 7.

List the JAR-OPS general requirements for Flight Data Recorder containers.

QUESTION 8.

Describe the fundamental difference between Type I Flight Data Recorders and Type II and IIA Flight Data Recorders.

QUESTION 9.

For aircraft with a maximum certified take-off mass greater than 5700 kg the minimum flight time during which a cockpit voice recorder shall be capable of retaining information is …………………

QUESTION 10.

Inputs to the cockpit voice recorder are typically from ………………… .

QUESTION 11.

Define the difference between Absolute pressure and Gauge pressure.

QUESTION 12.

The two principal types of direct reading pressure gauge are ………………….. .

QUESTION 13.

The two devices most commonly used for transmission of remotely-sensed pressure are …………… .

QUESTION 14.

Describe the coloured arcs, and their meaning, on piston engine oil pressure and MAP gauges.

QUESTION 15.

The device used to sense piston engine cylinder temperature is, typically ……………….. .

QUESTION 16.

Typical applications for temperature gauges in aircraft engine monitoring systems are ………………

QUESTION 17.

An air temperature sensor is said to have a recovery factor of 0.88. What does this mean.

QUESTION 18.

Describe the coloured arcs, and their meaning, on piston engine oil temperature gauges carburettor air temperature gauges.

QUESTION 19.

List the four types of tachometer in common usage, together with the types of engine with which they are usually associated.

QUESTION 20.

Describe the coloured arcs, and their meaning, usually found on RPM gauges.

QUESTION 21.

Briefly describe the salient features of independent and integrated fuel flow measurement systems.

QUESTION 22.

Where is the fuel flowmeter typically located in large turbine engine fuel systems and what information is typically displayed.

QUESTION 23.

There are two significant disadvantages to the resistive method of tank contents measurement. List these, and describe briefly how they are overcome in the capacitive method.

QUESTION 24.

An aircraft fuel tank has a capacity of 2,600 litres. If the capacitance measured in the tank is 100pF when the tank is empty, and 200pF when it's full, what will be the capacitance when it contains 1.950 litres of fuel.

QUESTION 25.

Describe the function of the two-phase motor in a capacitance tank contents gauging circuit.

QUESTION 26.

Turbo-propeller power plants are fitted with torquemeters, which are usually either hydro-mechanical or electronic. Briefly describe each type.

QUESTION 27.

The yellow arc on a torquemeter gauge indicates ....

QUESTION 28.

Flight Hour meters may be operated from the engine tachometer or by an electric clock. Describe the operating principles of each type.

QUESTION 29.

In which plane of rotation does a gas turbine engine vibration monitor usually sense vibration.

QUESTION 30.

The vibration monitor indicator is calibrated in units of ....

QUESTION 31.

Briefly describe the basic EICAS system?

QUESTION 32.

Describe the operation modes of a basic EICAS system?

QUESTION 33.

Describe the operating modes of a basic ECAM system?

QUESTION 34.

Which of the four ECAM modes are available in flight?

QUESTION 35.

Which of the EICAS modes are available in flight?

ANSWERS:

ANSWER 1.

The overspeed warning system provide a distinct aural warning any time the maximum operating speed of V_{MO}/M_{MO} is exceeded. The warning can be silenced only by reducing speed below V_{MO}/M_{MO}.

JAR Ref: 002 03 05 00

Page Ref: 022 28-1

ANSWER 2.

Input data for the mach/overspeed warning system is obtained from the Central Air Data Computer, the Flight Management Guidance Computer (for aircraft weight and C of G), and the position of slats and flaps.

JAR Ref: 022 03 05 00

Page Ref: 022 28-4

ANSWER 3.

The regulatory margin between stall warning and stall is for the warning to occur 5% or 5 knots CAS above CAS above the stall speed, whichever is the greater.

JAR Ref: 022 03 06 00

Page Ref: 022 28-4

ANSWER 4.

A simple pneumatic stall warning system comprises a slotted plate on the wing leading edge, connected to a plenum chamber and a reed-operated horn in the cockpit. At high angles of attack the slot moves above the stagnation point into an area of low pressure, causing air to be drawn through the reed/horn assembly such that a noise is emitted.

ANSWER 5.

The three stages of operation of a stall system are:

Auto ignition to prevent engine flame-out

Stall warning – stick shaker motors vibrate the control column to simulate the effects of buffet

Staff identification – horns sound, amber lights illuminate and the pneumatic stick pusher is activated, accompanied by red warning lights.

JAR Ref: 022 03 06 00

Page Ref: 022 29-2

ANSWER 6.

The Flight Data Recorder shall be capable of retaining the data recorded during at least the last 25 hours of its operation.

JAR Ref: 022 03 07 00

Page Ref: 022 29-2

ANSWER 7.

The flight Data Recorder container is to:

Be painted a distinctive orange or yellow colour

Carry reflective material to facilitate its location

Have a securely attached, automatically-activated underwater location device

Be shock, temperature and fire proofed

Be installed as far aft as possible, to reduce the probability of shock damage

Receive its electrical power from a bus that provides maximum reliability for recorder operation

Have an aural or visual means of pre-flight checking recorder operation

JAR Ref: 022 03 07 00

Page Ref: 022 29-5

ANSWER 8.

Type 1 Flight Data Recorders are capable of recording at least 32 parameters; Type II and IIA FDR's are capable of recording at least 15 parameters.

JAR Ref: 022 03 07 00

Page Ref: 022 29-6

ANSWER 9.

For aircraft with a maximum certified take-off mass greater than 5700 kg the cockpit voice recorder shall be capable of retaining information recorded during at least the last 2 hours of operation.

JAR Ref: 022 03 08 00

Page Ref: 022 30-1

ANSWER 10.

Input to the voice recorder is from the area microphone, from the Captain's and First Officer's received audio and from hot microphone inputs to the Audio Management Unit.

JAR Ref: 022 03 08 00

Page Ref: 022 30-6

ANSWER 11.

Absolute pressure is pressure above absolute zero, that is the difference between the pressure being measured and the pressure in a complete vacuum. Gauge pressure is the difference between the pressured being measured and ambient atmospheric pressure.

JAR Ref: 022 04 01 00

Page Ref: 022 31-1

ANSWER 12.

The two principal types of direct reading pressure gauge are the Bourdon tube type and the capsule operated type.

JAR Ref: 022 04 01 00

Page Ref: 022 31-2 & 3

ANSWER 13.

The two devices most commonly used for transmission of remotely-sensed pressure are the ratiometer and the synchronous, or Desynn remote reading system.

JAR Ref: 022 04 01 00

Page Ref: 022 31-4 & 5

ANSWER 14.

The coloured arcs on piston engine oil pressure gauges are RED for minimum and maximum pressures and GREEN for the normal operating range. The coloured arcs on a piston engine MAP gauge are BLUE for the cruise power region within which a lean mixture may be used and GREEN showing the range within which a rich mixture is required.

JAR Ref: 022 04 01 00

Page Ref: 022 31-6 & 7

ANSWER 15.

The device used to sense piston engine cylinder head temperature is, typically, a surface contact type thermocouple.

JAR Ref: 022 04 02 00

Page Ref: 022 31-7

ANSWER 16.

Typical applications for temperature gauges in aircraft piston engine monitoring systems are oil temperature, cylinder head temperature, carburettor air temperature and exhaust gas temperature. In gas turbine engines oil temperature and either turbine inlet or exhaust gas temperature gauges are essential requirements. Air inlet temperature is also usually indicated.

JAR Ref: 022 04 02 00

Page Ref: 022 31-6 & 7

ANSWER 17.

Temperature sensors are not usually capable of sensing, or recovering, the full temperature increase due to ram rise and the extent to which they do is known as the recovery factor. A sensor that measures SAT plus 88% of the ram rise would be said to have a recovery factor of 0.88.

JAR Ref: 022 04 02 00

Page Ref: 022 31-9

ANSWER 18.

The face of the oil temperature gauge is marked with a GREEN sector to indicate the safe normal operating temperature range; RED line indicate the minimum and maximum safe operating temperatures. The carburettor air temperature gauge is typically calibrated from –50°C to +150°C. There are three coloured arcs on the face of the gauge. YELLOW, from –10°C to +15°C, indicates the icing hazard range. GREEN, from +15°C to +40°C, is the normal operating range. RED, above +40C indicates detonation hazard.

JAR Ref: 022 04 02 00

Page Ref: 022 31-9

ANSWER 19.

Smaller aircraft piston engines commonly use mechanical tachometers, in which a drag cup is driven directly from the engine by means of a flexible drive. RPM measurement in larger piston engines is often by means of an electric tachometer, in which the output of an engine-driven alternator powers a synchronous motor and drag cup. Gas turbine engine RPM may be measured using an electric tachometer in which the output of an engine-driven alternator is converted to pulsed DC and drives the RPM gauge pointer through a torquemotor and potentiometer. Turbo-fan engines usually employ a magnetic tacho probe method of RPM measurement.

JAR Ref: 022 04 03 00

Page Ref: 022 31-10 & 11

ANSWER 20.

RPM gauge displays usually incorporate coloured arcs and limit markings, with a GREEN arc indicating the normal operating range of engine speeds and a RED line indicating maximum permissible RPM.

JAR Ref: 022 04 03 00

Page Ref: 022 31-12

ANSWER 21.

Independent fuel flow system use a flow sensor in the form of a spring-loaded vane in the fuel feed line, the deflection of which increases in proportion to the rate of flow and is converted into an electrical signal by means of a potentiometer. The output of the potentiometer operates a milliammeter, the face of which is calibrated in rate of fuel flow. Integrated flow measurement systems incorporate an integrator which computes fuel flow against time to display fuel used in addition to fuel flow.

JAR Ref: 022 04 04 00

Page Ref: 022 31-12

ANSWER 22.

In large turbine engine fuel systems the flowmeter is usually situated in the high pressure fuel line between the FCU and the burner nozzles. The associated gauge typically indicates both flow rate and fuel used and may incorporate a low fuel flow warning.

JAR Ref: 022 04 04 00

Page Ref: 022 31-12

ANSWER 23.

The disadvantages of the resistive method of tank contents measurement are the false indications that arise from: (a) fuel movement in the tank and (b) fuel expansion due to increased temperature. The capacitive system of tank contents measurement overcomes (a) by using a number of measuring probes and an ”averaging” system in the electrical supply to the tank contents gauge. Expansion due to temperature increase is offset by the decreased permittivity of the fuel, so that the dielectric constant remain essentially the same despite an increase in fuel level.

JAR Ref: 022 04 05 00

Page Ref: 022 31-18 & 19

ANSWER 24.

The fuel has twice the capacitance of air and, at 1,950 litres, the tank is three-quarters full. Thus the capacitance will be 100 + 75 = 175 pF.

ANSWER 25.

The two-phase motor in a capacitance tank contents gauging circuit drives both the balance potentiometer and the pointer of the tank contents gauge.

JAR Ref: 022 04 05 00

Page Ref: 022 31-16

ANSWER 26.

A hydro-mechanical torquemeter operates on the principle of a closed hydraulic system in which pistons are moved in cylinders an amount proportional to the turning moment created by the revolving propeller shaft . The resultant increase in hydraulic pressure operates a pressure gauge calibrated to indicate torque. An electronic torque measuring system uses strain gauges located in the drive train from engine to propeller. As the twisting moment due to torque increases the strain gauge is increasingly distorted and its electrical output increases; this is used to drive a torquemeter in the cockpit.

JAR Ref: 022 04 06 00

Page Ref: 022 31-18 & 19

ANSWER 27.

The yellow arc on a torquemeter gauge indicates the onset of negative torque.

JAR Ref: 022 04 06 00

Page Ref: 022 31-19

ANSWER 28.

Piston engine tachometer-operated flight hour meters are usually calibrated to record one hour of flight based upon the cruise RPM of the engine. Electric clock flight hour meters are activated in some cases by closing the battery master switch, in others only when the aircraft wheels are off the ground, and in yet others only when the engine is running.

JAR Ref: 022 04 07 00

Page Ref: 022 31-19

ANSWER 29.

The vibration sensor is a spring-suspended electro-magnet mounted so that it is sensitive to radial vibration.

JAR Ref: 022 04 08 00

Page Ref: 022 31-20

ANSWER 30.

The vibration monitor indicator is calibrated in units of Relative Amplitude, i.e. the amplitude of the sensed vibration relative to a fixed datum.

JAR Ref: 022 04 08 00

Page Ref: 022 31-21

ANSWER 31.

The system comprises a suite of sensors, two computers, two display units, an EICAS control panel, a display select panel, a maintenance panel, discrete caution and warning lights and a set of standby engine indicators. The sensors collect data from various points around the aircraft and transmit it to the computers. The computers process the sensor data and generate display information. Only one computer is necessary to drive the system. The upper display shows primary engine data and warnings and cautions. The lower display show engine secondary data or other system information depending on pilot selection. If one display fails the other will show a engine primary data and other information in a compressed format.

If both of the displays or both of the computers fail the standby engine indicators show essential engine related data.

ANSWER 32.

The basic EICAS system has three modes of operation. These are Operational, status, and maintenance. In operational mode the upper screen shows primary engine data whilst the lower screen remains blank during normal operations. In the event of an abnormal condition the upper screen displays an appropriate alert massage and the lower screen displays secondary information relating to the condition. In status mode the lower screen shows the status of selected systems. The maintenance mode is not available in flight and must be selected from the maintenance panel. In this mode the system shows maintenance related information.

ANSWER 33.

The ECAM system has four modes of operation. These are normal, advisory, failure and manual. In normal mode the system displays information relating to the current stage of flight. This information is shown in checklist form on the left screen and in diagrammatic form on the right screen. In advisory mode is selected automatically in the event of a non-emergency change in the status of any of the monitored systems. Failure mode takes precedence over all others and is selected automatically if any monitored system parameter exceeds pre-set limits. The left screen shows an appropriate warning and a list of the necessary corrective actions. The right screen shows the situation in diagrammatic form. In manual mode the crew can call up a range of selected diagrams, accompanied by system status messages.

ANSWER 34.

All four ECAM modes are available in flight.

ANSWER 35.

Only the operational and status modes are available in flight. The maintenance mode is not available on the flight deck and can be used only on the ground.

Electronic Displays

85. The conventional engine and system instrumentation display of a large multi-engine aircraft consisted of a large panel with a mass of gauges, usually of the circular scale variety, mounted in rows upon it. To assist the flight engineer in monitoring the display the gauges would be arranged in a logical manner with, for example, the oil pressure gauges for the four engines mounted side-by-side. Thus, the engineer was able to monitor the display at a glance by ensuring that each of the four gauge pointers was in the same position. When system indications were normal the actual gauge reading was unimportant, merely the normality was subconsciously noted.

86. Electronic "glass cockpit" engines and system displays follow this logic by only displaying essential information to the pilots. That is, engine information of fundamental importance such as rpm, thrust and turbine temperature (known as primary engine information) and any engine or system abnormalities. Thus, the information displayed to the pilots is limited to that which they need to know for the safe operation of the aircraft at any given time and the display size can be accordingly limited to small, easily monitored cathode ray tube (CRT) displays.

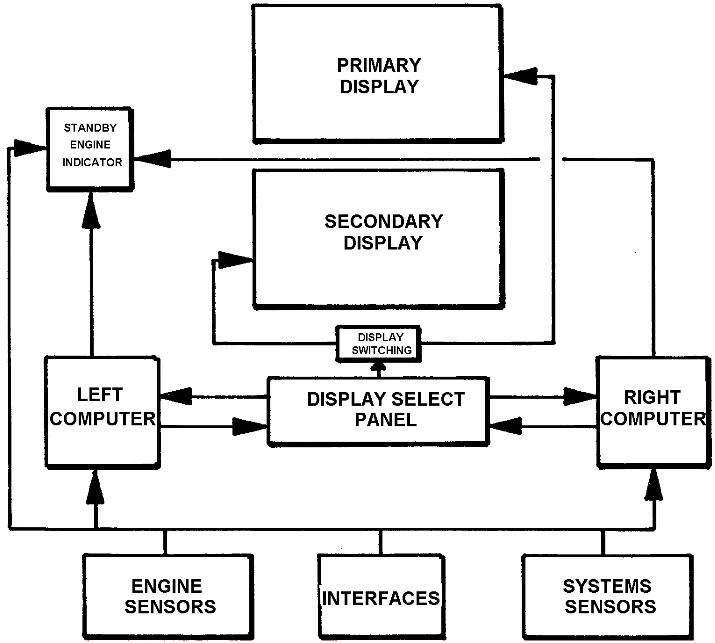
87. The most widely used electronic display system is the engine indicating and crew alerting system (EICAS), in which engine and system information is continuously recorded, and displayed as required. An intermediate electronic/conventional system is used in the early Airbus aircraft, known as electronic centralised aircraft monitoring (ECAM), in which the engine instrumentation display uses conventional gauging, but systems information is displayed electronically.

Engine Indicating and Crew Alerting Systems

88. The EICAS consists of two CRT displays, a display selection panel and two computers. The computers receive inputs from engine and system sensors and generate the analogue/digital displays of primary and secondary information, which appear on the CRT screens. Only one computer is operating at a time, the other being a standby. Figure 31-19 shows a block schematic diagram of an EICAS.

FIGURE 31-19

EICAS Schematic System



89. The two display units are mounted one above the other, the upper unit displaying primary engine information such as EPR, N_1 spool speed and EGT. Warning and caution messages also appear on this screen in the event of abnormal situations. The lower screen displays secondary engine operating parameters such as oil pressures, temperatures and quantities, fuel flow, vibration, N_2 spool speed, etc.

90. The displays are in colour, with colour coding as follows:

RED - Warning messages, operating limits

AMBER - Cautionary and advisory messages

WHITE - Analogue scales, digital read-outs

GREEN - Selected and target values (EPR, N_1, etc)

CYAN - Parameter titles

MAGENTA - In-flight engine starting and cross-feeding

91. The system has two functional display modes for flight crew use, plus a third mode for use by ground engineers only. These are known as operational, status and maintenance modes. The first two are selected from the Display Select Panel illustrated at Figure 31-20. Maintenance mode can only be selected on the maintenance control panel and is not available in flight.

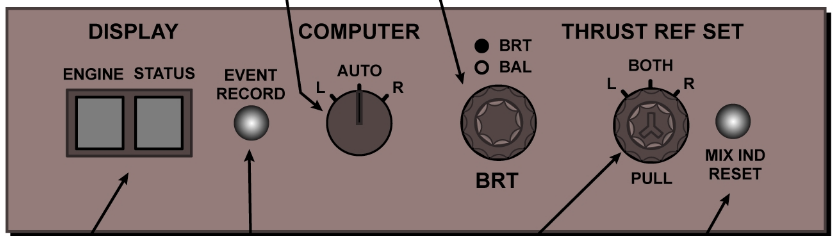
FIGURE 31-20
Display Select Panel

COMPUTER SELECTION

Auto-Left computer controls displays. Automatic switching to Right computer if Left fails

BALANCE AND BRIGHTNESS CONTROLS

Outer knob- Balance between screens
Inner knob- Display brightness



DISPLAY SELECT SWITCHES

Push to select required lower panel display. Push again to page-on

Push to record information for maintenance

THRUST REFERENCE SETTING

Outer knob sets EPR BUG for left, Right, or Both engines.
Inner knob sets thrust referance-
Push for Autoset
Pull for Manual

Push to reset Maximun values of,
N1, EGT, N2 and N3

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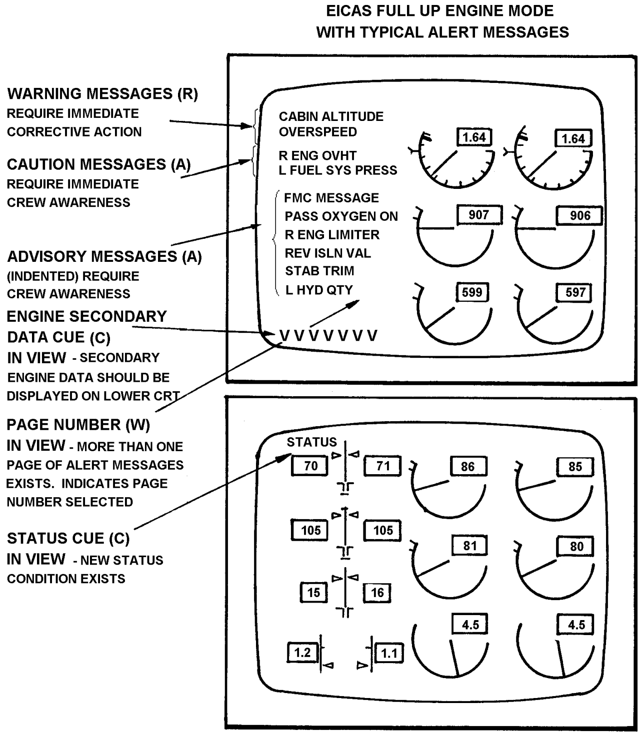
Operational Mode

92. The normal in-flight mode, this displays primary engine operating information on the upper screen. The lower screen will remain blank under normal operating conditions. In the event of abnormal conditions alert messages will be displayed on the upper screen with relevant secondary information displayed in analogue/digital form on the lower screen. An example of an operational mode display is illustrated at Figure 31-21.

FIGURE 31-21

Operational Mode Display



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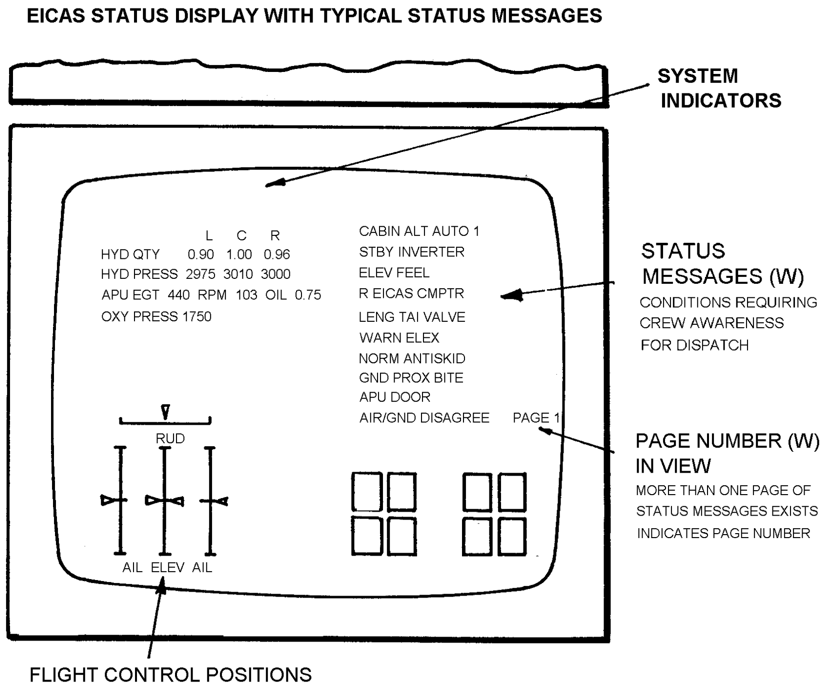
Status Mode. This display is primarily used on the ground to determine the aircraft readiness for dispatch. Status messages indicate system or equipment faults affecting dispatch and cross-refer to the aircraft's Minimum Equipment List. The status display appears on the lower (secondary) screen as illustrated at Figure 31-22. In this example system states appear at the upper left of the screen, flying control positions lower left and status messages indicating equipment faults on the right hand side of the screen.

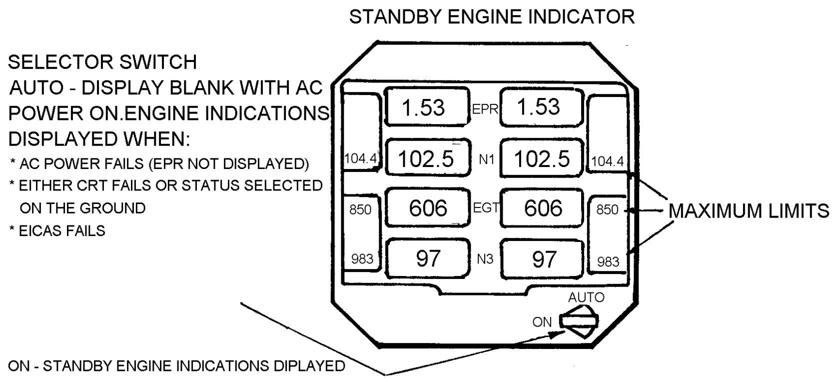
FIGURE 31-22
Status Mode Display



93. If one CRT fails, a compacted display of engine indications and crew alerting messages appears on the operable CRT screen, accompanied by an EICAS DISPLAY advisory message. Under these circumstances Status Mode is not available in flight.

94. If both EICAS computers and/or CRT's fail a standby engine indicator is activated. This provides a limited light-emitting diode (LED) display of primary engine operating parameters as illustrated at Figure 31-23.

FIGURE 31-23
Standby Engine Indicator

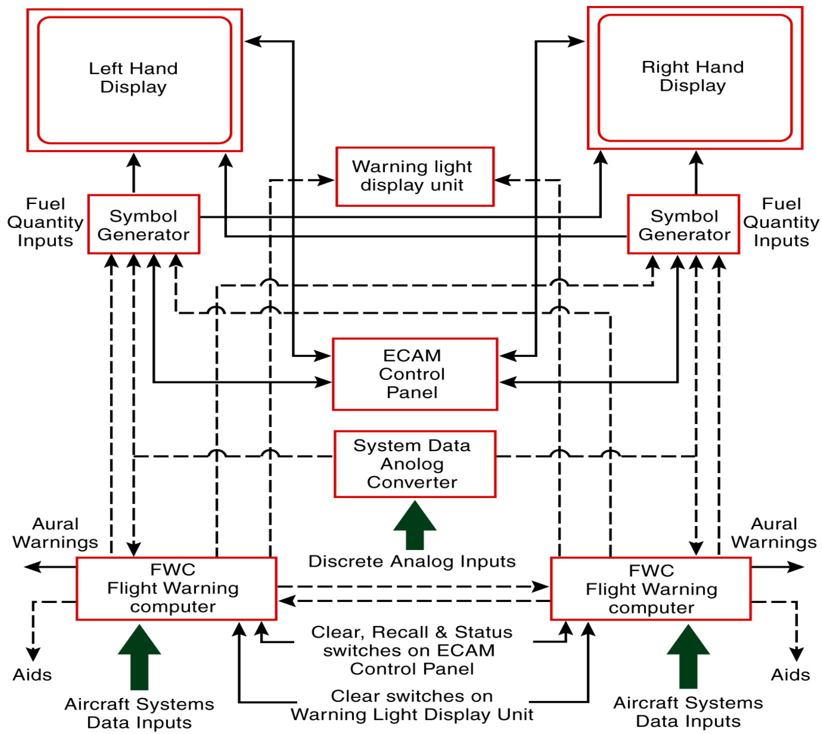


Electronic Centralised Aircraft Monitoring (ECAM)

95. This is a system that was introduced in the Airbus A310, where traditional engine monitoring instruments are retained and the electronic displays are of systems operation in checklist and schematic form. In the A320 and subsequent Airbus types the ECAM equipment has been developed and includes engine operating data on one of its two display units, in much the same manner as EICAS. The following description is of the A310 equipment.

96. The system comprises two CRT display units mounted side-by-side. The left-hand unit provides systems status information, warnings and corrective actions in a sequenced checklist layout. The right-hand unit provides graphic or schematic information associated with the messages on the left display. Figure 31-24 shows a block schematic diagram of the ECAM system.

FIGURE 31-24
ECAM Schematic



97. ECAM operates in one of four possible display modes. These are NORMAL, ADVISORY, FAILURE and MANUAL.

NORMAL mode. The normal mode of operation is flight related and the displays are appropriate to the existing phase of aircraft operation e.g. pre-flight, take-off, climb, cruise, descent, approach and post-landing.

98. The left screen displays a memo list of system states and the right screen displays similar information in diagrammatic form. For example, the left screen might list the aircraft systems (hydraulic, air conditioning, electrical etc) in use and the right screen might display a schematic of the main electrical distribution system with circuit breakers in their present positions.

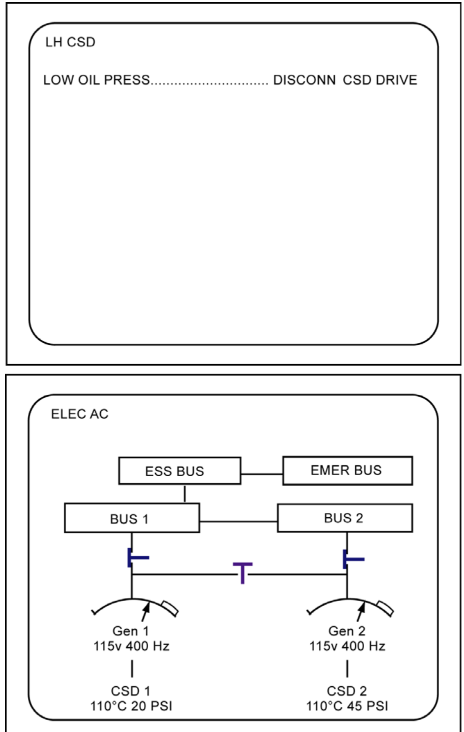
ADVISORY mode. An advisory mode may be automatically selected if a change of system status of a non-emergency nature occurs, such as switching from APU to main generators.

FAILURE mode. Failure mode takes precedence over all other modes and is automatically selected if any monitored parameter exceeds pre-determined limits. A warning message is displayed on the left-hand screen, accompanied by a list of the necessary corrective crew actions and an aural alert. If the warning applies to a single system it appears underlined (for example <u>L.H. CSDU</u>) followed by the nature of the problem (for example LOW OIL PRESSURE) and the corrective action required (in this case DISCONNECT CSD DRIVE).

99. In this example the ECAM failure mode has alerted the crew to the fact that the left engine constant speed drive unit is experiencing low oil pressure and advised the crew that, unless the CSDU is disconnected, it is likely to seize totally. In cases where other sub-systems may be affected the titles of these appear "boxed" (in this example LH ELECTRICAL will be boxed, since this CSDU is driving the left engine AC generator). Meanwhile the right-hand screen will display a schematic of the appropriate system with, given the above example, CSDU oil pressures and temperatures displayed.

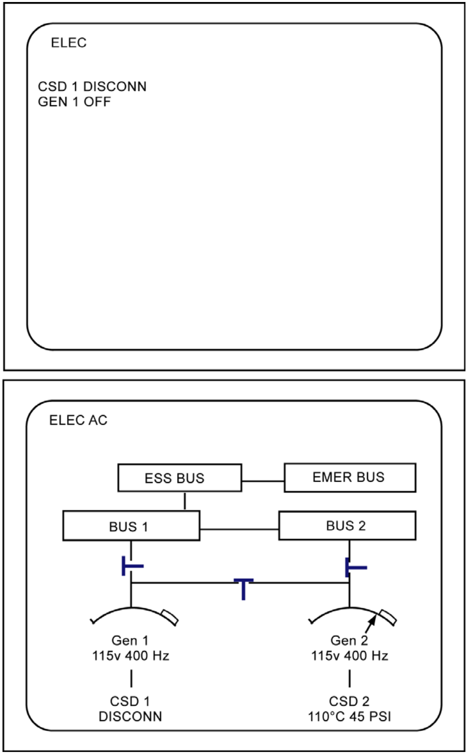
100. The above example of an ECAM display in failure mode is illustrated at Figure 31-25.

FIGURE 31-25
Failure Mode -
Alert Display



101. When the pilots have taken the necessary corrective action, the left and right displays will change to show the current system status, as illustrated at Figure 31-26.

FIGURE 31-26
Failure Mode - Display after Corrective Action



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MANUAL mode. The manual mode is the fourth mode of operation of ECAM and allows the crew to call up a series of diagrams of the aircraft's systems, accompanied by system status messages.

FIGURE 234

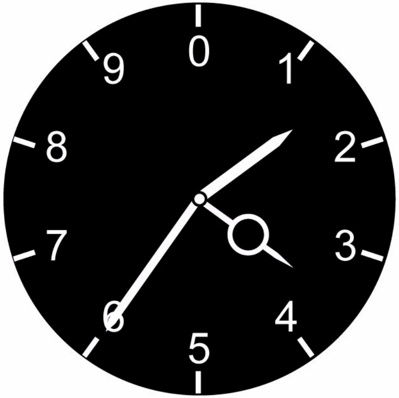


FIGURE 235

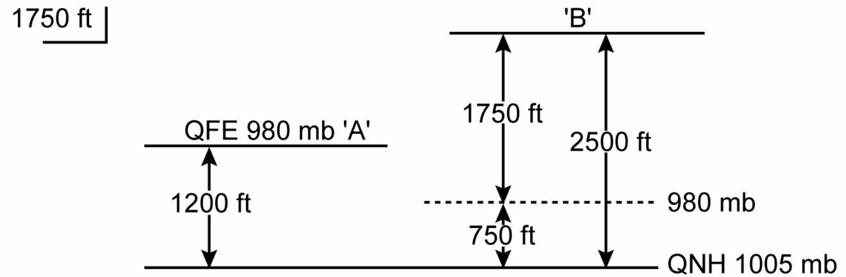


FIGURE 236

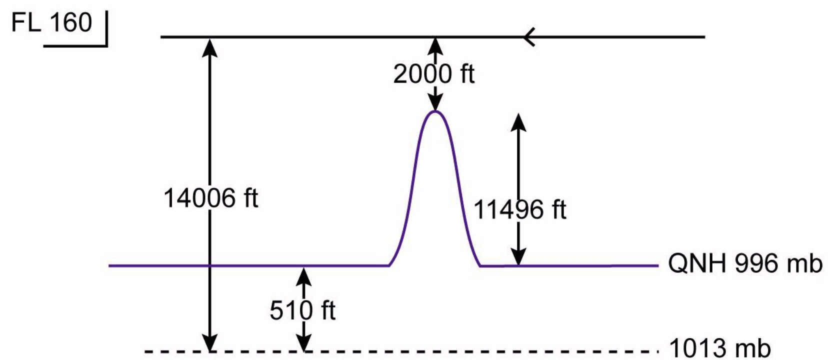


FIGURE 237

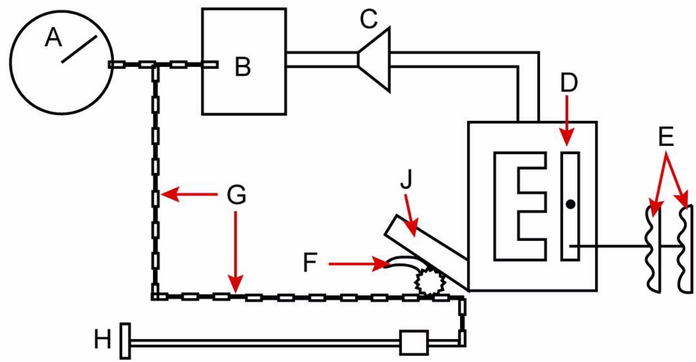


FIGURE 239

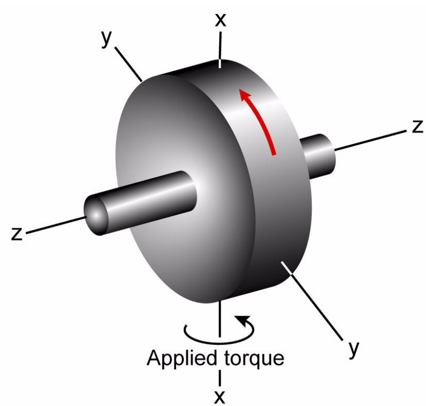


FIGURE 240

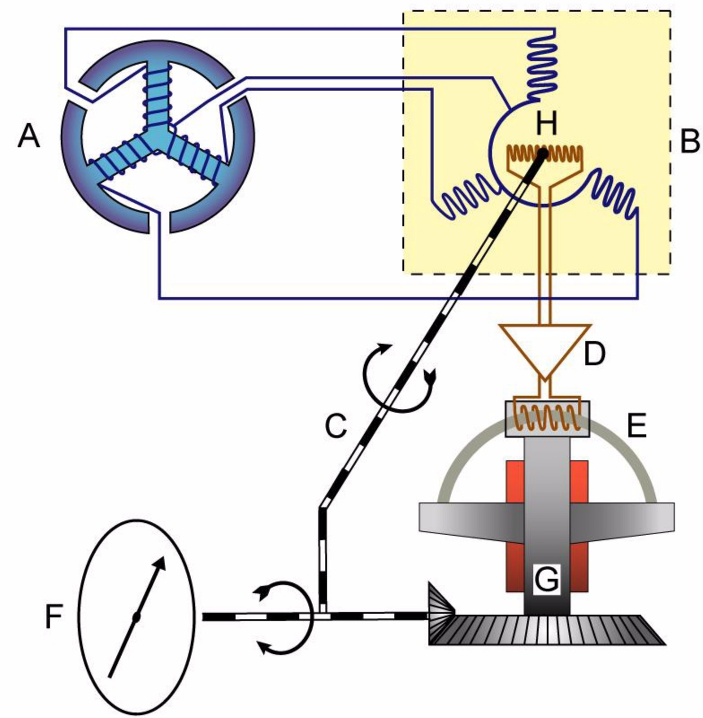
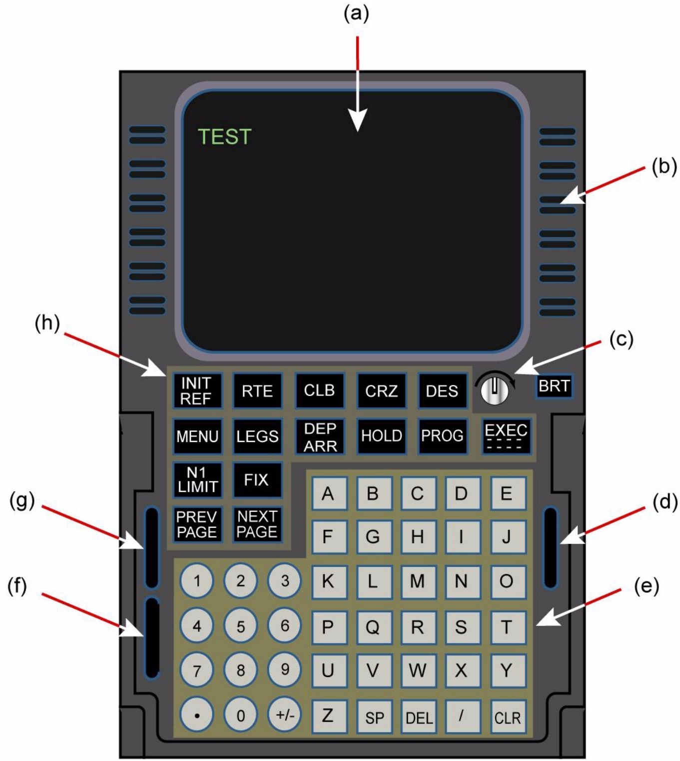


FIGURE 241



(a)
(b)
(c)
(d)
(e)
(f)
(g)
(h)
TEST
INIT REF
RTE
CLB
CRZ
DES
BRT
MENU
LEGS
DEP ARR
HOLD
PROG
EXEC
N1 LIMIT
FIX
PREV PAGE
NEXT PAGE
A B C D E
F G H I J
K L M N O
P Q R S T
U V W X Y
Z SP DEL / CLR
1 2 3
4 5 6
7 8 9
• 0 +/-

FIGURE 242

FIGURE 243

FIGURE 244

FIGURE 351

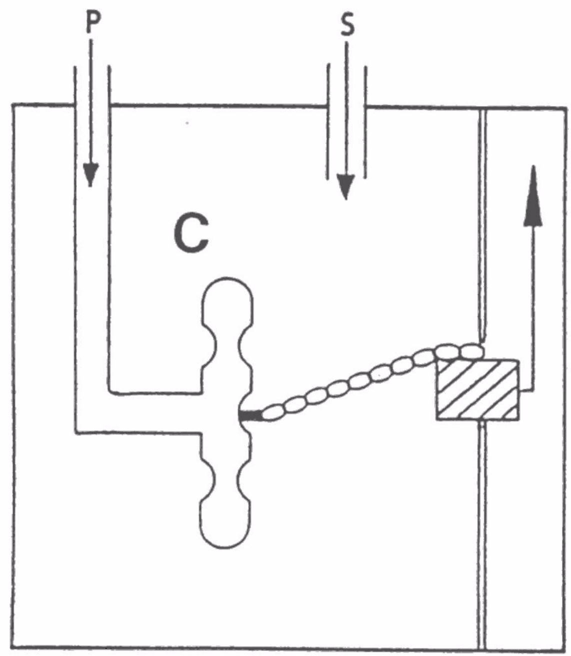


FIGURE 352