



# INSTRUMENTATION

## ATPL GROUND TRAINING SERIES

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# *Introduction*

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Introduction

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**1**

Characteristics and General Definitions

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## Units

<b>Distance:</b>	Metres, Kilometres	
	Nautical miles	(1 NM = 6080 ft or 1852 m)
	Feet	
	Statute miles	(1 mi = 5280 ft or 1609 m)
<b>Time:</b>	Hours	
	Minutes	
	Seconds	
<b>Speed:</b>	Knots	
	Miles per hour	
	Kilometres per hour	
<b>Mass:</b>	Kilogram	
	Pound	
<b>Pressure:</b>	Pascals - hectopascal	(1 Atmos = 1013.25 hPa)
	Bar - millibar	(1 Atmos = 1013.25 mb)
	Inches of mercury	(1 Atmos = 29.92 inHg)
	Millimetres of mercury	(1 Atmos = 760 mmHg)
	Pounds per square inch	(1 Atmos = 14.7 psi)
<b>Temperature:</b>	Celcius	(MSL = +15°C)
	Kelvin (Absolute)	(MSL = +288K, 288°A)
	Fahrenheit	(MSL = +59°F)
<b>Volume:</b>	Litres	
	Pints	
	Gallons	Imperial and US (1 Imp Gallon = 1.2 US Gallon)
<b>Angles:</b>	Degrees	
	Minutes	
	Seconds	
<b>Position:</b>	Latitude	
	Longitude	(Shown in Deg Min Sec)



## Introduction

Pilots receive information about the state of their aircraft and its speed, altitude, position and attitude through instruments and displays. These can vary from the simplest of dials and pointers to modern electronic displays (the so-called 'glass cockpits'), depending on the vintage and the complexity of the aircraft, and a simple dial can seem very different in appearance and sophistication from a modern cathode ray tube or liquid crystal screen. However, certain problems of **range, resolution, accuracy and reliability** are general characteristics of all instrumentation systems.

## Measuring Range Versus Accuracy

It is often necessary to show a large operating range, yet still indicate with accuracy over the whole range. For instance, an airliner might be limited to a maximum permitted airspeed of 350 knots, so perhaps the instrument would be designed to display up to, say, 380 or 400 knots. However, certain speeds are critical to flight safety and need to be read with extreme accuracy – ideally to the nearest knot. If we put the whole range on a single revolution of the instrument the division representing one knot will very small and will be difficult to read accurately.

### Circular Scale (Linear)

A simple indicator showing the change of value of the parameter to be measured over a range of 0 to 30 units is shown in [Figure 1.1 \(Linear\)](#). The accuracy with which these values need to be measured will govern the spacing of the graduation.

### Circular Scale (Non-linear)

Some instruments are required to show changes of parameters more accurately at certain parts of the scale. The example in [Figure 1.2 \(Non-linear\)](#) shows a rate of climb indicator where low rates of climb/descent are more easily read than high rates. This is a **logarithmic** scale.

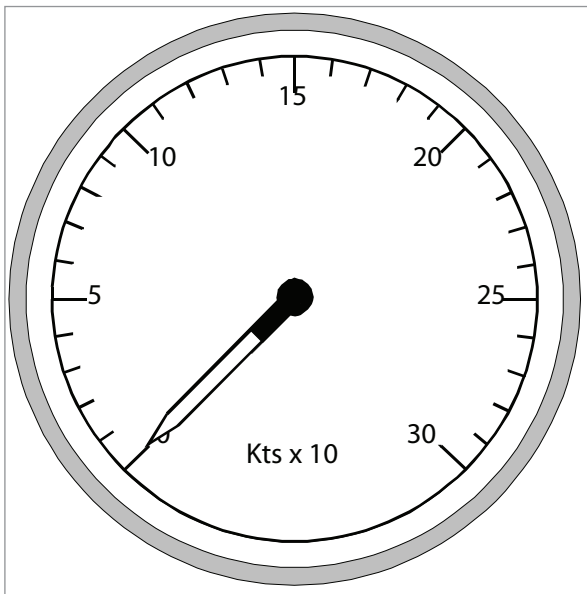


Figure 1.1 Circular scale - linear

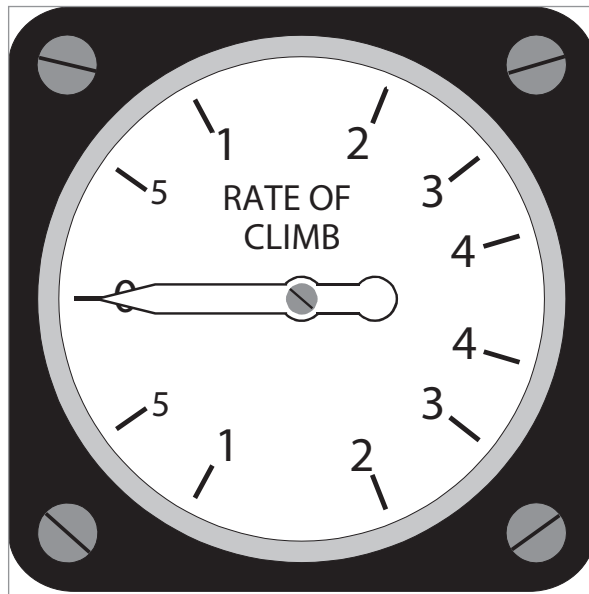


Figure 1.2 Circular scale - non-linear

## High Range Long Scale Displays

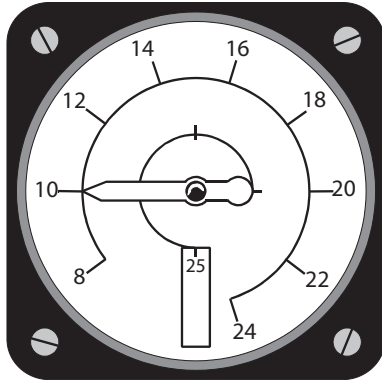


Figure 1.3 Single pointer airspeed indicator

Where the instrument needs to show changes over a high range of values and these changes need to be read with a fair degree of accuracy,  $360^\circ$  of movement of the pointer may not be sufficient. The pointer may make more than one revolution to cover the required range, as on the airspeed indicator shown in [Figure 1.3](#), though this type of display may lead to some confusion.

[Figure 1.3](#) indicates an airspeed of 300 kt.

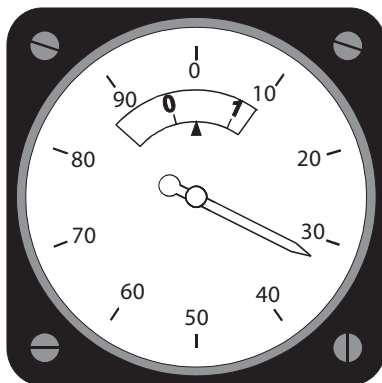


Figure 1.4 An airspeed indicator

Another solution is to have a pointer moving over a fixed scale (tens of knots) with a moving scale indicating larger units (hundreds of knots).

[Figure 1.4](#): The small pointer and dial, inset in the top of the instrument, indicates 100s. The long pointer and main dial indicates 10s.

This indicator shows an airspeed of 33 kt.

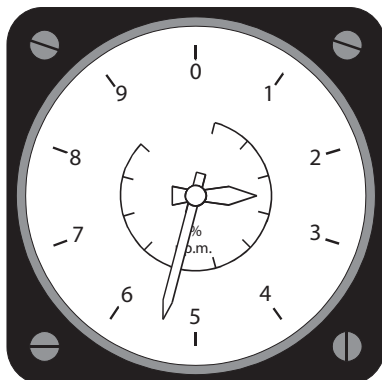


Figure 1.5 A revolution counter

A less confusing display uses two concentric pointers moving over two separate scales, as shown on the revolution counter.

[Figure 1.5](#): The small needle, and inner scale, reads 10s of units. The large needle, and outer scale, reads units.

This indicator shows  $25\frac{1}{2}$  % rpm for a turbine engine.



Figure 1.6 A three pointer airspeed indicator

A further solution, shown in *Figure 1.6*, is to display information in a similar fashion to a clock, with pointers showing hours, minutes and seconds. This system is used on many altimeters. The long pointer will cover 1000 feet in one revolution, so each division of the scale represents 100 feet. The middle pointer will cover 10 000 feet per revolution, each division marking 1000 feet and the smallest pointer will cover 100 000 feet, each division representing 10 000 feet.

*Figure 1.6* indicates a height of 25 950 ft.

## Ergonomy

**Ergonomy** (also known as **human engineering**) is the science of relationships between people and machines. An ergonomic device interacts smoothly with peoples' bodies and actions. In an aviation context this can mean designing the shape and position of controls, levers and knobs so that they are easily controlled and unlikely to lead to an incorrect selection. For instruments or instrument systems it means designing instruments that are unlikely to be misread and locating them in a layout that facilitates easy and correct interpretation of the information displayed. Standard layouts came to be adopted.

### Location

The 'flying' instruments which covered the handling of the aircraft were arranged in the layout of the 'basic six'. Other instruments tended to be scattered around the cockpit in positions most convenient to the designer and manufacturer, seldom to suit the needs of the pilot.

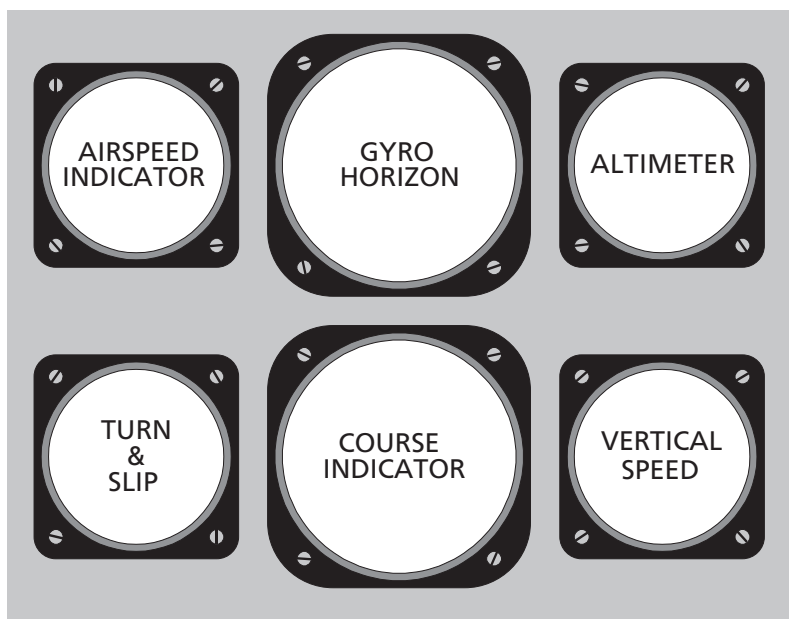


Figure 1.7 The 'basic six' instrument layout

Since the introduction of the 'basic six' developments in aircraft instruments and operations led to the introduction of the **'basic T'**.

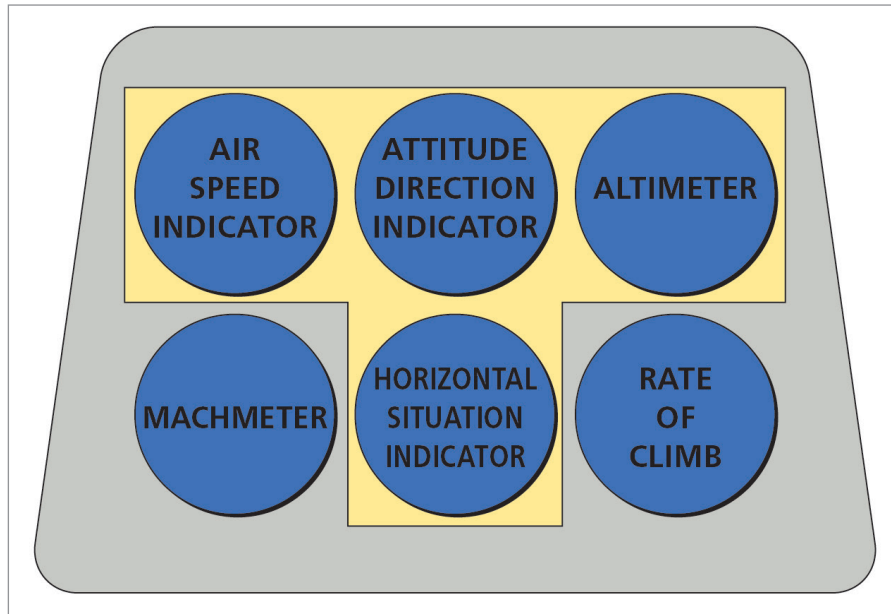
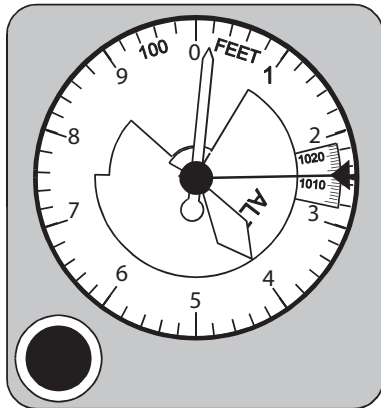


Figure 1.8 The 'Basic T' instrument layout

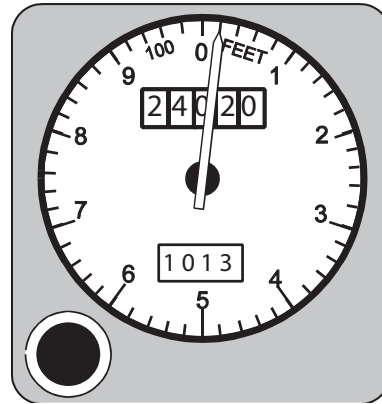


Figure 1.9 The basic instrument panel of a Piper PA 34 Seneca

Presentations can be in **analogue** or **digital** form. Analogue is, typically, a pointer on a dial whereas digital is a row of numbers. Look at the 2 types of altimeter display at [Figure 1.10](#) and [Figure 1.11](#).



*Figure 1.10 An analogue altimeter*



*Figure 1.11 A combined digital analogue altimeter*

With the 3-pointer analogue system, the altitude information (24 020 feet) is harder to absorb at a single glance than with the digital display. The digital numbers are much easier to read. However, we note that one pointer still remains in the design of the mainly-digital presentation. This is because the human eye and brain cannot easily interpret rate information from moving numbers. Whilst the altimeter is primarily designed to show altitude, with a separate instrument (the vertical speed indicator) to show rate of change of altitude, pilots pick up a lot of secondary information about vertical rate from the angular rate of the altimeter pointer.

## Electronic Displays

Historically, instruments have been located on **instrument panels**, though this is now changing with modern electronic displays. Compare the traditional layout ([Figure 1.12](#)) with the modern electronic (glass) layout in [Figure 1.13](#) & [Figure 1.14](#) below.



*Figure 1.12 Traditional Analogue Displays*



Figure 1.13 Modern electronic (glass) displays - Boeing B787

With modern electronic systems, although the displays have to be on the flight deck where the crew can see and operate them, the computing units and power units are located remotely in some other part of the aircraft, usually in a separate compartment called the **Avionics Bay** or the **Electrics and Electronics (E&E) Bay**.



Figure 1.14 Electronic instrument panel - light aircraft

## Readability

A readable instrument should be designed with an **eye reference point** in mind. This is the anticipated position that the pilot's eye will occupy when viewing the instrument under normal conditions. If the instrument has a design where there is a reference mark or index with a scale behind it, it is important that the eye, the index and the scale are all in line. Otherwise, there is an error known as **parallax**, which is simply caused by viewing the instrument from slightly to one side instead of from the front.

## Coloured Arcs

A standardized system of colour coding for operating ranges for conventional non-electronic instrument is widely used. These are:

- Green:** Normal operating range
- Yellow or Amber:** Cautionary range
- Red:** Warning, or unsafe operating range

Figure 1.15 is an example of the gauges, fitted to a Piper Warrior, showing the colour coding.



Figure 1.15 Red, yellow and green operating ranges

For more complex instruments, usually electronic displays, CS-25 sets out the following colour standardization.

- White:** Present status
- Blue:** Temporary situation
- Green:** Normal operating range
- Yellow or Amber:** Cautionary range
- Red:** Warning, or unsafe operating range





## Chapter

# 2

## Pitot and Static Sources

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## Introduction

An aircraft at rest on the ground in still air is subject to normal atmospheric pressure, which bears equally on all parts of the aircraft. This ambient pressure is known as static pressure. An aircraft in flight, whilst still subject to the static pressure at its flight level, experiences an additional pressure on the leading edges due to the resistance of the air to the aircraft's movement.

This additional pressure is **dynamic** pressure, and its value depends on the speed of the aircraft through the air and on the density of the air. The leading edges, therefore, encounter a **total** pressure consisting of static plus dynamic pressures. This total pressure is also known as pitot pressure.

It is not possible to isolate dynamic pressure by direct measurement because it cannot be separated from the associated static pressure. The instruments which require dynamic pressure therefore measure total (pitot) pressure and also static pressure and then subtract static from pitot within the instrument to derive dynamic pressure.

$$\text{PITOT} = \text{STATIC} + \text{DYNAMIC}$$

or rearranging:

$$\text{DYNAMIC} = \text{PITOT} - \text{STATIC}$$

The following instruments require inputs of static and pitot:

Static Only	Pitot and Static
Altimeter	Airspeed Indicator (ASI)
Vertical Speed Indicator (VSI)	Machmeter

Inside an aircraft, pressure and temperature are seldom the same as outside the aircraft so pitot and static pressures must be sensed by devices mounted on the **outside** of the aircraft.

## Pitot/Static Heads

An open-ended tube parallel to the longitudinal axis of the aircraft is used to sense the total pressure (static plus dynamic). This device is a 'pitot tube' mounted in a 'pitot head'.

The open end of the tube faces into the moving airstream, the other end leading to the airspeed capsules in the ASI and Machmeter.

The moving airstream is thus brought to rest in the tube, so generating the extra (dynamic) pressure which together with the static pressure already in the tube provides the required total (pitot) pressure.

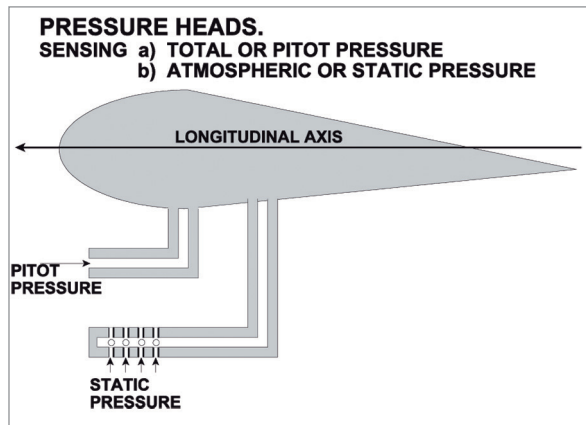


Figure 2.1

A 'static head' consists of a tube with its forward end sealed but with holes or slots cut in the sides. These slots do not face into the airflow and therefore, in theory, they sense only the static pressure. In fact, of course, there will be a suction effect and the sensed static pressure will be slightly lower when the aircraft is moving. This pressure supplies the static 'line' to the pressure instruments. A pressure sensing system consisting of separate pitot and static heads is shown in [Figure 2.1](#).

The static and pitot sources may be combined in one 'pressure head', the static tube surrounding the pitot tube, with separate

pressure lines leading to the pressure instruments. [Figure 2.2](#) illustrates an example of this type.

## Requirements of a Pitot Tube

The pitot tube must be positioned **outside the boundary layer**, so it usually consists of a head on a strut if mounted on the side of the fuselage, or it can be a tube placed on the nose, ahead of the fuselage. The opening must be designed to be **parallel to the airflow** in the **normal flight attitude**.

The air can either be brought to rest in the pitot probe against a **stagnation wall**, which is simply a pressure-measuring flat surface, and then transmitted up the pitot pipelines to the ASI and Machmeter. Alternatively, the pitot pressure can be passed directly up into the pitot pipelines to the instruments, which is more usual in elementary aircraft.

Measurement of dynamic pressure is **essential** to safe flight. At too low a speed, the aircraft will stall. At too high a speed it will be overstressed. The ASI is critical to flight safety and so it cannot be allowed to block because of ice. An electric anti-icing **heater** coil is usually incorporated. Any errors due to the heating effect may be reduced by design and calibration. However, if water is drained, ice should not be able to form. **Drain holes** are therefore provided. These will, of course, cause some loss of pressure, but this can be suitably calibrated and allowed for in the design.

## Requirements of a Static Source

The static source, whether a simple hole or a combined probe, should have its opening at **right angles** to the airflow, so that only static pressure is sensed, with **no component of dynamic pressure**. Some static sensors, especially those in the combined pitot/static probe may be fitted with electric heating.

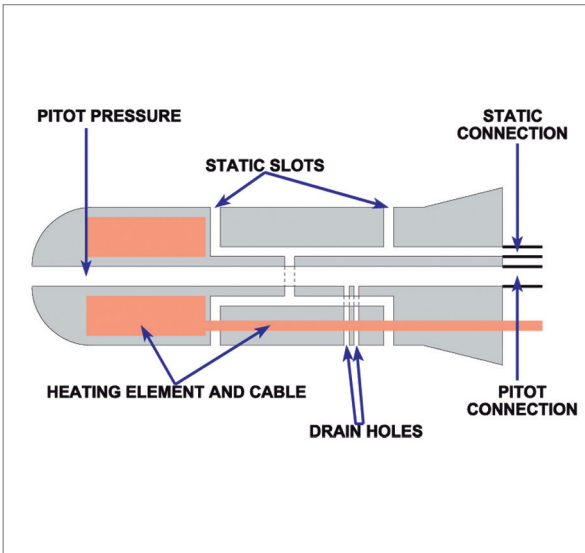


Figure 2.2 A combined pitot/static pressure head

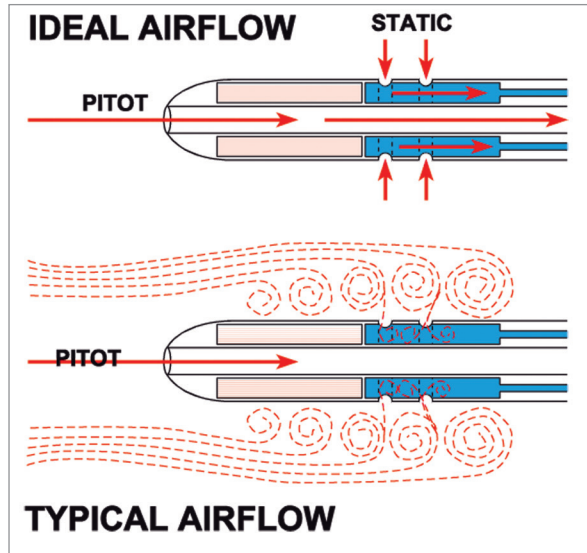


Figure 2.3 How turbulence affects the value of static pressure

## Position Error

As previously stated, if the aircraft has forward motion, the static pressure sensed will be slightly lower than if the aircraft is stationary, due to suction. As the speed increases, due to turbulent airflow in the region of the pitot/static heads, this error becomes greater. Turbulence will also affect the pitot reading. The error involved is called **Position Error** (or alternatively 'pressure' error). At large angles of attack (which are usually associated with lower airspeeds) the pressure head is inclined at an angle to the airstream so that position error is usually greater. Flight manuals may list different values of position error for different flap settings.

Position error depends mainly on the positioning of the pressure head, the airspeed, and the aircraft attitude. Turbulence produced in the airstream by the pressure head itself affects the value of static pressure sensed rather than the pitot pressure because the turbulence is downstream of the pitot opening. This is shown diagrammatically in [Figure 2.3](#).

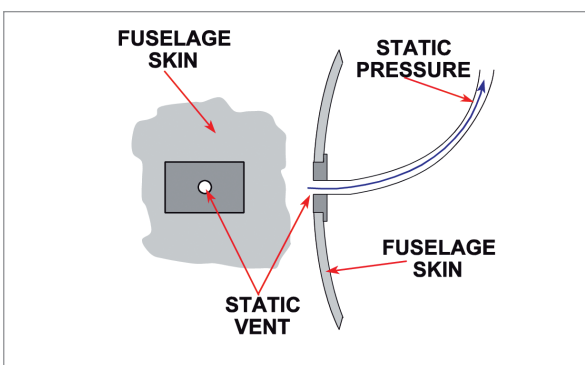


Figure 2.4 A static vent

Because of this, the **Static Vent**, was introduced as a source of static pressure instead of the static head, pitot pressure then being sensed by a simple pitot head. A static vent separated from the pitot head is shown in [Figure 2.4](#).

There is usually some place on the airframe, usually on the side of the fuselage, where true (or nearly true) static pressure is obtained over the whole speed range of the aircraft. A flat metal plate is fitted at this position, the

static line from the pressure instruments terminating at a small circular hole - the static vent - in this plate. A similar vent is positioned on the opposite side of the fuselage and **the two are interconnected** for transmission of static pressure to the instruments so that **errors produced by yawing are largely eliminated**.

### Advantages of the Static Vent

- The airflow in the region of the vents is less turbulent and the static pressure measured is more accurate.
- Errors produced when side-slipping or yawing are reduced.
- Duplication of static vents either side of the fuselage reduces errors due to side-slip or yawing (cross balancing of static vents).

### High Speed Probes

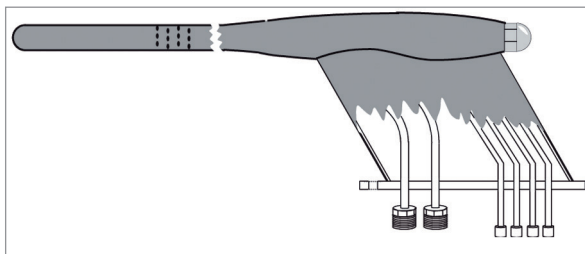


Figure 2.5 A high speed pitot/static probe

The shock waves associated with flight at high mach numbers can produce significant errors in pressure sensed by a static vent. Modern high speed aircraft may accordingly be fitted with a more sophisticated combined pitot/static pressure head in order to keep position error within acceptable limits. The choice of location for a probe, or vent, is dependent upon the aerodynamics of the aircraft. Typical locations are: ahead of a wing tip, under a

wing, ahead of the vertical stabilizer tip, at the side of the fuselage nose section, and ahead of the fuselage nose section.

### Manoeuvre-induced Error

Manoeuvre-induced errors are caused by short-term fluctuations of pressure at the static vents and delays in the associated pipelines transmitting pressure changes to the instruments.

Even servo altimeters and air data computer systems suffer from this type of error as they utilize the same static vents as the simple pressure instruments. Change in angle of attack, and turbulence due to lowering (or raising ) flaps and landing gear are the prime causes of the error-producing changes in airflow over the static vents.

Most commonly, manoeuvre-induced error appears as a marked lag in pressure instrument indications.

The errors are usually **more significant during changes of pitch attitude** than during yawing or rolling movements so that the worst effects are at the start of the climb or descent and on levelling out.

**Overshooting** (referred to as **go-around**) and **flight in rough air** are particularly vulnerable.

The errors are unpredictable both in size and in sense so that pressure instruments cannot be relied upon to indicate accurate instantaneous values or accurate rates of change.

This particularly applies to vertical speed indicators. In-flight manoeuvres should therefore be carried out using gyroscopic instruments as the primary reference. A manoeuvre-induced error may be present for some time after movement of the control surfaces has ceased, values of three seconds at low altitude increasing to 10 seconds at 30 000 feet (longer for VSIs) being quite common.

## Full Pitot/Static System

The transmission from the probes to the instruments usually takes the form of **pipelines** in elementary and older systems, but in modern aircraft it is usually carried by **electrical wires**. Both pitot and static pipelines will have in-built **water traps**. Modern systems often have electronic pressure **transducers** at the pitot and static sources with **built-in error correction**. The measurement is analogue but **analogue/digital interface units (A/D IFUs)** convert it into digital form for onward use. This may be to stand-alone pressure instruments but, more usually in modern aircraft, it is to a device known as the **air data computer**. If it is in digital form, **data digital buses** may be used for transmission instead of wires.

Pitot systems are not usually cross-coupled. The left pitot source goes to the left pitot instruments and the right pitot source goes to the right pitot instruments. Modern flight instrumentation systems may **compare** the outputs and give a **warning** in the event of a discrepancy in excess of, say, 5 knots, but they do not cross-feed pitot pressure.

Static systems, however, are almost invariably cross-coupled. Both the left and the right static system will have its own static vent on each side of the fuselage. Each static system will take an input from its own left and right static vents and the mixed static pressure for that system will pass up to the static instruments for that side. Large aircraft will also have a standby pair of static vents (left and right) for the standby ASI and altimeter. (It is not normal to have a standby VSI or Machmeter). Cross-coupling the left and right vents on these systems reduces error caused by yawing and side-slipping.

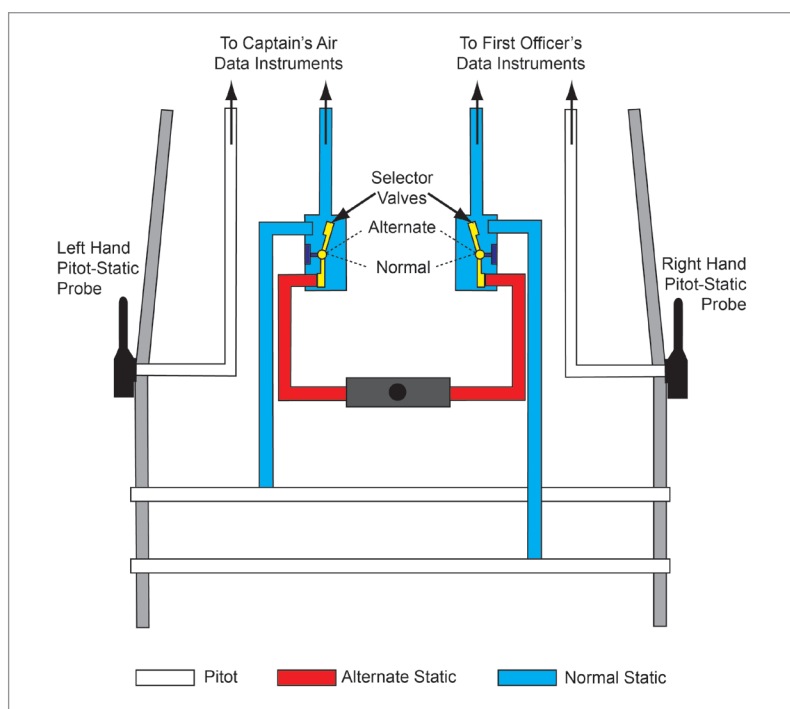


Figure 2.6 Emergency static source

For light aircraft, an **alternate static source** is normally provided in the event of the static head/vents becoming blocked. The selector switch will be in the cabin.

The alternate static source may be to the outside of the aircraft or from inside the cabin (in unpressurized aircraft only).

The alternate static pressure will be **less accurate** than the primary (blocked) static vent/head, since that would have been in the optimum position. The alternate static pressure sensed is likely to be **lower** than ambient due to aerodynamic suction. This is generally true whether the alternate source is external or from within an unpressurized cabin. Instruments using static are calibrated to take account of the pressure (position) error at their normal supply so, when alternate (standby) pressure systems are used, correction values for the instruments concerned may be found in the Operating Data Manual for the aircraft.

### Pitot Covers and Static Vent Plugs

Pitot and static openings are highly sensitive. Any dirt, dust and sand in them can distort their measurements, often drastically. Also, as they tend to be warmer at night than the rest of the aircraft, insects can be attracted to them as resting places. It is therefore necessary to cover them when the aircraft is not in use.

Pitot covers are canvas or rubber closed tubes which fit over the outside of the pitot probe. Static plugs are usually made of rubber and are shaped like small corks. It is clearly essential that they not be left in place in flight, therefore removing them is included in the preflight external checks. They normally have a very conspicuous streamer or ribbon up to a metre long attached to them to ensure that they are not overlooked.

### Pitot and Static Heaters

All aircraft have pitot heaters and, on combined pitot/static probes, the static opening will consequentially receive some heating. Modern systems have an **alert warning** if the heaters are **not switched on in flight**.

It is essential to test the operation of the pitot heater prior to flight. This can be done by switching it on for about 30 seconds and **carefully** feeling the pitot probe to check that it has got warm. On other aircraft it may be possible to check that on switching on there is a rise in the ammeter current, or that the magnetic compass is deflected. It should then be switched off again so that it does not burn out. Make sure that the pitot covers and static plugs are removed **before** turning the heaters on to check them or you may burn or melt them!

It is normally part of the pre-take-off checks to ensure that heaters are switched on again and part of the after-landing checks to switch them off.

### Preflight Checks of the Pitot/Static System.

- All covers and plugs removed and stowed.
- All tubes, holes, slots free of obstructions.
- Pitot head heater operating.



## Questions

1. **A pitot head is used to measure:**
  - a. dynamic minus static pressure
  - b. static plus dynamic pressure
  - c. static pressure
  - d. dynamic pressure
  
2. **A static vent is used to measure:**
  - a. dynamic pressure minus static pressure
  - b. dynamic pressure plus static pressure
  - c. atmospheric pressure
  - d. dynamic pressure
  
3. **A pressure head is subject to the following errors:**
  - a. position, manoeuvre-induced, temperature
  - b. position, manoeuvre-induced
  - c. position, manoeuvre-induced, density
  - d. position, manoeuvre-induced, instrument
  
4. **Given:**  
**Pt = total pressure**  
**Ps = static pressure**  
**Dynamic pressure is:**
  - a.  $P_t - P_s$
  - b.  $(P_t - P_s) / P_t$
  - c.  $(P_t - P_s) / P_s$
  - d.  $P_t / P_s$
  
5. **Manoeuvre induced error:**
  - a. is caused by pressure changes at static probes or vents
  - b. is likely to be greatest when yawing after engine failure
  - c. is combined with instrument and position error on a correction card
  - d. lasts for only a short time at high altitude
  
6. **Position error:**
  - a. may be reduced by the fitting of static vents
  - b. will usually decrease with an increase in altitude
  - c. will depend solely on the attitude of the aircraft
  - d. will usually decrease as the aircraft approaches the speed of sound
  
7. **Fitting static vents to both sides of the aircraft fuselage will:**
  - a. reduce the position error
  - b. balance out errors caused by side-slipping or yawing
  - c. require a calibration card for each static vent
  - d. enable a greater number of instruments to be fitted

8. Where an alternate static source is fitted, use of this source usually leads to:
- a. a temporary increase in lag error
  - b. a lower pressure error than with normal sources
  - c. an increase in position error
  - d. no change in position error



## Answers

1	2	3	4	5	6	7	8
b	c	d	a	a	a	b	c

## Chapter

# 3

## Air Temperature Measurement

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## Introduction

Why does a pilot need to know the temperature of the air in which he is flying? There are several reasons:

- Avoidance of icing conditions
- Because temperature affects air density, which, in turn, affects:
- Engine power and aircraft performance
- Measurement of speed
- Measurement of altitude

## Avoidance of Icing Conditions

The formation of ice on aircraft, particularly in cloud, can be very rapid. Icing can be very dangerous, because the following can occur:

- Loss of lift (ice will distort the aerofoil shape)
- Increase in drag (ice will distort the aerofoil shape)
- Increase in mass (can be as much as ten tons with thick icing on large aircraft)
- Freezing of control surfaces so that they cannot be moved
- Loss of engine power or total engine failure due to intake or carburettor icing
- Chunks of ice flying off the propellers and hitting the side of the fuselage

Many forms of weather are unpleasant or uncomfortable (turbulence, lightning, etc) but ice is particularly dangerous. There are two ways to minimize the risk. One is to avoid flying in cloud. The other is to climb or descend in order to avoid the temperature bands particularly associated with icing. These will vary from aircraft type to aircraft type but will be known for each particular type.

## Engine Power and Aircraft Performance

To see why air temperature affects engine power, consider how any aviation engine (jet or piston) works. Air is drawn in through the intake and mixed with vapourized fuel which is burnt in a cylinder or a combustion chamber where its volume is greatly expanded. This expansion is partly caused by the addition of combustion products (i.e. fuel), but mainly caused by a large temperature rise. In order to achieve correct combustion, the fuel/air ratio has to be carefully controlled. If the air is dense, more fuel will be provided and more power will be available. However, if the air is less dense, less fuel will be provided, in order to maintain the correct mix. Therefore less power will be produced. This can have a considerable effect, particularly on take-off performance. Pilots therefore need to be able to measure air temperature in order to calculate engine power and performance.

## Measurement of Speed

We cannot measure airspeed directly, so we measure air pressure. However, pressure is dependent on both relative speed and air density. Air temperature affects air density and therefore affects the calculation of airspeed.

## Measurement of Altitude

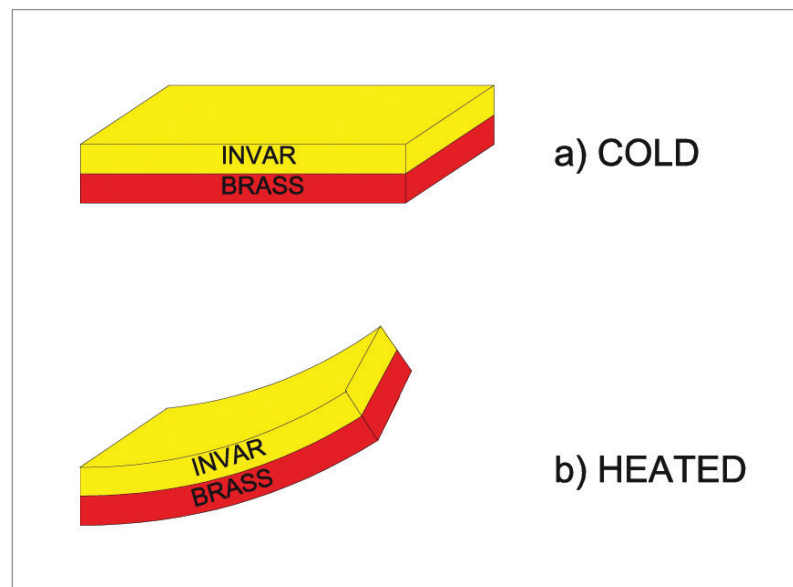
The rate of pressure change with altitude varies with temperature. Altimeter indication can be in error unless corrected for temperature differences from normal. This can have serious safety implications when flying near high ground in cloud.

## Air Temperature Thermometers

Air temperature thermometers can be divided into 2 basic types: Direct Reading or Remote Reading.

**Direct Reading.** These work on the principle of differential coefficients of expansion with temperature. Invar is a nickel steel alloy notable for its uniquely low coefficient of thermal expansion. A bimetallic strip is constructed consisting of invar and brass bonded together as shown in [Figure 3.1](#).

When this strip is heated, the brass, having a higher coefficient of expansion than the invar, will expand more than the invar so that the strip curls as shown. How much the strip curls depends on the temperature rise to which the strip is subjected.



*Figure 3.1 A bimetallic strip*

The strip is drawn out into a helix, to give greater pointer movement for a given temperature rise, and is incorporated into an instrument with a dial as shown in [Figure 3.2](#).



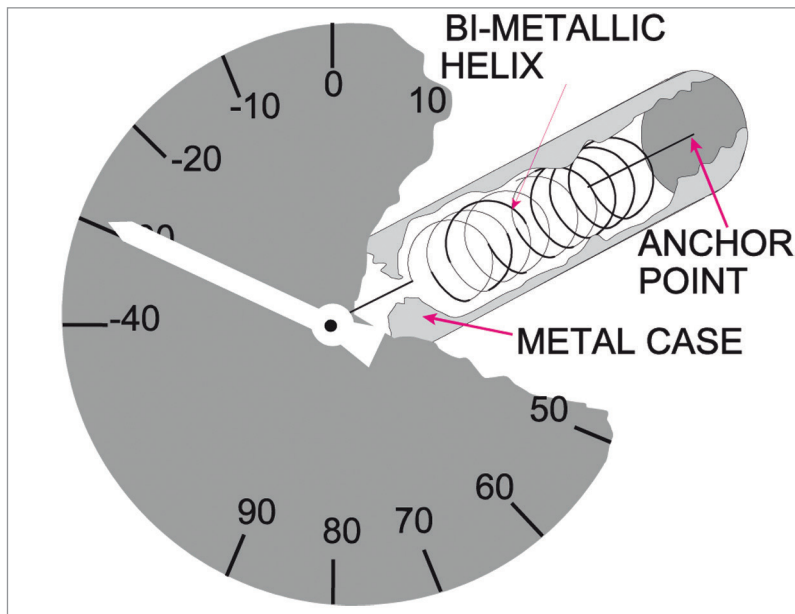


Figure 3.2 A bimetallic helix thermometer

The thermometer is mounted on the windscreen or fuselage with the tube protruding out into the air stream and the dial visible to the pilot, as shown in [Figure 3.3](#).

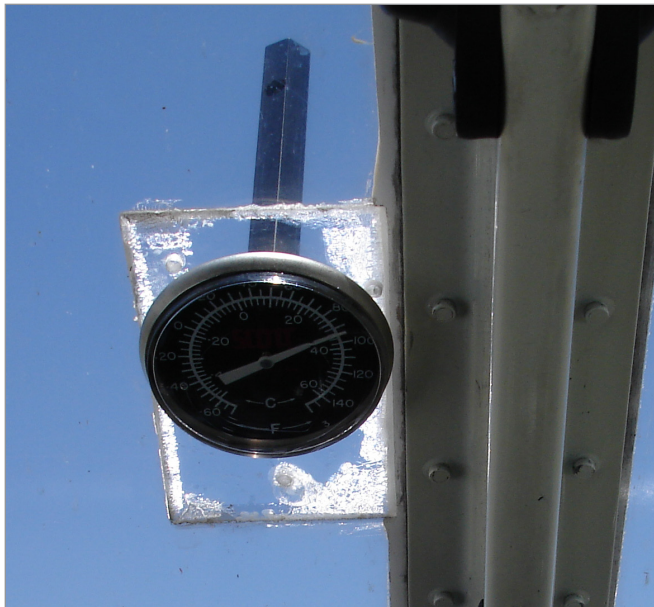


Figure 3.3 Thermometer placement on a PA28 Warrior

**Remote Reading.** However, the direct reading thermometer is not a workable solution for a larger aircraft. Firstly, a device that penetrates the windscreens would weaken the structure to an unacceptable extent at the pressures associated with higher speeds. Secondly, it is desirable to have the temperature information in an electrical form so that it can be fed to other instruments and systems.

## Total Air Temperature Probe

Figure 3.4 and Figure 3.5 show typical air temperature probes.

The probe is in the form of a small strut and air intake made of nickel-plated beryllium copper, which provides good thermal conductivity and strength. It is fixed to the fuselage at a point which keeps it away from the aircraft's boundary layer.

As the air flows through the tube, separation of water particles is achieved by causing the airflow to turn through a right angle before passing round the sensing element.

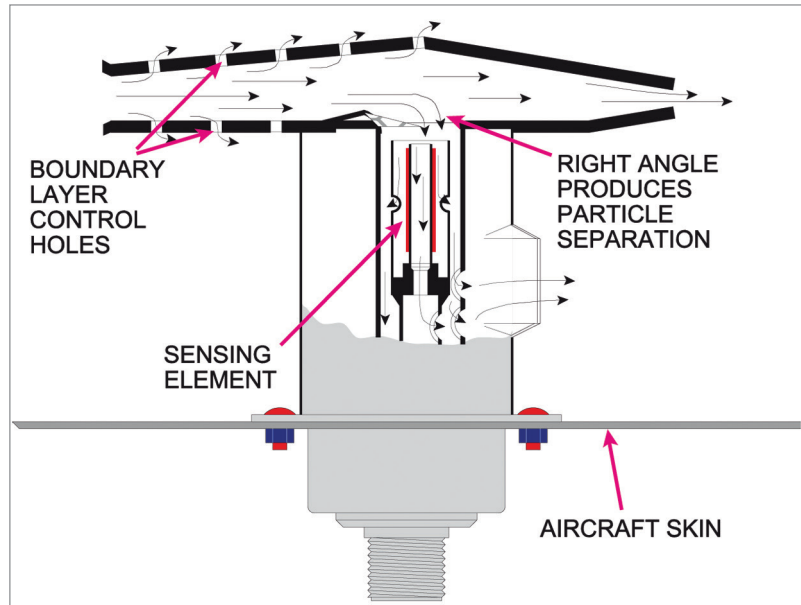


Figure 3.4 A total temperature probe

The bleed holes in the intake casing permit boundary air to be drawn off because of the higher pressure inside the intake.



Figure 3.5 A total temperature probe

A resistance wire made of pure platinum is used as the sensor. It has a very high thermal conductivity and a rapid response to change.

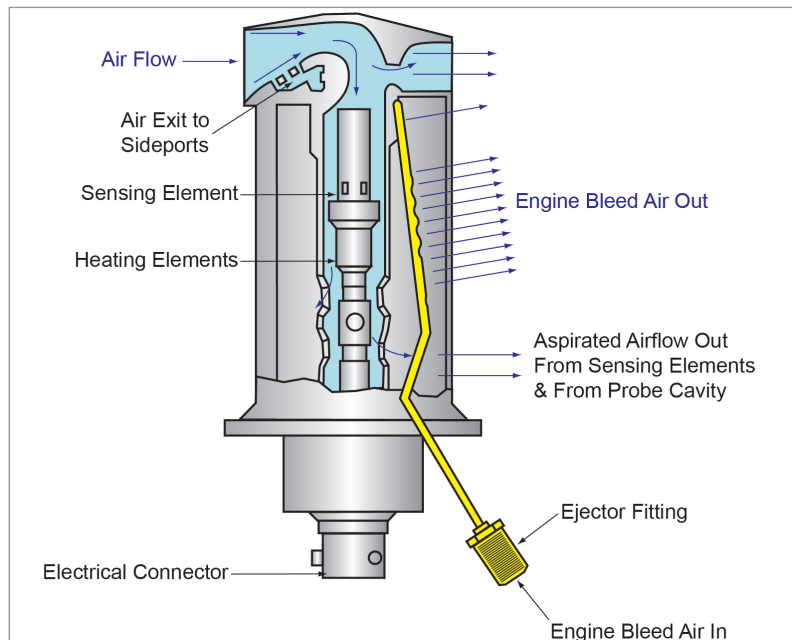
An inbuilt heating element prevents the formation of ice. It is self-compensating – as the temperature rises, so does the heater resistance, reducing the heater current.

The heater can have a small effect on the temperature readings. However, this introduces an error of less than 1°C, which is not significant.

It is also necessary to measure air temperature on the ground. Modern aircraft do not use full power for take-off in order to avoid

unnecessary thermal stress to the engines. The take-off is carried out using the minimum power necessary to ensure safety, but no more. Runway length, weight, altitude and temperature are taken into consideration in the calculation of the required power and it is perfectly normal to carry out a take-off at, say, 93% power.

In order to measure air temperature on the ground an air to air ejector (aspirator) is used, as shown in *Figure 3.6*. Bleed air from either an APU or a running engine creates a negative differential pressure within the casing so that outside air is drawn through it even when the aircraft is stationary in order to prevent the temperature of stagnant, heat-soaked air from within the casing being measured.



*Figure 3.6 Aspirated TAT probe - suction effect from positive compressor air pressure*

In the above diagram, the engine bleed air is positive pressure from one of the engine compressor stages. Although it is blowing, not sucking, the flow is arranged such that the rearward movement of the engine bleed air creates a suction effect past the sensing element.

## Errors

Air temperature gauges are subject to the following errors:

- Instrument error - imperfections in manufacture
- Environmental errors - solar heating of the probe (overcome by shielding in the strut)  
Ice accretion on the probe (overcome by use of the heater)
- Heating error - adiabatic (compression) and kinetic (friction) heating

Of these, by far the greatest is heating error – the effects of compression and kinetic heat.

## Heating Error – The Effects of Compression and Kinetic Heat

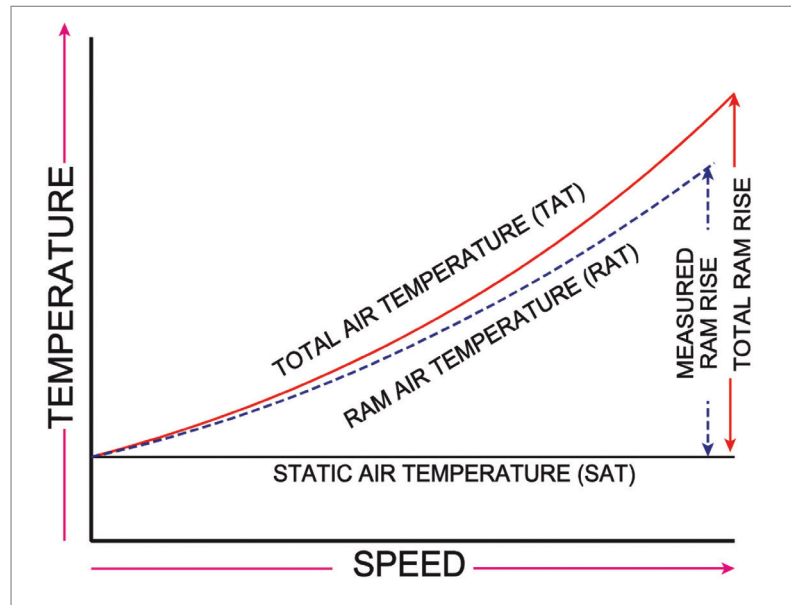


Figure 3.7 The effect of speed on measured temperature

As aircraft speed increases, the measured temperature exceeds that of the still air. This is caused by a combination of kinetic (friction) and adiabatic (compression) heating.

- **Kinetic heating.** Kinetic heating is a contributor to the temperature rise effect in the Direct Reading thermometer. As the speed increases, more and more molecules of air per second impact against the flat plate at the surfaces of the probe. This causes a temperature rise due to friction.
- **Adiabatic heating.** Adiabatic heating is the main contributor to the temperature rise effect in the Remote Reading (or Total Head) thermometer. It is caused by a conversion of energy, not a direct application of heat. For example, boiling a kettle is a direct application of heat, whereas pumping a bicycle pump causes a rise in temperature without any input of heat because pressure energy is converted into temperature energy. In the Remote Reading temperature probe, the outside airflow which may be several hundred knots is brought virtually to rest in the platinum measurement chamber very rapidly. The energy of the moving air is released in the form of a temperature rise.

There is a known relationship between the relative speed and the temperature rise. The laws of thermodynamics are such that the combination of kinetic and adiabatic heating will always add up to a figure known as the **Total Ram Rise**.

Unfortunately, any measurement process always has leaks and inefficiencies, so we do not measure the full ram rise. The amount of ram rise actually sensed is called the **Measured Ram Rise**. If we could measure the Total Ram Rise, we would measure the **Total Air Temperature**. However, in practice, what we measure is the **Ram Air Temperature**, which is lower.

- **Static Air Temperature (SAT).** Sometimes written as  $T_s$ . SAT is the temperature of the undisturbed air through which the aircraft is about to fly.

- **Total Air Temperature (TAT).** Sometimes written as  $T_t$ . TAT is the maximum temperature attainable by the air when brought to rest adiabatically.
- **Ram Air Temperature (RAT).** The RAT is the measured air temperature.
- **Ram Rise.** Ram rise is the difference between the SAT and the TAT (Total Ram Rise) or the difference between the SAT and the RAT (Measured Ram Rise).
- **Recovery Factor.** The percentage of the Ram Rise sensed and recovered by a TAT probe is called the Recovery Factor (usual symbol is  $K_r$ ).

### Ram Rise – Application of Recovery Factor

Assume an aircraft experiencing a SAT of  $-60^\circ\text{C}$  at a speed where the (theoretical) Total Ram Rise would be  $30^\circ\text{C}$ . Assume a Recovery Factor of 0.9.

SAT	=	$-60^\circ\text{C}$
Total Ram Rise	=	$+30^\circ\text{C}$
(Theoretical) Total Air Temperature	=	$-30^\circ\text{C}$
Recovery Factor	=	0.9
Measured Ram Rise ( $+30 \times 0.9$ )	=	$+27^\circ\text{C}$
Ram Air Temperature	=	$-33^\circ\text{C}$

The gauges would actually indicate the Ram Air Temperature of  $-33^\circ\text{C}$ . A correction factor of  $-27^\circ\text{C}$  would be applied by one of several methods to give SAT.

Theoretically, uncorrected air temperature gauges give RAT, not TAT. However, in many aircraft, the gauges are incorrectly labelled 'TAT'.

SAT is sometimes called COAT (Corrected Outside Air Temperature)

TAT is sometimes called IOAT (Indicated Outside Air Temperature)

OAT on its own is usually taken to mean SAT.

### Summary

$$\text{TAT} = \text{SAT} + \text{Ram Rise}$$

$$\text{IOAT} = \text{COAT (or OAT)} + \text{Ram Rise}$$

## Correction of TAT/RAT to SAT

The indicated RAT can be converted to SAT by the following methods:

- Rapid formula for use in the air
- CRP5 or similar navigation computer
- Accurate formula
- Data Tables
- Air Data Computer

## Rapid Formula for Use in the Air

This formula is a good simple approximation:

$$\text{Ram Rise} = \left( \frac{v}{100} \right)^2$$

where  $v$  = TAS in knots.

So, if TAS is 200 knots the ram rise is 4°C, if TAS is 300 knots the ram rise is 9°C, if TAS is 400 knots the ram rise is 16°C, and so on.

This formula should not be used in JAA exams because it is not quite as accurate as the CRP5 solution, but it is good enough for practical use in the air when a quick solution is required and it is not convenient to find a Nav Computer or look up tables.

## Navigation Computer

On the slide rule face of the navigation computer there is a blue scale

The outer side is TAS and the inner is Ram Rise. In the above example a TAS of 400 knots gives a Ram Rise of 17°C.

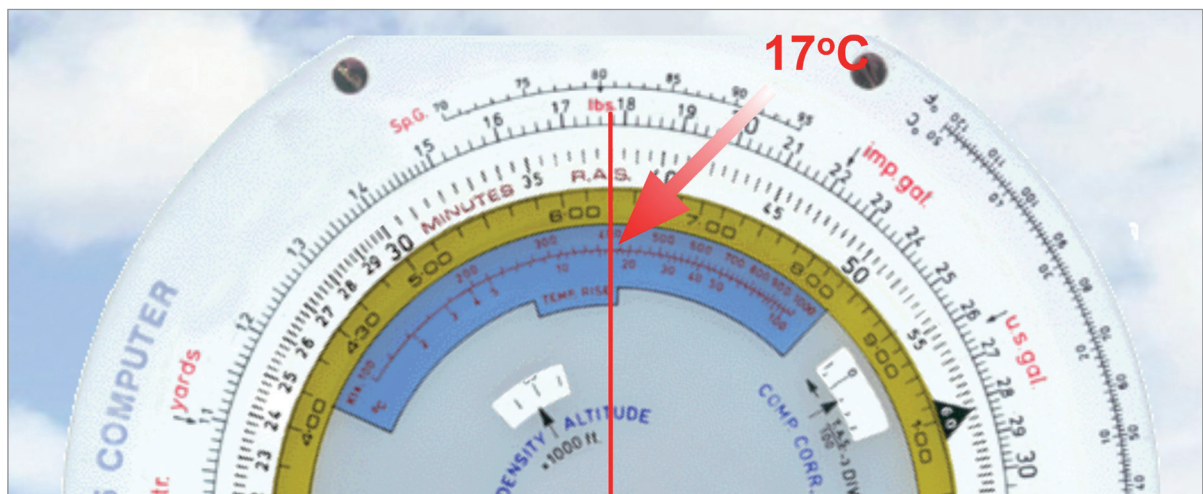


Figure 3.8 Use of the blue scale for ram rise on the CRP-5

## Accurate Formula

The accurate formula is:  $SAT = \frac{TAT}{1 + 0.2 K_r M^2}$

where M = Mach Number

However, the formula assumes that temperature is quoted in degrees Absolute ( $^{\circ}A$ ), or Kelvin (K).

## The Absolute, or Kelvin, Temperature Scale

When Celsius designed his temperature scale, he chose the boiling point of water as  $100^{\circ}C$  and the freezing point of water as  $0^{\circ}C$ . This seemed reasonable, given the state of physics at the time. However, the problem with the normal Celsius temperature scale is that it changes sign at  $0^{\circ}C$  (the freezing point of water). Numbers above zero are positive whilst numbers below are negative. But to measure the relative levels of hotness in objects in a meaningful way, we should start from a baseline of absolute zero – no heat at all.

This occurs at  $-273^{\circ}C$  on the Celsius scale. So  $-273^{\circ}C$  is  $0^{\circ}A$  or 0 K on the kelvin scale. One kelvin, or one degree Absolute, is the same amount of temperature change as one degree Celsius. Therefore the freezing point of water occurs at 273 K, the boiling point of water at 373 K, and so on.

Assume the following:

Indicated TAT (actually RAT) =  $-20^{\circ}C$

Mach No = M 0.73 (typical cruise speed for a B737 in Long Range Cruise)

Recovery Factor = 0.98 (which is a typical value for a modern TAT probe)

$K_r$  is determined by flight testing and is published in the operating instructions for the aircraft.

A TAT of  $-20^{\circ}C$  is 253 K. Substituting into our formula:

$$SAT = \frac{TAT}{1 + 0.2 K_r M^2}$$

$$SAT = \frac{253}{1 + (0.2 \times 0.98 \times 0.73^2)}$$

$$SAT = 253 / (1 + 0.1044)$$

$$SAT = 253 / 1.1044 = 229 \text{ K}$$

229 K is  $-44^{\circ}C$  on the Celsius scale.

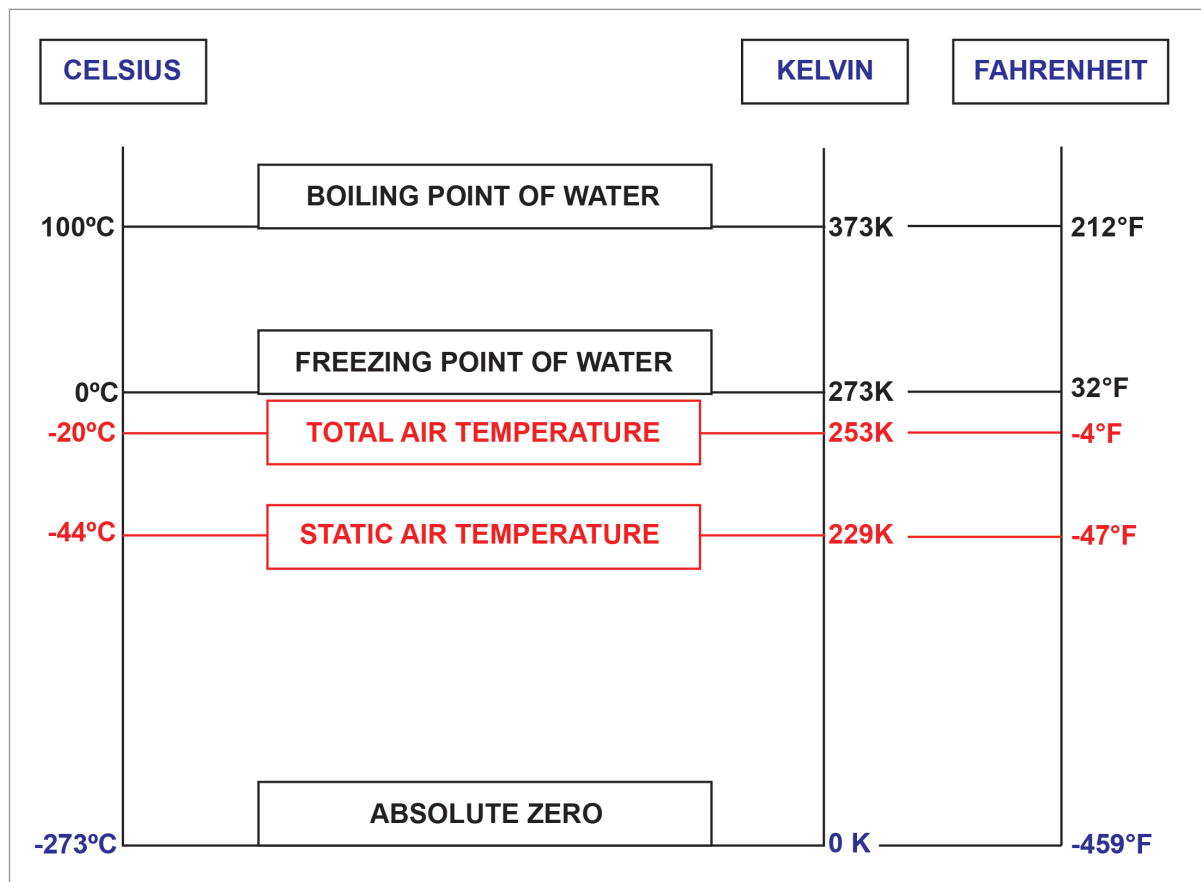


Figure 3.9 Relationship between Celsius and Kelvin – TAT / SAT

## Calibration

As a result of the variable changes in the atmosphere, due to temperature and pressure, it is necessary to have a standard calibration for the instruments.

The conditions used for calibration are usually those assumed for the:

**International Standard Atmosphere  
ISA**



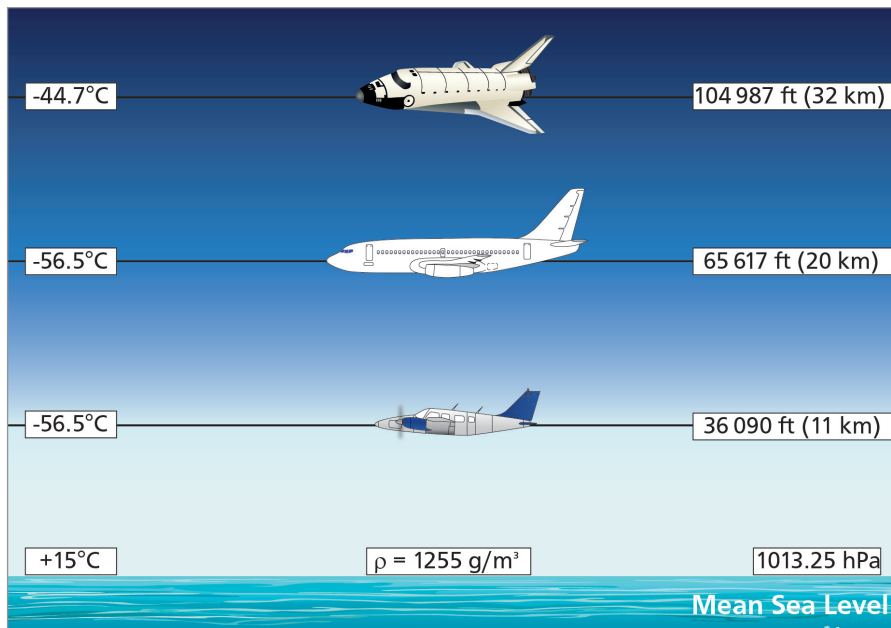


Figure 3.10 The International Standard Atmosphere (ISA)

The relevant assumptions are:

**At Mean Sea Level**

Pressure 1013.25 hectopascals      Temperature +15°C      Density 1225 g/m<sup>3</sup>

**From MSL up to 11 km (36 090 feet)**

Temperature falling at 6.5°C per km (1.98°C/1000 feet)

**From 11 km to 20 km (65 617 feet)**

A constant temperature of - 56.5°C

**From 20 km to 32 km (104 987 feet)**

Temperature rising at 1°C per km (0.3°/1000 feet)

With these assumptions, the pressure corresponding to any given level in the ISA can be calculated from the calibration formulae.

Graphs or tables can be produced showing height in terms of pressure under standard conditions. These tables can be used for the manufacturer's calibration of the altimeter scale.

Any discrepancies, if within certain agreed tolerances, would be listed over the operating height ranges as instrument errors. (The calibration is carried out with increasing and decreasing readings so that the amount of lag at calibration conditions can be determined).

## Questions

1. **The difference between static air temperature and total air temperature is known as:**
  - a. corrected outside air temperature
  - b. the total ram rise
  - c. the recovery factor
  - d. hot ramp radiation
  
2. **A direct reading aircraft thermometer usually consists of a bimetallic helix protruding into the airstream. Movement of the pointer over the temperature scale will depend upon:**
  - a. difference in electrical resistance of the two metals
  - b. increase in pressure as airspeed increases
  - c. increase in adiabatic cooling as airspeed increases
  - d. different coefficients of expansion of the two metals
  
3. **A remote reading thermometer depends on ..... to indicate changes in temperature:**
  - a. change of electrical resistance of the two metals
  - b. change of electrical resistance with temperature
  - c. change of electrical resistance with pressure
  - d. change of electrical capacitance with temperature
  
4. **Aircraft air temperature thermometers are shielded to protect them from:**
  - a. solar radiation
  - b. accidental physical damage on the ground or hailstones in flight
  - c. airframe icing
  - d. kinetic heating
  
5. **At a true airspeed of 500 knots, what is the ram rise?**
  - a. 50°C
  - b. 25°C
  - c. 5°C
  - d. 16°C
  
6. **An air temperature probe may be aspirated in order to:**
  - a. prevent icing
  - b. measure air temperature on the ground
  - c. compensate for moisture level at the ramp position
  - d. reduce the effects of solar radiation
  
7. **Total Air Temperature is:**
  - a. the maximum temperature attainable by the air when brought to rest adiabatically
  - b. the temperature indicated on the air temperature thermometer plus the ram rise
  - c. the static air temperature minus the recovery factor
  - d. the recovery factor plus the ram rise

8. Which of these formulae gives the total temperature ( $T_t$ ) from the static temperature ( $T_s$ ):

- a.  $T_t = T_s(1 + 0.2 M^2)$
- b.  $T_t = T_s(1 + 0.2 K_r M^2)$
- c.  $T_t = T_s / (1 + 0.2 K_r M^2)$
- d.  $T_t = T_s(1 - 0.2 M^2)$

## Answers

1	2	3	4	5	6	7	8
b	d	b	a	b	b	a	b

## Chapter

# 4

## The Airspeed Indicator (ASI)

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## Principle of Operation

An aircraft on the ground in still air is subject only to atmospheric (static) pressure (S). However, in flight, the leading edges of an aircraft are subject to an additional (dynamic) pressure. This results in a total (pitot) pressure (P) on the leading edges of dynamic pressure plus static pressure.

$$\text{Pitot} = \text{Dynamic} + \text{Static}$$

The pitot head senses pitot pressure and the static/vent senses static pressure. These two pressures are fed to the airspeed indicator, a differential pressure gauge, which measures their difference (the dynamic pressure). Dynamic pressure is related to airspeed, because:-

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

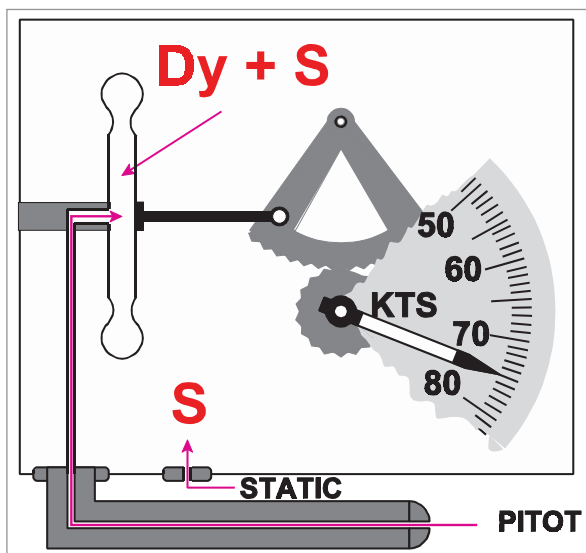
where V is true airspeed (TAS) and  $\rho$  is the density of the surrounding air.

The ASI measures airspeed by measuring dynamic pressure, displaying the result (usually in knots) on a suitably calibrated scale. (1 knot is 1 nautical mile per hour).

## Construction

It is therefore necessary to construct a device in which static pressure is subtracted from pitot (total) pressure in order to isolate dynamic pressure. Dynamic pressure cannot be directly measured; it must be established from the difference between pitot and static pressure.

Static pressure is fed into a hermetically-sealed instrument case, the pitot pressure being piped to a thin metal capsule capable of expansion and contraction. The layout is shown in *Figure 4.1*.



*Figure 4.1 A functional ASI*

Static pressure is present on both the inside and the outside of the metal walls of the capsule and so cancels. Therefore the pressure differential between the inside and outside of the capsule is  $(Dy + S) - S$  which is Dynamic. Expansion or contraction on the capsule will therefore be proportional to the changes in dynamic pressure produced by changes of airspeed.

The capsule movements are transmitted by a temperature-compensated magnifying linkage to the pointer indicating airspeed on the face of the ASI.

## Calibration

Dynamic pressure depends not only on the speed of the aircraft but also on the air density.

This density varies with temperature and pressure and therefore with **altitude**. The ASI is calibrated to read true airspeed for the air density of 1225 grams per cubic metre which would be produced by the ISA MSL pressure of 1013.25 hPa and temperature + 15°C (dry air conditions). No allowance is made in the calibration for the change in density which occurs with change of altitude.

As the ASI is only calibrated at MSL, and ISA conditions, flight at any other height will cause errors.

## ASI Definitions and Errors

**Instrument Error.** Manufacturing imperfections and usage result in small errors which are determined on the ground under laboratory conditions by reference to a datum instrument.

A correction card can be produced for the speed range of the instrument.

**Position Error.** Alternatively known as 'pressure' error, this arises mainly from the sensing of incorrect static pressure, and is described more fully in the section entitled Pressure Heads.

Position errors throughout the speed range are determined by the aircraft manufacturer during the test flying programme for a particular aircraft type.

It is not unusual to compile a joint correction card for position and instrument errors and place it in the aircraft near the ASI concerned.

**Manoeuvre-induced Errors.** These are associated chiefly with manoeuvres involving change in angle of attack, giving transient errors and a lag in the indication of changes in airspeed.

## Calibrated Airspeed (CAS)

The pilot corrects the IAS for Instrument and Position Error from the correction card to give Calibrated Airspeed (CAS). For instance, for a combined correction of +2, if the pilot wants a speed of 100 knots CAS, he flies at 98 knots IAS. Calibrated Airspeed is also known in some older publications (and on the Pooley CRP-5 Navigation Computer) as Rectified Airspeed (RAS). However, this term is no longer in general use.

**CAS is IAS corrected for Instrument and Position (Pressure) Error.**

Aircraft stress and stall limits are determined by dynamic pressure, not speed. There are quoted limiting 'speeds' to avoid overstress or control problems with the aircraft as a whole, and also to avoid overstress of various services such as flaps and undercarriage. In addition, there are also stall 'speeds' for various configurations. These are functions of dynamic pressure. As pressures, they could be quoted in bars, psi or pascals. They are not speeds. However, because the ASI is calibrated in 'knots' (even though this is only a true speed for a specific air density), dynamic pressure is measured in knots of CAS. CAS will be equally valid as a measure of stall or stress limits whatever the aircraft's true airspeed or altitude and whatever the air density. The display of simple ASIs is therefore IAS and that of Air Data Computers is IAS or, in those with a computed correction for Instrument and Pressure Error, it is CAS. This is in order to allow



pilots to be able to fly their aircraft safely without overstressing or stalling them. (Air Data Computers calculate and indicate speed, altitude, temperature and other air parameters).

For navigation, and flight planning, the CAS is of no significant use. What the pilot requires is the speed of the aircraft relative to the ground. In the absence of any wind (this effect is discussed in General Navigation - book 10) this speed, relative to the ground, is known as the True Airspeed (TAS).

## Equivalent Airspeed (EAS)

Whenever Density Error is present, Compressibility Error exists. It is possible to correct CAS just for Compressibility Error without correcting for Density Error. The resultant dynamic pressure is then called **Equivalent Airspeed (EAS)**.

**Equivalent Airspeed is CAS corrected for Compressibility Error only.**

EAS is the most accurate value of dynamic pressure. It has been corrected for Instrument, Pressure and Compressibility error, all of which are forms of measurement error. EAS is the most accurate measure of the dynamic pressure over the wing. In practice the difference between EAS and CAS is not great unless altitude becomes significant. However, at high elevation airports, particularly with high-performance airliners which may have high take-off and landing speeds, CAS (and therefore IAS) is higher for the same EAS.

**At constant weight, regardless of altitude, an aircraft always lifts off at a constant EAS.**

All limit speeds are calculated from EAS, the errors are then re-introduced to display the speeds as IAS.

## True Airspeed (TAS)

**Density Error.** Unless the air round the aircraft is at the calibration density of 1225 grams per cubic metre, which can only occur near sea level, the ASI cannot correctly indicate TAS. Dynamic pressure is proportional to density, and so at altitude, where density is less, the dynamic pressure generated by a given TAS will be less than for the same TAS in flight at sea level. ASI capsule expansion will therefore be proportionately less and the speed indicated will be less than the true speed.

Summarizing, the ASI under-reads the true speed at altitude as density is less than 1225 grams/cu metre, this discrepancy being called 'density error'. If the density is greater than ISA at MSL, the ASI will over-read the true speed.

**EAS + Density Error = TAS**

There is no single instrument that gives a direct reading of density. Density has to be calculated using the interaction of pressure and temperature. However, pressure is a function of altitude. By combining altitude and temperature information, the density correction is derived by the navigational computer. The major factor in the density calculation is the pressure value.

The calculation of the errors, using a navigation computer, is not required for the Instrumentation EASA exam. If you are interested in the calculations, full details of how to calculate the errors can be found in the Oxford ATPL Book 10 - General Navigation - Chapter 6.

**Compressibility Error.** Air is compressible and the pressure produced in the pitot tube is higher than it would be for an ideal incompressible fluid, for which the dynamic pressure is  $\frac{1}{2}\rho V^2$ .

The ASI is calibrated to allow for this complex compressible flow formula assuming a density of 1225 g/cu metre. If the density is any lower, the standard correction for compressibility will be in error.

Because of this, the instrument will over-read, IAS and CAS will be too high, and a subtractive compressibility correction will have to be applied

The correction, which exceeds 20 knots if TAS is near the speed of sound, can be obtained from graphs or tables, or it can be applied by high speed navigation computers.

If the TAS is less than 300 knots the error is small enough to be ignored in the calculation of TAS from IAS.

## Airspeed Indicator Errors - Application of Corrections

What you see on the instrument is called **Indicated Airspeed (IAS)**. **Instrument Error** and **Pressure (or Position) Error** are then applied to give **Calibrated Airspeed (CAS)**.

In practice, we do not have separate cards for the Instrument Error and the Pressure Error. It is simpler to combine both corrections onto a single card. **IAS** is therefore corrected for both **Instrument and Pressure Error** by the correction card, to give **CAS**.

**CAS** is then corrected for compressibility error to give **Equivalent Airspeed (EAS)**, to allow for the fact that the density of air is rarely precisely 1225 grams per cubic metre, which is the value for which the ASI is calibrated. (The ISA atmosphere at Mean Sea Level). In practice, as an airline pilot, you are unlikely to deal with EAS and it is rarely encountered outside scientific and test flying. The ASI is already calibrated to allow for compressibility at ISA at MSL and, under those conditions, no compressibility correction is necessary. Compressibility correction is small at True Airspeeds (TAS) lower than 300 knots and no correction is considered necessary.

You will therefore always calculate the **Density Error** correction first to give **True Airspeed (TAS)**. If the TAS you find is 300 knots or less, no further correction is necessary. If the TAS is greater than 300 knots, **Compressibility Error** correction must be applied.

### Limiting Speeds

Some are usually shown on the ASI

$V_{NO}$  = The maximum normal operating limit speed.

$V_{NE}$  = The Never Exceed speed

$V_{so}$  = The stall speed or the minimum steady flight speed in the landing configuration.

$V_{s1}$  = The stall speed or the minimum steady flight speed in a specified configuration.

$V_{FE}$  = The maximum Flap Extension speed

$V_{YSE}$  = best rate of climb speed with one engine failed

The following are also important, but are not shown on the ASI

$V_{LO}$  = The maximum Landing Gear Operation speed (up or down).

$V_{LE}$  = The maximum speed Landing Gear Extended speed

**ASI ACCURACY TOLERANCE - (CS-25) -  $\pm 3\%$  or 5 kt whichever is the greater.**

## ASI Colour Coding

Some ASIs incorporate coloured markings on the dial - these 'range markings' consist of coloured arcs and radial lines.

**The White Arc** denotes the flap operating range, from stall at maximum AWW in the landing configuration (full flap, landing gear down, wings level, power-off) up to  $V_{FE}$  (maximum flaps extended speed).

**The Green Arc** denotes the normal operating speed range, from stall speed at maximum all-up weight (flaps up, wings level) up to  $V_{NO}$  ('normal operating limit speed' or 'maximum structural cruising speed') which should not be exceeded except in smooth air. Operations at IASs in the green arc should be safe in all conditions, including turbulence.

**The Yellow Arc** denotes the caution range, which extends from  $V_{NO}$  (normal operating limit speed) up to  $V_{NE}$  (the never exceed speed). The aircraft should be operated at IASs in the caution range only in smooth air.

**A Red Radial Line** denotes  $V_{NE}$ , the never exceed speed.

Optionally for piston engined light twins:

**A blue radial line** denotes the best rate of climb speed for one engine out, maximum weight, at mean sea level ( $V_{YSE}$ ).

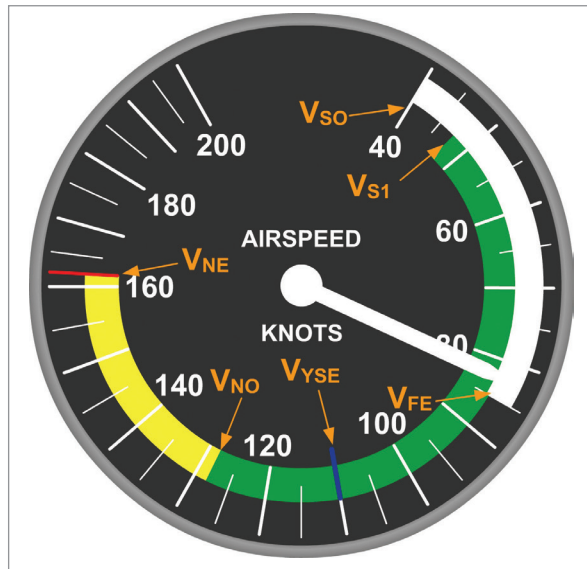


Figure 4.2 Coloured arcs on ASI

## Pitot and Static Blockages

**Pitot Head.** If the pitot head becomes blocked, the ASI reading will, in general, remain unchanged.

In level cruise, a blockage (probably ice, but possibly insects) will lock in the previous pitot pressure and any change in actual airspeed will not be registered.

If altitude is changed with a blocked pitot head and clear static source, the IAS will decrease during a descent because the pressure locked inside the capsule remains constant while the static pressure of the air surrounding the capsule increases. Therefore (pitot – static) results in an under-reading. Conversely the IAS increases during a climb with blocked head.

**With a blocked pitot source, the ASI under-reads in a descent.**

**Static Head.** A static head is more exposed to icing conditions and is therefore more likely to become obstructed than is a static vent.

A blocked static source during descent will mean that the 'old' (higher altitude) static pressure surrounding the capsule will be lower than it should be, so that if the pitot supply is normal the ASI will over-read. This could be dangerous in that the aircraft is nearer the stall than the ASI is indicating.

Note that a climb with blocked static source and normal pitot air will result in the ASI under-reading.

**With a blocked static source, the ASI over-reads in a descent.**

If the alternative static source is selected an error may occur. This error will be due to position error. Any dynamic, or turbulence, effects would usually result in a higher static pressure and thus produce an under-reading. This error is known and would be documented in the Flight Manual.

**Summary.** A useful mnemonic for examinations is **PUDSOD**, which stands for

**'Pitot Blocked: – Under-reads in Descent    Static Blocked: – Over-reads in Descent'**

## Pitot and Static Leaks

**Pitot Leaks.** A leak in the pitot tube causes the ASI to under-read because of the loss of dynamic pressure.

**Static Leaks.** Static leaks can occur either inside or outside the pressure cabin. A leak in the static tube where the outside pressure is lower than static, which occurs with almost all unpressurized aircraft, will cause the ASI to over-read, but probably not significantly. With a pressurized cabin, if the leak occurs within the pressure compartment, the cabin altitude will be incorrectly sensed as static pressure, usually rendering the ASI useless.

## Serviceability Checks

The following checks of the ASI and pressure supply system should be made before flight:-

- Pressure head cover(s) and static vent plug(s) removed and stowed aboard the aircraft.
- Pitot tube(s), holes/slots in static head(s) and/or static vent(s) should be checked free from obvious obstructions such as insects.
- Pitot head heater operative (if fitted).
- Dial glass clean and undamaged.
- The instrument should indicate airspeed in the correct sense shortly after starting the take-off run.

## Questions

1. **A leak in the pitot total pressure line of a non-pressurized aircraft to an airspeed indicator would cause it to:**
  - a. over-read in a climb and under-read in a descent
  - b. over-read
  - c. under-read in a climb and over-read in a descent
  - d. under-read
  
2. **A pitot blockage of both the ram air input and the drain hole with the static port open causes the airspeed indicator to:**
  - a. read a little low
  - b. read a little high
  - c. react like an altimeter
  - d. freeze at zero
  
3. **If the static line to the ASI becomes blocked during a long descent, a dangerous situation could arise due to the ASI:**
  - a. over-reading, this indicated speed falsely showing the aircraft to be further from the stalling speed than it actually is
  - b. under-reading, this indicated speed falsely showing the aircraft to be closer to the stalling speed than it actually is
  - c. under-reading, this indicated speed possibly leading to the operation of flaps and/or landing gear at speeds in excess of safety speeds
  - d. over-reading, this indicated speed possibly leading to the operation of flaps and/or landing gear at speeds in excess of safety speeds
  
4. **An aircraft maintaining a constant CAS and altitude is flying from a cold airmass into warmer air. The effect of the change of temperature on the speed will be:**
  - a. CAS will decrease
  - b. EAS will increase
  - c. TAS will increase
  - d. TAS will decrease
  
5. **The airspeed indicator is calibrated to:**
  - a. conditions of the International Standard Atmosphere at all heights
  - b. conditions of the International Standard Atmosphere at MSL
  - c. an air density of 1013.25 g/m<sup>3</sup>
  - d. indicate correctly in any atmosphere
  
6. **Dynamic pressure is equal to:**
  - a.  $\frac{1}{2} \rho V^2$
  - b.  $\frac{1}{2} V \rho^2$
  - c.  $(\frac{1}{2} \rho V)^2$
  - d.  $\frac{1}{2} (\rho V)^2$

7. **Excluding blockages, the full list of errors of the ASI is:**
- a. instrument error, position error, density error, manoeuvre induced error
  - b. instrument error, position error, temperature error, compressibility error, manoeuvre induced error
  - c. instrument error, position error, barometric error, temperature error, lag, manoeuvre induced error
  - d. instrument error, position error, density error, compressibility error, manoeuvre induced error
8. **Some ASIs have coloured arcs and lines marked on their dials. A yellow arc and a white arc indicate:**
- a. cautionary range and normal operating range
  - b. flap operating speed range and normal operating range
  - c. cautionary range and flap operating speed range
  - d. flap operating speed range and cautionary range
9. **If the static line to the ASI becomes blocked during a climb, the ASI reading will:**
- a. increase, no matter what the actual airspeed is
  - b. progressively under indicate the value of airspeed
  - c. progressively over indicate the value of airspeed
  - d. stick at the airspeed showing at the moment of blockage



## Answers

1	2	3	4	5	6	7	8	9
d	c	a	c	b	a	d	c	b



## Chapter

# 5

## The Pressure Altimeter

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## Principle of Operation

The pressure altimeter is a simple, reliable, pressure gauge calibrated to indicate height. The pressure at a point depends on the weight of the column of air which extends vertically upwards from the point to the outer limit of the atmosphere.

The higher an aircraft is flying, the shorter is the column of air above it and consequently the lower is the atmospheric pressure at the aircraft.

In other words, the greater the height, the lower the pressure, and by measuring the pressure the altimeter measures height.

Unfortunately, the relationship between pressure and height is not a linear one, so that calibration of the altimeter scale is not a simple matter.

The situation is further complicated by high and low pressure weather systems which produce pressure differences in the horizontal plane. Furthermore, the temperature of the air at the surface and the temperature lapse rate in the air above vary considerably; this affects pressure.

## Some Definitions

The pilot should be familiar with the following definitions.

### *Height*

The vertical distance of a level, point or object considered as a point, measured from a specified datum.

### *Elevation*

The vertical distance of a fixed (non-moving) point or object measured from MSL.

### *Altitude*

The vertical distance of a moveable object measured from MSL.

### *Pressure Altitude*

This is the altitude of the aircraft with reference to the pressure level of 1013.25 hPa.

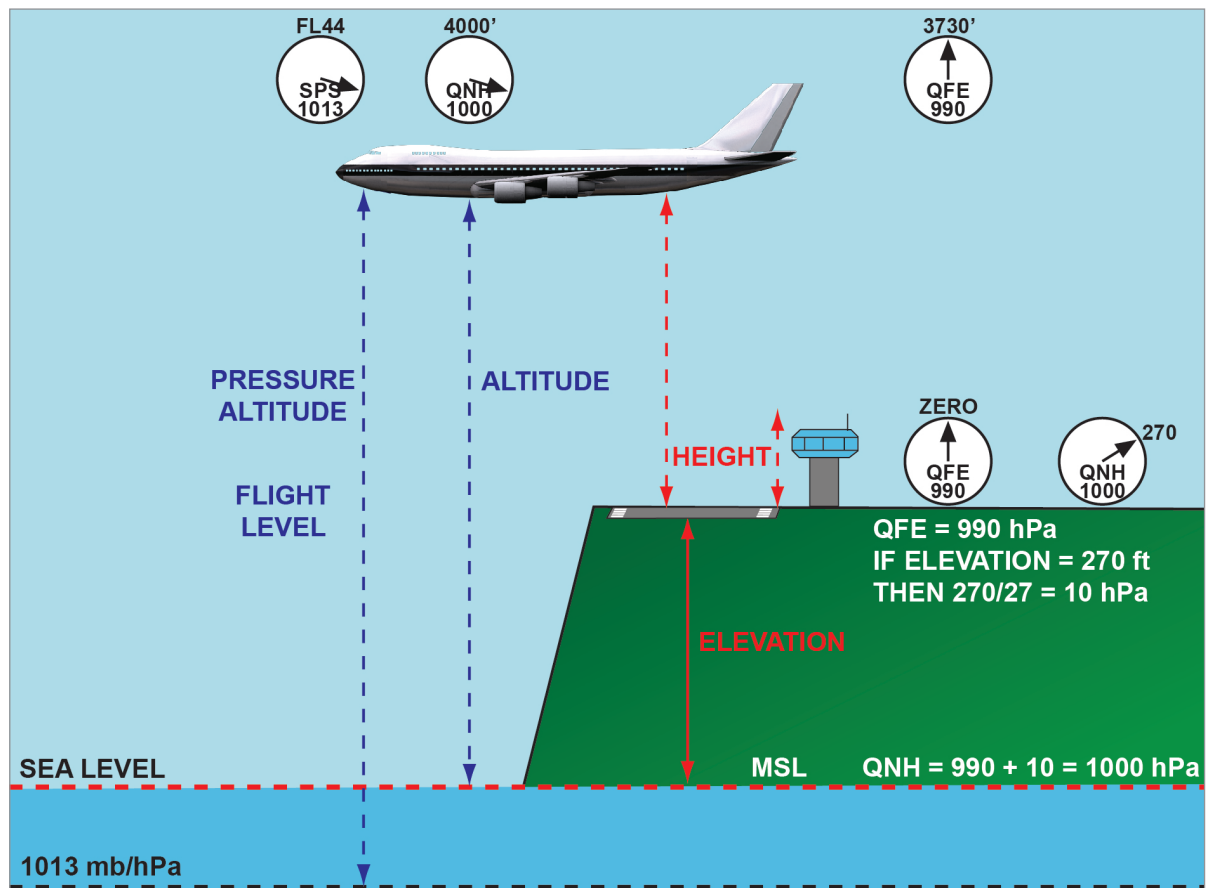


Figure 5.1 Terminology

### True Altitude

True Height means the height of the aircraft vertically above the surface immediately below. Used more often in connection with radio or radar altimeters than with pressure altimeters.

### Calibration

The altimeter is calibrated in accordance with the Standard Atmosphere (ISA) over its entire operating range. (This range is usually from 5000 ft below sea level up to 80 000 ft).

**Note 1:** The pressure altimeter is calibrated to give a linear presentation of the non-linear atmospheric distribution. This is achieved by the use of a variable magnification lever system and the dynamic design of the capsules.

**Note 2:** Temperature compensation is achieved by the use of a bimetal compensator connected in the lever/linkage system.

**Note 3:**  $1013.25 \text{ hPa} = 29.92 \text{ inHg} = 14.7 \text{ psi}$

## Simple Altimeter

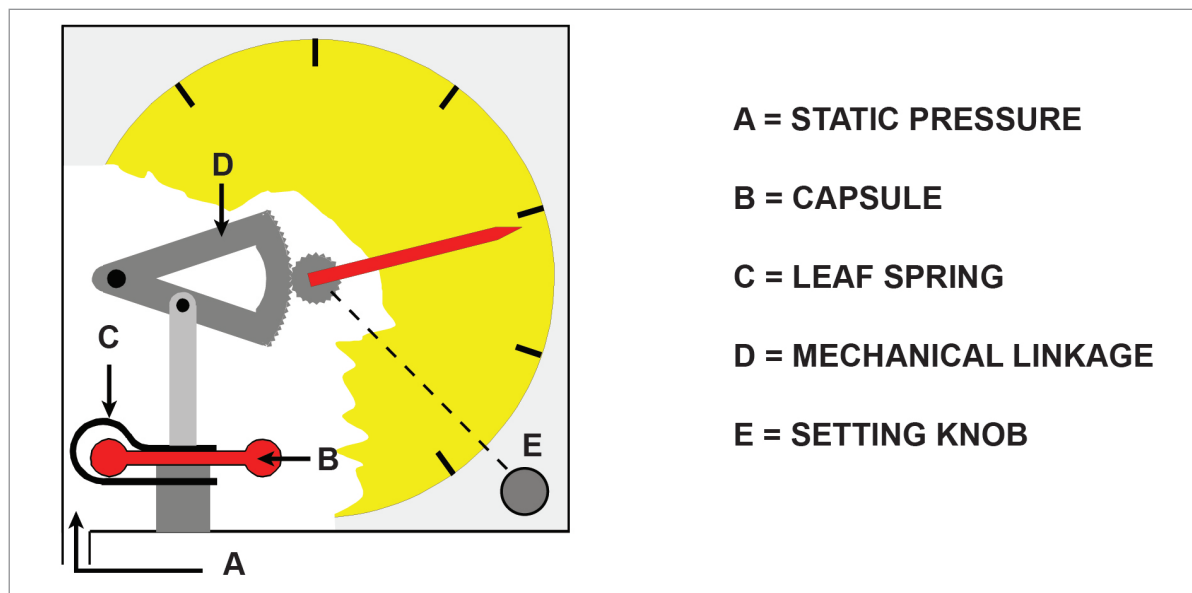


Figure 5.2 Simple altimeter

Static pressure is fed into the case of the instrument from the static source. As height increases, static pressure decreases and the capsule expands under the control of a leaf spring. A mechanical linkage magnifies the capsule expansion and converts it to a rotational movement of a single pointer over the height scale. The linkage incorporates a temperature-compensating device to minimize errors caused by expansion and contraction of the linkage and changes in spring tension due to fluctuations in the temperature of the mechanism.

Figure 5.2. shows the basic linkage, but the actual arrangements are much more complex.

The simple altimeter has a setting knob which is geared to the pointer. With this knob the pointer can be set to read zero with the aircraft on the ground so that when airborne the altimeter indicates approximate height above aerodrome level. Alternatively the pointer can be set (before flight) to the aerodrome elevation so that when airborne the instrument shows approximate height above mean sea level.

## Sensitive Altimeter

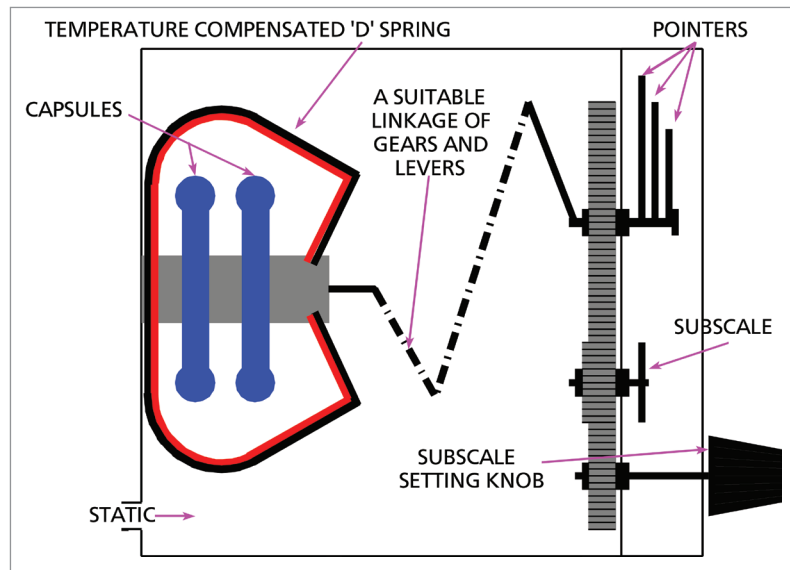


Figure 5.3 Sensitive altimeter

The Single Pointer Single Altimeter was not accurate enough, and was developed into the Sensitive Altimeter illustrated in *Figure 5.3*.

The principle of operation is similar to that of the simple altimeter but there are the following refinements:-

- A bank of two or three capsules gives the increased movement necessary to drive three pointers. These are geared 100:10:1, the smallest indicating 100 000 feet per revolution, the next 10 000 feet per revolution and the largest 1000 feet per revolution.
- Jewelled bearings are fitted, reducing friction and the associated lag in indications.

**Note:** Some altimeter systems employ "Knocking/Vibrating" devices to help overcome initial inertia of the internal gear train when transmitting movement from the capsules to the pointers.

- A variable datum mechanism is built in. This, with the aid of a setting knob, enables the instrument to be set to indicate height above any desired pressure datum.

The variable datum mechanism is used as follows:-

The pilot turns the knob until the desired pressure level (say, 1005 hPa) appears on a pressure subscale on the face of the instrument.

As the knob is turned, the height pointers rotate until, when the procedure is completed with the subscale showing the desired 1005, the altimeter indicates the aircraft's height above this pressure level.

If for instance the aerodrome level pressure happened to be 1005 hPa, the altimeter would be reading height above the aerodrome. Further details of the procedural uses of the pressure subscale are given later in this chapter. The subscale setting only changes when the pilot turns the knob. A change in altitude or surface pressure has no direct effect on the reading of the subscale. As the pilot alters the subscale setting, the altimeter pointers move, but the design of the mechanism is such that the reverse does not apply (for example, during a climb, the pointers rotate but the subscale setting remains unchanged). British altimeters have a subscale setting range between 800 to 1050 hectopascals.

Reading Accuracy

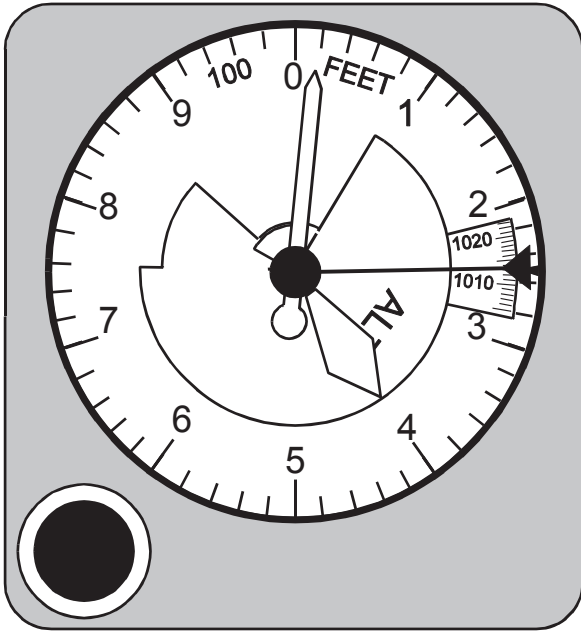


Figure 5.4 Three pointer altimeter indicating 24020 ft

The simple altimeter is not sensitive, recording perhaps 20000 ft for each revolution of its single pointer. The three-pointer instrument gives a much more sensitive indication of height and change of height but suffers from the severe disadvantage that it can be easily misread.

It is not difficult for the pilot to make a reading error of 10000 ft, particularly during a rapid descent under difficult conditions with a high flightdeck workload.

Accidents have occurred as a result of such misreading. Various modifications to the pointers and warning systems have been tried with the object of preventing this error, including a striped warning sector which appears as the aircraft descends through the 16000 foot level.

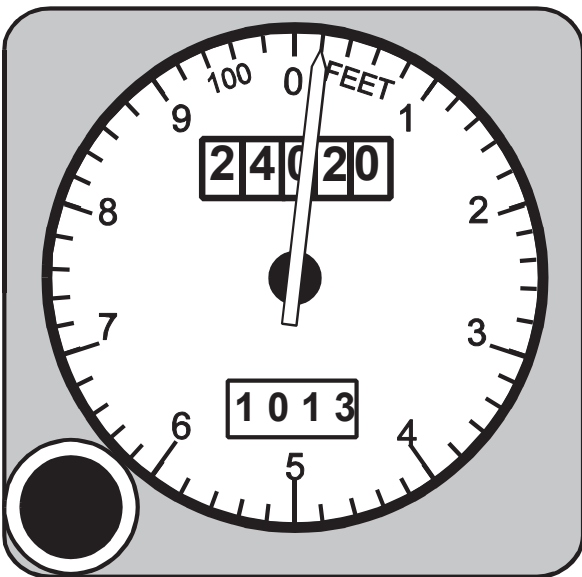


Figure 5.5 Counter / pointer altimeter

The greatest advance has been the introduction of the counter-pointer altimeter, illustrated in Figure 5.5, which gives a much more positive indication than the three-pointer dial drawn in Figure 5.4.

With further reference to Figure 5.5, it will be realized that though the digital counters give an unambiguous indication of the aircraft's height, they give a relatively poor display of the rate of change of height.

For this reason the instrument also has a single pointer which makes one revolution per 1000 feet, giving the clear indication of change of height which is extremely important to the pilot, particularly on the final approach in instrument conditions.

## Examples of Altimeters



A Sensitive Altimeter Reading 265 ft

Altimeter Reading  
12850 ft or  
3917 m

Electronic Display fitted to a Boeing 737

Figure 5.6 Altimeter types



## Servo-assisted Altimeters

Though at least one counter-pointer altimeter driven directly by pressure capsules has been produced, most instruments of the type are servo-assisted. This servo-assistance not only gives the altimeter an increased operating range but also improves the instrument accuracy, particularly at high levels.

At high altitude the change in pressure corresponding to a given change in height is much smaller than at low altitude.

This means that for a given height change, capsule movement at high altitude is relatively small, and frictional resistance in the linkage of an unassisted altimeter causes correspondingly greater errors and more lag. With servo-assistance, the requisite power is available to overcome the frictional resistance with consequently enhanced instrument accuracy.

The principle of the servo altimeter is that the small movements of the capsules are detected by a very sensitive electromagnetic pick-off. This produces an electric current which is amplified and used to drive a motor which rotates the counters and pointer.

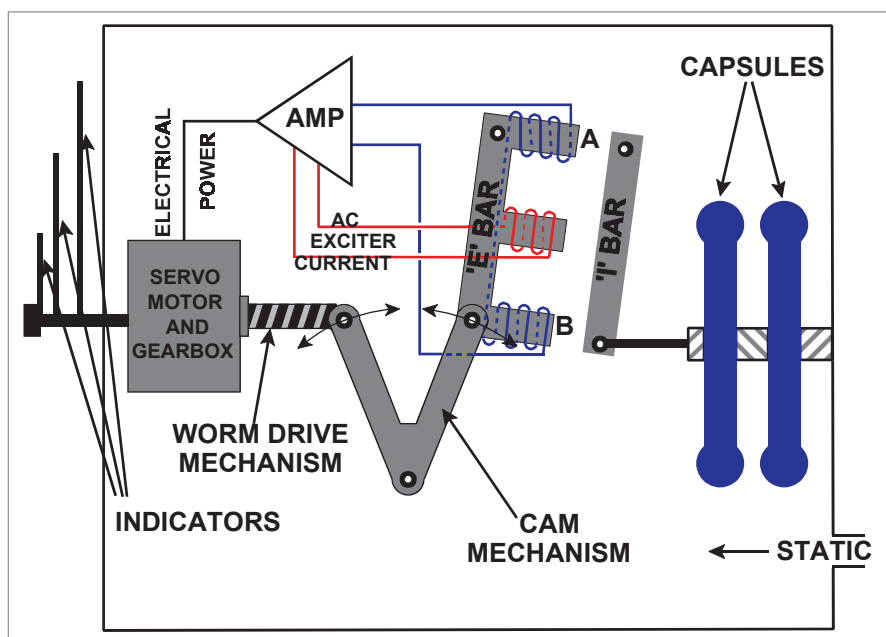


Figure 5.7 Servo Assisted Altimeter Schematic

AC is fed to the middle leg of the E bar, setting up alternating magnetic fields in the outer legs 'A' and 'B'.

The coils on these two legs are wound  $180^\circ$  out of phase. The exciter therefore induces a current in each leg, but since these are  $180^\circ$  out of phase and of equal strength, they cancel each other out when the I bar is equidistant from the legs of the E bar (that is when no pressure change acts on the capsules).

With a change of pressure the capsules expand or contract, moving the I bar on its pivot, closing the gap between the I Bar and E Bar at one end and opening it at the other.

This causes an imbalance of magnetic fields and therefore of the currents induced in the 'A' and 'B' coils. The imbalance causes an error signal which is passed to the amplifier, where it is amplified and rectified, and thence to the servomotor.

The servomotor drives the counter-pointer system of the altimeter and at the same time, via a cam drive, re-aligns the E Bar with the I Bar.

Once re-aligned, the error signal ceases and the altimeter indicates the correct height.

In this system the only work required of the capsules is to move the I Bar, eliminating the effects of friction and manufacturing imperfections in the gearing of a conventional altimeter.

This type of altimeter is sensitive to very small pressure changes and therefore more accurate than the sensitive altimeter, particularly at high altitudes where pressure changes (per unit height increment) are very small. The lag experienced in other types of altimeter with rapid changes of height is greatly reduced.

The normal instrument error is approximately equivalent to the effect of 1 hPa change of pressure (about 30 ft at MSL, 50 ft at 20 000 ft, or 100 ft at 40 000 ft).

**The tolerance at MSL (CS-25) is +/- 30 ft per 100 kt CAS**

## Tolerances

These values are for example only, and do not have to be learnt.

**Typical Simple Altimeter** (range zero to 35 000 feet)

Height (feet)	0	35 000
Tolerance (feet)	$\pm 100$	$\pm 1000$

**Typical Sensitive Altimeter** (range zero to 80 000 feet)

Height (feet)	0	40 000	80 000
Tolerance (feet)	$\pm 70$	$\pm 600$	$\pm 1500$

**Typical Servo Altimeter** (range zero to 100 000 feet)

Height (feet)	0	40 000	60 000	100 000
Tolerance (feet)	$\pm 30$	$\pm 100$	$\pm 300$	$\pm 4000$

## Altimeter Errors

The errors which affect altimeters are many and the extent of some of them varies with altimeter type. Much effort is expended on improving instrument accuracy, and the permissible tolerances of modern altimeters are smaller than with earlier types.

There are other errors caused by deviation of the actual atmosphere from standard conditions, and also the difficulty in sensing correctly the outside air pressure. A list of the main errors follows.

### *Position (or Pressure) Error*

This is largely due to the inability to sense the true static pressure outside the aircraft, as described in the chapter on Pressure Heads. The error is usually small but increases at high Mach numbers (and, consequently, at high altitudes usually associated with high Mach numbers).

### *Instrument Error*

Manufacturing imperfections, including friction in the linkage, cause errors throughout the operating range. The errors are kept as small as possible by adjustments within the instrument, and the calibration procedure ensures that they are within permitted tolerances. Residual errors may be listed on a correction card.

*Note: With the sensitive altimeter the error increases with altitude, which also explains why the decrease of accuracy with altitude is less serious with the servo altimeter.*

### *Manoeuvre-induced Error*

This is caused by transient fluctuations of pressure at the static vent during change of, mainly, pitch attitude and delays in the transmission of pressure changes due to viscous and acoustic effects in the static pipeline. This is discussed more fully in Chapter 2 dealing with pressure heads.

### *Barometric Error*

Providing the altimeter has a pressure subscale, and the local pressure is set on it, the altimeter will indicate height AMSL (though still subject to the other errors). If the local surface pressure has changed since the pressure value was set, a 'barometric' error of roughly 30 feet per hectopascal will result. If pressure has fallen, the altimeter over-reads.

### *Example Problem*

Exam questions sometimes include the term 'height involved'. This complicates matters. Think carefully when answering.

An aircraft flies from 'A' to 'B' at a constant indicated altitude of 10 000 feet with the pressure at 'A' of 1025 hPa set on the subscale THROUGHOUT THE FLIGHT. On arrival overhead 'B', where the pressure is 995 hPa, what will be the true altitude (assuming that there are no other errors, and assuming that 1 hPa corresponds to 30 feet)?

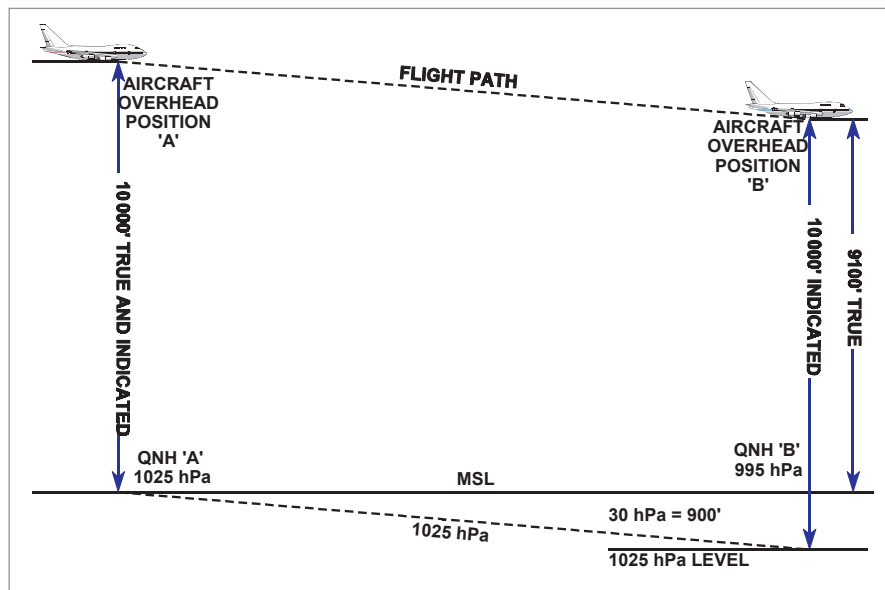


Figure 5.8 Datum diagram

### Solution

The altimeter indicates the height above the 1025 hPa pressure datum set on the subscale. At 'A', 1025 hPa is the MSL pressure so the aircraft is actually 10 000 feet above MSL. When it gets to 'B' where MSL pressure is 995 hPa, the 1025 hPa level will be below sea level, remembering that pressure decreases as height increases. The amount involved is  $1025 - 995 = 30$  hPa, or  $30 \times 30 = 900$  feet. The altimeter is now indicating 10 000 feet above a datum which is 900 feet below MSL. The true altitude (actual height AMSL) of the aircraft must therefore be  $10\,000 - 900 = 9100$  feet. The altimeter indicates 10 000 feet but the true altitude is 9100 feet. The instrument is over-reading, and the aircraft is closer to the surface than is indicated. This is a potentially dangerous situation, occurring in flight from HIGH TO LOW pressure causing the altimeter to read HIGH. A datum diagram such as that shown in [Figure 5.8](#) helps to sort out this type of problem.

### Remember:

**Pressure always decreases as altitude increases.**  
**The altimeter indicates height above the datum set on the subscale.**

### Time Lag

With many types of altimeter the response to change of height is not instantaneous. This causes the altimeter to under-read in a climb and over-read in a descent. The lag is most noticeable when the change in altitude is rapid and prolonged. In the laboratory calibration of the sensitive altimeter, the lag between increasing readings and decreasing readings should not exceed 150 feet. With servo-assisted altimeters there is said to be no appreciable lag unless the rate of change of height exceeds 10 000 feet per minute. This is because the servo altimeter does not suffer from the linkage friction which causes a much larger error in the sensitive altimeter.

## Temperature Error.

Even with no other errors at all, the pressure altimeter will not indicate true altitude (height AMSL) unless the surface temperature and lapse rate of the column of air are those assumed in the calibration.

When flying in colder air (with an air density greater than ISA at that altitude), the altimeter will over-read.

Where the temperature at cruising level deviates from standard, an approximate correction can be made with most navigational computers. The correction can only be approximate since temperatures in the rest of the column of air are not known. The correction is considered too inaccurate to be worth making at heights above 25 000 feet.

## Example Problem

The indicated altitude is 10 000 feet with local pressure set and a COAT (corrected outside air temperature) of  $-25^{\circ}\text{C}$ . Will the true altitude be more or less than the indicated value?

## Solution

The ISA temperature at 10 000 feet would be about  $-5^{\circ}\text{C}$ , so the aircraft is flying in colder-than-standard conditions (ISA minus  $20^{\circ}$ ). Pressure decreases more rapidly in cold than in warm air. Therefore, assuming a constant surface pressure, the pressure at a given true altitude in the cold air will be less than at the same altitude in 'standard' air. The altimeter in the cold air will interpret this lower pressure as a higher altitude and will therefore over-read the true altitude. Using the computer set indicated altitude, 10 000 feet, against a COAT of  $-25^{\circ}\text{C}$  in the Altitude window. Read off the true altitude, about 9250 feet, on the outer scale against 10 000 feet on the inner scale. (An approximation of 4 feet per  $1^{\circ}\text{C}$  away from ISA per 1000 feet above sea level, may be used as a rough guide). Students should use the mathematical method for exam purposes.

Thus, in **flight from HIGH TO LOW TEMPERATURE** the altimeter would read **HIGH**. This is **potentially unsafe**, and is comparable to the case of barometric error illustrated in the previous worked example where with **flight from HIGH TO LOW PRESSURE** the altimeter also reads **HIGH**.

## Temperature Error Correction

Values to be added by the pilot to published altitudes (feet)

Aerodrome Temp $^{\circ}\text{C}$	Height above the elevation of the altimeter setting source													
	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
<b>-40</b>	40	60	<b>80</b>	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

Note:- The table is based on aerodrome elevation of 2000 ft; however it can be used operationally at any aerodrome.

Example: Decision height is 400 ft.  
 Aerodrome temperature is  $-40^{\circ}\text{C}$ .  
 From table correction = 80 ft.  
 Revised decision height = 480 ft

## Standard Datum Settings

### *Standard Setting*

When 1013.25 hPa is set on the subscale, the altimeter reading is called "Pressure Altitude". If 1013.25 hPa is set, an aircraft would normally fly Flight Levels. A Flight Level is the aircraft's height, above 1013.25 hPa, expressed in 100s of feet. Flight Levels only occur at 500 ft intervals. (e.g. 4500 ft will be FL45, or FL360 will be 36 000 ft.)

### *QFE*

This is aerodrome level pressure, which when set on the subscale, will cause the altimeter of an aircraft on the ground to read zero, assuming there is no instrument error. In flight, with QFE set, the altimeter will indicate height above the aerodrome QFE reference datum, provided ISA conditions exist between aerodrome level and the aircraft and there are no other altimeter errors. In practice, QFE is used mainly for circuit-flying and gives a good indication of height above the aerodrome, any errors involved being only small.

### *QNH*

This is the equivalent MSL pressure calculated by Air Traffic Control from the aerodrome level pressure assuming ISA conditions prevail between aerodrome level and MSL. With QNH set on the subscale, the altimeter of an aircraft on the aerodrome indicates aerodrome elevation, that is, the height AMSL (if there is no instrument error). In flight the altimeter will indicate altitude but this will only be the true altitude if the mean temperature in the column of air beneath the aircraft is the same as in ISA conditions (assuming there are no other altimeter errors). If conditions are different from standard, the indicated altitude, sometimes called QNH altitude, may deviate considerably from true altitude. The navigational computer can be used to make an approximate correction for this temperature error.

### *Regional QNH*

More correctly called 'lowest forecast QNH', this setting, forecast by the Met. Office, is used to ensure safe terrain clearance. It is the value below which QNH is forecast not to fall in a given period and area.

The value should be lower than the actual QNH anywhere in the area, and if set on the subscale, regional QNH will cause the altimeter to under-read (so erring on the safe side - the altimeter showing aircraft to be lower than it actually is).

### Blockages and Leaks

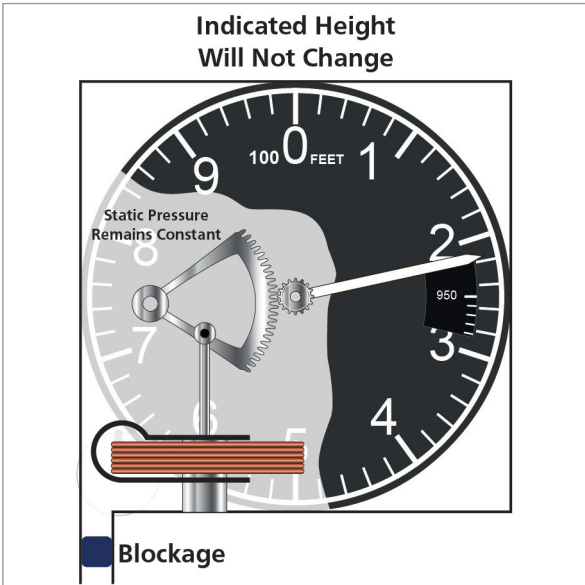


Figure 5.9 Static feed blocked

If the static source becomes blocked, the altimeter will not register any change in height - the height at which the blockage occurred will still be indicated regardless of any climb or descent. On many aircraft, an alternative source of static pressure will be available.

Should the static line fracture in a pressurized aircraft, the altimeter will show the (lower) cabin altitude rather than aircraft altitude

A fracture in the static line within an unpressurized aircraft will normally result in the altimeter over-reading, due to the pressure in the cabin being lower than ambient due to aerodynamic suction. See Chapter 2 Emergency Static Source.

If the aircraft is CLIMBING then the altimeter will UNDER-READ.

If the aircraft is DESCENDING then the altimeter will OVER-READ.

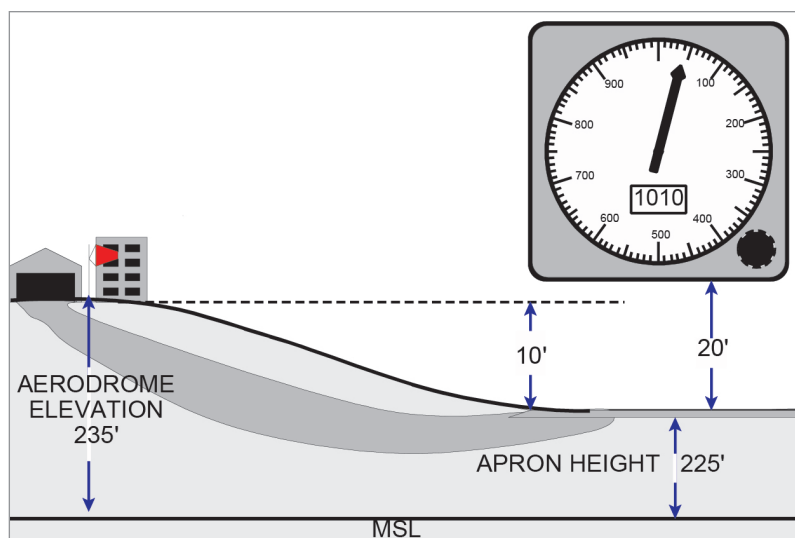
The amount of the error will increase as the aircraft moves away from the height at which the blockage occurred.

### Density Altitude

Density altitude can be defined as the altitude in the standard atmosphere at which the prevailing density would occur, or alternatively, as the altitude in the standard atmosphere corresponding to the prevailing pressure and temperature. It is a convenient parameter in respect of engine performance figures.

### Preflight Altimeter Checks

In the UK, the apron is the designated location for pre-flight altimeter checks (the apron being the loading and unloading and/or parking area). Apron elevation is displayed in the flight clearance office of the aerodrome concerned and is also published in the AGA section of the UK Air Pilot.



## Example Problem

Calculate the instrument error from the following details of a preflight altimeter check:

Aerodrome elevation	235 feet
Apron elevation	225 feet
Height of altimeter above apron	20 feet
Altimeter reading with QFE set	40 feet

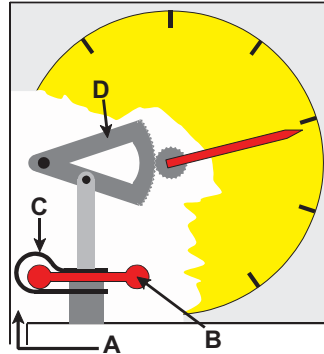
## Solution

The apron is 10 feet below the stated aerodrome elevation so assuming the QFE to be for the aerodrome level, an altimeter on the apron should read (-10) feet. However, the instrument is positioned in the aircraft 20 feet above the apron so it should show  $(-10) + 20 = +10$  feet. Its actual reading is +40 feet so it is over-reading by 30 feet, an instrument error of +30 feet.



Questions

1. The diagram below shows a simple altimeter. The parts labelled A, B, C and D are:



- a. pitot pressure inlet, linkage mechanism, bellows, quadrant  
 b. air inlet, temperature compensator, leaf spring, linkage mechanism  
 c. static pressure inlet, partially evacuated capsule, linkage mechanism, subscale setting device  
 d. static pressure inlet, partially evacuated capsule, leaf spring, linkage mechanism
2. In the International Standard Atmosphere, the mean sea level pressure is ....., the lapse rate of temperature ..... between MSL and ..... and is isothermal up to ..... The numbers missing are:
- a. 1225 hPa; 2° per 1000 ft; 37 000 ft; 66 000 ft  
 b. 1013.25 hPa; 1.98°C per 1000 ft; 36 090 ft; 65 617 ft  
 c. 1013.25 hPa; 1.98°C per 1000 ft; 36 090 ft; 104 987 ft  
 d. 1225 hPa; 1.98°C per 1000 ft; 36 090 ft; 104 987 ft
3. An aircraft taking off from an airfield with QNH set on the altimeter has both static vents blocked by ice. As the aircraft climbs away the altimeter will:
- a. read the airfield elevation  
 b. indicate the aircraft height amsl  
 c. read the height of the aircraft above the airfield  
 d. show only a very small increase in height
4. When flying from low pressure to high pressure, without resetting the altimeter datum, the barometric error of an altimeter will cause the instrument to:
- a. read the true altitude, providing a correction is made for temperature  
 b. over-read the true altitude of the aircraft  
 c. indicate a higher altitude than the correct one  
 d. under-read the true altitude of the aircraft

5. The errors affecting the pressure altimeter are:
- instrument position, manoeuvre induced, density, temperature, lag
  - instrument, pressure, manoeuvre induced, density, temperature, lag
  - instrument, position, manoeuvre induced, temperature, barometric, lag
  - instrument, pressure, lag, barometric, temperature, compressibility
6. An altimeter with ..... set on the subscale will indicate ....., but with ..... set, the altimeter will show .....
- 1013; pressure altitude; QNH; altitude
  - QNE; pressure altitude; QNH; height above airfield datum
  - QFE; height above the airfield datum; 1013; height amsl
  - QNH; height above touchdown; 1013; height amsl
7. An aircraft has one altimeter set to QFE and one to aerodrome QNH 1000 hPa. If the airfield elevation is 300 ft, immediately before take-off the altimeter with QFE set will read ..... and the other ....., If the QFE altimeter is set to 1013 when passing through the transition altitude 3000 ft, it will read ....., (Assume 1 hPa = 30 ft).
- 300 ft; zero; 2610 ft
  - zero; 300 ft; 3390 ft
  - zero; 300 ft; 3690 ft
  - zero; 300 ft; 2610 ft



## Answers

1	2	3	4	5	6	7
d	b	a	d	c	a	b

Chapter

6

The Vertical Speed Indicator

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## Introduction

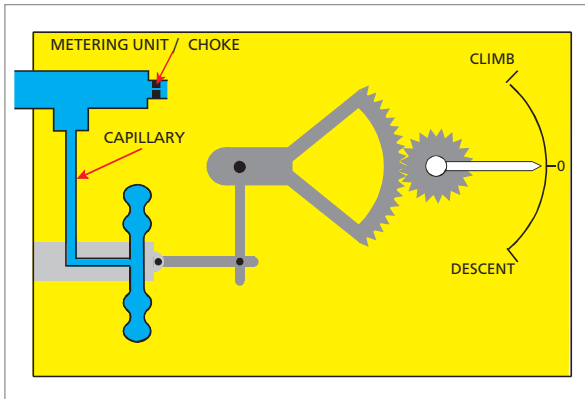


Figure 6.1

A pilot can get some idea of his rate of climb or descent from the angular rate of change of the altimeter pointer. However, there are times when a more accurate indication is necessary, such as achieving a certain height loss within a specified time on airways or in setting up a smooth rate of descent on a glide path on an instrument approach.

The Vertical Speed Indicator (VSI) displays rate of climb or descent. The instrument senses rate of change of static by comparing the present static pressure with the static pressure measured 4-6 seconds earlier.

The VSI can also be known as the Rate of Climb and Descent Indicator (RCDI).

## Principle

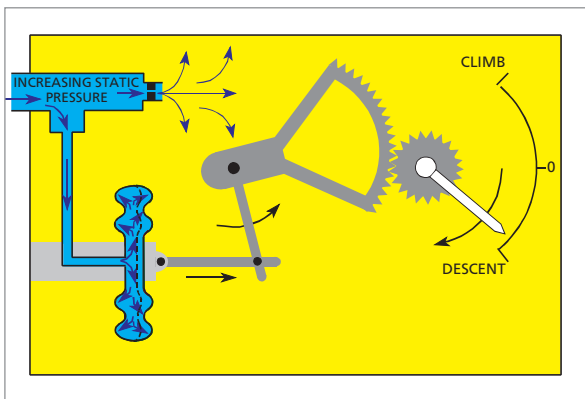


Figure 6.2

When an aircraft departs from level flight, the static pressure will change. The VSI measures the pressure difference between each side of a restricted choke/metering unit.

In level flight the pressures on each side of the choke are the same, during a climb or descent, air fed to the choke immediately responds to the change of atmospheric pressure but the choke transmits this change at a lower rate.

## Construction

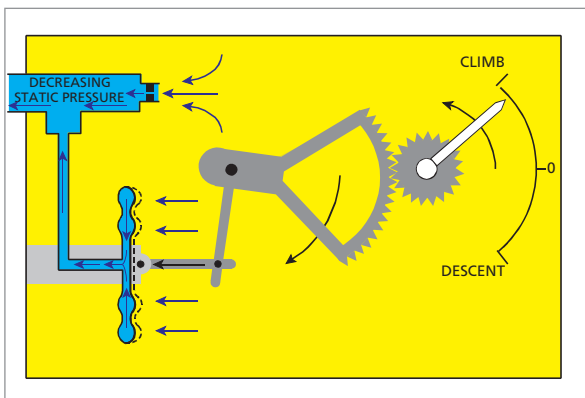


Figure 6.3

A capsule within an airtight case is fed with static pressure. The case is also fed with static pressure but through a restricted choke, thus if the static pressure is changed the pressure surrounding the capsule changes at a slower rate than that within the capsule, as shown in *Figure 6.2* and *Figure 6.3*. For example, if the aircraft is climbing, the pressure in the capsule will be less than that in the case, the consequent compression of the capsule is converted by a suitable linkage to a pointer indication of rate of climb.

## VSI Metering Unit

The restrictor (or choke, or metering unit) is more complicated than a simple hole. The static pressure change is actually hectopascals/min, but the pilot needs to see an indication of feet/min, whatever the altitude. However, there are more feet to a hectopascal at higher altitudes than lower altitudes, and therefore the rate of pressure change with altitude at different altitudes needs to be compensated for.

This is achieved by a combination of different types of hole (called 'capillary' and 'orifice'). Use of a suitable combination of these gives a near-constant indication in feet/minute, whatever the actual altitude and therefore, whatever the actual pressure differential needed.

## Display



Figure 6.4 The VSI Display

## The Errors of the VSI

**Instrument Error.** Due to manufacturing imperfections.

**Position (or Pressure) Error.** If the static pressure is subject to position error the VSI will wrongly indicate a climb or descent when speed is suddenly changed; this is most noticeable during take-off acceleration.

**Manoeuvre-induced Error.** Any short term fluctuations in pressure at the static vent during attitude changes will cause the instrument to indicate a false rate of climb or descent.

Additionally with most VSIs, the linkage includes a small counterbalance weight, the inertia of which causes delays in the indications of changes in vertical speed during manoeuvres.

**Time Lag.** The pointer takes a few seconds to steady because of the time taken to build up a steady pressure difference on climb or descent. There will also be a time lag on levelling out because of the time taken for the pressures to equalize. This error is most noticeable after a prolonged climb or descent, especially at a high rate.



Any blockages of the static line or vent will cause the needle to return to zero. If the supply of air to this instrument is blocked it is probable that the other pressure instruments (ASI, altimeter and Machmeter) will also be affected.

## The Instantaneous Vertical Speed Indicator

To overcome the problem of lag, the Instantaneous Vertical Speed Indicator (IVSI) incorporates an accelerometer unit (sometimes called a dashpot or dynamic vane) which responds quickly to a change of altitude.

The figure below shows an IVSI at the beginning of a descent.

The piston in the vertical acceleration pump immediately rises in the cylinder and causes a temporary increase of pressure in the capsule. The capsule expands and the pointer will give an instant indication of descent.

As the initial acceleration is turned into a steady rate of descent, the piston will slowly descend to its original position, but by this time the correct differential pressure between the capsule and the case will have been set up and the correct rate of descent will continue to be shown.

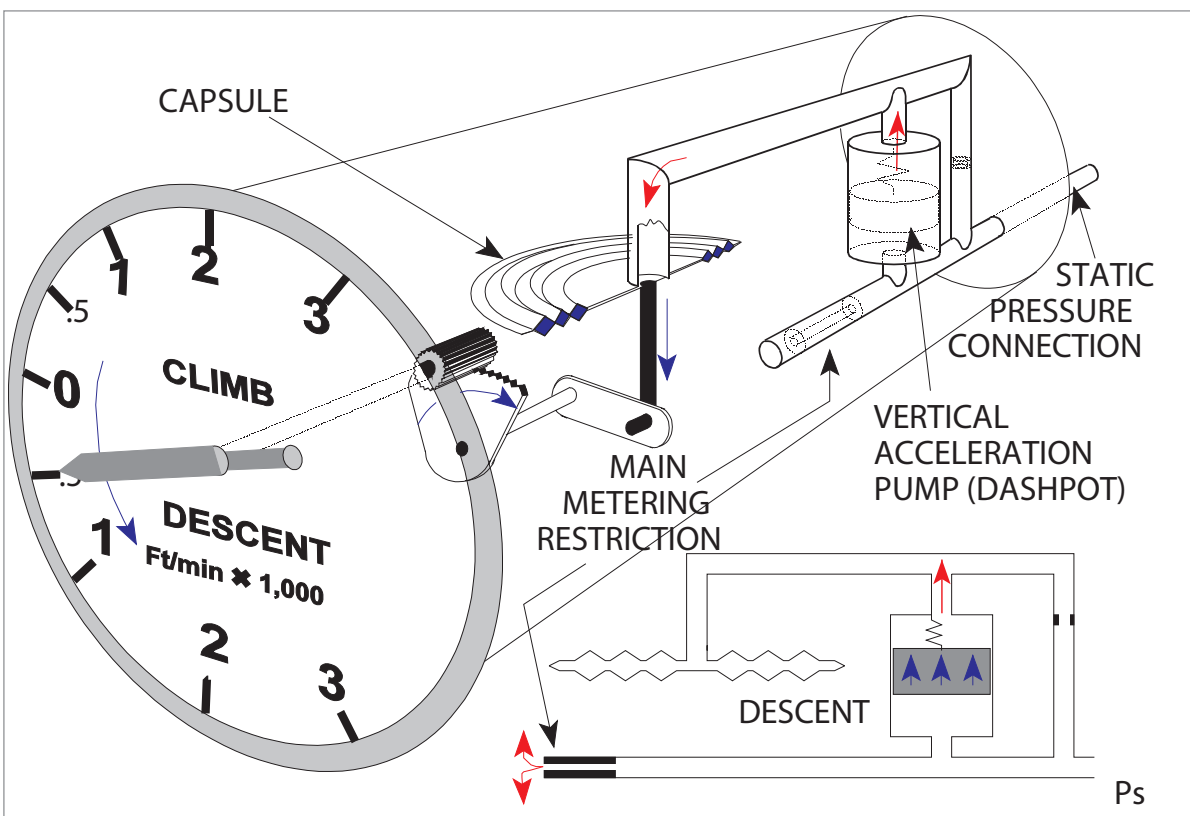


Figure 6.5 The Instantaneous Vertical Speed Indicator

**Errors Peculiar to the IVSI.** Because of the sensitivity of the dashpot assembly, the instrument tends to overreact to turbulent flying conditions and small fluctuations should be ignored.

In a steep, level turn, the piston will tend to sink towards the bottom of the cylinder and there will be a false indication of a climb.

## Presentation

Two types of presentation are available, a linear scale and a logarithmic scale, this latter presentation being more easily read at the lower rates of climb/descent. This is easily discerned from the figures below.

It should be noted that diaphragm overload stops may be fitted to prevent damage to the instrument should the rate of climb/descent exceed the maximum to which the instrument is calibrated. On some instruments a zeroing screw is fitted.

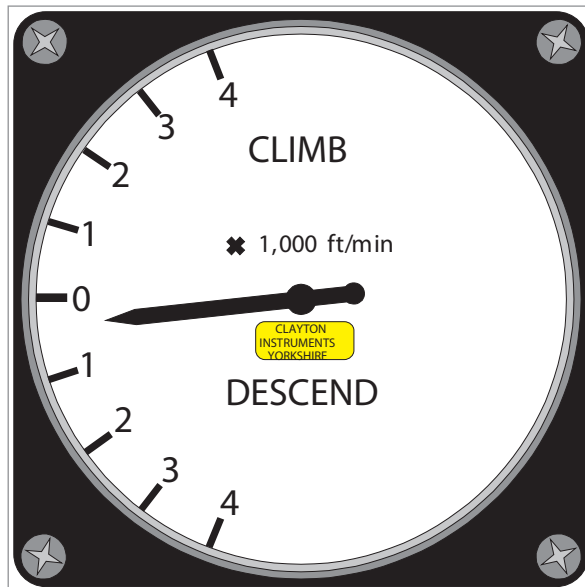


Figure 6.6 Linear Scale



Figure 6.7 Logarithmic Scale

## Serviceability Checks

### On the Ground

- The instrument should read zero, or the error should be within the permissible limits
  - +/- 200 feet per minute at temperatures  $-20^{\circ}\text{C} + 50^{\circ}\text{C}$
  - +/- 300 feet per minute outside these temperatures
- There should be no apparent damage to the instrument.

### In the Air

The accuracy of the instrument may be checked against the altimeter and a stopwatch during a steady climb/descent and the instrument should indicate zero climb or descent when in level flight.



## Questions

1. **The vertical speed indicator indications may be in error for some seconds after starting or finishing a climb or descent. The error is a result of:**
  - a. a combination of time lag and manoeuvre induced errors
  - b. a combination of position error and manoeuvre induced errors
  - c. manoeuvre induced errors only
  - d. a combination of time lag and instrument error
2. **The advantage of having the VSI dial presentation in logarithmic spacing rather than in linear spacing is that:**
  - a. at low rates of climb or descent the pointer movement is much larger and so is more easily read
  - b. readings are instantaneous
  - c. a greater range of rates of climb and descent is shown
  - d. the internal mechanism is simplified by deletion of the calibration choke
3. **In the IVSI, lag error:**
  - a. is overcome by feeding a sample of static pressure to the case and delaying it to the capsule
  - b. is overcome by using a special dashpot accelerometer assembly
  - c. is overcome by the use of logarithmic presentation
  - d. is only overcome when initiating a climb or descent
4. **Because the VSI measures rates of change of static pressure and not actual values of static pressure, position error:**
  - a. never affects VSI indications
  - b. may cause errors in the VSI during the take-off run
  - c. may cause errors in VSI indications whenever airspeed is change
  - d. may cause errors in VSI indications whenever airspeed is changed, even if there is no change in position error
5. **When entering a steep turn, an IVSI is likely to show:**
  - a. no change in altitude
  - b. a climb
  - c. a descent
  - d. a slight descent at high airspeed only
6. **If the static vent becomes blocked during a climb:**
  - a. the VSI will stop at the rate of climb of the aircraft at the time of blockage
  - b. the VSI will indicate a decreasing rate of climb
  - c. the VSI will return to zero
  - d. the VSI will indicate an increasing rate of climb
7. **In conditions of clear air turbulence:**
  - a. the standard VSI is more sensitive
  - b. the IVSI is more sensitive
  - c. both types will react the same
  - d. the vertical acceleration pump will not be affected

8. **Change of temperature as an aircraft climbs or descends:**
- a. will affect VSI readings whenever temperature lapse rate differs from standard conditions
  - b. is compensated at the metering unit by means of a capillary and orifice
  - c. has no effect on the VSI as only static pressure is used in this instrument
  - d. may be allowed for by use of tables or computer
9. **Permissible limits of accuracy of the VSI are ..... when ..... within a temperature range of ..... and ..... outside this range.**
- a. + 250 fpm, on the ground, -20°C to +50°C, +/- 300 fpm
  - b. + 200 fpm, at any height, -20°C to +30°C, +/- 300 fpm
  - c. + 250 fpm, at any height, -20°C to +50°C, +/- 300 fpm
  - d. + 200 fpm, on the ground, -20°C to +50°C, +/- 300 fpm

## Answers

1	2	3	4	5	6	7	8	9
a	a	b	b	b	c	b	b	d

# Chapter 7 The Machmeter

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## High Speed Flight

In high speed aircraft the **Machmeter is an essential instrument**. As an aircraft approaches the local speed of sound the airflow over some parts of the fuselage or wings may be accelerated up to the speed of sound and a **shock wave** will form. These shock waves cause more drag, less lift, Mach tuck, buffeting, and reduction in control effectiveness or loss of control. (Mach tuck is a downward-pitching sudden change of trim which can be severe).

In order to avoid danger associated with flight at high Mach numbers, a limiting Mach number will be specified for each aircraft, based on flight trials. This must not be exceeded. It is known as  $M_{MO}$ .

The Machmeter therefore displays the present Mach number so that the pilot can keep his speed below the particular  $M_{MO}$  for his aircraft and avoid the problems associated with high speed flight.

## Speed of Sound

The **speed of sound** is not constant but **varies with air temperature**. A formula for calculating the local speed of sound (LSS) is:

$$LSS = 38.95 \sqrt{T}$$

where,

LSS is given in knots,

38.95 is a constant,

and

T is the **absolute temperature**, ( $0^{\circ}\text{C} = 273^{\circ}\text{A} = 273 \text{ K}$ )

Therefore the higher the air temperature, the higher the speed of sound, and vice versa. Since temperature normally reduces as altitude increases, the speed of sound normally reduces as altitude increases.

In ISA conditions at mean sea level ( $+15^{\circ}\text{C}$ ) the speed of sound is 661 knots, while at 30 000 feet ISA ( $-45^{\circ}\text{C}$ ) the speed of sound will have reduced to 589 knots.

## Machmeter Principle of Operation

The Machmeter uses two capsules and linkages to indicate the aircraft's True Airspeed (TAS) as a proportion of the local speed of sound (LSS)

The first capsule is an Airspeed Capsule which will expand and contract as a result of changes in the dynamic pressure.

$$\text{Mach Number} = \frac{\text{TAS}}{\text{LSS}}$$

The second capsule is a sealed Altimeter Capsule which will expand and contract as the static pressure inside the instrument case changes.

However, MN is proportional to....

$$\frac{\frac{D}{\rho}}{\frac{S}{\rho}}$$

As density ( $\rho$ ) cancels out, we can see that:-

MN is proportional to.....  $\frac{D}{S} = \frac{P-S}{S}$

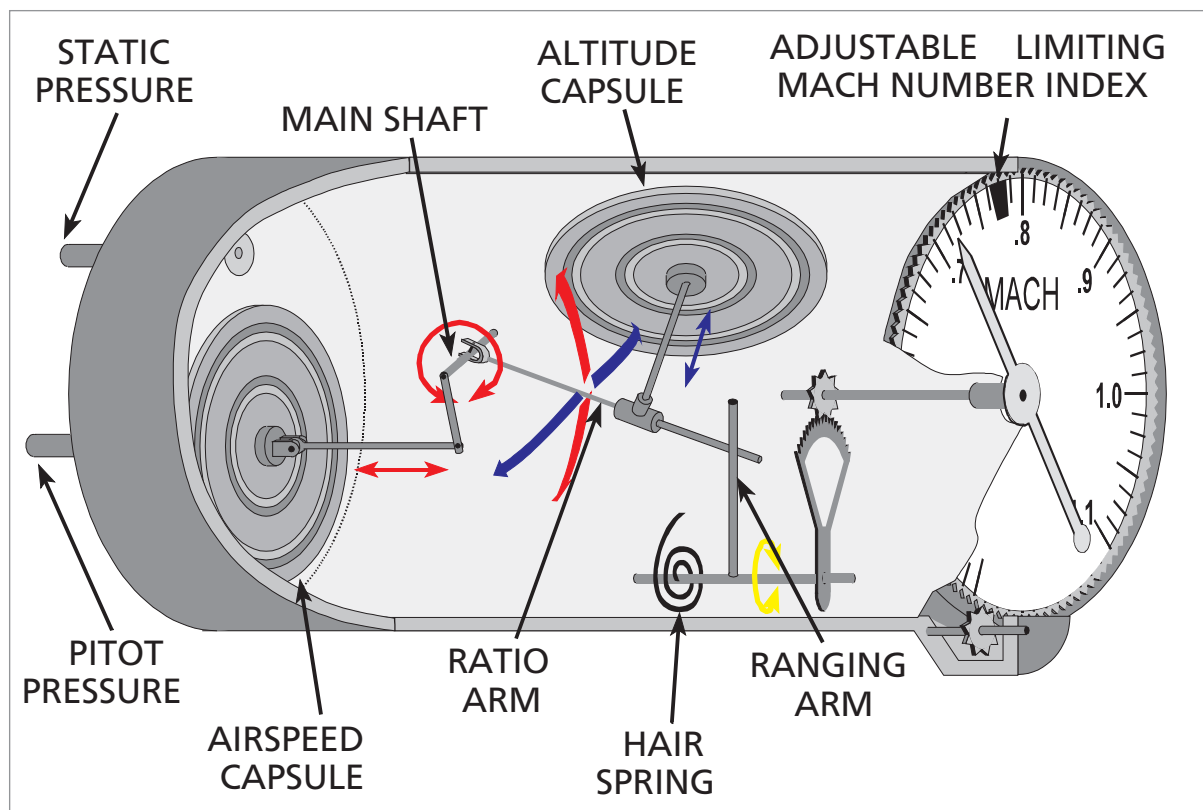


Figure 7.1 The Machmeter

## Machmeter Construction

*Figure 7.1* shows the parts of a Machmeter. It consists of a simple aneroid **altitude capsule** and an **airspeed capsule** which are connected at the ratio arm.

Static pressure enters the case of the instrument, while pitot pressure is fed directly into the airspeed capsule. Expansion or contraction of the airspeed capsule is transmitted via the airspeed link and **main shaft** to the **ratio arm**, but the position of the ratio arm is also governed by expansion or contraction of the altitude capsule.

A spring-loaded **ranging arm** transmits the movement of the ratio arm to the pointer mechanism.

Basically, if either or both **capsules expand** (due to an increase in IAS and/or altitude) then the ranging arm will rotate out of the diagram and the indicated **Mach Number will increase**. If airspeed or altitude reduce then the ratio and ranging arms move back into the paper, and a lower Mach number is displayed.

An **adjustable index** on the instrument scale can be positioned by means of a small knob. This index can be set to the **limiting Mach number** for the aircraft type (in straight and level flight), to provide a visual warning to the pilot.

## Machmeter Errors

The Machmeter suffers from **instrument, position and manoeuvre induced errors only**. It does **not** suffer from temperature or density errors, as these errors cancel out. In addition since compressibility error depends on dynamic/static pressure, and the instrument is calibrated to this ratio, compressibility error is calibrated out.

### *Position Error*

The instrument uses the same sources of pitot and static pressure as the ASI and therefore suffers from position error caused by disturbed airflow at the pitot head and/or static vent. At low Mach numbers below, careful design and positioning of the pressure sources ensure that position error on modern jet aircraft is small. However, at higher Mach Numbers, changes in airflow may cause position error to become bigger and possibly change its sign. If the sign of the position error is such that the Machmeter under-reads, the error could become dangerous at high Mach numbers. The normal arrangement in modern jet transport aircraft is to allow for instrument and position error such that the Machmeter **always over-reads**.

### *Manoeuvre Induced Error*

The Machmeter will suffer an additional, unpredictable error whenever the aeroplane manoeuvres. This is due to the unpredictable changes in the airflow over the static source.

## Blockages

**Static Source Blocked.** If the static source is blocked, the pressure in the Machmeter case will not alter when the aircraft climbs or descends.

If a blockage occurs in a **climb**, the altitude capsule will not move. Assuming a constant IAS (and therefore a constant dynamic pressure) the airspeed capsule will contract as the static component of pitot pressure reduces. The Machmeter will therefore **under-read**.

If a blockage occurs in a **descent**, at a constant IAS, the airspeed capsule will expand due to the increasing static component of pitot pressure. The Machmeter will therefore **over-read**.

**Pitot Source Blocked.** Assuming a climb or descent at a constant IAS (and therefore a constant dynamic pressure) the Machmeter will **over-read in the climb** and **under-read in the descent**. In the climb the airspeed capsule will expand in error because the static component of pitot in the capsule will be greater than the static in the case. In the descent the static component of pitot will be too small and therefore the airspeed capsule will contract.

*Note:* It may be noticed that the Machmeter blockage errors are the same as the ASI blockage errors.

## Abbreviations

**MMR** Machmeter reading, the uncorrected reading

**IMN** Indicated Mach number, MMR corrected for instrument error (the values quoted in Flight Manuals are normally IMN)

**TMN** True Mach number, IMN corrected for position error  $M_{MO}$ . There is much less risk of an over-speed condition arising when this is available.

## Machmeter Summary

Mach number = TAS/LSS.

Speed of sound is proportional to the square root of the absolute temperature, and therefore decreases with the decrease in temperature normally encountered with increase in altitude.

While climbing at a constant Mach number, TAS decreases and CAS decreases more rapidly, the LSS also decreases.

While climbing at a constant CAS, TAS and Mach number increase but the LSS decreases.

Remember that in calculations involving the **Jet Standard Atmosphere**, the temperature is assumed to be +15°C at MSL with a lapse rate of 2° per 1000 with **no** upper limit (i.e. no tropopause).

## Climb at a Constant CAS in Standard (ISA) Atmosphere

If we were to climb at 330 kt CAS from sea level to 36 000 ft in the standard atmosphere,

- TAS will increase from 330 kt to 593 kt, and
- Mach number will increase from M 0.5 to M 1.05.

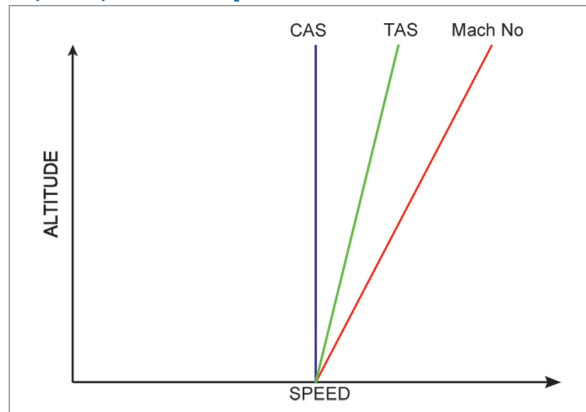


Figure 7.2

The rapid rise of Mach number (in this case far exceeding  $M_{MO}$ ) is the reason why high performance aircraft are flown on CAS (or IAS) for the first part of the climb before transferring to a constant Mach number for the rest of the climb. Similarly in the descent at constant CAS, TAS and Mach number reduce, with Mach number reducing at a greater rate.

This is shown diagrammatically in [Figure 7.2](#). For a constant CAS (blue line) as altitude increases, TAS (the green line) increases, and Mach number (the red line) increases at a greater rate. The navigation computer can also be used to show the relationship between CAS, TAS and Mach number but also gives us an idea of magnitude.

Now consider a descent at M 0.8 from 40 000 ft to sea level in the jet standard atmosphere on the navigation computer.

At 40 000 ft M 0.8 is 450 kt TAS, at sea level it has increased to 528 kt. The CAS has increased more markedly from 242 kt at 40 000 ft to 528 kt at mean sea level. This would exceed  $V_{MO}$ . Therefore although Mach number is used at altitude, CAS will be used in the descent.

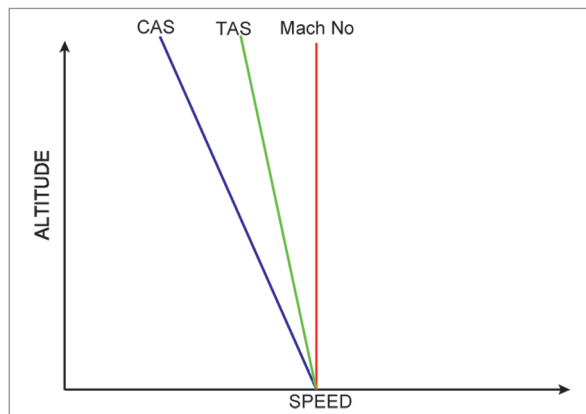


Figure 7.3

**Note:** You will have probably noticed by now that the relationship of CAS, TAS and Mach number as an aeroplane climbs or descends through the standard atmosphere remains the same. That is [Figure 7.2](#). and [Figure 7.3](#). are the same - just tilted to one side or the other. Therefore when considering the climb/descent through an isothermal layer and an inversion only the constant TAS figure will be shown.

## Descent at a Constant Mach Number in Standard Conditions

During a descent in the ISA, the LSS will be increasing (as temperature increases). Therefore if Mach number is being kept constant the TAS must be increasing (Mach number = TAS/LSS). During the descent air density increases and if TAS is also increasing the CAS must also increase at a greater rate (Dynamic Pressure =  $\frac{1}{2} \rho V^2$ ). This is shown in [Figure 7.3](#). Similarly in a climb at constant Mach number the TAS or CAS both reduce.

## Climb and Descent through an Isothermal Layer

### Constant Mach number

An isothermal layer is a layer of air in which the temperature does not change. Therefore the LSS will not change, and for a constant Mach number the TAS will not alter. The CAS will change however due to density error, reducing during the climb and increasing during a descent.

### Constant CAS

Climbing at a constant CAS, the TAS and Mach number will both increase (at the same rate).

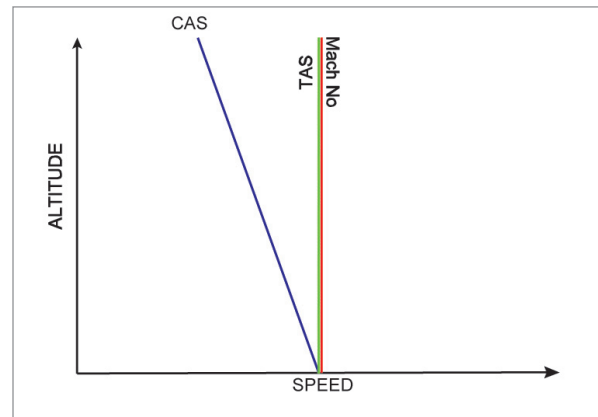


Figure 7.4

## Climb and Descent through an Inversion

### Constant Mach Number

In an inversion the temperature of the air will increase (get warmer) as altitude increases. Therefore in a climb the LSS will increase, and for a constant Mach number the TAS will increase.

(Mach number = TAS/LSS).

CAS will reduce as air density reduces. Descending at a constant Mach number the TAS will reduce and the CAS will increase.

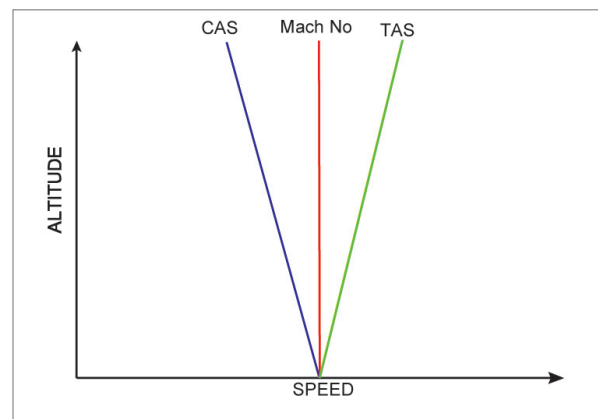


Figure 7.5

### Constant CAS

Climbing at a constant CAS, the TAS and Mach number will both increase (TAS at a greater rate than Mach number).

## Climb/Descent Summary

In summary;

TAS will always increase when an aeroplane climbs at a constant CAS.

Climbing at a constant TAS the CAS will always reduce.

This is because pressure has a greater effect on air density than temperature.

Climbing at a constant CAS the Mach number will always increase.

Climbing at a constant Mach number the CAS will always reduce.

This is because the CAS/TAS density error dominates over the change in LSS due to temperature variation.

## Example Problems Associated with the Machmeter

**Problem 1:** What is the speed of sound at FL380 in ISA conditions?

**Solution 1:**

In the ISA atmosphere FL380 is above the tropopause and therefore the temperature will be -56.5°C or 216.5 K

$$\begin{aligned} \text{LSS} &= 38.95 \sqrt{T} \\ &= 38.95 \sqrt{216.5} \\ &= \mathbf{573 \text{ knots}} \end{aligned}$$

These calculations can also be completed on the navigation computer. Place the Mach number index arrow against the temperature (in °C), locate M 1.0 (the blue 10 on the navigation computer) on the inner Mach number scale and read off the TAS on the outer scale.

**Problem 2:** Determine the TAS corresponding to M 0.70 at JSA MSL (+15°C or 288 K).

**Solution 2:**

Using the computer, set the Mach number index against +15°C in the Airspeed window. Against 7 (for M 0.7) on the inner scale, read off the answer (463 knots) on the outer scale.

Alternatively calculate TAS from the formula

$$\begin{aligned} \text{TAS} &= \text{Mach number} \times \text{LSS} \\ &= 0.7 \times 38.95 \sqrt{288} \\ &= 0.7 \times 661 = \mathbf{463 \text{ knots}} \end{aligned}$$

**Problem 3:** Calculate without using a computer the altitude in the JSA atmosphere at which a TAS of 450 knots corresponds to Mach 0.80

**Solution 3:**

$$\begin{aligned} \text{Mach Number} &= \frac{\text{TAS}}{\text{LSS}} & \text{LSS} &= \frac{\text{TAS}}{\text{MN}} \\ \text{LSS} &= \frac{450}{0.8} = 562.5 \text{ kt} \end{aligned}$$

$$\text{However, } \text{LSS} = 38.95 \sqrt{T} \quad \sqrt{T} = \frac{\text{LSS}}{38.95} = \frac{562.5}{38.95} = 14.44$$

$$T = 14.44^2 = 209^\circ$$

**T = 209° Absolute which is equal to - 64°C**

- 64°C occurs at **FL395** in the JSA which has no tropopause.

**Problem 4:** If a decrease of 0.12 in the Mach number results in a decrease of 80 knots in the TAS, what is the local speed of sound?

**Solution 4:**

$$\text{Mach Number} = \frac{\text{TAS}}{\text{LSS}} \qquad \text{LSS} = \frac{\text{TAS}}{\text{MN}}$$

$$\text{LSS} = \frac{80}{0.12} = 667 \text{ kt}$$

**Problem 5:** An aircraft is flying at FL360 with a TAS of 467 knots at Mach 0.8 when the temperature difference from JSA is +9. What is the temperature difference at FL320 if Mach 0.8 still gives a TAS of 467 knots?

**Solution 5:**

At FL360 in JSA temp would be - 57°C. JSA +9 would be - 48°C

However if Mach No. and TAS remain the same, then we must be flying in an **isothermal layer**, so the temperature of - 48°C and the LSS would remain the **same**.

If the temperature at FL320 is also - 48°, temp deviation from standard must be + 1° as JSA should be - 49°C.

## Mach / Airspeed Indicator

Since many commercial aircraft require indications of both IAS and Mach number, it is sensible to combine both instruments. The basic principals of both instruments still apply.

Errors

The combined instrument will have the errors of both the Machmeter and the airspeed indicator, namely; instrument, position, manoeuvre induced, density and compressibility errors.

## Construction

**There are two types of Mach/Airspeed Indicator:**

- A self-contained instrument fed from pitot and static sources.
- A combined instrument fed from the Air Data Computer.

**Note that:**

- The airspeed pointer moves clockwise over a fixed scale.
- From M 0.5 the Mach number is read off the same pointer as it moves over a moving Mach number scale. This scale rotates anticlockwise beneath the pointer as Mach number increases.
- A second striped needle may be present to mark  $V_{MO}$ .



If the aircraft is fitted with an Air Data Computer (ADC) it will measure pitot pressure, static pressure and Total Air Temperature and then electronically, send the information to any instruments and other computers which require it. The advantages related to the combined Mach/Airspeed instrument are:

- The ability to correct for instrument and position errors to give Calibrated Airspeed (CAS) instead of IAS.
- The use of a digital displays for both Mach number and CAS.

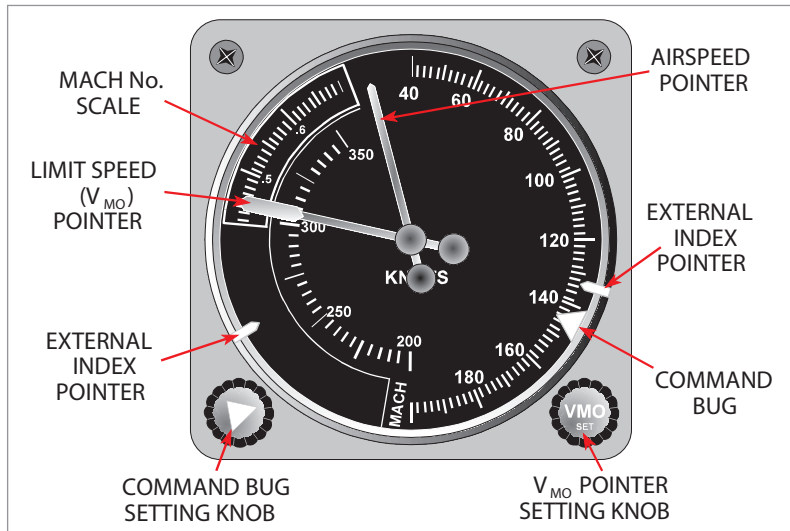


Figure 7.6 A Mach/Airspeed Indicator fed from Pitot and Static Sources

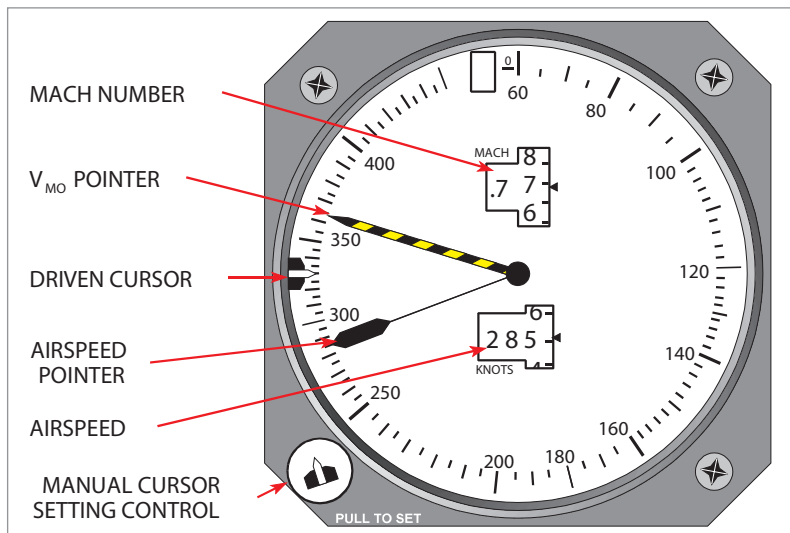


Figure 7.7 A Combined Instrument fed from the Air Data Computer

## Questions

1. **The local speed of sound is equal to:**  
(K = Constant)
  - a.  $K \sqrt{\text{temperature } (^{\circ}\text{F})}$  knots
  - b.  $K \sqrt{\text{temperature (K)}}$  knots
  - c.  $K \sqrt{\text{temperature } (^{\circ}\text{C})}$  knots
  - d.  $K \sqrt{\text{temperature (K)}}$  metres per second.
  
2. **At FL350 with a JSA deviation of -12, the true airspeed when flying at M 0.78 is:**
  - a. 460 kt
  - b. 436 kt
  - c. 447 kt
  - d. 490 kt
  
3. **When climbing at a constant mach number below the tropopause through an inversion:**
  - a. the CAS and TAS will both increase
  - b. the CAS and TAS will both decrease
  - c. the CAS will decrease and the TAS will increase
  - d. the CAS will increase and the TAS will decrease
  
4. **When descending below the tropopause under normal conditions (increasing temperature) at a constant CAS:**
  - a. both TAS and mach number will decrease
  - b. both TAS and mach number will increase
  - c. the TAS will decrease and the mach number will increase
  - d. the TAS will increase and the mach number will decrease
  
5. **Cruising at FL390, M 0.84 is found to give a TAS of 499 kt. The ISA deviation at this level will be:**
  - a. -17
  - b. +17
  - c. +19
  - d. -19
  
6. **The errors to which the Machmeter is subject are:**
  - a. instrument error, position error, compressibility error and manoeuvre induced error
  - b. instrument error, position error and manoeuvre induced error
  - c. instrument error, position error, barometric error, temperature error and manoeuvre induced error
  - d. instrument error, position error, density error and manoeuvre induced error

7. The relationships between TAS, mach number (MNo) and local speed of sound (LSS) is:
- a.  $LSS = \frac{MNo}{TAS}$
  - b.  $MNo = \frac{LSS}{TAS}$
  - c.  $TAS = MNo \times LSS$
  - d.  $MNo = LSS \times TAS$
8. The Machmeter gives an indication of mach number by measuring the ratio:
- a.  $\frac{\text{pitot pressure}}{\text{static pressure}}$
  - b.  $\frac{\text{static pressure}}{\text{dynamic pressure}}$
  - c.  $\frac{\text{dynamic pressure}}{\text{pitot pressure}}$
  - d.  $\frac{\text{dynamic pressure}}{\text{static pressure}}$
9. An aircraft is flying at FL350 with a JSA deviation of +8. The Mach No. is 0.83 and the TAS 485. If the aircraft descends to FL300 and maintains the same Mach No. and TAS, the JSA deviation will now be:
- a. +8
  - b. -2
  - c. +2
  - d. -18

## Answers

1	2	3	4	5	6	7	8	9
b	b	c	a	b	b	c	d	b

## Chapter

# 8

## Air Data Computer

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Introduction

In many large aircraft currently in service, the conventional pressure instruments which show altitude, airspeed and Mach Number(MNo) are replaced by indicators displaying information generated by a central computer, the Air Data Computer (ADC). The computer unit and displays, together with the sensors of the basic data of pitot pressure, static pressure and air temperature, and a power-pack, form the aircraft's Air Data System (ADS). Whilst such a system is self-contained, its outputs are essential to the operation of the aircraft's Automatic Flight Control System (AFCS). ADS outputs may also be used in the altitude transponder, flight data recorder, navigation computer and more.

A number of different aircraft types may use the same basic Air Data Computer and this device will need to be integrated into the aircraft systems and this is achieved by a "Configuration Module". The module can be calibrated to take into account differences in pressure/temperature gathering efficiencies due to positioning of the gathering probes and this information can then be entered into the computer to obtain the most accurate indications possible.

The standard ADS instruments show altitude, vertical speed, airspeed and MNo. Additional instruments can display Total Air Temperature (TAT), Static Air Temperature (SAT) and TAS. A schematic diagram of a conventional ADS is shown in Figure 8.1.

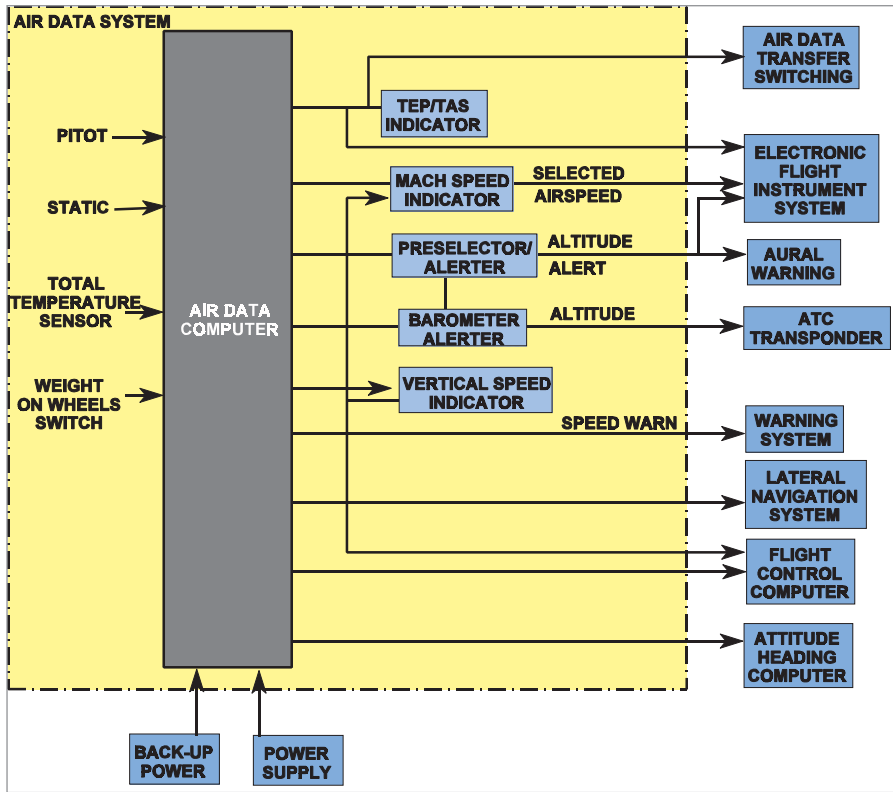


Figure 8.1 Conventional System

**Notes:** The weight on wheels switch decouples the stall warning system when the aircraft is on the ground.

AOA may also be an input to the ADC for use in some aircraft systems.

## Pitot - Static System

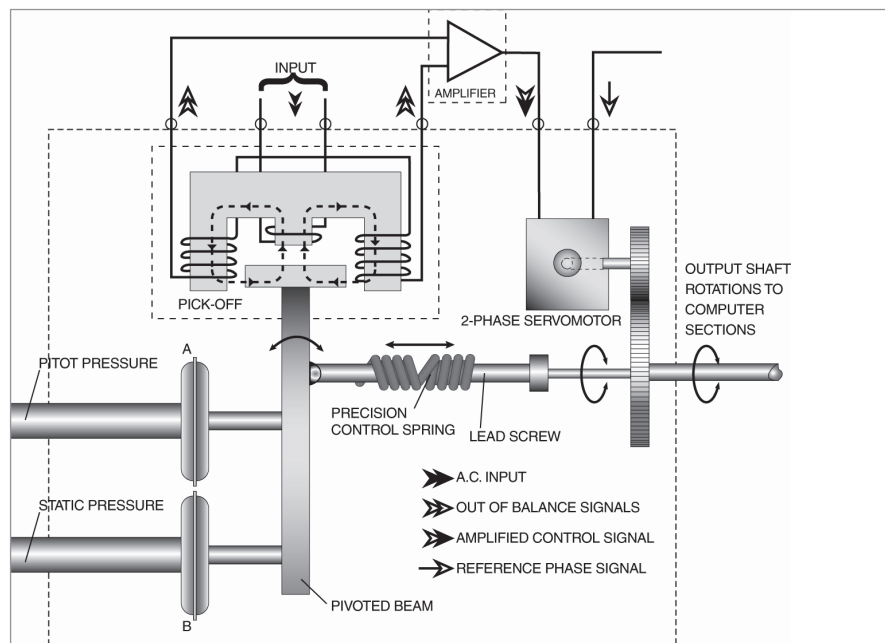
In a typical aircraft, identical sets of air data instruments are provided on the Captain's and First Officer's instrument panels. Each set of instruments is connected to one of two (allowing redundancy) ADCs fed from independent pitot and static sources, which can be cross-connected, as shown in *Figure 8.3*. In addition to the indicators powered by the two ADCs there is a standby barometric altimeter and a standby airspeed indicator, fed direct from pitot and static sources separate from those used for the ADCs. Each of the three independent pitot-static systems makes use of cross-coupled static vents located on each side of the fuselage. This arrangement is designed to reduce error due to side-slip or yaw.

## Air Data Computer

The Air Data Computer in current aircraft is a device that uses analogue or digital computing techniques to convert pressure and temperature data into electrical signals which are transmitted to the display instruments and to other systems.

The two types of ADC system found are described as either **Analogue** or **Digital** from the method of assessment and transmission of information used.

The analogue type uses continuous physical variables, such as voltage or pressure, to assess and represent the measurements obtained. The illustration at *Figure 8.2* shows an airspeed assessment device from an **Analogue ADC** indicating the inputs of static and pitot pressure. The pressures are joined together mechanically and, using a **Pressure Transducer**, transmitted forward for use through the rotation of a shaft driven by a 2-phase servomotor which in turn is connected to a CX synchro where angular position can be measured and read off as an airspeed.



*Figure 8.2 Analogue ADC - airspeed assessment*

An analogue Air Data Computer may internally be split into the following modules for assessment and onward transmission of data obtained through the temperature, static and pitot pressure gathering devices - **Altitude**, **Computed Airspeed**, **Mach Speed**, **True Airspeed** and using data from the altitude module via a **Rate of Climb** module will give vertical speed.



The relationships between TAS, Mach No., temperature, pitot and static pressures can be expressed as mathematical formulae. The ADC resolves these formulae continuously to produce the required outputs from pressure and temperature inputs in the form of shaft rotations or electrical signals.

The Digital system uses digital data (binary data) in its assessment and transmission of information. The **Analogue to Digital Converters**, at the input side of the ADC, use measurements of pressure, temperature and AOA and change them from the analogue form to digital form for use within the ADC and onward transmission to the flight deck.

## System Redundancy

Provision for blockages and / or failure of an ADC is made through change-over cocks that permit an alternative static source to be connected to the computer or by the use of electrical switching that enables the Captain's instrument to be fed from the First Officer's ADC and vice versa. These arrangements are illustrated in [Figure 8.3](#) and [Figure 8.4](#).

In some aircraft the ADS is designed so that the outputs from each computer are not directed exclusively to instruments on one side of the panel. By mixing the sources of air data to each side, the possibility of an undetected malfunction is reduced.

In the event of total failure of both ADCs due perhaps to loss of power supply, the flight can be continued by reference to the standby instruments.

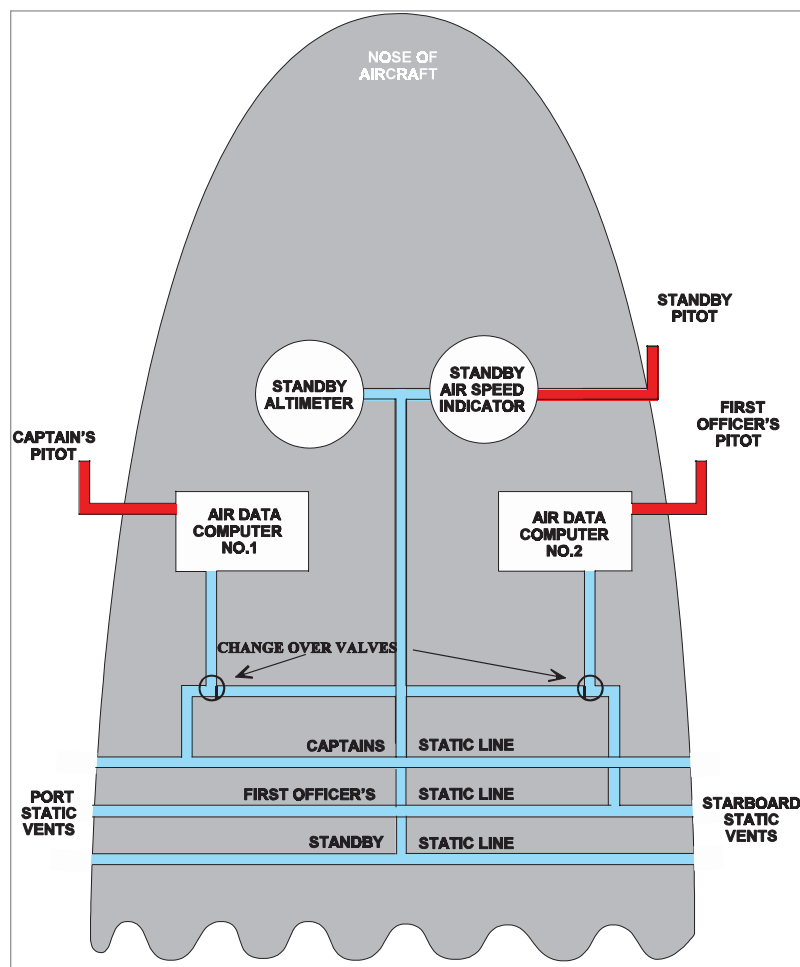


Figure 8.3

## Failure Warning

A comparison monitor can be incorporated to compare the outputs of the ADCs and to give automatic warning to the pilot of malfunction. With a purely mechanical system, comparison between left-hand and right-hand instruments must be carried out visually. A warning flag will appear on the appropriate ADS instrument if there is loss of valid data or if an internal failure occurs. In addition, a light will illuminate either on the instrument warning panel or on the central warning system indicator.

## Built-in Test Equipment (BIT or BITE)

There is no provision made for the manual input of data into the ADC in the event of any failure, but the Built-in Test Equipment will give prompt indication of any malfunction that might occur. (See 'Failure Warning' below). In any ADC there will be three types of BITE process:

### *Power Up BITE*

This functions when power is applied to the ADC on start-up or after a break. A check is made on the Microprocessor, the Memory Store and the Air Data functions.

### *Continuous BITE*

This is an automatic check of all stages of input and output carried out throughout the operation of the ADC about once every second.

### *Maintenance BITE*

This enables maintenance crew to carry out checks on the ground using a Test or Test/History switch (current or post failures).

## Advantages of an Air Data System.

An ADS has certain advantages when compared with conventional mechanical instruments:

### *Improved Displays*

Electrically-served instrumentation allows the manufacturer complete freedom to design new displays that are easier to read and unambiguous. These include digital, moving tape and combined displays.

### *Reduced Instrument and Lag Errors*

The major cause of instrument error in conventional mechanical instruments is friction loss within the linkage. The limited response rate of such linkages gives rise to lag error. Both problems are largely overcome with ADSs by the use of servomotors.

### *Error Correction*

Computation of height, airspeed and other variables within one computer permits error corrections to be applied through especially shaped cams appropriate to the particular aircraft. For example, position error correction (PEC) can be calculated within the Mach No. computer channel for additional use within the height and airspeed channels.

### *Central Source for Other Systems*

The ADC provides not only the conventional information displayed on the instrument panel but also air data in many forms as required for other systems.

### *Clean Design*

The use of electrically-driven instruments reduces the amount of pneumatic plumbing required behind the instrument panel to only those lines connected to the standby airspeed indicator and altimeter. In addition to space saving and easier maintenance, the use of shorter pitot/static line reduces error-producing acoustic effects.

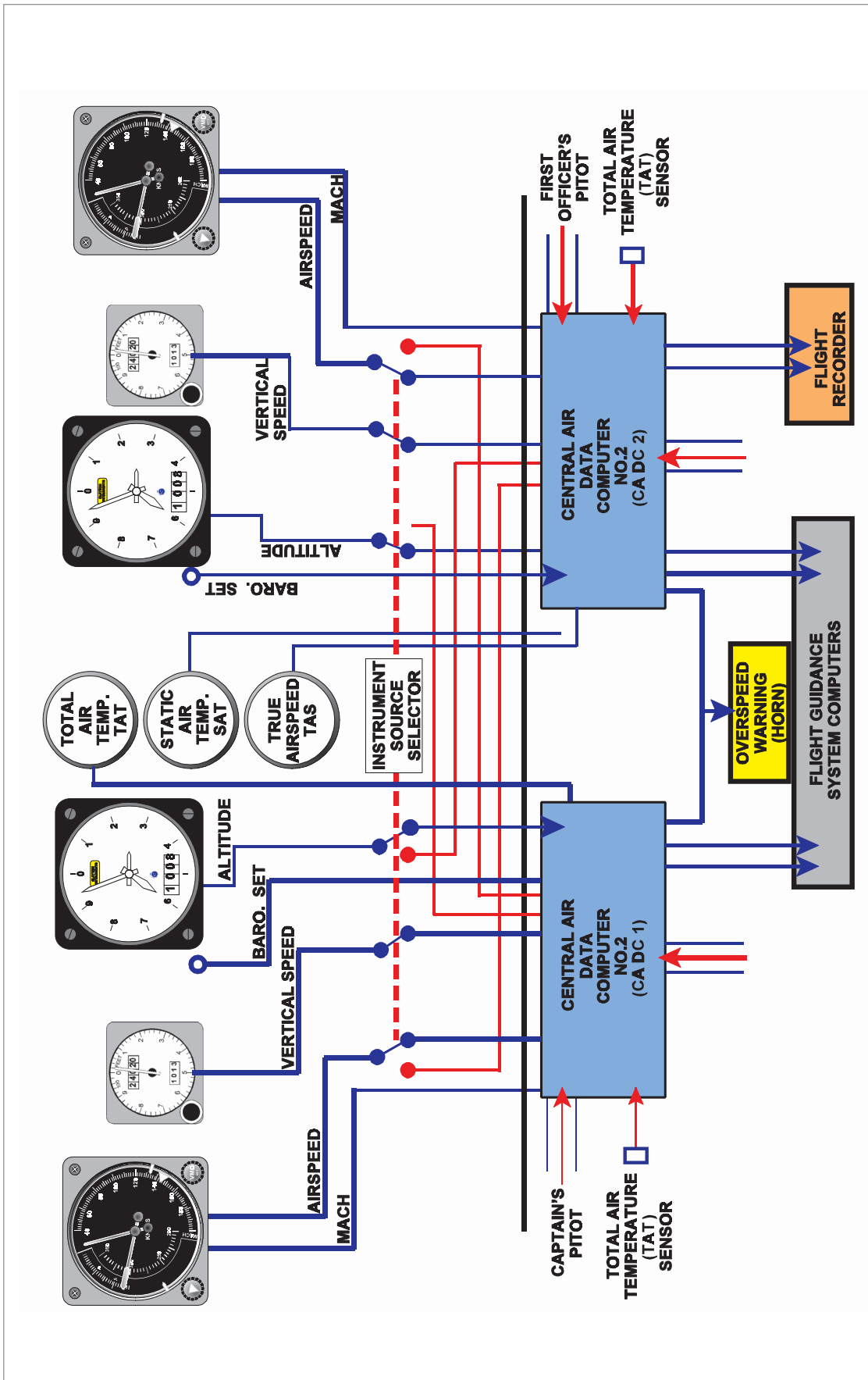


Figure 8.4 Combined Air Data System



Chapter  
**9**  
Terrestrial Magnetism

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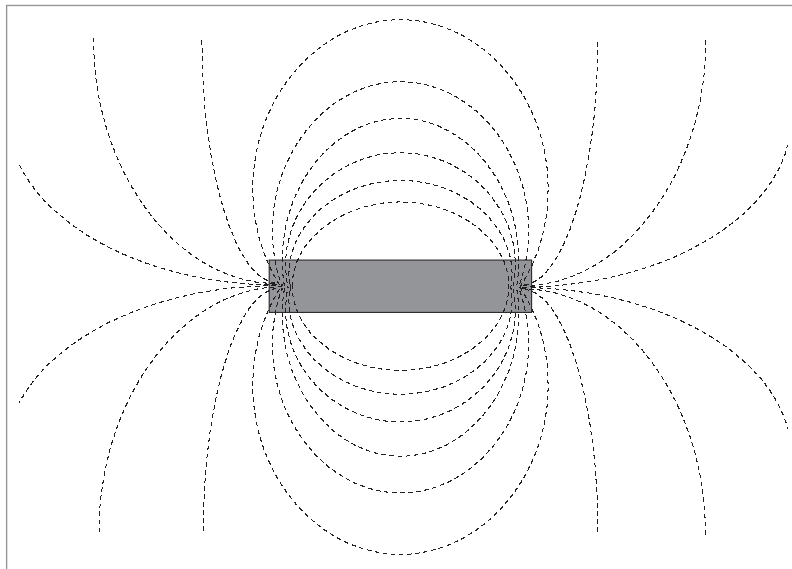
## The Magnet

For thousands of years the oxide of iron called magnetite has been observed to attract small pieces of iron. This property is known as 'magnetism'.

Another property for which magnetite was known was its north-seeking capability; if mounted on wood and floated in water it would swing round and align itself in a roughly north-south direction, so acting as a primitive compass. In more recent history it was found that some metallic elements and alloys (mainly 'ferrous' - iron and steel) could be given these properties, bars of such magnetized material being known as 'magnets'.

## Magnetic Field

The field of a magnet is the space around it in which its magnetic influence is felt. This may be illustrated by placing a piece of card over a bar magnet and scattering iron filings on it. When the card is shaken or tapped the filings will take up the field pattern as shown in *Figure 9.1*.



*Figure 9.1*

## Poles of a Magnet

From *Figure 9.1* it can be seen that the 'lines of force' traced by the iron filings converge towards small areas near the ends of the magnet.

These two areas are called the 'poles' of the magnet and are where the properties of magnetism are most strongly displayed. Magnets are made in various shapes but each magnet always has two poles.

A unit pole cannot exist. If a magnet is cut into two pieces, each piece will have two poles.

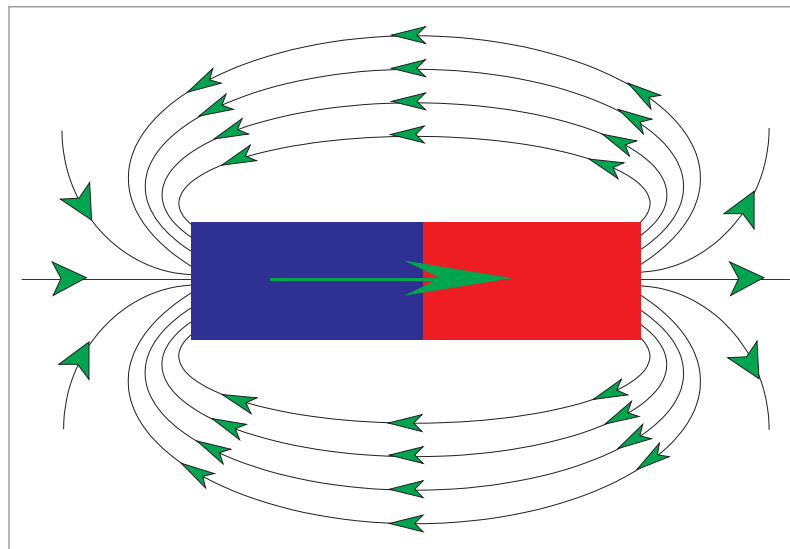
## Red and Blue Poles

A freely suspended bar magnet (or compass needle) in the earth's magnetic field will align itself roughly north-south.

The end which points north is known as a north-seeking or **red** pole. The other end is a south-seeking or **blue** pole.

By convention, magnetic lines of force are directed out from the red pole and back in to the blue pole as shown in *Figure 9.2*.

Again referring to *Figure 9.2*, for convenience the magnet has been divided into two halves, one half containing the red pole, the other half containing the blue pole.



*Figure 9.2 The red and blue poles*

## Attraction and Repulsion Rules

If two bar magnets are placed in a line, end to end, so that the blue pole of one faces the blue pole of the other, a repulsion can be felt. If both magnets are turned around, so that red pole is close to red pole, then again the ends try to move apart. If, on the other hand, the blue pole of one magnet is placed close to the red pole of the other, an attraction is felt.

The rule is:

Like poles repel each other

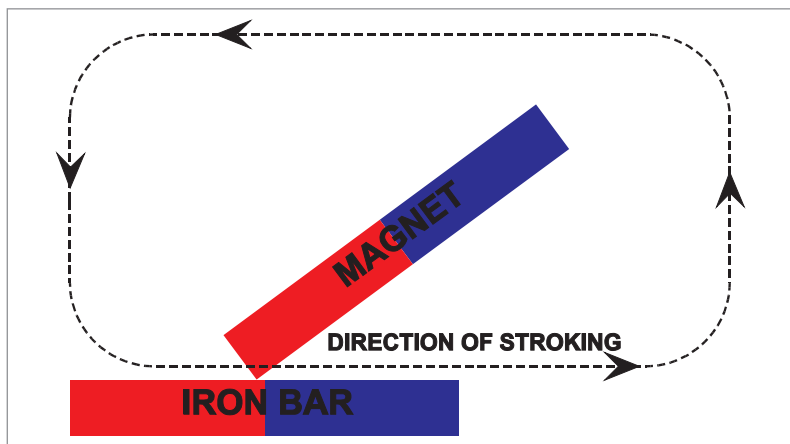
Unlike poles attract each other



## Methods of Magnetization

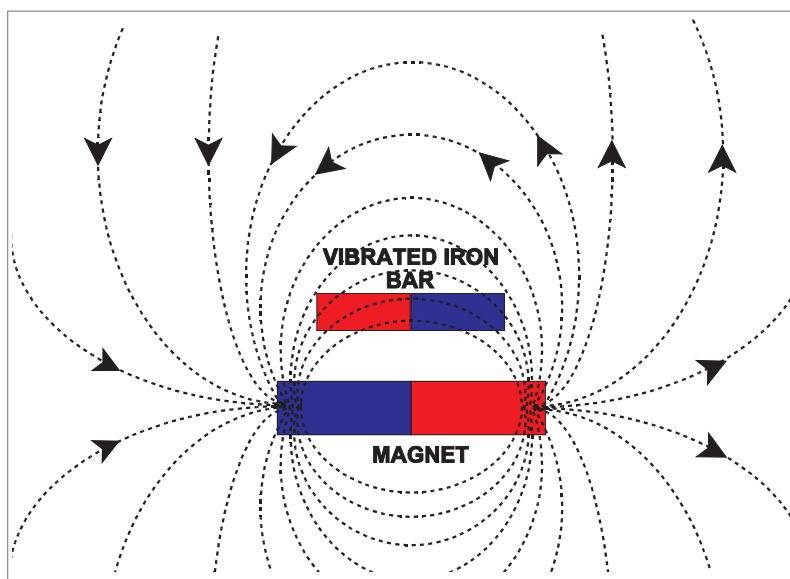
Magnetism may be induced in an unmagnetized bar of iron by one of the following methods:

- By stroking the bar repeatedly in the same direction with one end of a magnet, a process in which the end of the bar last touched by the red end of the magnet is left as a blue pole. *Figure 9.3* depicts the process and shows the resulting polarity of the iron bar.



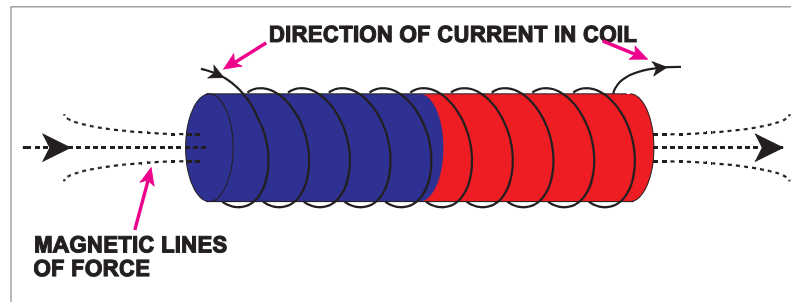
*Figure 9.3 Making a magnet by stroking*

- By aligning the iron bar with the lines of force of a magnetic field and subjecting it to vibration or hammering. Such agitation during manufacture (in the earth's magnetic field) is the main cause of aircraft magnetism. *Figure 9.4* shows the polarity of the induced magnetism in the iron bar; it is such that there is continuity in the pattern of lines of force, as usual directed **in** to a blue pole, **out** from a red pole. The example is analogous to an aircraft being manufactured on a northerly heading in the earth's field and acquiring a permanent red pole in the nose and blue pole in the tail.
- In the case of iron simply by subjecting to a magnetic field. The induced polarity is shown in *Figure 9.4*.



*Figure 9.4 Making a magnet by vibrating or hammering*

- By placing the specimen within a solenoid (a cylindrical coil of wire) carrying a direct current. This is the most satisfactory method as the current flowing in the coil produces a concentrated magnetic field along the axis of the coil so that a high degree of magnetism can be induced in the iron. (Note that the amount of magnetism which can be induced is not unlimited because, at a certain level, the iron becomes magnetically 'saturated'). *Figure 9.5.* shows the polarity of the magnetism induced in the bar inside the solenoid. (If the current flow were reversed the induced magnetic polarity would be reversed).



*Figure 9.5 Magnetization by solenoid*

## Methods of Demagnetization

Three ways of removing most or all of the magnetism from a magnetized item are listed below.

**Shock.** A magnetized bar of iron can be placed at **right angles** to the earth's magnetic field and hammered.

**Heat.** If the specimen is heated to about  $900^{\circ}\text{C}$ , it loses its magnetism and this does not return as the specimen cools.

**Electric Current.** The component is placed inside a solenoid carrying alternating current, the amplitude of which is gradually reduced to zero. The strong alternating magnetic field produced by the alternating current keeps reversing the direction of magnetization (that is the **polarity** of the magnetism) in the specimen. Not only is the polarity being reversed, but the intensity of magnetization is being reduced as the current is reduced. The specimen's magnetism is very quickly reduced to zero or very nearly zero.

## Magnetic and Non-magnetic Materials

Magnetic materials are 'ferrous' metals iron and steel, steel being iron alloyed with substances such as carbon, cobalt, nickel, chromium, and tungsten. These metals are called 'ferromagnetic' and in an aircraft they may be magnetized and produce deviation in the aircraft's compasses.

Many materials used in aircraft construction are non-magnetic and do not affect the compass.

Examples of such non-ferrous substances are aluminium, duralumin, brass, copper, plastic and paint.

### Hard Iron and Soft Iron

Ferromagnetic material can be broadly divided into two classes, **hard iron** and **soft iron**. The words hard and soft do not refer to the physical properties of the material but to their magnetic characteristics.

A strong magnetizing field is required to produce magnetic saturation in hard iron.

Hard iron magnetism is said to be ‘permanent’, meaning that the material, typically steel containing cobalt or chromium, remains magnetized for an indefinite period after it has been removed from the magnetizing field.

Such a substance is suitable for permanent magnets. Soft iron magnetism is called ‘temporary’ (or ‘transient’ or ‘induced’) the substance being easy to saturate magnetically with only a weak magnetizing field but retaining little or no magnetism when the field is removed. Nearly pure iron behaves in this way.

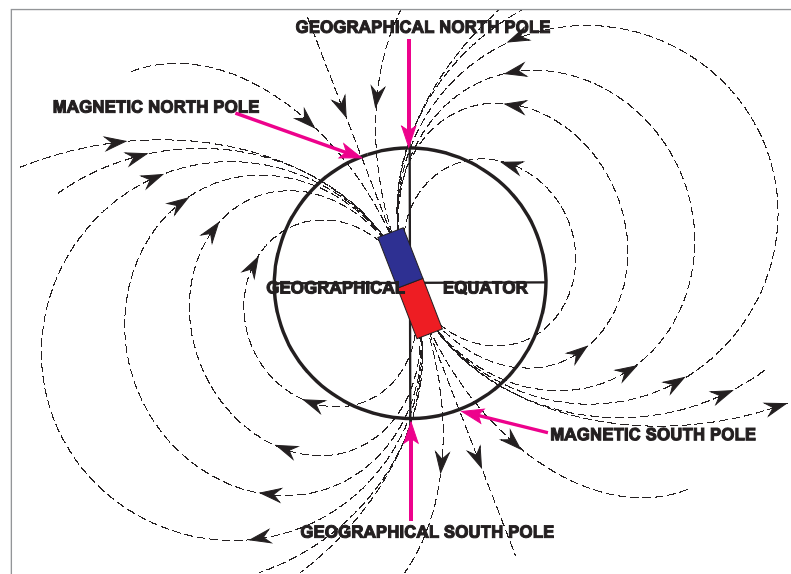
Some materials exhibit magnetic characteristics which lie somewhere between those of hard iron and soft iron. These substances can be magnetized but this ‘sub-permanent’ magnetism is lost partly or wholly over a period of time.

DESCRIPTION	METAL	EASE OF MAGNETISM	RETENTION OF MAGNETISM
HARD IRON	COBALT AND TUNGSTEN STEEL	HARD	CONSIDERABLE LENGTH OF TIME
SOFT IRON	SILICON IRON PURE IRON	EASY	PRACTICALLY NIL

## Terrestrial Magnetism

The earth behaves as though a huge permanent magnet were situated near the centre producing a magnetic field over the surface.

*Figure 9.6* shows that the poles of this hypothetical earth-magnet do not lie on the earth's spin axis, this lack of symmetry giving rise to magnetic variation. The magnetic poles are not stationary but are currently moving at between 6 and 25 NM per year. The north magnetic pole is moving faster than the south magnetic pole. Currently (2015) the north magnetic pole is located north of Alaska at  $86^{\circ}\text{N } 153^{\circ}\text{W}$ , and the south magnetic pole is south of Australia at  $64^{\circ}\text{S } 136^{\circ}\text{E}$ .



*Figure 9.6 Earth's magnetism*

## Magnetic Variation

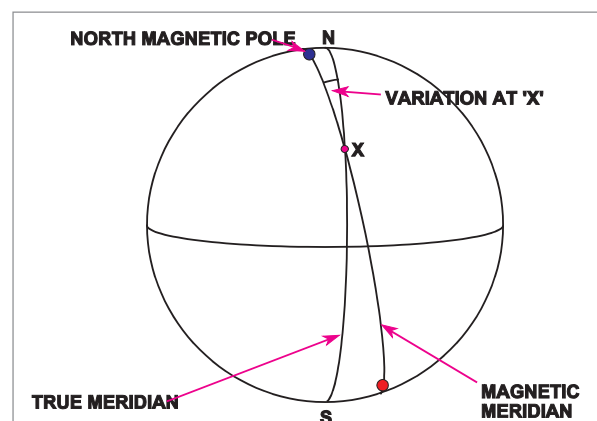
The direction of the earth's field at any given point can be indicated by a freely-suspended magnet. Such a magnet will align itself roughly in a north-south direction with its red pole towards the north magnetic pole.

The longitudinal axis of the magnet defines the direction of the magnet meridian at the point.

The **magnetic meridian** is the direction of the horizontal component of the earth's field at a point on the earth's surface.

The angle, measured in the horizontal plane, between the magnetic meridian at a point and the true meridian at the point is known as the magnetic **variation**.

Variation is designated west or east depending on whether the magnetic pole lies to the west or to the east of true north.



*Figure 9.7 Magnetic variation*

Variation can have any value from zero to 180°, the latter occurring on the true meridian linking north geographical with north magnetic pole, similarly in the southern hemisphere.

### Magnetic Dip

Except near the 'magnetic equator', where the lines of force are parallel to the surface, one end of the freely-suspended magnet will dip below the horizontal, pointing to the nearer pole.

To the north of the magnetic equator, the magnet's red pole will be lower whereas to the south the blue pole will be lower. The angle, measured in the vertical plane, between the axis of the magnet and the horizontal is called the **angle of dip**.

Fairly closely following the geographical equator (in the main 10° of latitude of it) is the 'magnetic equator', which can be represented on a chart by a line joining points on the earth where the angle of dip is zero.

If the freely-suspended magnet is moved either north or south of the magnetic equator the dip gradually increases, reaching about 66° in the United Kingdom. Over the earth's magnetic poles the dip is 90° and the magnet is then vertical.

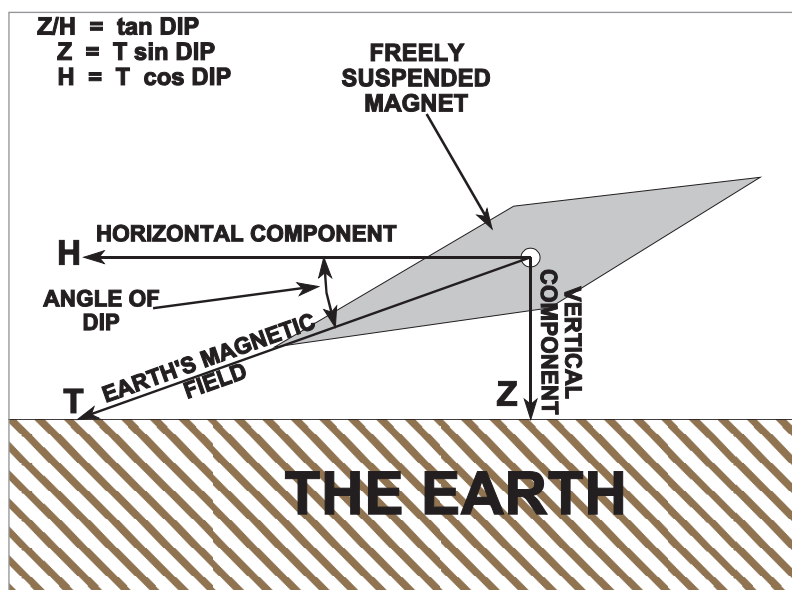


Figure 9.8 Resolution of the Earth's field

### Field Strength

The total force T exerted at a point by the earth's field acts in the direction taken up by a freely-suspended magnet influenced **only** by the earth's field. The total force, angle of dip and magnetic variation at a point are sometimes known as the 'magnetic elements' for that place. It is convenient to resolve this total force T into its horizontal and vertical components H and Z respectively. *Figure 9.8* demonstrates this resolution.

## Directive Force

The horizontal component  $H$  of the earth's field is known as the **directive force** because it is the component which aligns the magnetic compass needle with the magnetic meridian, so providing a directional reference. When either of the earth's magnetic poles is approached, this component approaches zero strength, while the value of  $Z$  approaches that of  $T$ . Over the pole, with dip  $90^\circ$  and zero directive force  $H$ , the magnetic sensor (compass) becomes useless.

In the region of the magnetic equator the strength of the directive force  $H$  approaches the value of  $T$ , while  $Z$  approaches zero as does the angle of dip.

It becomes apparent that the directive force  $H$  decreases as the angle of dip increases, and vice versa and [Figure 9.9](#) serves to illustrate this.

In fact, the relationship between  $H$  and dip angle is not quite as simple as it appears, because of irregularities in the pattern of the earth's field and variations with position and time of the total magnetic force  $T$ .

The strength of the horizontal component  $H$  at a latitude about  $60^\circ\text{N}$  of the magnetic equator is very roughly half the value of  $H$  at the magnetic equator.

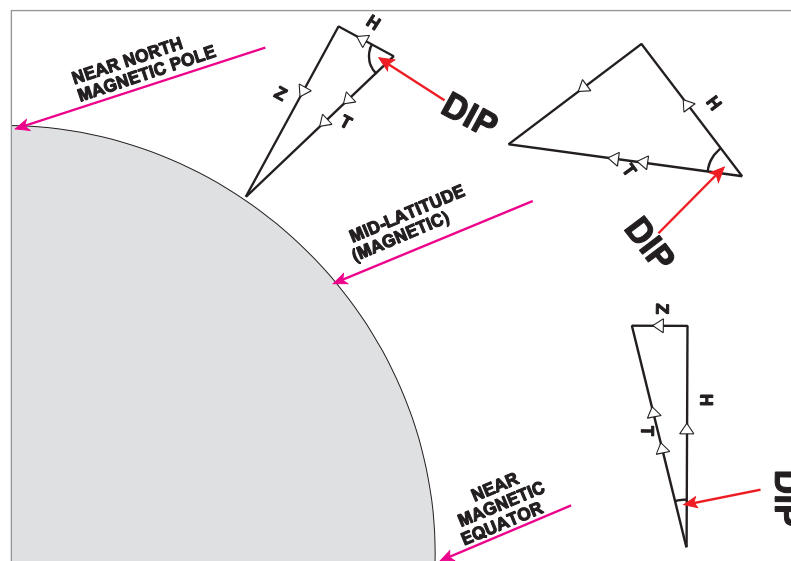


Figure 9.9 The Effect of Latitude on the Components of Dip

## Regular Changes in Earth Magnetism

**Secular Change.** The earth's field not only lacks symmetry but is also subject to several known periodic changes.

Of these, the **secular** changes are the most significant and are produced by the slow movement of the magnetic poles about the geographic poles, the period of this cycle being apparently about 960 years.

The north magnetic pole is moving slowly westward, this wander mainly affecting magnetic variation.

In the UK the value of westerly variation is currently decreasing at a rate of 7 minutes per annum, and the predicted variation in London in the year 2240 is zero.

The annual rate of change of variation is shown on navigation charts so that the variation printed against the isogonals can be readily up-dated.

Other regular changes occur diurnally, annually, and over an eleven-year period, this latter cycle apparently being related to the eleven-year cycle of sunspot activity. These changes, unlike the secular type mentioned earlier, are not of sufficient magnitude to affect normal navigation.

## Unpredictable Changes in Earth Magnetism

Magnetic 'storms' of varying intensity and lasting for as long as three days occur at irregular intervals. These phenomena appear to be produced by unusually large sunspots.

The main effect of these magnetic storms is a temporary but significant change in magnetic variation. The alteration is unlikely to exceed  $2^\circ$  in the UK but in the Arctic and Antarctic the change may exceed  $5^\circ$  and last for as long as an hour. The value of the directive force H can also change and in high latitudes may fall below the minimum required for efficient compass operation.

## Questions

1. The red pole of a freely suspended magnet will point towards ..... and at latitude  $60^\circ\text{N}$  will point ..... at an angle known as the angle of .....
  - a. the nose of the aircraft, downwards, deviation
  - b. the North magnetic pole, downwards, variation
  - c. the nearest pole, downwards, declination
  - d. the North magnetic pole, downwards, dip
  
2. If the total force of the earth's field at a point is  $T$  and the horizontal and vertical components  $H$  and  $Z$ , the value of  $H$  is found by the formula:
  - a.  $H = T \sin \text{dip}$
  - b.  $H = Z \tan \text{dip}$
  - c.  $H = T \cos \text{dip}$
  - d.  $H = T \tan \text{dip}$
  
3. The directive force of the earth's magnetic field:
  - a. varies with the heading of the aircraft
  - b. increases as the magnetic variation increases
  - c. increases as magnetic latitude increases
  - d. is greatest at the magnetic equator
  
4. The slow change in the earth's magnetic variation is known as the ..... change and is caused by .....
  - a. annual, westerly movement of the magnetic pole
  - b. diurnal, easterly movement of the magnetic pole
  - c. secular, westerly movement of the magnetic pole
  - d. annual, sunspot activity
  
5. Soft iron is comparatively ..... to magnetize whilst hard iron is ..... to demagnetize.
  - a. easy; difficult
  - b. easy; easy
  - c. difficult; easy
  - d. difficult; difficult
  
6. Which of the following materials are classed as ferromagnetic:
  - a. iron, steel, carbon-fibre
  - b. nickel, iron, steel
  - c. copper, iron, carbon steel
  - d. iron, cobalt steel, chromium steel
  
7. The magnetic moment of a magnet:
  - a. is the product of pole strength and effective length
  - b. varies inversely as the square of the distance between the poles
  - c. varies directly as the square of the distance between the poles
  - d. decreases as the magnet length increases





## Answers

1	2	3	4	5	6	7
d	c	d	c	a	d	a

Chapter  
**10**

The Direct Indicating Compass

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## The Magnetic Compass

A compass is an instrument designed to indicate direction on the surface of the earth, relative to some known datum. The magnetic compass uses the horizontal component of the earth's field as its directional datum. Unfortunately, the earth's field is normally not aligned with the true meridian - the most desirable datum from which to measure direction. The angular difference between true and magnetic meridians is called the **magnetic variation** discussed in the previous chapter.

The purpose of a magnetic 'steering' compass in an aircraft is to indicate heading, the direction in which the aircraft is pointing.

Magnetic influences - iron/steel components, electric currents - distort the earth's field so that the compass magnet assembly deviates from the magnetic meridian. This is called **compass deviation**.

## Direct Indicating Magnetic Compass

This chapter deals with the direct indicating or direct reading magnetic compass, where the pilot directly reads his heading in relation to the pivoted magnet assembly.

The basic type of direct reading magnetic compass used in aircraft is the vertical card design.

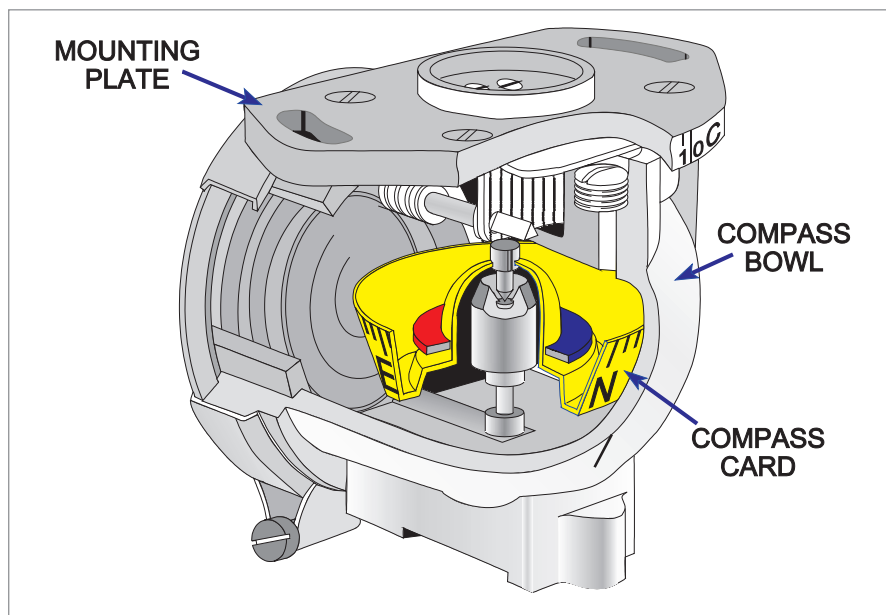


Figure 10.1 A vertical card compass

## The Vertical Card Compass

The vertical card compass - which is also known as the B-type or E-type - is the direct reading compass in general use. It is usually the main magnetic heading reference in light aircraft and the standby compass in larger aircraft.

It consists of a circular compass card attached directly to the magnet assembly. This combined unit is suspended in liquid within the compass bowl. A vertical lubber line on the glass window of the bowl enables the heading to be read off the compass card.

## Compass Requirements

The direct reading magnetic compass contains a pivoted magnet which must be able to align itself, and remain aligned, with the horizontal component of the earth's magnetic field. For the compass to succeed certain requirements must be satisfied. The most important of these are that the magnet system must be:

- Horizontal
- Sensitive
- Aperiodic

## Horizontality

In order to measure direction in the horizontal, the magnets must lie as nearly as possible in the horizontal plane during normal straight and level flight. A freely suspended magnet assembly would align itself with the earth's total field so the magnets would only be horizontal at the magnetic equator.

To achieve horizontality, the magnet assembly is '**pendulously suspended**', the centre of gravity of this assembly being lower than its supporting pivot, as shown in [Figure 10.2](#).

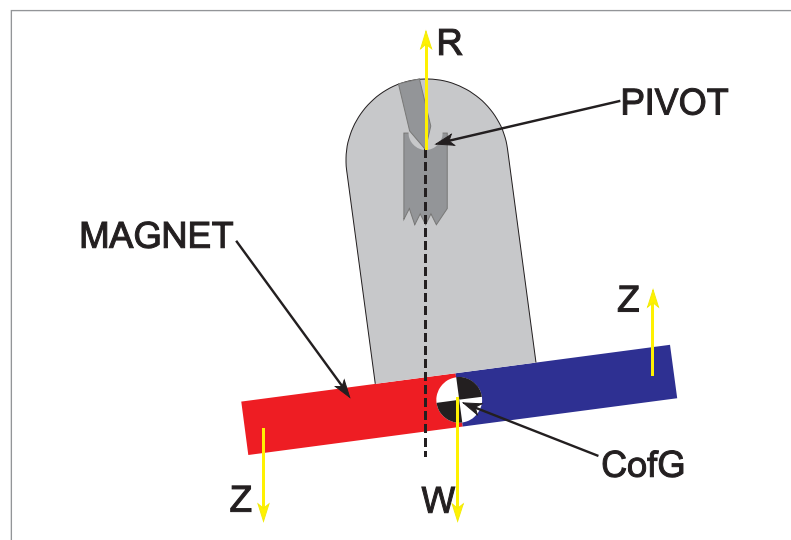


Figure 10.2

In this way, the tilting effect caused by the vertical component  $Z$  of the earth's field is opposed by the weight of the magnet assembly, this equilibrium being achieved at the cost of only a very slight residual tilt of the magnets (north-seeking ends down) - by about  $2^\circ$  in mid-latitudes - in the northern hemisphere. (South-seeking end down in the southern hemisphere). [Figure 10.2](#) shows the two turning couples involved.

One is produced by  $Z$  which exerts a downward force on the red (north-seeking) end of the compass magnet and an upward force on the blue end.

The other couple is produced by the weight  $W$  acting downwards through the centre of gravity (displaced because of the tilt) and the reaction  $R$  acting upwards through the pivot.

For equilibrium, the magnet takes up the amount of tilt necessary to make the couples balance. (A third - very weak - couple produced by the horizontal component, H, of the earth's field, opposing the tilt has been omitted for simplicity).

## Sensitivity

The magnet system is required to seek the horizontal component H of the earth's field in all areas except near the magnetic poles - where the horizontal component is inadequate.

The notes on magnetism show that the ability of a pivoted magnet to align itself with an external field - its sensitivity - depends on the strength of the external field and on the magnetic moment of the magnet. The weak external field (H) at a place cannot be changed, but the magnetic moment of the magnet can be increased - by increasing the magnet's length and/or pole strength.

It is, however, undesirable to increase the magnet length so the pole strength is increased by using **two, four or six short magnets** or a circular magnet, made of an alloy which will accept and retain the high degree of magnetism required.

Sensitivity is further increased by reducing friction. This is achieved in three ways:

- By using an **iridium-tipped pivot** in a **jewelled** cup.
- By **lubricating the pivot** with the liquid which fills the compass bowl.
- By reducing the **effective weight** of the magnet assembly acting down through the pivot, because the liquid that the magnet assembly is displacing is denser than air.

## Aperiodicity

The magnetic assembly is required to be aperiodic or '**dead beat**', which means that it should settle down quickly on a steady indication after being displaced by turbulence or manoeuvres.

Any tendency to oscillate must be quickly 'damped out'. The desired aperiodicity is achieved as follows:

- **Several short magnets** are used instead of one longer one. This keeps the mass of the assembly near the centre, so reducing the moment of inertia and consequently making any oscillations easier to damp out. Light alloy is utilized wherever possible in order to minimize the weight of the assembly framework.
- The primary purpose of the liquid in the compass bowl is to act as a **damping liquid** on the compass assembly. The grid ring compass dampens oscillations more rapidly than the vertical card compass, due to addition of **damping wires**. These wires are attached to the magnet assembly and also pass through the damping liquid.

## The Compass Liquid

The liquid mentioned earlier is essential to the design of the compass. Two difficulties may arise. Firstly, the liquid is likely to expand or contract with change of temperature; this is overcome by incorporating an expansion chamber or 'Sylphon tube'.

Secondly, errors occur in sustained turns as a result of 'liquid swirl'. Liquid swirl occurs due to the viscosity of the liquid; because of this the liquid chosen should have a low viscosity to minimize liquid swirl. Liquid swirl is discussed later in this chapter.

Various liquids, including alcohol, have been used. The main properties required of a compass liquid are:

- Low coefficient of expansion
- Low viscosity
- Transparency
- Low freezing point
- High boiling point
- Non-corrosiveness

## Deviation

Deviation is produced by the iron/steel components in the aircraft. It is the angle between the local magnetic meridian and the direction in which the compass magnets are lying.

Deviation is named easterly (or plus) if the north-seeking (red) ends of the magnets point to the east of magnetic north. If the north-seeking ends point to the west of magnetic north, deviation is said to be westerly (or minus).

Deviation varies with heading so it has to be measured on a series of different headings. This is usually done by conducting a **compass swing** (which is fully covered in the chapter on aircraft magnetism). Once deviation has been reduced as far as possible, the residual deviation is recorded on a compass deviation card, which is located in the aircraft.

During the swing, normal flying conditions should be simulated as far as possible, with engines running, electrical/radio services switched on, and the aircraft in a level flight attitude.

It is obviously most important that no ferromagnetic objects such as tools, or watches should be placed near the compass as this would introduce unknown amounts of deviation. Furthermore, ferromagnetic payloads should be stowed as far away from the compass as permissible within the loading limits. With exceptionally large ferromagnetic loads, a compass swing may have to be carried out before flight with the load aboard.

## Accuracy

**EASA (Part-25) Requirement:  $\pm 10^\circ$**



## Acceleration and Turning Errors

Direct reading compasses are subject to large errors during linear acceleration or deceleration, or during a turn.

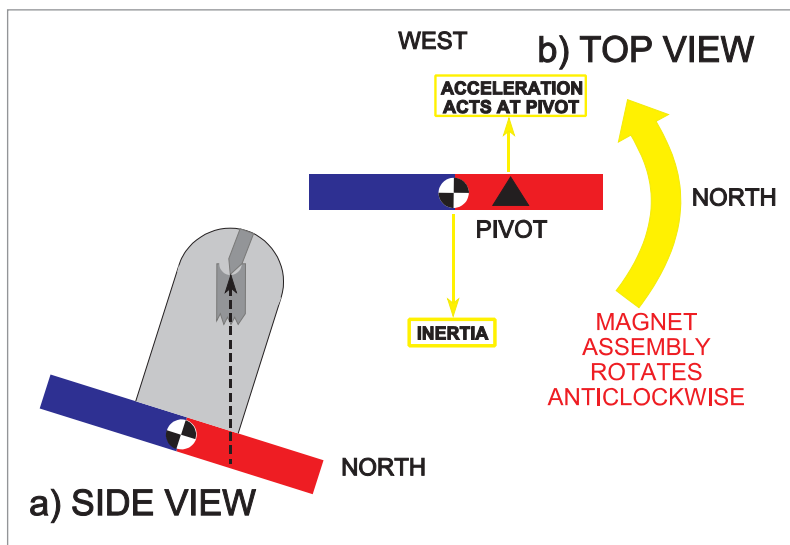
Most manoeuvres which cause the centre of gravity (CG) of the magnet assembly to move away from its normal position, almost directly below the pivot, will produce an error.

However, if the manoeuvre displaces the centre of gravity north or south of its usual position so that CG and pivot are still in the plane of the magnetic meridian, the magnet assembly merely changes its north-south tilt angle, with no rotation in azimuth and consequently no error.

Note also that turning and acceleration errors only occur where there is a significant vertical component (Z) in the earth's field, so that except for a small liquid swirl effect in turns, the errors are non-existent near the magnetic equator.

The north-seeking end of the compass magnet should remain pointing in the same direction - magnetic north - whether the aircraft is moving in a straight line or turning.

Acceleration and turning errors occur however when the north-seeking end of the magnet is displaced from magnetic north and therefore an incorrect heading will be shown on the compass card which is attached to the magnet. *Figure 10.3* shows a pendulously suspended magnet (with residual dip) in the northern hemisphere.



*Figure 10.3 A Pendulously Suspended Magnet in the Northern Hemisphere*

Note that the vertical line through the pivot point is now closer to the nearer (north) magnetic pole than the magnet's centre of gravity. Consider an aircraft, (and therefore the magnet assembly) being accelerated towards the west, as shown in *Figure 10.3*. The magnet is attached to the aircraft at the pivot point. However, while the pivot is being accelerated the magnet's inertia, which acts at the magnet's centre of gravity, will try to maintain its state of uniform motion. The result will be that the magnet will rotate (in this case anticlockwise) and the incorrect heading will be shown.

## Errors Caused by Linear Acceleration

The size of the acceleration error depends on a number of factors which includes aircraft heading. Acceleration / deceleration errors are **maximum** on **East** and **West** (M) headings and **zero** on **North** and **South** (M) headings.

The error is caused by inertia acting on a magnet which has residual dip due to the effect of the vertical component Z on the magnet.

### Acceleration on 270°M (NH)

Figure 10.4 shows an aircraft accelerating on a magnetic heading of 270°M in the northern hemisphere, such as occurs during take-off on runway 27. Since the magnet assembly is pendulously suspended, its inertia will cause it to swing back behind the pivot point which is offset to the north of the magnet's centre of gravity. This displacement enables a turning couple produced by the earth's vertical component Z to rotate the magnet assembly anticlockwise round the pivot.

The angle measured clockwise from the north-seeking end round to the aircraft's nose increases. The compass reading will therefore **increase**, so indicating an **apparent turn towards north**. Thus, according to the compass, the aircraft is now heading, say, 280° whereas its real heading is in fact still 270° - the compass is **over-reading**.

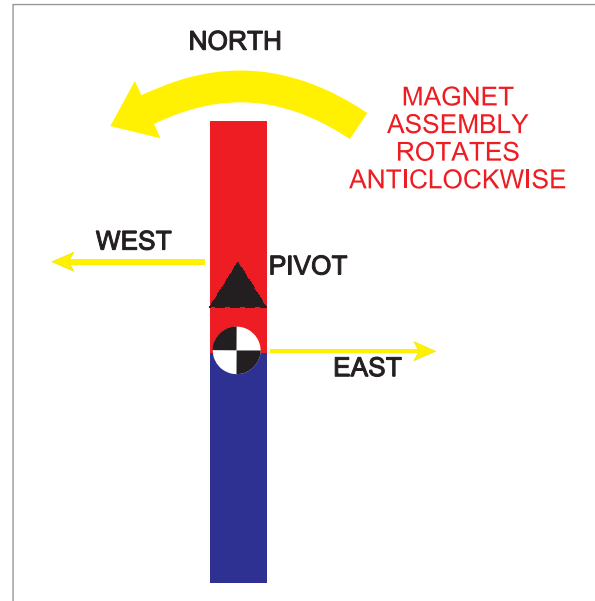


Figure 10.4 Acceleration on 270°M (Northern hemisphere)

### Acceleration on 090°M (NH)

Figure 10.5 shows an aircraft accelerating on a magnetic heading of 090°M in the northern hemisphere. Since the magnet assembly is pendulously suspended, its inertia will cause it to swing back behind the pivot point. This displacement enables a turning couple to rotate the magnet assembly **clockwise** round the pivot. The angle measured clockwise from the north-seeking end round to the aircraft's nose reduces. The compass reading will therefore **decrease**, so indicating an **apparent turn towards north**.

Thus, according to the compass, the aircraft is now heading, say, 080° whereas its real heading is in fact still 090° - the compass is **under-reading**.

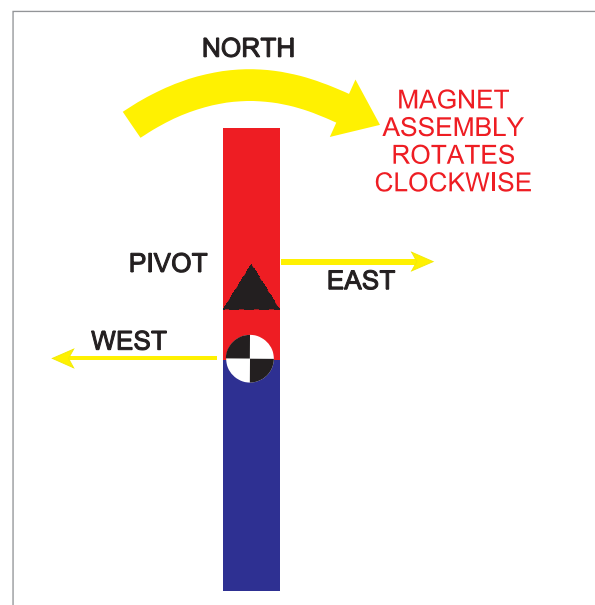


Figure 10.5 Acceleration on 090°M (Northern hemisphere)

## Deceleration on 090°M (NH)

Figure 10.6 shows an aircraft decelerating on a magnetic heading of 090°M in the northern hemisphere.

Since the magnet assembly is pendulously suspended, its inertia will cause it to swing forwards ahead of the pivot point.

This displacement enables a turning couple to rotate the magnet assembly anticlockwise round the pivot.

The compass reading will therefore **increase**, so indicating an **apparent turn towards south**. Thus, according to the compass, the aircraft is now heading, say, 100° whereas its real heading is in fact still 090° - the compass is **over-reading**.

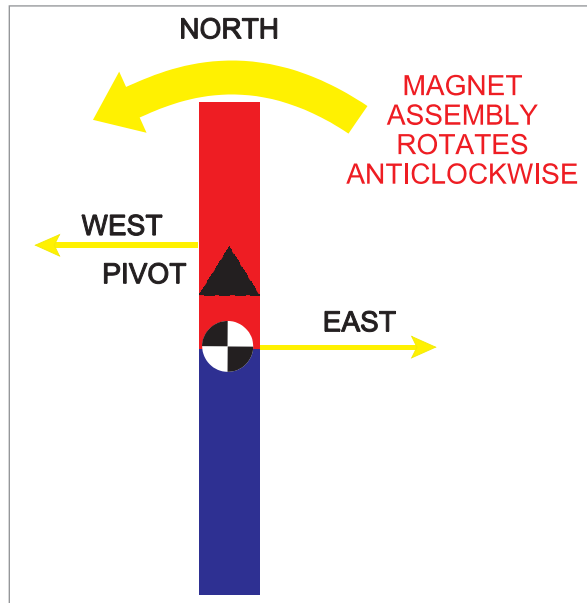


Figure 10.6 Deceleration on 090° M (Northern hemisphere)

## Acceleration on 270°M (SH)

Figure 10.7 shows an aircraft accelerating on a magnetic heading of 270°M in the southern hemisphere. The inertia will cause the magnet assembly to swing back behind the pivot point which is now offset to the south of the magnet's centre of gravity. This displacement enables a turning couple to rotate the magnet assembly clockwise round the pivot.

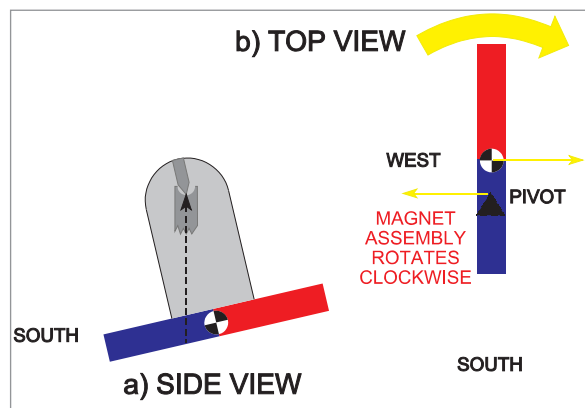


Figure 10.7 Acceleration on 270°M (Southern hemisphere)

The compass reading will therefore **decrease**, so indicating an **apparent turn towards south**. Thus, according to the compass, the aircraft is now heading, say 260°, whereas its real heading is in fact still 270° - the compass is **under-reading**.

### Acceleration on a Northerly Heading (NH)

Figure 10.8 shows an acceleration on a northerly heading (northern hemisphere).

The CG lags and the north-south tilt of the magnet assembly changes, but the magnets are tilting in the vertical plane of the magnetic meridian through the pivot - so no error occurs.

With deceleration on north/south headings there is again no error, only a reduced N/S tilt due to the inertial forward swing of the magnet assembly.

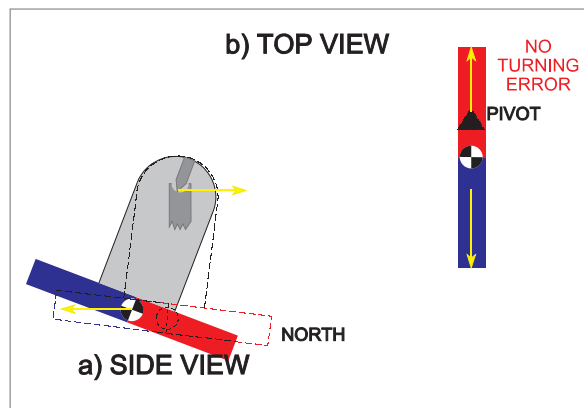


Figure 10.8 Acceleration on  $360^{\circ}M$   
(Northern hemisphere)

### Summary of Acceleration Errors

**Acceleration errors** are **zero** on N/S magnetic headings (in both hemispheres), increasing to **maximum** on headings  $090^{\circ}M$  and  $270^{\circ}M$ . **Acceleration** causes an **apparent turn** towards the **nearer pole** (apparent turn north in the northern hemisphere, apparent turn south in the southern hemisphere). **Deceleration** causes an **apparent turn** towards the **further pole** (apparent turn south in the northern hemisphere, apparent turn north in the southern hemisphere).

Whenever the **magnet** assembly is displaced **clockwise**, the **readings** will **decrease** and the compass will **under-read**.

Whenever the **magnet** assembly is displaced **anticlockwise**, the **readings** will **increase** and the compass will **over-read**.

The size of a linear acceleration error depends on the:

- heading
- magnitude of the acceleration
- design of the magnet system
- magnetic latitude (which affects the relative strengths of H and Z).

The errors are **maximum** near the magnetic poles, decreasing to **zero** at the **magnetic equator**.

## Turning Errors

Turning errors are **maximum** when turning through **north** and **south**, and ignoring liquid swirl **zero** when turning through **east** and **west**.

The basic theory of turning errors is much the same as that for linear acceleration errors.

Due to the earth's vertical component of the magnetic field, **Z**, the **compass's CG** will be **displaced** from almost beneath the pivot point away from the nearer pole. In a **turn**, the aircraft accelerates towards the centre of the turn, and therefore an **acceleration force** acts through the **pivot** towards the centre of the turn, while the opposing **centrifugal force** due to **inertia** acts outward through the **CG**.

This results in the **magnet** assembly tending to 'swing out' from the turn, **rotating** the magnet assembly around the pivot point and producing a **turning error**.

Turning errors are usually more significant than acceleration errors for the following reasons:-

- They are inherently of greater magnitude because greater displacement of the magnet assembly is likely in turns.
- Turns occur more often and are likely to be more prolonged than linear accelerations.

### Turning from 045° to 315° (NH)

Consider an aircraft executing a **left-hand turn** in the northern hemisphere as it passes through 000°M.

The magnet's cg is displaced from beneath the pivot point away from the north pole due to the vertical component of the earth's magnetic field. Because of inertia the **magnet** assembly will be thrown out of the turn **rotating** the magnet assembly **anticlockwise**.

If there was no turning error the magnet would remain stationary and the aircraft rotate 90° around it - resulting in the pilot seeing 90° passing beneath the compass's lubber line.

However, the **aircraft** is turning port and the **magnet** assembly **rotates** in the **same (anticlockwise) direction**.

Although the aircraft has turned 90° around the compass, the magnet has been displaced and rotated in the same direction by a number of degrees (say 20°). The pilot will therefore only see 70° pass beneath the lubber line and the compass is termed **sluggish**.

Whenever the magnet rotates anticlockwise it will **over-read**.

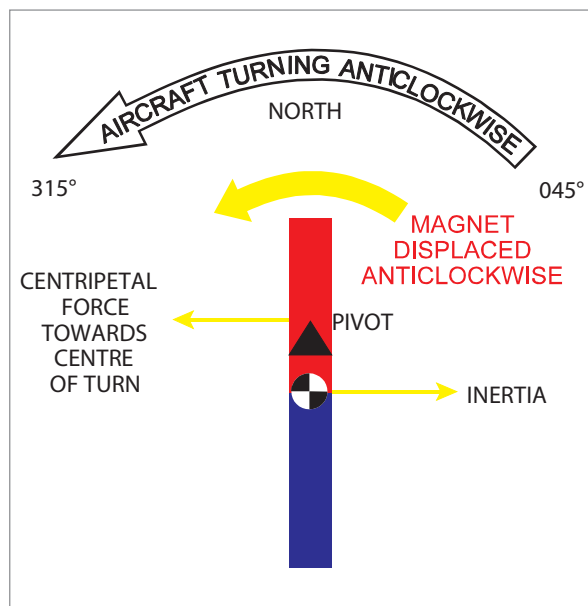


Figure 10.9 Turning from 045° to 315° (Northern Hemisphere)

This means that if the pilot stops the turn at 315° **indicated** the **actual** heading will be numerically smaller such as 295° - therefore the **turn must be stopped early** (such as 335°) to achieve the correct heading.

This can also be described as **undershooting** the required heading (note 'undershoot' is referring to turning through a smaller **angle**, and should not be confused with 'under-read' which means that the **numerical** heading indicated is too small).

If the pilot deliberately undershoots, rolling out when the compass reads about 335°, he should observe, when the wings are levelled, the compass 'catch up' and settle on 315°.

### Turning from 315° to 045° (NH)

Consider an aircraft **turning right** through north in the northern hemisphere as it passes through 000°M, the magnet's CG is displaced from beneath the pivot point away from the north pole due to the vertical component of the earth's magnetic field.

Because of inertia the **magnet** assembly will be thrown out of the turn **rotating** the magnet assembly **clockwise**.

**Note:** The aircraft and the magnet assembly are again rotating in the same direction (but this time clockwise) and therefore the compass will again be sluggish.

Whenever the magnet rotates clockwise it will **under-read**. This means that if the pilot stops the turn at 045° **indicated** the **actual** heading will be numerically larger such as 065°.

Therefore the **turn must be stopped early** (such as 025°), or the pilot should **undershoot the indication**, to achieve the correct heading.

### Turning from 135° to 225° (NH)

Now consider an aircraft **turning right** as shown in *Figure 10.11* through south in the northern hemisphere as it passes through 180°M, the magnet's CG is displaced from beneath the pivot point away from the nearer pole (the north pole) due to the vertical component of the earth's magnetic field.

Because of inertia the **magnet** assembly will be thrown out of the turn **rotating** the magnet assembly **anticlockwise**.

The **aircraft** is turning **clockwise** (right) but the **magnet** assembly is rotating anticlockwise.

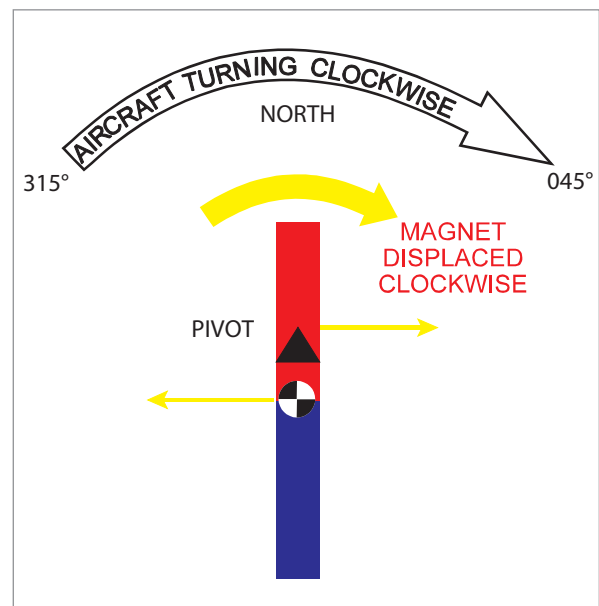


Figure 10.10 Turning from 315° to 045° (Northern hemisphere)

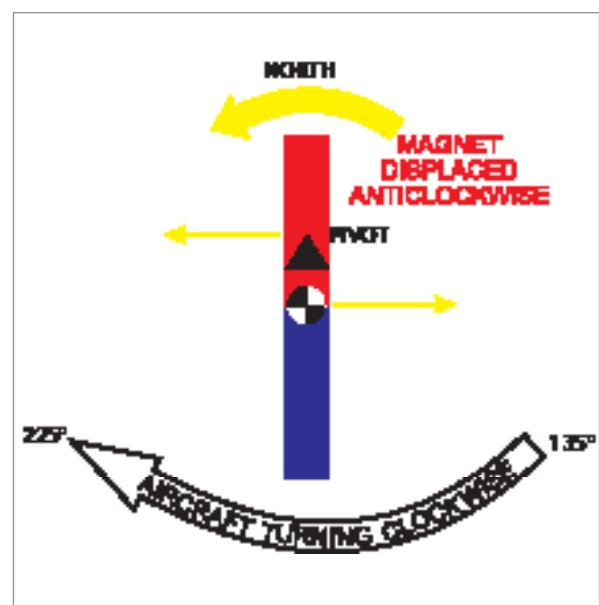


Figure 10.11 Turning from 135° to 225° (Northern hemisphere)

Therefore the aircraft and the magnet are now **rotating in opposite directions**. Although the aircraft has turned  $90^\circ$  around the compass, the magnet has been displaced and rotated in the opposite direction by a number of degrees (say  $20^\circ$ ). The pilot will therefore see  $110^\circ$  pass beneath the lubber line and the compass is termed 'lively'.

Whenever the magnet rotates anticlockwise it will **over-read**. This means that if the pilot stops the turn at  $225^\circ$  **indicated** the **actual** heading will be numerically smaller, such as  $205^\circ$ .

Therefore the **turn must be stopped late** (such as  $245^\circ$ ), or the pilot should overshoot, to achieve the correct heading.

### Turning from $135^\circ$ to $225^\circ$ (SH)

Now consider an aircraft **turning right** as shown in *Figure 10.12* through south in the southern hemisphere as it passes through  $180^\circ$ M, the magnet's CG is displaced from beneath the pivot point away from the nearer pole (the south pole). Because of inertia the **magnet** assembly will be thrown out of the turn **rotating** the magnet assembly **clockwise**.

The **aircraft** and the **magnet** assembly now are **rotating in the same direction** (clockwise) and therefore the compass will again be **sluggish**.

Whenever the magnet rotates clockwise it will **under-read**. This means that if the pilot stops the turn at  $225^\circ$  **indicated** the **actual** heading will be numerically larger such as  $245^\circ$ . Therefore the **turn must be stopped early** (such as  $205^\circ$ ), or **undershoot**, to achieve the correct heading.

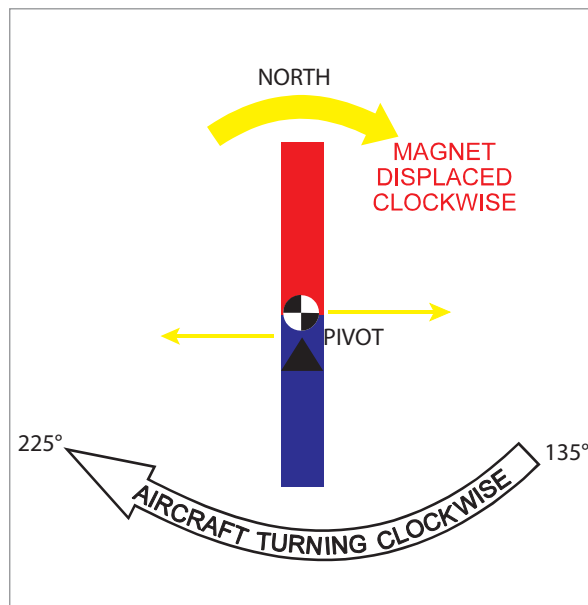


Figure 10.12 Turning from  $135^\circ$  to  $225^\circ$  (Southern hemisphere)

Remember that when the wings are levelled, the compass will 'catch up' and settle on  $225^\circ$ .

### Turning through East or West

Consider a turning aircraft passing through the magnetic headings of  $090^\circ$  and  $270^\circ$ .

The magnets are not horizontal but their tilt is north-south, that is in the vertical plane of the magnetic meridian through the pivot.

There is no rotational couple acting round the pivot, so there is no turning error.

**Turning errors are zero when passing through east or west.**

### Other Notes on Turning Errors

It is **easier to steer a southerly** rather than a **northerly** heading in the northern hemisphere, firstly because on south the compass does not indicate the wrong direction of turn as it can on north, and secondly because the 'lively' nature of the indications reduces the risk of over-correcting small steering errors.

### Magnitude of Turning Errors

There are many factors affecting the severity of turning errors. They are **worst at high latitudes** where **Z is strong** and H is weak.

Other relevant variables include rate of turn, duration of turn, speed of the aircraft, the headings involved and the design of the compass.

### Turning Errors - Liquid Swirl

The effect known as liquid swirl was mentioned earlier. During a turn, the liquid in contact with the inside of the bowl tends to be dragged around with the bowl, so producing in the liquid small eddies which drift inwards from the circumference and **deflect the magnet assembly in the direction of turn**. Therefore the liquid tends to swirl - and rotate the magnet assembly with it - in the same direction as the aircraft's turn.

Accordingly, when turning through **north** in the **northern hemisphere** it will **increase** the magnitude of the **turning error** (in which the assembly turns in the same direction as the aircraft).

The size of the turning error when turning through **south** in the **northern hemisphere** (where the assembly turns in the opposite direction to the aircraft) **will be reduced**.

In the southern hemisphere the swirl effect will be in the opposite sense.

***Note:** At the magnetic equator where there is no vertical component Z in the earth's field, liquid swirl is the sole source of turning error; with most compasses the effect is only slight.*

### Summary of Turning Errors

These are **maximum** when passing through magnetic **north** or **south**, decreasing to **zero** when passing through **east** or **west**.

The **error increases** with **increase** in **magnetic latitude**.

At the magnetic equator the only turning error is due to liquid swirl.

**Whenever the pilot turns through the nearer pole** (north in the northern hemisphere, or south in the southern hemisphere):

- the aircraft and compass **rotate** in the **same direction**
- the **compass** will be **sluggish**
- the pilot should **undershoot** the turn / **roll out early**
- **liquid swirl** will **increase** the turning error

**Whenever the pilot turns through the further pole** (south in the northern hemisphere, or north in the southern hemisphere):

- the aircraft and compass **rotate** in the **opposite direction**
- the **compass** will be **lively**
- the pilot should **overshoot** the turn / **roll out late**
- **liquid swirl** will **reduce** the turning error



At the **MAGNETIC EQUATOR**, there is **NO TURNING ERROR** because there is no "dip".

Remember, that it is a displacement of the **MAGNETS** in a **CLOCKWISE** direction when viewed from above which causes the compass to **UNDER-READ**, and a displacement in an **ANTICLOCKWISE** direction which causes the compass to **OVER-READ**.

## Questions

1. In a standby direct reading compass there is:
  - a. a non-pendulously mounted magnet system
  - b. a single pendulously mounted bar magnet
  - c. a circular magnet or pair of bar magnets pendulously mounted
  - d. a low magnetic moment system, either of circular or bar configuration
  
2. The main requirements of a direct reading magnetic compass are that it should be:
  - a. horizontal, sensitive, periodic
  - b. easily read, floating in a transparent liquid, quick to react to change in aircraft heading
  - c. positioned directly in front of the pilot, easily corrected for magnetic deviation, aperiodic
  - d. aperiodic, horizontal, sensitive
  
3. For a position in the southern hemisphere, the effect of acceleration errors are greatest on headings:
  - a. 180° and 360°
  - b. 045° and 225°
  - c. 135° and 315°
  - d. 090° and 270°
  
4. An aircraft in the southern hemisphere is turning from a heading of 090°C to 360°C using a DGI. At the end of the turn the compass will read ..... than 360° and liquid swirl will ..... this effect.
  - a. more; increase
  - b. less; increase
  - c. more; decrease
  - d. less; decrease
  
5. In a standby compass the magnet system is immersed in a transparent liquid. The purpose of this liquid is to:
  - a. increase sensitivity, increase aperiodicity
  - b. increase sensitivity, decrease aperiodicity
  - c. increase sensitivity at high latitudes, lubricate bearings
  - d. increase sensitivity, reduce liquid swirl
  
6. To improve the horizontality of a compass, the magnet assembly is suspended from a point:
  - a. on the centre line of the magnet
  - b. below the centre of gravity
  - c. above the centre of gravity
  - d. varying with magnetic latitude

7. The magnitude, and sense, of turning error shown by a direct reading compass varies with:

1. the design of the compass
2. the direction of the turn
3. the rate of turn
4. which hemisphere the aircraft is in
5. the heading of the aircraft
6. the amount of dip at the aircraft's latitude

Of these statements:

- a. only 1, 2, 5 and 6 are correct
- b. only 1, 3, 5 and 6 are correct
- c. only 2, 4 and 5 are correct
- d. all are correct

8. During a sustained turn ..... the nearer magnetic pole, the effect of liquid swirl will ..... compass turning error.

- a. away from; increase
- b. towards; not affect
- c. away from; not affect
- d. towards; increase

9. When carrying out a turn at the magnetic equator there will be:

- a. no turning error
- b. a tendency to under-read turns through south and over-read turns through north
- c. a tendency to under-read turns due to liquid swirl
- d. no turning error when turning through east or west

## Answers

1	2	3	4	5	6	7	8	9
c	d	d	d	a	c	d	d	c

Chapter  
**11**  
Gyroscopes

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## Gyroscopes

The simplest form of gyroscope (or gyro) consists of a rapidly spinning disc (called a **rotor**). Any rapidly spinning symmetrical rotor exhibits gyroscopic properties – even if it has not been specifically designed as a gyro. For instance, the earth acts as a gyro, and so do spinning tops and bicycle wheels. Most aircraft gyros are discs between 2 and 5 cm in diameter, spinning at speeds between 4000 and 55 000 rpm, depending on their design.

The shaft about which the rotor spins is called the **axis**. Gyros are defined in their orientation as either horizontal or vertical by reference to the spin axis, not the rotor.

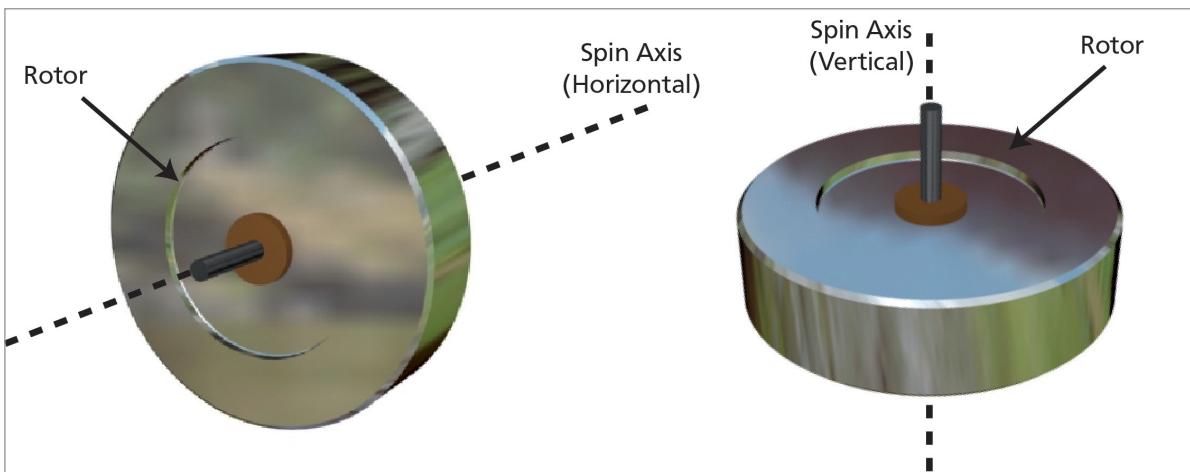


Figure 11.1 Gyro parts and orientation

Gyros have 2 basic properties which make them important as the basis of aircraft attitude and direction instruments. These are **rigidity** and **precession**.

### Rigidity

**Rigidity is the gyro's property of maintaining its axis in a fixed direction in space unless subjected to an external force.** It is caused by the inertia of the spinning mass.

In order for a gyro to be able to maintain a fixed direction, it must be attached to the airframe in such a way that the aircraft has freedom to manoeuvre without disturbing the orientation of the gyro. This is achieved by using suspension devices which allow this freedom. These are called **gimbals**. Gyros may have one or two gimbals.

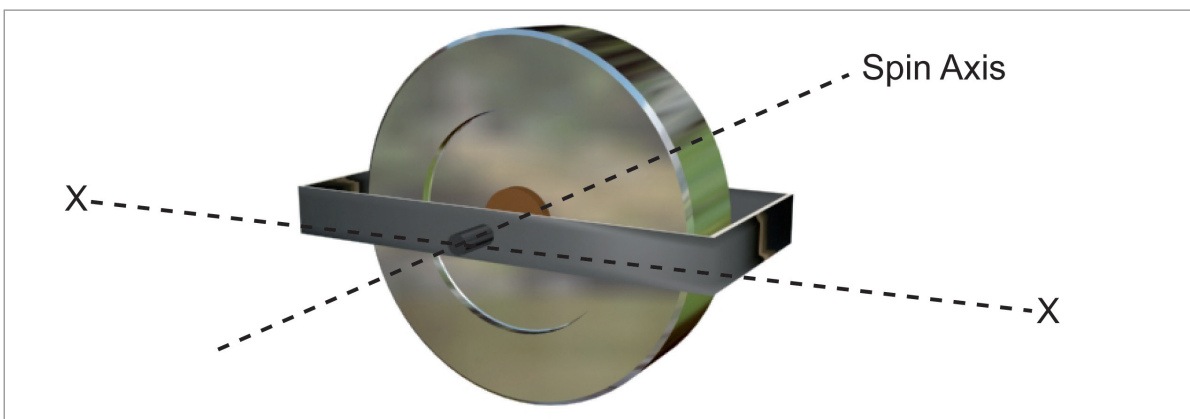


Figure 11.2 One gimbal, one degree of freedom

In *Figure 11.2*, assume that the aircraft is travelling left to right across the page, i.e. its longitudinal axis is also the gyro axis XX (This is not the same thing as the spin axis). The aircraft is free to bank whilst the gyro remains in the same orientation (spin axis horizontal and in a fixed direction). However, if the aircraft were to yaw, the gyro and its gimbal would be forced out of its original orientation.

The aircraft is free to pitch or bank without affecting the gyro, but not to yaw. How we describe this depends on the convention being used. Some references will describe this as a '2-degrees of freedom system'.

However, the convention used by EASA does not regard the spin axis as a degree of freedom, because no measurement can be made in pitch.

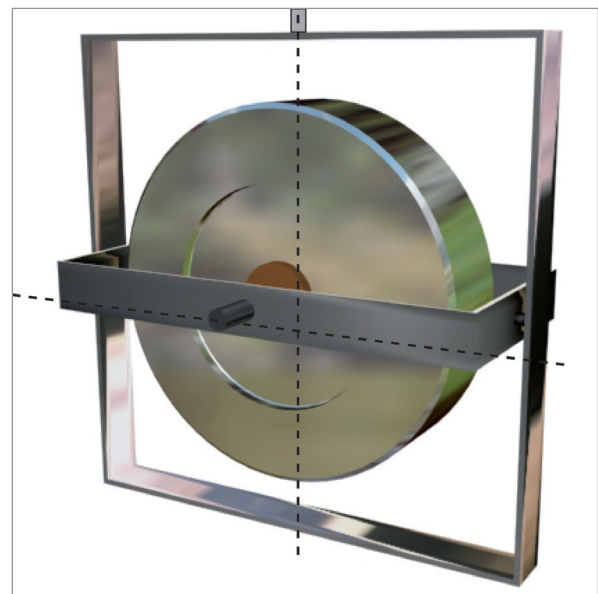
*Figure 11.3*, right, shows a 2-gimbal system:

In *Figure 11.3* the aircraft is free to pitch and bank as before but, with the second gimbal, it is also free to yaw without disturbing the gyro.

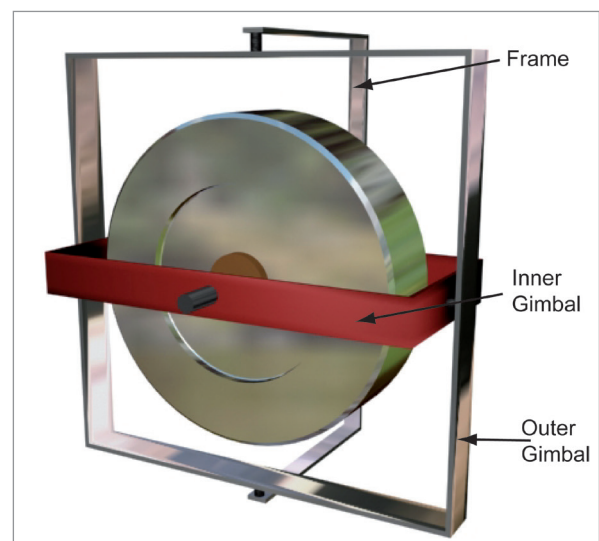
The definition of rigidity includes the words 'fixed direction in space'. Rigidity (also called 'gyroscopic inertia') is subject to Newton's Laws of Motion, which apply to all of space, not just the earth. In theory, if there were no other errors and the gimbaling system allowed complete freedom, the axis of the gyro would point to a fixed point in space (for instance, a distant star), irrespective of the rotation of the earth or the motion of the aircraft over the earth.

Over short periods of time, this difference between earth orientation and space orientation may not be important, particularly if it is swamped by mechanical errors or if the gyro is maintained by mechanical means to some earth reference. However, it becomes important when considering accurate freely-gimballed gyros over long periods.

On simple aircraft, gyros do not normally have more than two gimbals. In *Figure 11.4*, right, we have an inner and an outer gimbal, but the third structure, on the outside of the outer gimbal, is the frame. It is attached to the aircraft and moves with the aircraft.



*Figure 11.3 Two gimbals, two degrees of freedom*



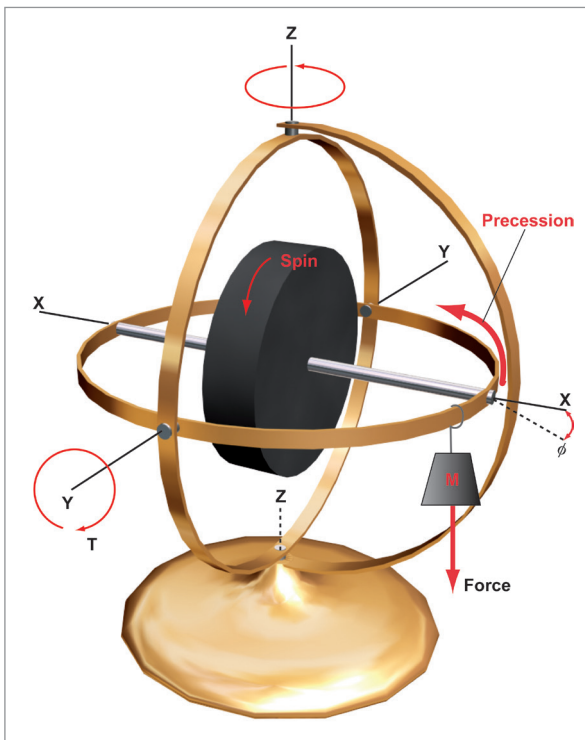
*Figure 11.4 Two gimbals and a frame*



## Precession

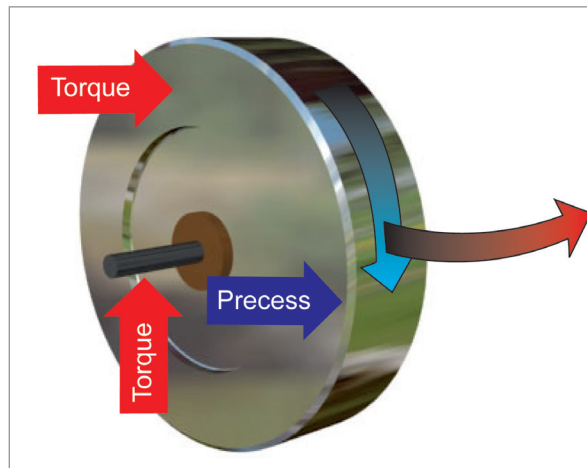
In *Figure 11.5*, the gyro is rotating clockwise. Assume that an upward force (torque) acts on the spin axis. This is the equivalent of the force going into the rotor at the 12 o'clock position. However, the reaction will not be to make the gyro rotate backwards about the horizontal axis from the point where the torque is applied. Instead, the torque is precessed through  $90^\circ$  in the direction of rotation and the rotor moves inwards to the page about the vertical axis, as though the force had been applied at the 3 o'clock position to a stationary rotor.

Now we put this situation into a gimballed gyro, as in *Figure 11.6* below:



*Figure 11.6 Gyroscopic precession*

remains in the same relative position on the gimbal ring, the gyro spin axis will continue to rotate at a constant velocity.



*Figure 11.5 Torque precessed through  $90^\circ$  in direction of rotation*

In *Figure 11.6* the gyro is spinning about its spin axis  $XX$ . A small mass  $M$  is applied on the inner gimbal in line with the  $XX$  axis. The mass acts to pull the inner gimbal down, producing a torque about the  $YY$  axis. Effectively, this is a force being applied to the 6 o'clock position of the rotor.

Initially the gyro axis tilts through a small angle  $\phi$ . The spin axis is no longer pointing to the original fixed point in space. After that, no further movement takes place about the  $YY$  axis.

The torque is then precessed through  $90^\circ$  in the direction of the gyro's rotation. The rotary motion takes place at the 3 o'clock position and the spin axis starts to turn at a constant velocity about the  $ZZ$  axis.

If the torque at  $M$  is withdrawn, the precession ceases.

If the torque application at  $M$  continues and

## The Relationship Between Precession and Rigidity

The rigidity of a gyro depends on two properties:

- Moment of Inertia (a combination of the mass and the effective radius at which the mass operates)
- Rotor rpm

**Moment of Inertia.** Moment of inertia is a measure of how big and how heavy the gyro is. A gyro with a greater radius will have a larger moment of inertia than a smaller one with the same mass. A gyro with a greater mass will have a larger moment of inertia than one of the same radius but less mass.

In order to minimize weight (mass) but get a greater moment of inertia, the mass is often concentrated at the rim of a gyro. A bicycle wheel, which uses spokes, is a clear example.

**Rotor rpm.** The faster the rotor spins, the greater the gyro's rigidity.

**The rigidity of a gyroscope is increased if the mass is increased, the effective radius at which the mass operates is increased, or if the rotor rpm is increased.**

**The rate of precession is directly proportional to the applied torque but inversely proportional to the moment of inertia and the rotor rpm rate.**

Effectively, this means that precession and rigidity are opposite characteristics. If a gyro has a lot of rigidity, it will not precess very much. If it precesses a lot, it cannot be very rigid.

## Wander

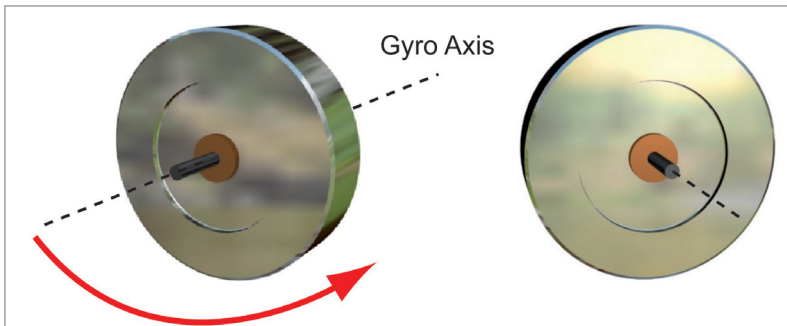


Figure 11.7 Horizontal wander of the gyro axis - drift

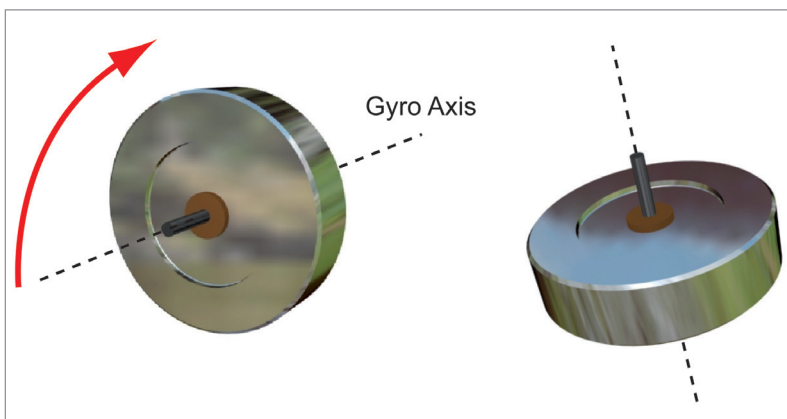


Figure 11.8 Vertical wander of the gyro axis – topple

Despite this property of rigidity, the orientation of a gyro axis may alter over time. Any departure of a gyro axis from its original orientation is called **wander**. Wander may be either **real** or **apparent** (the difference is explained below) or a combination of both. Wander is subdivided into **drift** and **topple**. If the gyro axis wanders in the **horizontal** plane, it is called **drift**, either real or apparent (Figure 11.7).

If the gyro axis wanders in the **vertical** plane, it is called **topple**, either real or apparent (Figure 11.8).

Horizontal axis gyros can both drift and topple – as seen in Figure 11.7 and Figure 11.8. However, vertical axis gyros can only topple. They can topple either forwards, backwards or sideways, as shown in Figure 11.9 – but they cannot drift.

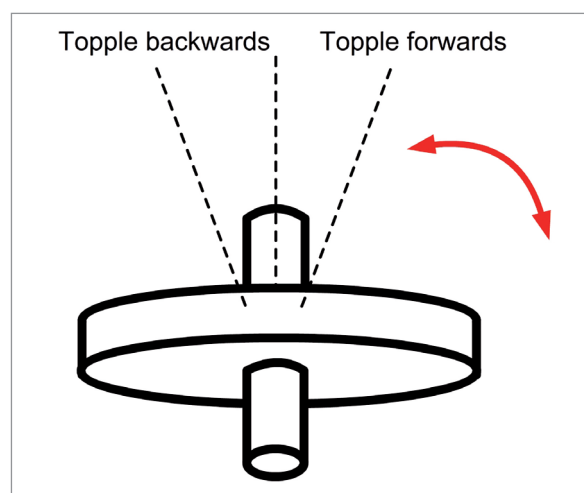
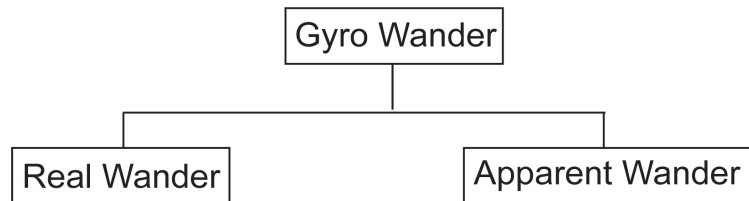


Figure 11.9 Vertical axis – topple, but no drift

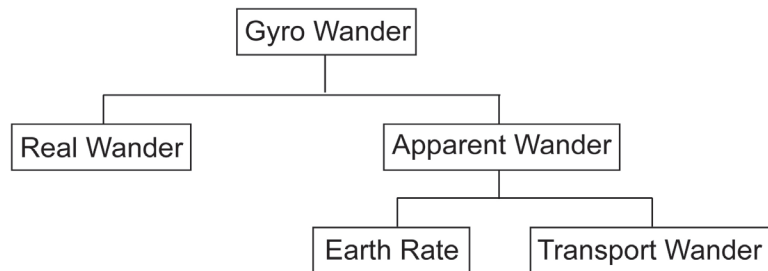
### Real and Apparent Wander

As was stated before, gyro wander, whether drift or topple, can be subdivided into real and apparent wander:

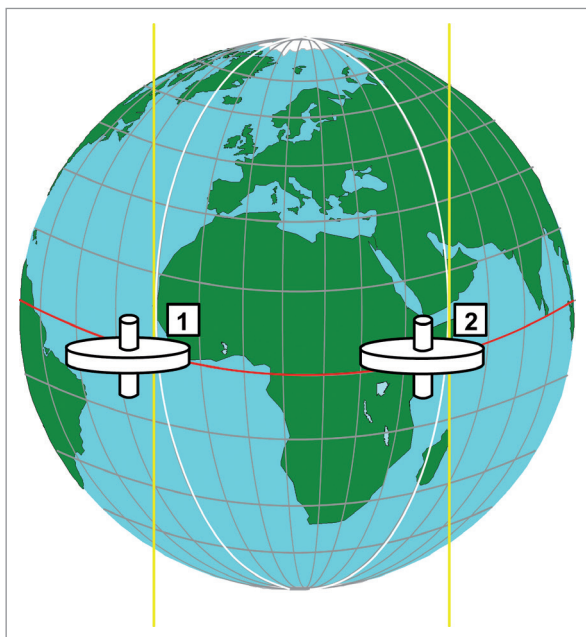


**Real Wander.** In real wander, the axis of the gyro moves with respect to inertial space. This departure from the original orientation is caused by **manufacturing imperfections**, such as **uneven rotor bearing friction, gimbal friction, imbalance in the mass of the rotor and unbalanced gimbals**. Real wander can be reduced by higher quality engineering and manufacturing. Depending on the application, the cost of reducing real wander may not be justified by the level of accuracy required. As with instrument error, gyros need to be as accurate as the need of the application in use, but over-engineering costs extra money. **Real wander** can also be known as **'random' wander**.

**Apparent Wander.** However, even if all real wander were eliminated and the gyro remained in the same orientation in space (for instance, pointing at a distant star) the direction indicated by a gyro would still change. This is because of changes in the observer's frame of reference.



One is caused by the rotation of the earth and is called **Earth Rate**. The other is caused by flight east or west at latitudes other than the equator and is called **Transport Wander**.



**Earth Rate.** Imagine an aircraft parked on the equator with a gyro axis pointing to true north, i.e. aligned with the local meridian.

As the earth rotates from position 1 to position 2, the gyro axis remains fixed in space and also remains aligned with the local meridian.

Figure 11.10 No horizontal component of earth rate at the equator

However, at a pole:

At the pole, the gyro remains aligned in space but the meridian direction (i.e. the direction of local true north) changes as the earth rotates. Therefore we get the full earth rate of  $15^\circ$  per hour. For anywhere between the equator and the pole:

**Earth Rate =  $15 \times \sin \text{latitude } ^\circ/\text{hour}$**

(The earth rotates one revolution ( $360^\circ$ ) in 1 day (24 hours). Therefore the earth's spin rate is  $15^\circ/\text{hour}$ . Latitude at the pole is  $90^\circ$  and  $\sin 90^\circ = 1$ . Latitude at the equator is  $0^\circ$  and  $\sin 0^\circ = 0$ ).

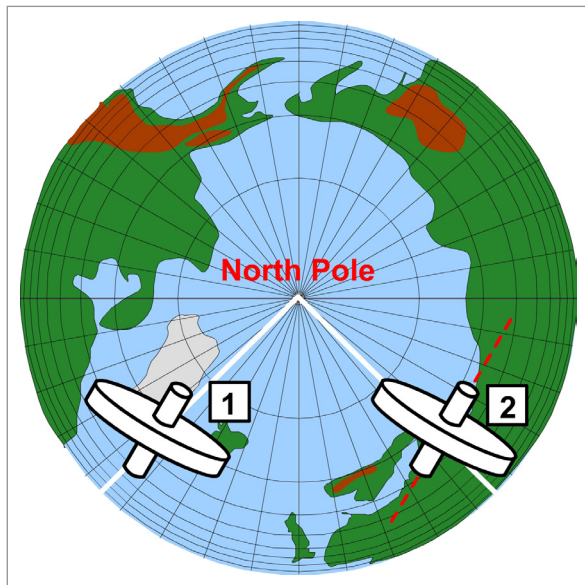
Whether earth rate is positive or negative depends on the earth hemisphere. In [Figure 11.11](#), we are in the northern hemisphere. At position 1, the aircraft and the gyro are both pointing north. At position 2, the gyro remains orientated to the original 'gyro north', but the local earth direction of the parked aircraft is up the local meridian, i.e. true north. If the gyro is taken as the datum, the aircraft is now on a heading of about  $280^\circ$ . The gyro heading has appeared to decrease. We call this a negative earth rate and so it has a value of  $-15 \times \sin \text{latitude } ^\circ/\text{hour}$ .

However, suppose the aircraft were parked in the southern hemisphere, as in [Figure 11.12](#) below.

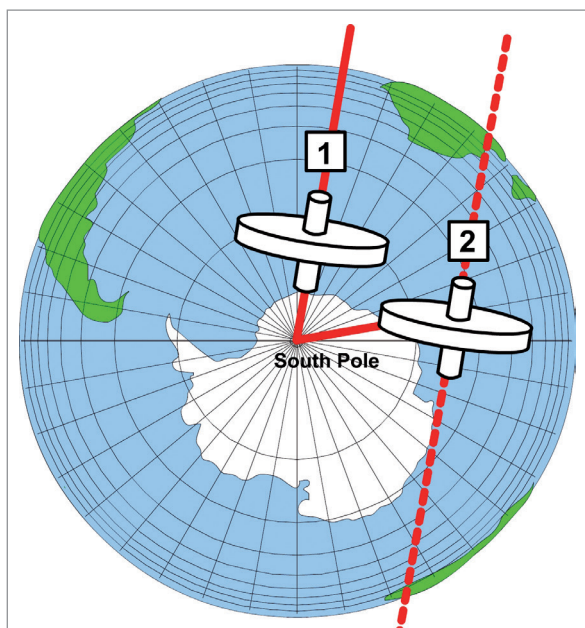
The earth rotates eastwards. As seen from above the N pole, looking down, ([Figure 11.11](#)) this is **anticlockwise**. However, when seen from the S pole looking upwards, as in [Figure 11.12](#), this is a **clockwise** rotation. If the aircraft is parked pointing northwards (12 o'clock in position 1), when the earth rotates, the aircraft is now still pointing in a direction of true north (outwards from the S pole), point about 3 o'clock in the above diagram, but if the gyro is taken as the datum the gyro heading is about  $080^\circ$ . The gyro heading has appeared to increase.

Therefore: **Earth rate is negative in the northern hemisphere**

and: **Earth rate is positive in the southern hemisphere**



*Figure 11.11 Full horizontal component of earth rate at a pole*



*Figure 11.12 Earth rate at the South Pole*

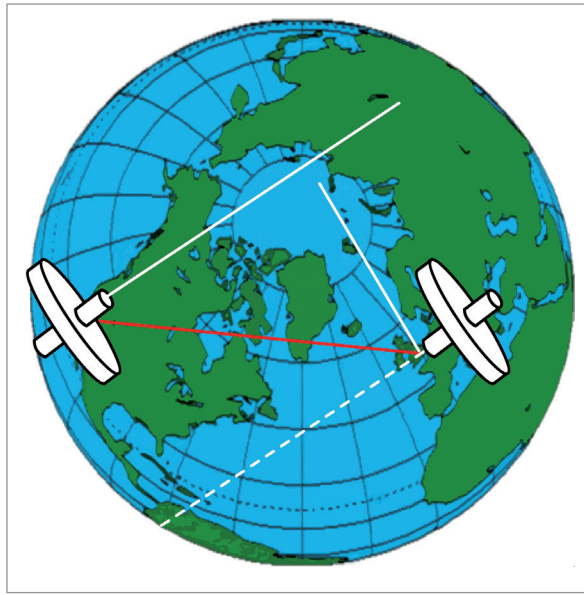


Figure 11.13 Horizontal transport wander

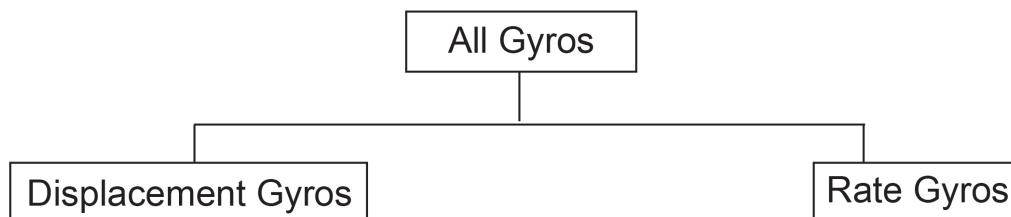
**Transport Wander.** Although transport wander takes place at the same time as earth rate, we shall explain them separately for simplicity. So, disregarding earth rate, imagine the earth stopped in space, not rotating.

The aircraft flies from Los Angeles to London (the red track in *Figure 11.13*). The gyro is aligned with the local meridian at Los Angeles (the local true north). This is the solid white line. When the aircraft arrives in London, the gyro is still aligned with the same direction in space (the dotted white line). However, the actual direction of true north from London is the line connecting London to the North Pole. The difference is the transport wander, which is in fact the difference in the alignment of the meridians at Los Angeles and London.

## Types of Gyro – by Function

Some gyros measure **angles**, e.g.  $10^\circ$  of pitch,  $5^\circ$  of bank,  $30^\circ$  of heading change, etc. These are called **displacement gyros**. Others measure **angular rate**, e.g. a turn rate of  $3^\circ$  per second. These are called **rate gyros**.

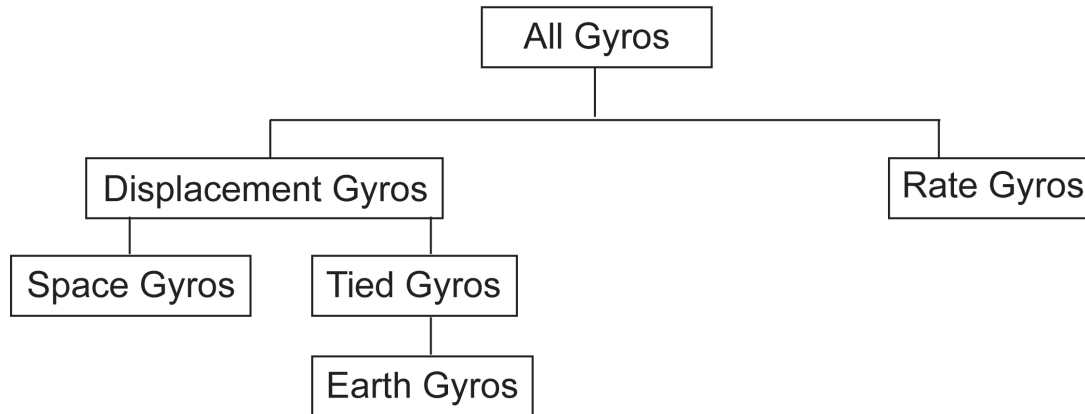
**Displacement gyros** have **2 gimbals and 2 degrees of freedom**. Examples are the **Directional Gyro Indicator (DGI)** and the **Artificial Horizon**. **Rate gyros** have **one gimbal and one degree of freedom** and are used in the **Rate of Turn Indicator** and in **yaw dampers**.



Displacement gyros can be subdivided into space gyros or tied gyros. **Space** gyros have **gyroscopic inertia with reference to a point in space**. They are **free to wander** and, if they do, nothing corrects them back to their original datum. Therefore they need to have a rate of real wander which is so low that it may be considered negligible for practical purposes. Space gyros need to be very accurate indeed and they are correspondingly expensive. They are used in Inertial Navigation Systems.

With **tied** gyros, if they wander, they are **restored back to some orientation** by an external force. Tied gyros are maintained in some particular attitude or direction rather than space. For instance, the directional gyro of a gyro-magnetic compass is slaved to remain oriented to the magnetic north.

An **earth gyro** is one which is maintained vertical or horizontal with respect to local **gravity**. The Artificial Horizon is an example. **Earth gyros are a subset of tied gyros**. Therefore, all earth gyros are tied gyros, but not all tied gyros are earth gyros.



## Types of Gyro – by Construction

**Tuned Rotor.** Tuned rotor gyros are the traditional ‘spinning disc’ type. These are fitted to all elementary and most intermediate types of training aircraft. The basic DGI, Artificial Horizon and Turn Meter covered in the next three chapters are all tuned rotors.

**Ring Laser Gyros.** Ring laser gyros (RLGs) emerged in the 1980s and are now used in nearly all modern airliners. They work by comparing 2 light paths round a glass prism. They offer greater reliability and accuracy but are more expensive.

**Fibre Optic Gyros.** Fibre optic gyros (FOGs) are an extension of the RLG principle. They have only recently become accurate enough for aircraft applications and the Airbus A380 is the first commercial aircraft to use them.

## Types of Gyro – Suction or Electric Power

Tuned rotor gyros may be either **air driven (suction)** or **electric** powered.

For the air driven type, an engine-driven vacuum pump or carburettor venturi pressure (or venturi tube on some light aircraft) reduces the pressure in the instrument case so that filtered replacement air is sucked in and led through a jet which blows onto ‘buckets’ cut in the periphery of the rotor to make it spin – the same principle as a water wheel.

In electric gyros, the rotor is part of an electric motor.

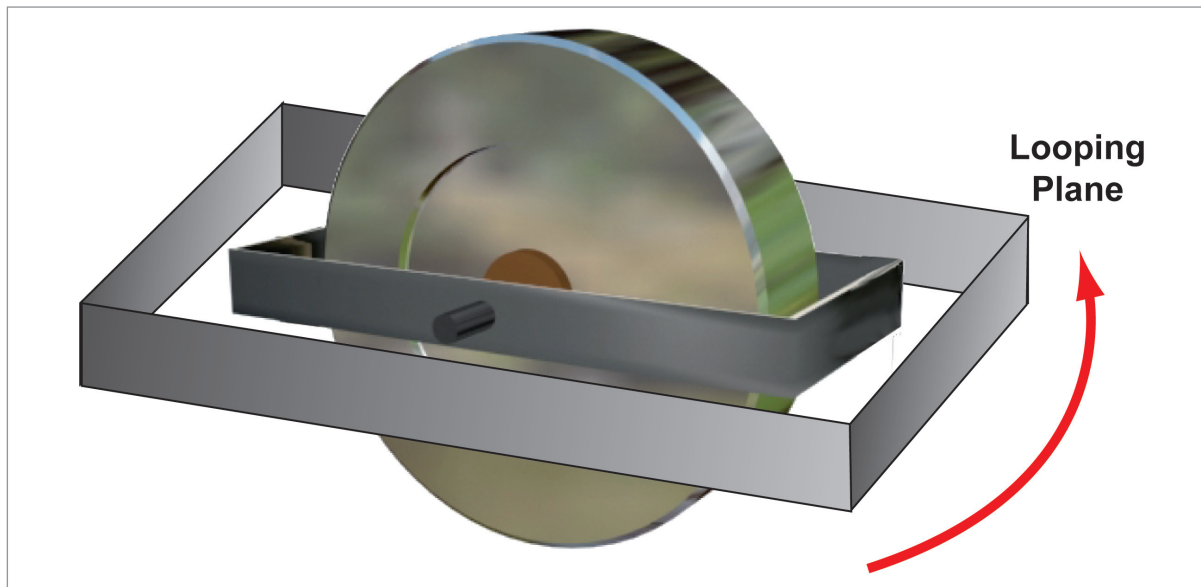
**Suction Gyros.** Suction gyros are **independent of electric power** and so are **not affected by electrical failure**. But moisture, dust, oil and grit in the airflow block the filter, giving **variable rotor rpm**. At high altitude, the engine manifold pressure available may be insufficient to maintain rotor speed. Furthermore, any atmospheric impurities which penetrate the filter can reduce bearing life and unbalance gimbals, thereby **impairing accuracy**.

**Electric Gyros.** Electric gyros are generally more **expensive and heavier** than the air driven type and they require power supplies. However, they can be **faster** and have **more moment of inertia**. The **rotor rpm can be more rapidly achieved** and then **maintained more accurately**.

Therefore, on non-electronic training aircraft, the **main gyro instruments** are **usually electric powered** for greater accuracy whilst the **standby instruments** are often **air driven** in order to still be available after a power failure.

**Alternate Power Sources.** Both types can suffer from failure of power sources. The electric type will have some type of 'flag' indicator to show the pilot if this happens so that he can select standby power or switch to the standby instrument. Pneumatically driven gyroscopes usually have an air pressure indicator (often called the 'suction' gauge) to show failure of the vacuum pump. Some air driven gyroscopes have an alternate power source which is manually selected.

**Gimbal Lock.** The ultimate case of gimbal interaction is **gimbal lock**, which occurs if the aircraft continues its bank to  $90^\circ$ . In this case, the inner and the outer gimbal take up the same orientation, as shown in *Figure 11.14*. The gimbals still give the freedom to continue any roll, but one degree of freedom has been lost. If the pilot were now to pull back on the control column (i.e. rotate in the looping plane), the gyro would now be forced out of its orientation. This would result in precession of the gyro, probably violently, usually described as 'toppling'.



*Figure 11.14 Gimbal lock*

For most unsophisticated aircraft, gimbal lock is a problem and will result in toppling, that is, temporary loss of the use of the gyro until it can be re-erected. For more complex aircraft, especially those with aerobatic manoeuvrability, it can be avoided, either by the use of a fourth gimbal or by a **gimbal flip** mechanism.

Gimbal flip incorporates a powerful torque motor. When the inner and outer gimbals are nearing a locked situation, the motor is triggered to rapidly flip the outer gimbal round by  $180^\circ$ , thereby restoring freedom.

One of the biggest limitations of Direct Indicating Magnetic Compasses is their susceptibility to turning and acceleration errors. Using a more stable datum than a light compass needle or magnetic assembly would reduce or eliminate these. What is required is a datum which has **rigidity in space (the property of pointing in a specific direction, once set up)** and which is stable enough not to be disturbed by the effects of aircraft manoeuvres. A **gyroscope** provides this.



As well as a **direction (heading)** reference, gyroscopes can also provide an **attitude** reference.

On simple aircraft, gyroscopes provide the basis of the:

- Directional Gyro Indicator (DGI)**
- Artificial Horizon**
- Turn and Slip Indicator (or Turn Co-ordinator)**

More complex aircraft use gyroscopes in:

- Gyro-magnetic compasses**
- Inertial Navigation Systems or Inertial Reference Systems**
- Yaw Dampers**
- Autopilots**
- Stabilization of radar scanners**
- and many other applications

## Questions

1. **Rigidity of a gyroscope depends on:**
  - a. weight, force applied and speed of rotation
  - b. rate of precession and the force applied
  - c. weight, rate of precession and speed of rotation
  - d. mass, radius of gyration and speed of rotation
  
2. **A force is applied to deflect a gyroscope. If the rpm of the gyro is then doubled, the precession rate will:**
  - a. remain as before
  - b. increase
  - c. decrease
  - d. cease altogether
  
3. **In gyroscopic theory the term 'topple' is defined as:**
  - a. real wander only, in the horizontal plane
  - b. real wander only, in the vertical plane
  - c. wander, real or apparent, in the vertical plane
  - d. wander, real or apparent, in the horizontal plane
  
4. **A force applied to the spinning axis of a rotor is precessed:**
  - a. through  $90^\circ$  in the direction of spin of the rotor
  - b. through  $90^\circ$  in the direction of spin of the rotor in the northern hemisphere and through  $90^\circ$  in the opposite direction in the southern hemisphere
  - c. through  $270^\circ$  in the direction of spin of the rotor
  - d. at a rate proportional to the speed of rotation of the gyro
  
5. **Real wander of a gyro can be caused by:**
  - a. asymmetrical friction at the spinning axis
  - b. rotation of the earth
  - c. increasing the rpm of the rotor
  - d. moving the gyro north or south of its present position
  
6. **A gyro with only one degree of freedom is known as a:**
  - a. tied gyro
  - b. earth gyro
  - c. space gyro
  - d. rate gyro
  
7. **A perfectly balanced space gyro at the equator has its spin axis aligned with true north. After 6 hours the axis will be aligned with:**
  - a. true east direction
  - b. true west direction
  - c. true north direction
  - d. true south direction

8. The main advantage of electric gyros are:
- a. light weight, high rpm, constant speed, inexpensive
  - b. high rpm, only require low voltage DC, constant speed, sealed casing
  - c. high rpm, high moment of inertia, rapid build-up of speed, constant RPM
  - d. sealed casing, constant speed, high precession rate, low cost
9. Apparent wander of a gyro can be caused by:
- a. rotation of the earth
  - b. clear air turbulence
  - c. gimbal friction
  - d. external torque

## Answers

1	2	3	4	5	6	7	8	9
d	c	c	a	a	d	c	c	a

Chapter

**12**

Directional Gyro Indicator (DGI)

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## Introduction

The **directional gyro indicator (DGI)**, often called the '**direction indicator**' (**DI**) provides a **stable directional reference** in azimuth for maintaining accurate headings and for executing precise turns. There is **no magnetic element** in the DI, so it is not north-seeking and must initially be **synchronized** with the **magnetic compass**. The synchronization must be checked at regular intervals because of real and apparent gyro wander (drift). The **DGI** does not therefore replace the compass; its stable, dead-beat indications are **complementary** to the north-seeking capability of the **compass**. Having no magnetic element, the DGI does not suffer from the compass turning and acceleration errors produced by the vertical component of the earth's magnetic field.

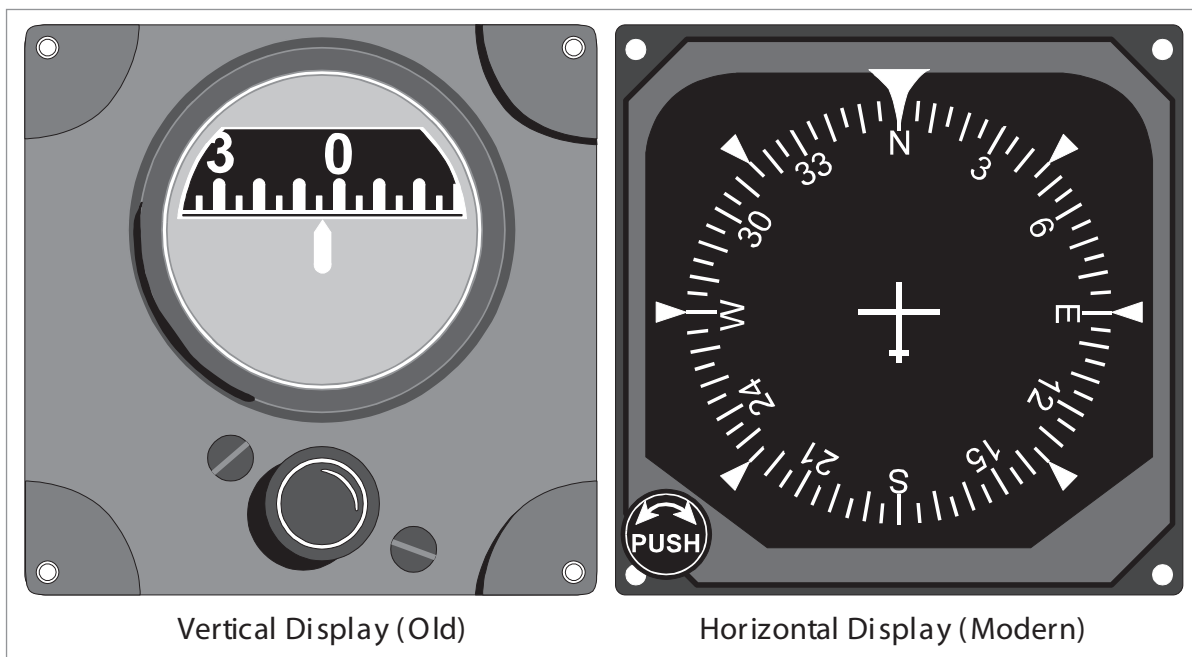


Figure 12.1 Two directional gyro indicators

## The Principle and Construction of the DGI

The DI employs a **tied gyro**, that is to say, a gyro having freedom of movement in three planes mutually at right angles but with the **rotor axis** maintained in the **yawing plane** of the aircraft. This means that the rotor axis is horizontal in level flight, and because of gyroscopic rigidity it provides the datum from which heading can be measured.

The rotor is mounted in the inner gimbal (on bearings mounted in the outer gimbal) which has restricted freedom to turn. The outer gimbal can rotate through  $360^\circ$  about the aircraft's vertical axis, on bearings in the case.

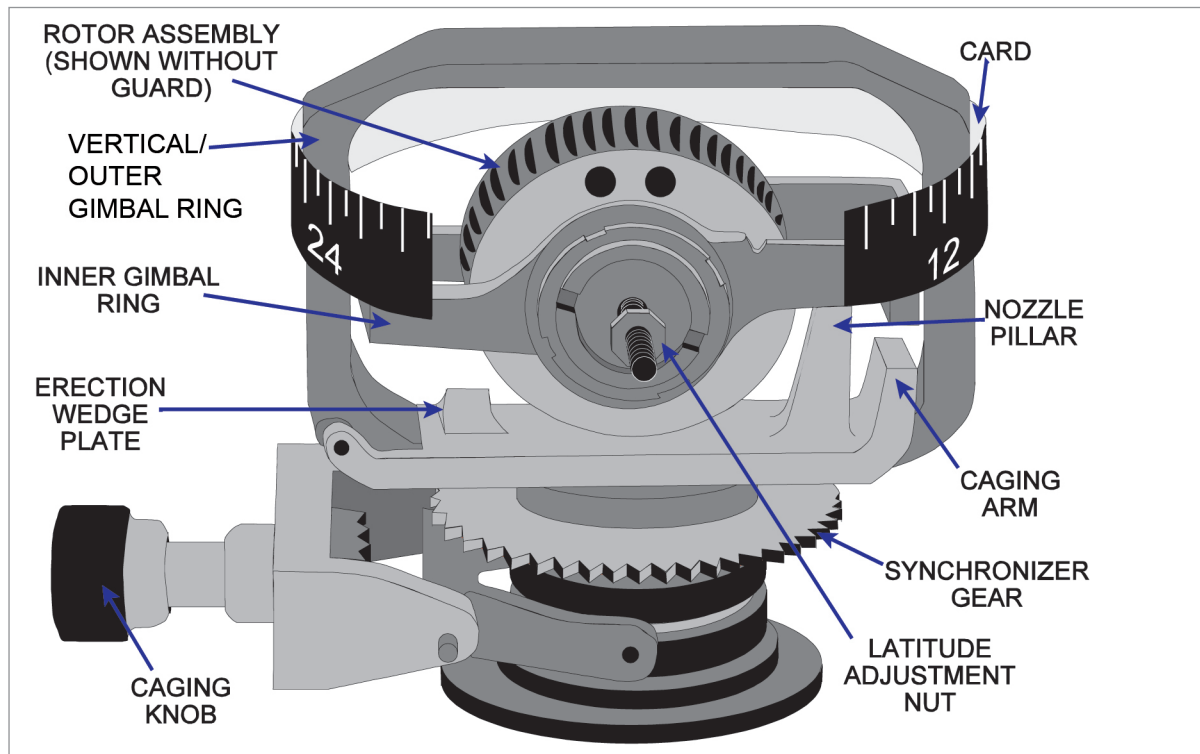


Figure 12.2 An air driven directional gyro

Note that the rotor axis, the inner gimbal axis, and the outer gimbal axis are mutually at right angles.

During a turn, the aircraft and instrument case turn on the vertical axis bearings of the outer gimbal whilst the gyro rotor, gimbals and indicating scale all remain fixed in azimuth because of gyroscopic rigidity.

Heading is indicated on the scale by a lubber line painted on a glass window in the instrument case. Some designs have a circular vertical-card indicating scale geared to the outer gimbal, in place of the cylindrical scale fixed to the outer gimbal in the earlier type shown in [Figure 12.1](#).

### The Control System - Suction Gyros

With earlier designs of DGI, the rotor is driven by twin jets of air applied from the outer gimbal ring. Suction is applied to the case of the instrument, and replacement air enters the case through a filter and is ducted to the jets on the outer gimbal which act on 'buckets' cut in the rotor.

The jets not only spin the rotor but also serve to maintain or tie the rotor axis in the yawing plane of the aircraft.

The rotor axis is lying in the yawing plane and therefore at right angles to the outer gimbal axis, the full force 'X' of the jets being used to drive the rotor ([Figure 12.3](#)). If the aircraft banks, gyroscopic rigidity keeps the rotor axis fixed in space and it is therefore no longer in the yawing plane.

The outer gimbal axis is no longer at right angles to the rotor axis, so the jet driving force 'X' acts at an angle to the plane of the rotor.



This force can now be resolved into two components, component 'Y' in the plane of rotation maintaining the spin of the rotor, and component 'Z' acting at 90° to the plane of rotation (Figure 12.4).

Because this is a gyro, the component 'Z' will precess the rotor as if the force had been applied at a point 90° around the circumference of the rotor in the direction of its spin.

The result will be as though a force 'Q' (Figure 12.4) was operating to re-erect the rotor with its axis in the yawing plane. If the heading is such that the rotor axis is aligned with the longitudinal axis of the aircraft, the application of bank alone (with no turn) will not displace the rotor axis from the yawing plane. This aspect is mentioned again in the paragraph on limitations.

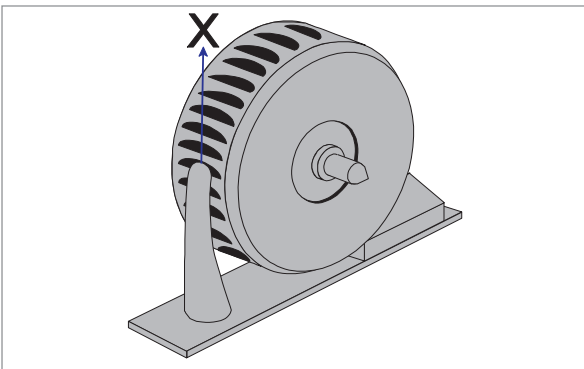


Figure 12.3 Rotor axis in yawing plane

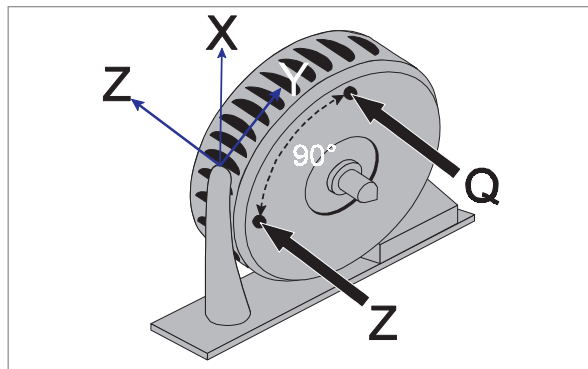


Figure 12.4 Rotor axis displaced

A second control system, which is usually combined with the above system works as follows: The jet of air spins the rotor and then flows round the outside of the rotor inside a metal case, as shown in Figure 12.5.

The air leaving the case is directed at a wedge plate fixed to the outer gimbal. When the gyro is correctly erected, this 'exhaust' jet is divided by the wedge plate into two equal streams producing equal reactions on the outer gimbal (R1 & R2). As soon as the rotor axis is displaced from the yawing plane the streams become unbalanced (Figure 12.6) and the reactions on the outer gimbal at the wedge plate also become unequal.

The resultant of these reactions applies a torque to the outer gimbal about the vertical axis of the gyro. This torque is instantaneously transmitted by the outer gimbal to the inner gimbal and is represented by force 'F' in Figure 12.6. This makes the rotor and inner gimbal precess.

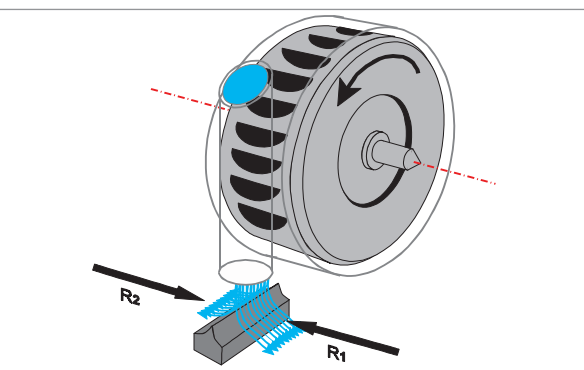


Figure 12.5 Rotor axis in yawing plane

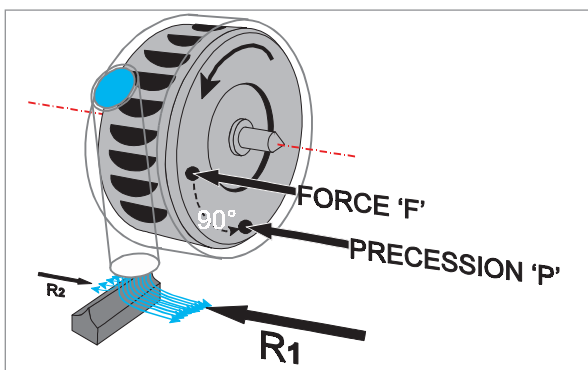


Figure 12.6 Rotor axis displaced

Thus an effective force 'P' acts to re-erect the rotor axis back into the yawing plane. If the gyro was displaced so far that the jet was nowhere near the wedge plate, then the first system would restore the gyro to its correct position.

The jet provides coarse adjustment and the wedge plate fine adjustment.

### The Caging Device

On the front of the instrument (see [Figure 12.2](#)) there is a caging knob which, when pushed in, moves a caging arm which locks the inner gimbal at right angles to the outer gimbal so locking the rotor axis in the yawing plane. At the same time a gear engages with the outer gimbal so that by turning the knob the gyro can be rotated and the scale reading synchronized with (usually) the compass reading.

The caging device is designed as described in order that:

- The DGI can be synchronized with the compass and reset as required.
- The gyro will not topple during synchronization.
- Toppling and possible damage to the instrument can be prevented by caging before manoeuvres in which pitch and roll limits may be exceeded.
- The gyro can be instantly re-erected and re-synchronized if it has toppled.

### DGI Limitations

If the aircraft exceeds the pitch or roll limits of 85° (55° in air driven gyro DIs) the gyro will topple as the inner gimbal comes up against the stops, the precession causing the outer gimbal and scale to spin rapidly. Exceptions to this can occur:

If the rotor axis is athwartships - 360° of aircraft rotation in the looping plane then being possible without toppling the gyro.

If the rotor axis is fore and aft - 360° of roll then being possible without toppling.

The actual indications on the scale at which these two situations can arise depend on the vintage and manufacture of the instrument.

### DGI Errors

There are several reasons why it is virtually impossible for a DGI to remain synchronized with the compass. The most significant errors are listed below and dealt with in subsequent paragraphs.

- Gimbaling errors.
- Random wander.
- Apparent wander due to earth's rotation.
- Errors resulting from varying rotor rpm.
- Apparent wander due to change of aircraft position (transport wander).

## Gimballing Errors

These are **errors** in the indications of the DGI which occur when **bank is applied**. If the errors during a 360° turn are plotted, an approximate double sine curve results, with zero error on four headings (90° apart) spaced between alternate positive and negative peaks (two of each). The curve becomes more complex if pitch changes are made during the turn. The actual readings on the DI at which the maximum errors occur depend on its make and mark.

The errors are small, provided deviations in attitude from the level position are only moderate, and they disappear as soon as level flight is resumed. They occur because of the geometry of the gimbal system, in that unless the instrument case (and the aircraft bolted to it) are able to rotate about one of the axes of the gyro, the outer gimbal itself must move - giving an error - if the rotor axis is to maintain its fixed direction.

## Random Wander

The gyro rotor axis may change its direction in space (real wander) or appear to change its direction (apparent wander) or suffer from both.

More details of real wander, which is mainly the 'random wander' due to **manufacturing imperfections**, are given in the chapter entitled 'Gyroscopes'. In the case of the DGI, gyroscopic rigidity is high and random wander (or drift) rates are low.

An air driven type with the rotor spinning at 10 000 rpm has a drift rate of about 1.6°/hour. A later design with rpm of 20 000 has a quoted drift rate of 1.2°/hour.

Lower rates of only a few degrees per hour are possible with electrically driven indicators. The random wander rates with gyroscopes used in inertial navigation systems may be less than 0.01°/hour.

## Apparent Wander (Due to Rotation of the Earth)

The apparent wander (or drift) of an azimuth gyro has already been mentioned briefly in the chapter on gyroscopes. It is now necessary to consider the magnitude of this wander.

An azimuth gyro (with the axis of the spinning rotor horizontal) is set up in gimbals and frame at the North (or South) Pole. The rotor axis will stay rigid in space (assuming zero real wander) while the earth rotates under it through 360° in one day or  $360/24 = 15^\circ$  in one hour.

An observer standing still watching the gyro will move (with the earth) once round it in 24 hours (*See Figure 12.7*).

If the gyro is the DGI, its reading will be **decreasing** (at the **North Pole**) at a rate of 15°/hour. At the South Pole the reading would **increase** at the same rate. This is the **maximum** rate of apparent wander due to the earth's rotation.

*Figure 12.8* shows a gyroscope set up on the ground at the **equator** with the axis horizontal and aligned north/south. In 24 hours, the observer and gyro will move with the earth once round the earth's axis of rotation. There is no change in the direction of the rotor axis relative to the meridian, so there is **zero apparent drift**. The **apparent drift rate** due to the earth's rotation is therefore a **function of latitude**, being maximum at the pole and zero at the equator.

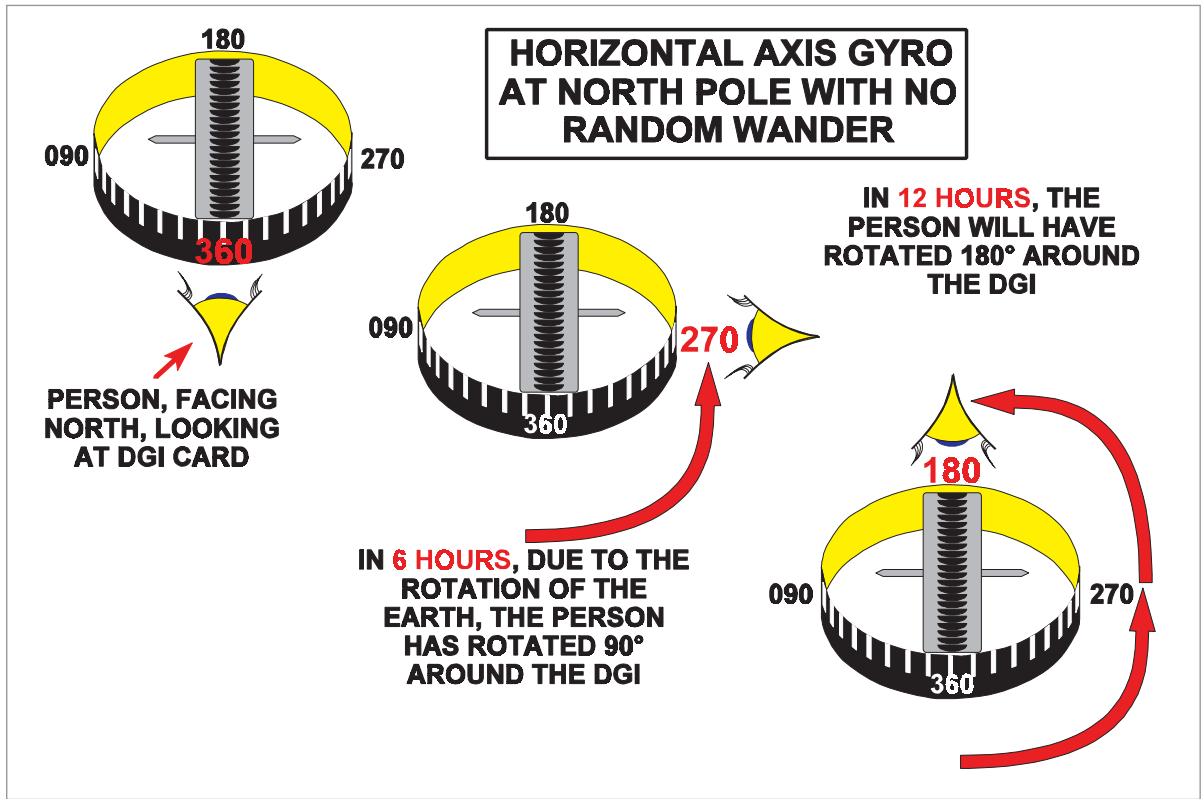


Figure 12.7 Apparent wander at the North Pole

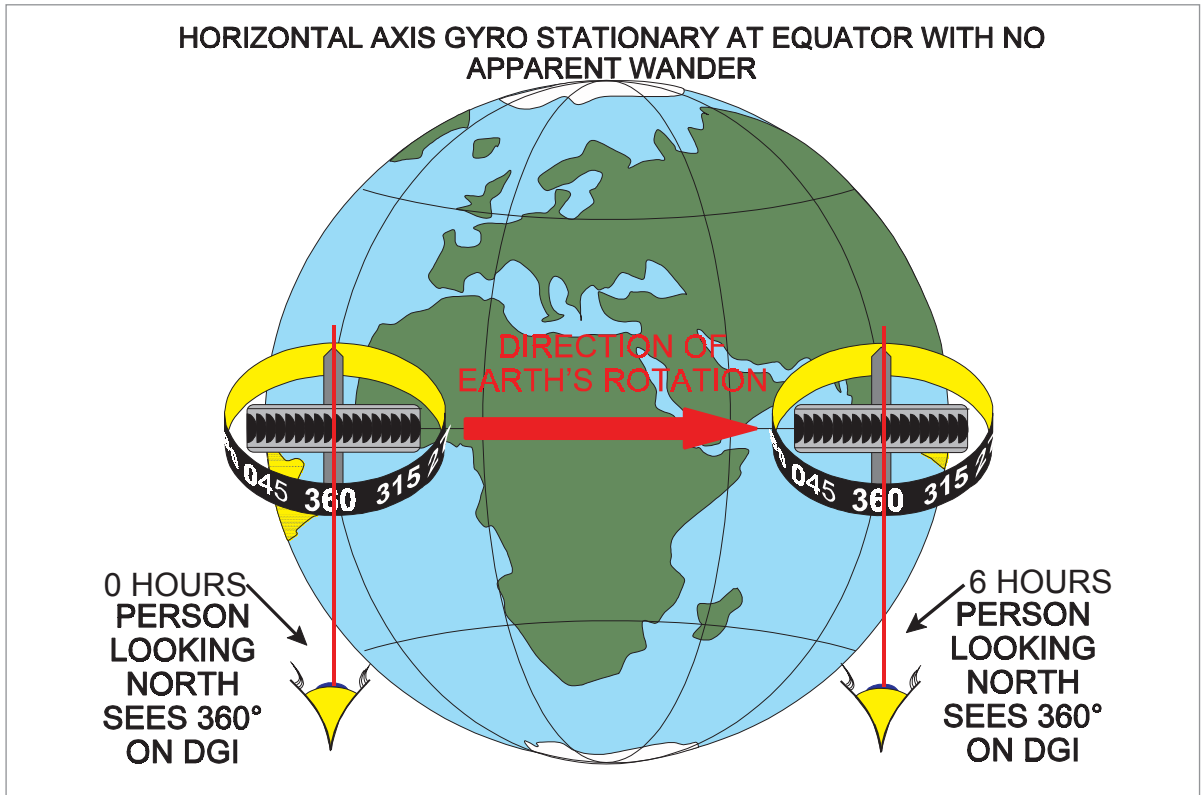


Figure 12.8 Apparent wander at the Equator

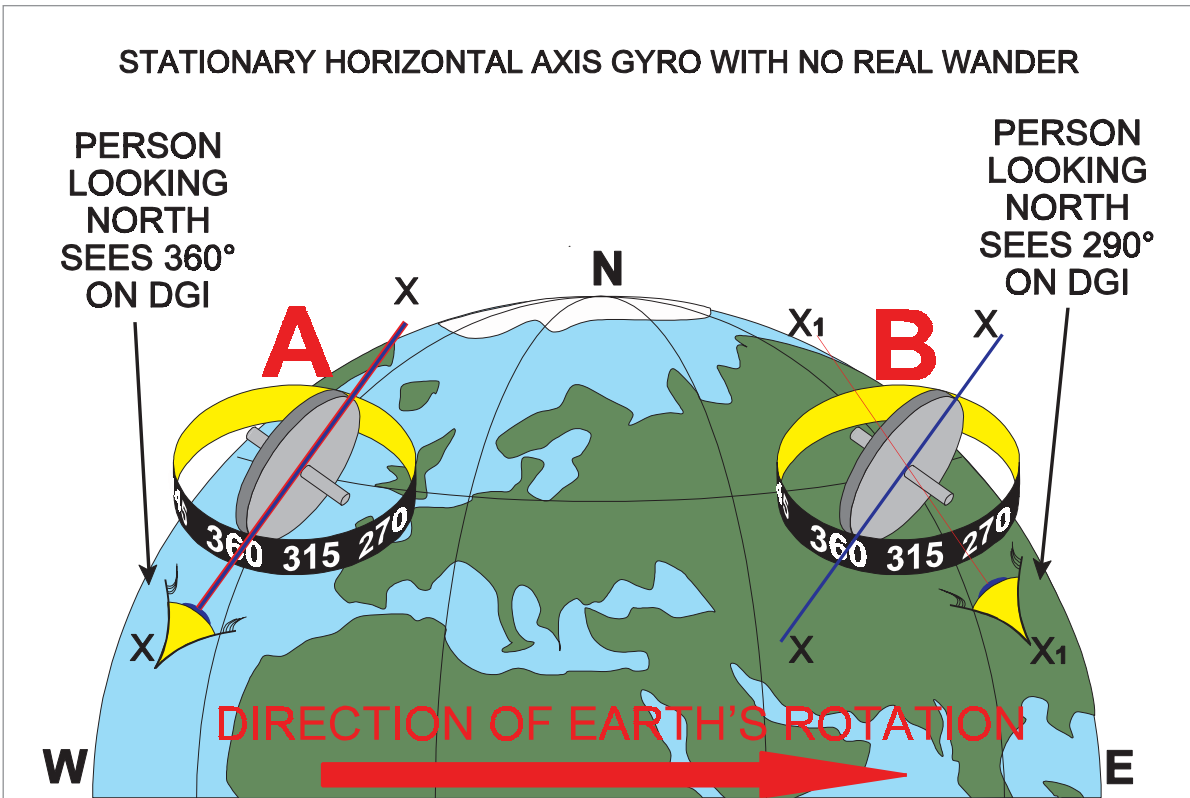


Figure 12.9 Apparent wander at intermediate latitudes in the northern hemisphere

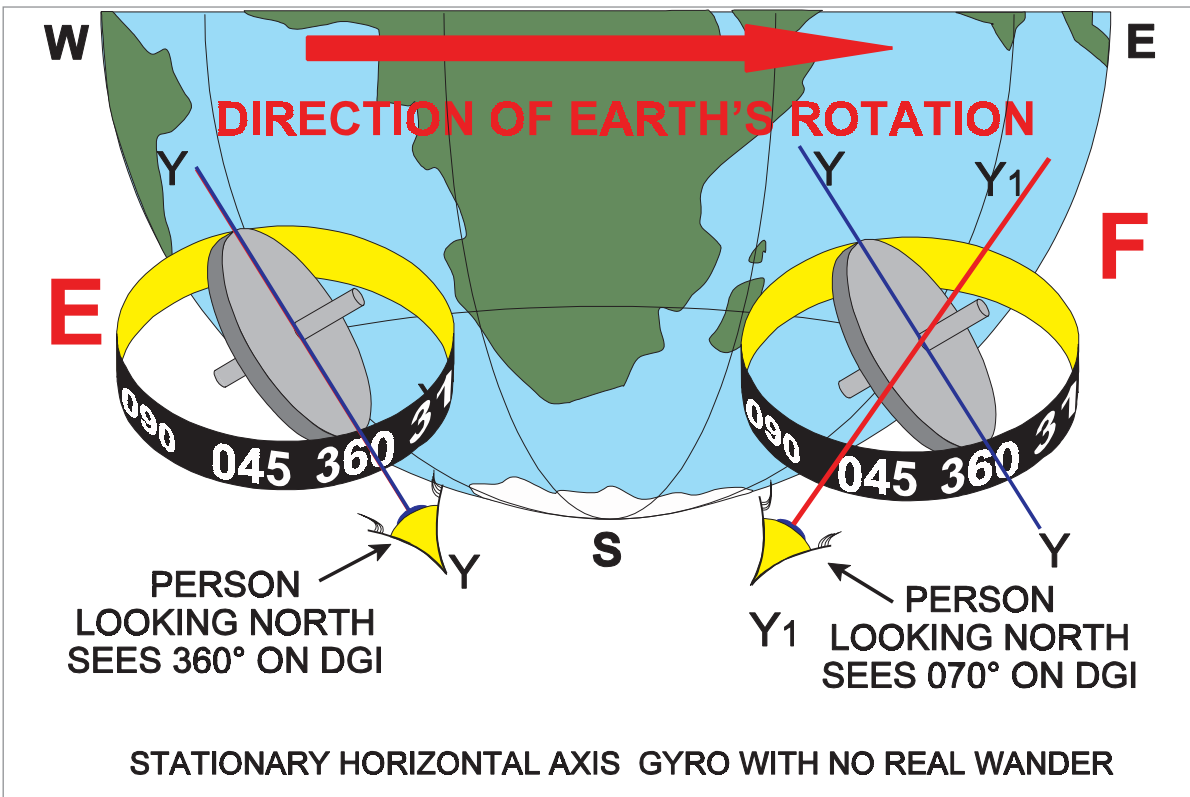


Figure 12.10 Apparent wander at intermediate latitudes in the southern hemisphere

Figure 12.9 demonstrates how the apparent drift due to the earth's rotation causes the reading of a DI to **decrease in the northern hemisphere**.

At A, an observer looking north at the DGI reads 360°. When the observer and his gyro have rotated with the earth to B, the observer will see a value some degrees west of north (a DGI reading of less than 360°) because gyroscopic rigidity is keeping the gyro rotor axis aligned with a fixed direction in space.

The rotor axis cannot remain aligned N/S with the meridian because the latter, except at the equator, is continually changing its direction in space - its spatial orientation - as the earth rotates.

As the observer and gyro continue to rotate with the earth, the readings will decrease further. Similarly, it can be seen that if an observer and gyro located in the **southern hemisphere** rotate with the earth from E, the readings of the DGI will **increase**.

Figure 12.11 shows graphically the variation of apparent drift with latitude. The drift rate is proportional to the sine of the latitude, so that assuming there is zero random drift and no compensation has been made:-

**Apparent drift rate =  $15 \times \sin \text{ lat}$  (degrees per hour)**

Note that this can only be correct if the gyro is 'stationary', meaning that it is not being moved or 'transported' from one place to another.

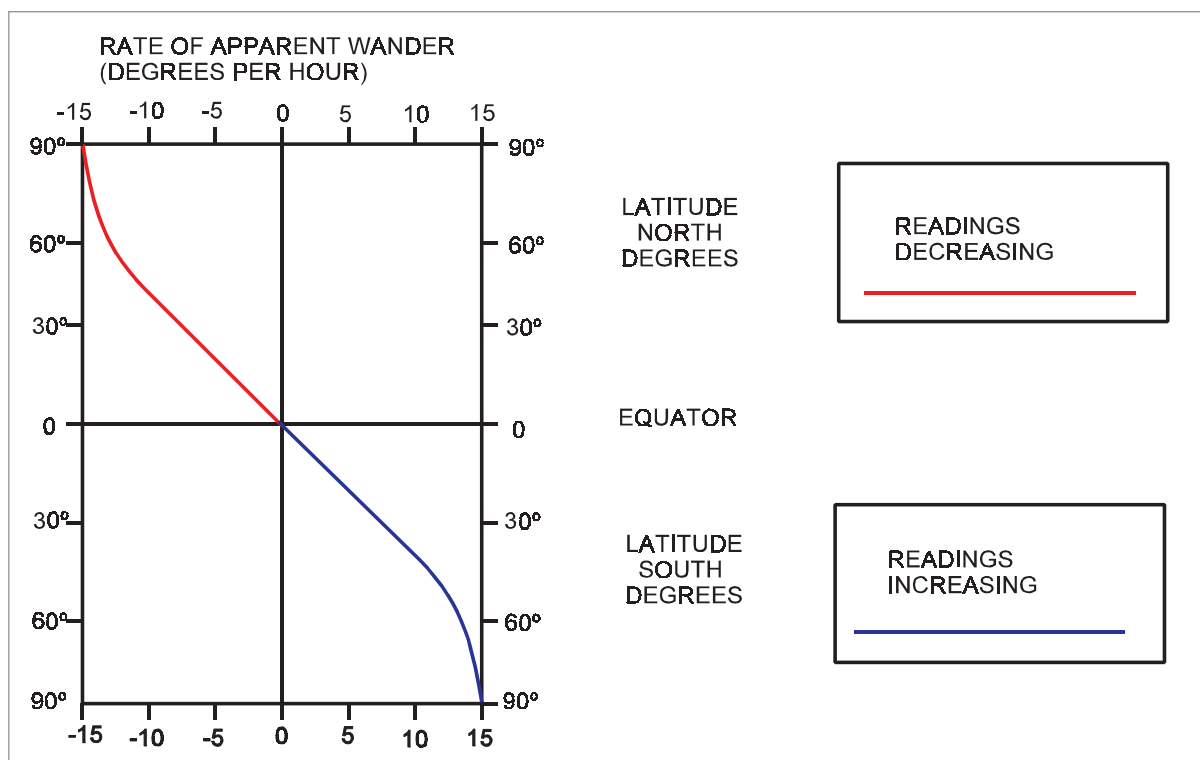


Figure 12.11 The rate of apparent wander on an uncorrected gyro

## Latitude Nut Correction

Compensation for apparent wander (drift) due to the rotation of the earth is by means of an adjustable **latitude rider nut** on a threaded stud fixed horizontally to the **inner gimbal**.

In its central position (*Figure 12.12*) the effect of the nut is cancelled by a counter-balance weight on the opposite side of the gimbal.

Screwed out a few turns, the nut applies a downward moment on the gimbal. This force, with the usual precession rule applied, produces (viewed from above) an anticlockwise precession of the gyro, including its scale, in azimuth. This would cause the reading in the window to increase.

Conversely, if the nut is wound in, clockwise precession occurs, making the readings decrease. Apparent drift due to the rotation of the earth can therefore be cancelled for a given latitude by using the latitude nut to produce an equal and opposite real drift. The ability to screw the nut in or out enables compensation to be made for increasing readings (southern hemisphere) or decreasing readings (northern hemisphere).

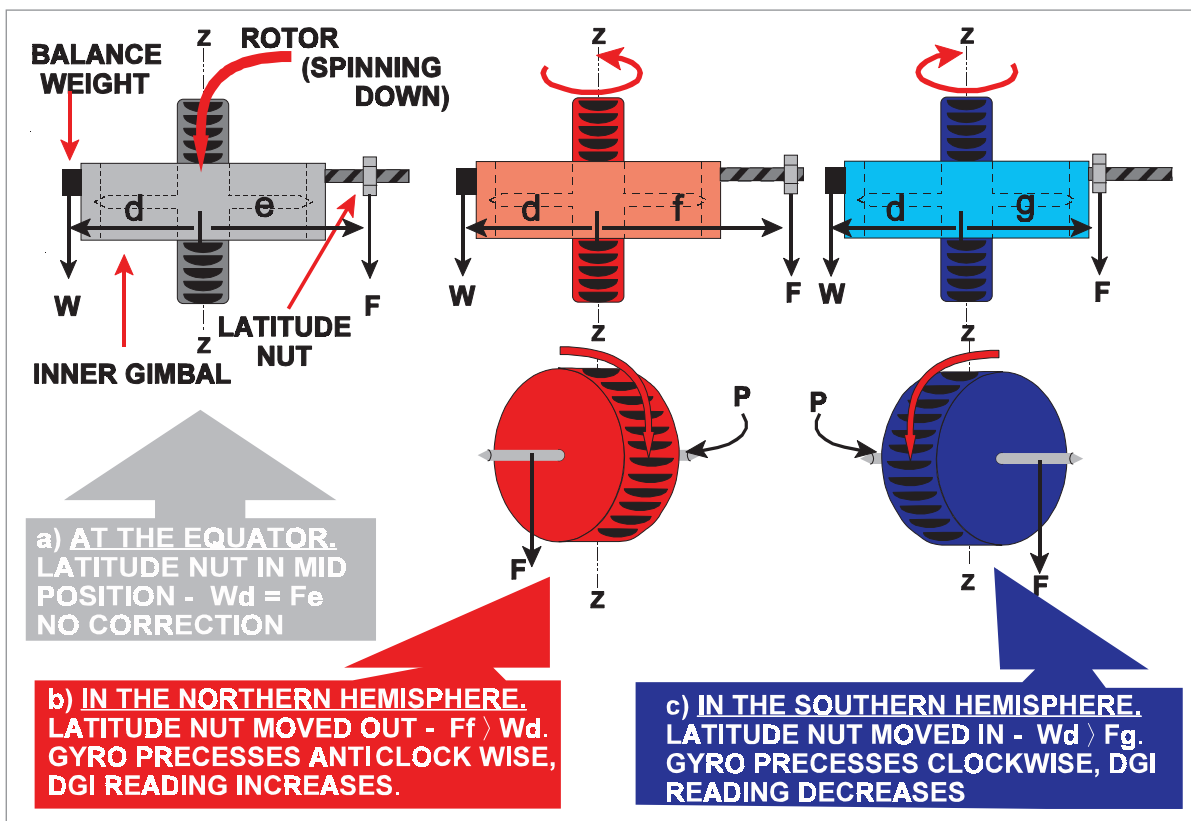


Figure 12.12 Compensation for Apparent Wander

The setting can only be changed under workshop conditions (not in the aircraft) so that compensation will only be correct for the chosen latitude. However, errors due to latitude changes are usually small compared with random wander errors of the DI.

Should the aircraft be moved to a new operating area involving a latitude change of the order of  $60^\circ$ , a DGI with the appropriate latitude correction would probably be substituted.

Figure 12.13 illustrates graphically the effect of compensating a gyro for the apparent drift of minus  $13^\circ/\text{h}$  at  $60^\circ\text{N}$  ( $15 \sin 60^\circ$ ).

The latitude nut introduces a real drift of plus  $13^\circ/\text{h}$  so that the resultant drift (assuming no random error) will be zero at  $60^\circ\text{N}$ . It will be realized that the compensation of  $+13^\circ/\text{h}$  applied for  $60^\circ\text{N}$  will now be present at all latitudes, and this is represented in the graph by renumbering the drift scale.

The drift in the region of  $60^\circ\text{N}$  is negligible after compensation but if the aircraft is now moved to an area south of the equator the drift values will be greater than if no compensation had been made (as a study of Figure 12.13 will show).

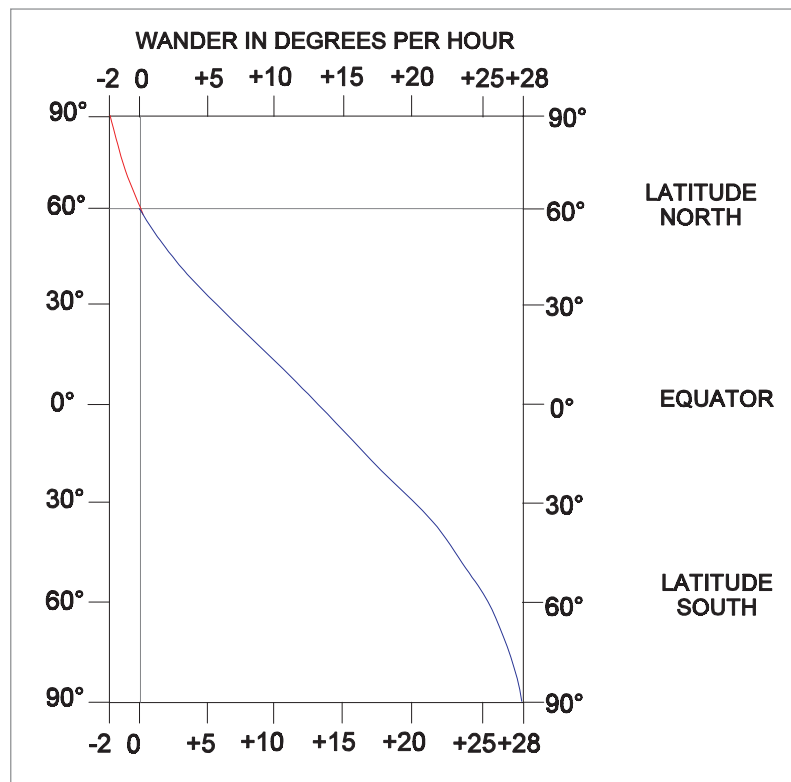


Figure 12.13 Gyro Corrected for  $60^\circ\text{N}$

Summarizing:

Flight north from the 'corrected latitude' gives a decreasing reading (a minus drift rate).

Flight south from the 'corrected latitude' gives an increasing reading (a plus drift rate).

Flight away from the 'corrected latitude' results in the drift rate increasing.

Flight towards the 'corrected latitude' results in the drift rate decreasing.



## Effect of Change of Aircraft Latitude on Compensated DGI

It has already been stated that the apparent drift rate due to the earth's rotation varies with the sine of the latitude (*Figure 12.11*).

If we consider an aircraft tracking due north, starting from the equator, the initial apparent drift rate of an uncorrected gyro is zero. As the flight progresses, the reading of the DGI decreases. By the time the aircraft reaches 30°N, the DGI reading is decreasing at a rate of 7½°/h, and at 60°N it is decreasing at 13°/h. It should be clear that in flight due north or south from the equator at constant ground speed the value of the apparent drift rate will increase from zero to a value of  $15 \sin \text{lat } ^\circ/\text{h}$  at the pole. The rate of increase of drift rate will not be linear - because of the sine function. The same applies if a compensated gyro is transported north or south of its latitude of correction.

## Errors Due to Unstable Rotor rpm

Since the rate of **precession** of a gyro **depends on rotor rpm**, over which no precise control is maintained in a suction-driven DGI, the latitude nut compensation is only approximate.

For instance, at high altitude with **inadequate suction**, the **rotor rpm** will be **lower** than the design value. This results in reduced gyroscopic rigidity and the latitude nut produces **too high a precession rate**, so **over-correcting** the apparent drift. Should rpm exceed the design figure, which is less likely to occur, the rigidity would increase and the latitude nut would produce a lower rate of precession so under-correcting the apparent drift.

## Transport Wander

At any latitude other than the equator, meridians (which define local north) are not parallel. If the gyro is aligned to one meridian, then flown east to west, the new meridian will be inclined to the old by transport wander.

## Drift Rate Calculations

### Example 1

An aircraft is stationary at 60°N. Calculate the hourly wander rate, for an uncompensated gyro.

### Solution 1

Apparent wander =  $-15 \times \sin 60^\circ (\text{decreasing})^\circ/\text{h} = -12.99^\circ/\text{h}$

## Questions

1. **A directional gyro indicator is basically a:**
  - a. horizontal axis earth gyro
  - b. horizontal axis tied gyro
  - c. vertical axis earth gyro
  - d. vertical axis tied gyro
  
2. **Apparent drift may be corrected in a DGI by:**
  - a. causing the gyro to precess in a clockwise direction (in the northern hemisphere)
  - b. attaching a bias weight to the inner gimbal which makes the gyro precess in azimuth in the same direction as apparent wander
  - c. correcting wander by means of air jets
  - d. attaching a bias weight to the inner gimbal which makes the gyro precess in azimuth in the opposite direction to apparent wander
  
3. **An air driven DGI is corrected for apparent wander at 56°N. If the aircraft is maintaining constant DGI readings:**
  - a. when flying north from 56°N the true heading of the aircraft will decrease
  - b. when flying east from 56°N the true heading will decrease
  - c. when flying south from 56°N the true heading will decrease
  - d. when flying west from 56°N the true heading will increase
  
4. **The formula used to calculate apparent wander of a directional gyro in the northern hemisphere is:**
  - a. +15 sine latitude in degrees for the time of running
  - b. +15 sine latitude in degrees per hour
  - c. -15 sine latitude in degrees per hour
  - d. 15 sine latitude in degrees per hour increasing
  
5. **Errors of the directional gyro are:**
  - a. acceleration error, turning error, altitude error, transport wander, rotor speed error
  - b. gimbaling error, random wander, apparent wander, rotor speed error, transport wander
  - c. gimbaling error, looping error, rolling error, rotor speed error, transport wander
  - d. transport wander, apparent wander, latitude error, turning error, acceleration error
  
6. **The spin axis of a directional gyro is maintained in ..... by means of ..... in an air driven gyro and by means of a ..... in an electrically driven gyro.**
  - a. the horizontal plane; air jets; wedge plate
  - b. the vertical plane; air jets; torque motor
  - c. the yawing plane; air jets; torque motor
  - d. the yawing plane; air jets; wedge plate

7. **The purpose of the caging knob is:**
- a. to prevent the gyro toppling
  - b. to reset the heading
  - c. to reset the heading and to prevent toppling
  - d. to prevent apparent wander
8. **In an air driven directional gyro the air jets are attached to:**
- a. the inner gimbal
  - b. the outer gimbal
  - c. the instrument casing
  - d. the rotor axis
9. **The limits of pitch and roll for a modern directional gyro are respectively:**
- a.  $55^\circ$  and  $85^\circ$
  - b.  $85^\circ$  and  $55^\circ$
  - c.  $55^\circ$  and  $55^\circ$
  - d.  $85^\circ$  and  $85^\circ$
10. **Gimballing error:**
- a. will disappear after a turn is completed
  - b. will remain until the gyro is reset
  - c. will only occur during a  $360^\circ$  turn
  - d. will be zero on only two headings during a  $360^\circ$  turn

## Answers

1	2	3	4	5	6	7	8	9	10
b	d	c	c	b	c	c	b	d	a

Chapter  
**13**

The Artificial Horizon

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## The Artificial Horizon Introduction

The **artificial horizon (AH)** provides the pilot with **information** in terms of the aircraft's attitude both in **pitch** and **roll**. It is a primary instrument, replacing the natural horizon in poor visibility. The attitude display consists of a miniature aircraft shape or 'gull-wing' (tail view) painted or engraved centrally on the inside of the glass face of the instrument, and therefore fixed to the instrument case and the actual aircraft. Behind this representation of the aircraft is the horizon bar, linked to the gyro in such a way that the bar is gyro-stabilized parallel to the true horizon. The artificial horizon may be suction or electrically driven. It is also known as a gyro horizon and attitude indicator.

## Construction

The artificial horizon uses an **earth gyro** in which the spin axis is maintained in, or tied to, the **vertical** by earth's gravity. This means that the plane of the rotor rotation is horizontal, so providing the stable lateral and longitudinal references required.

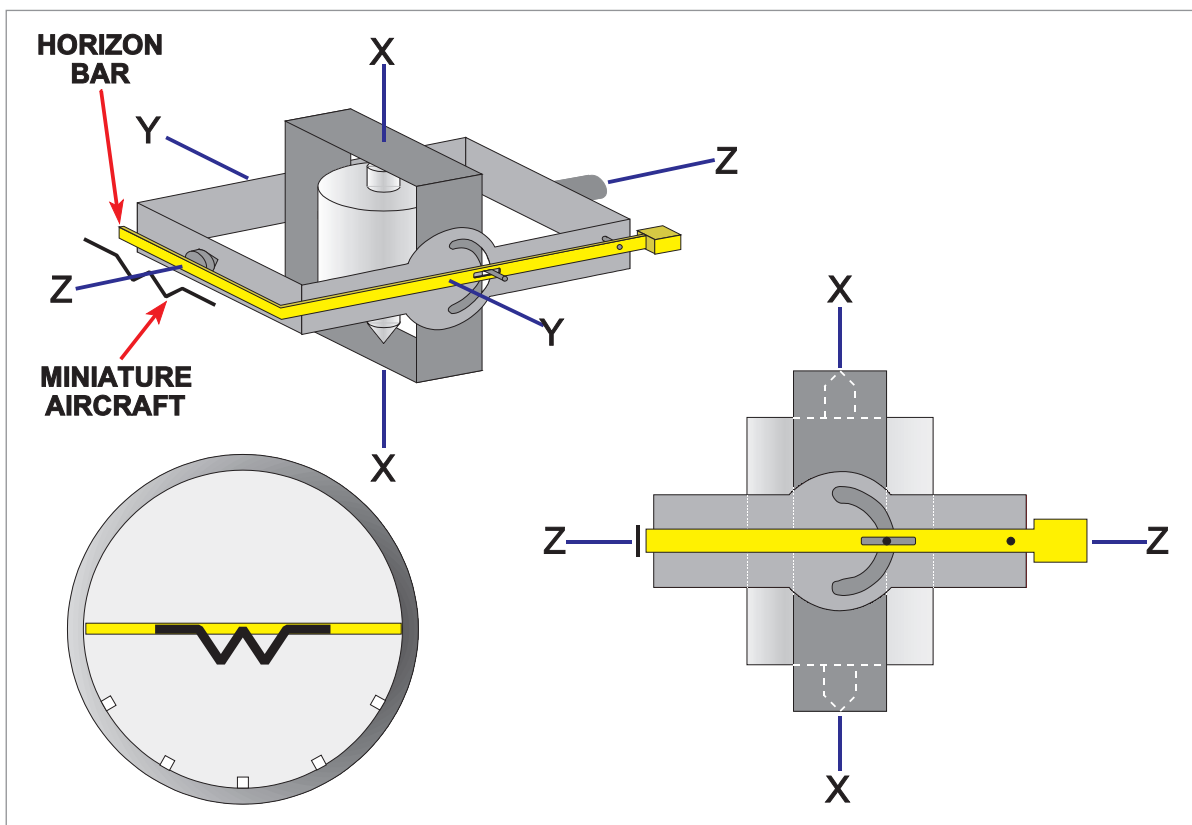


Figure 13.1

## Artificial Horizon Indications

Figure 13.1 shows the three axes of the gyro: XX, YY and ZZ. Because the gyro is tied to the vertical, note that the axis XX (the spin axis) will remain earth vertical and therefore the axis YY will be earth horizontal when the aircraft is straight and level.

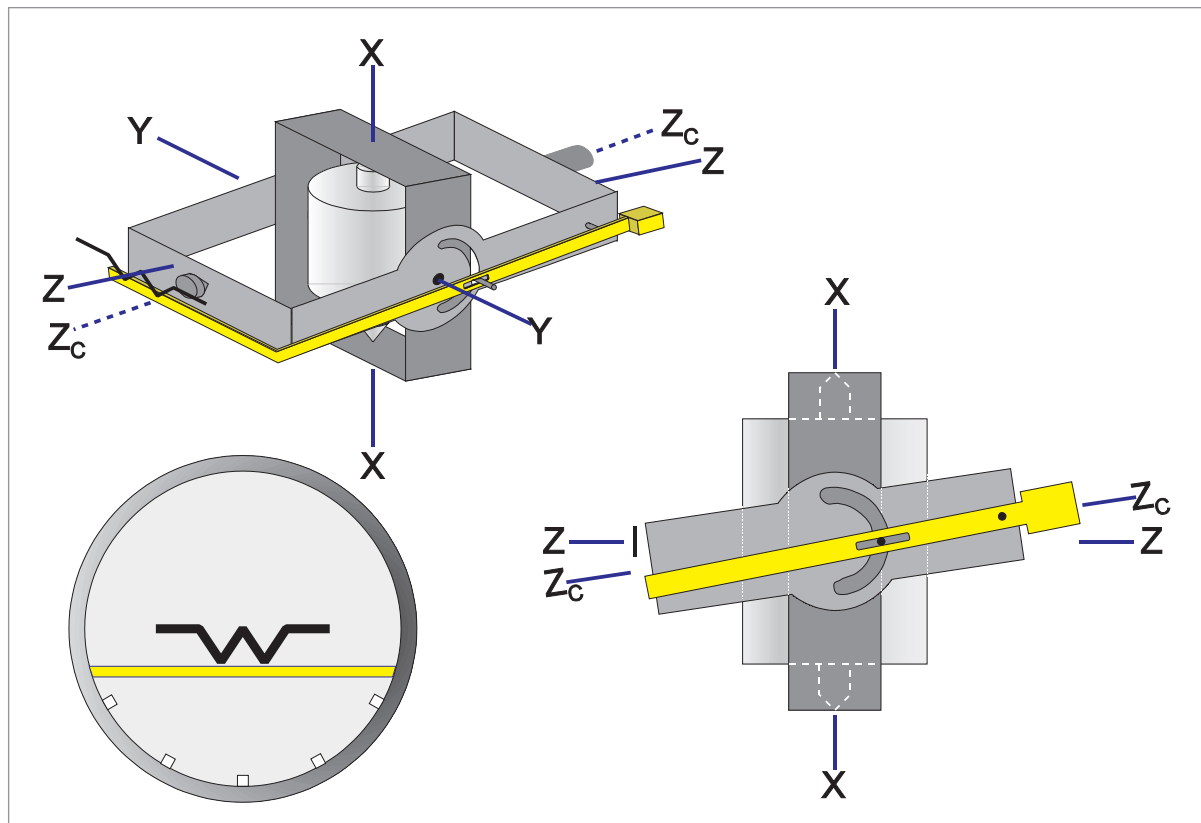


Figure 13.2 A nose-up attitude

### Pitch

Figure 13.1 shows the level-flight attitude display and two views of the instrument with the case removed.

In Figure 13.2 a nose-up attitude (of 10 degrees) is shown. The pitch-up movement has rotated the case together with the attached outer gimbal ring about the lateral axis  $YY$ . As this occurs, a guide pin protruding from the stabilized inner gimbal forces the horizon bar arm down. The horizon bar is now below the gull-wing producing the nose-up indication. (Figure 13.5 shows the gimbal rings and the pitch-indication linkage in greater detail) The three views in Figure 13.3 relate to a pitch-down situation. Note that the angle of pitch may be selected using the pitch markers shown.

### Roll

In roll, as with pitch, the rigidity of the vertical gyro provides the stable attitude reference. As the aircraft rolls (about the longitudinal axis -  $ZZ$  in the diagrams) the instrument case and the gull-wing will rotate about the stabilised gyro rotor and gimbal system.

The gyroscopic rigidity of the spinning rotor holds the horizon bar in the rolling plane so that the amount and direction of bank are displayed by the gull-wing relative to the horizon bar. A more accurate indication of the amount of bank is given by a pointer attached to the outer gimbal and showing bank angle on a scale painted on the face of the instrument.



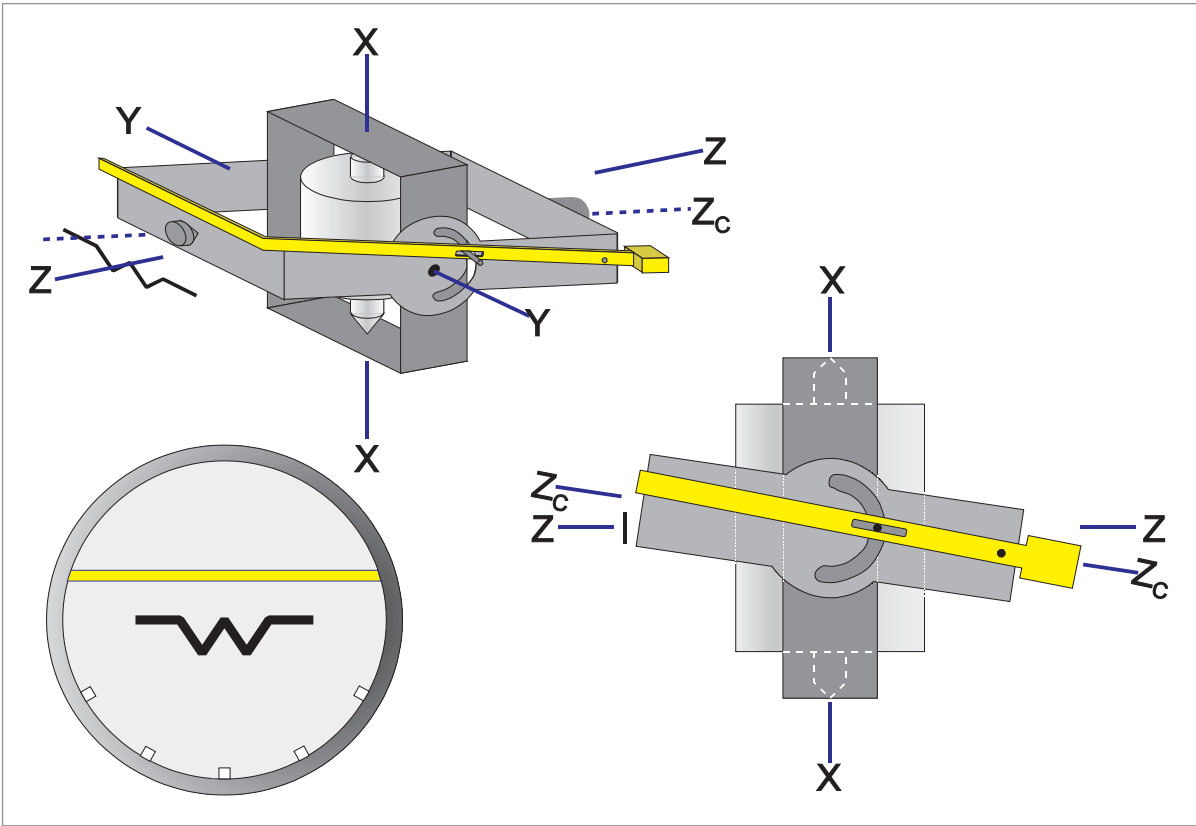


Figure 13.3 A nose-down attitude

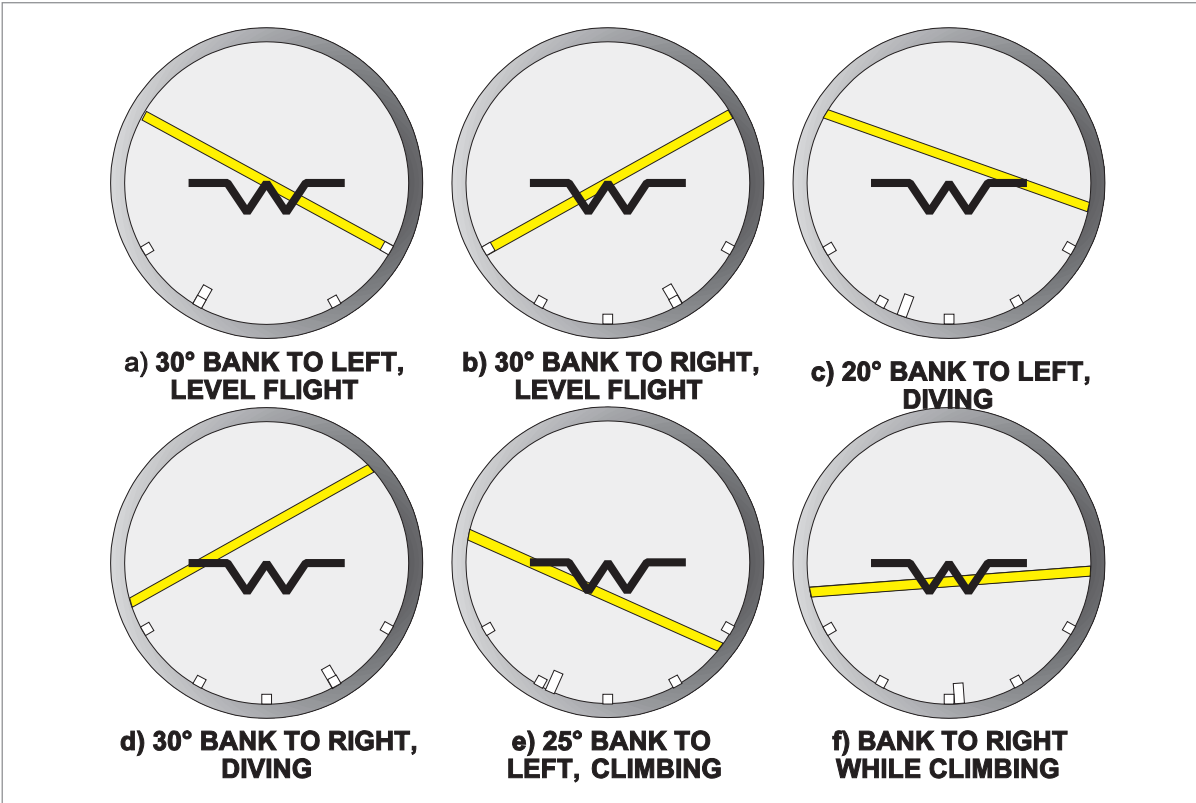


Figure 13.4

Figure 13.4 shows a number of artificial horizon displays with the aircraft in different attitudes.

## Limitations

The amount the case can move relative to the gyro is controlled by fixed stops. With **older designs**, typical limits are  $\pm 60^\circ$  in pitch and  $110^\circ$  each way in roll. In **modern instruments** there is **complete freedom in roll** and up to  $85^\circ$  (plus or minus) in pitch. If the limits are exceeded, the gyro 'topples', giving violent and erratic movements of the horizon bar. Unless a fast erection system is incorporated, accurate indications will not be obtained until the gyro has re-erected itself over a period of 10 to 15 minutes.

## Control Systems

The rotor assembly is made very slightly bottom-heavy in order to keep down the time taken for initial erection when the gyro is first started up, but a complex control system is required to maintain the rotor axis vertical in flight. A **suction or air driven** artificial horizon exhausts air through four slots which are normally half covered by four **pendulous vanes**. **Electric** artificial horizons use **levelling / mercury switches** and **torque motors**.

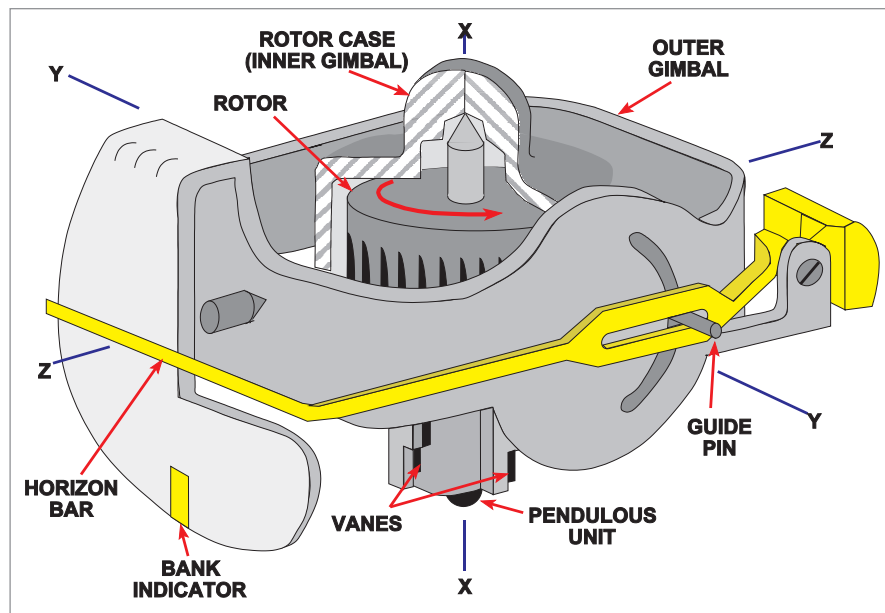


Figure 13.5 The air driven artificial horizon

## The Air Driven Artificial Horizon

In the air driven artificial horizon an engine-driven suction pump (or venturi tube in some light aircraft) is used to create a suction of about 4 inches of mercury in the instrument case. Replacement air, drawn in by this suction via a filter, is ducted through the outer and inner gimbals to enter the rotor case as a jet which spins the rotor at up to **15 000 rpm**. After driving the rotor, the air passes into the instrument case through slots at the base of the rotor housing.

### Control System

The control system of the air driven artificial horizon consists of four slots and four pendulous (hanging) vanes at the base of the rotor housing. The vanes hang down so that when the rotor axis is vertical each slot is half covered by its vane, and four equal jets of air emerge from the slots, fore and aft and left and right, as in [Figure 13.6](#). Because the four jets are of equal strength but in opposite directions no force is exerted on the gyro and therefore no precession occurs - the gyro rotor remaining vertical.

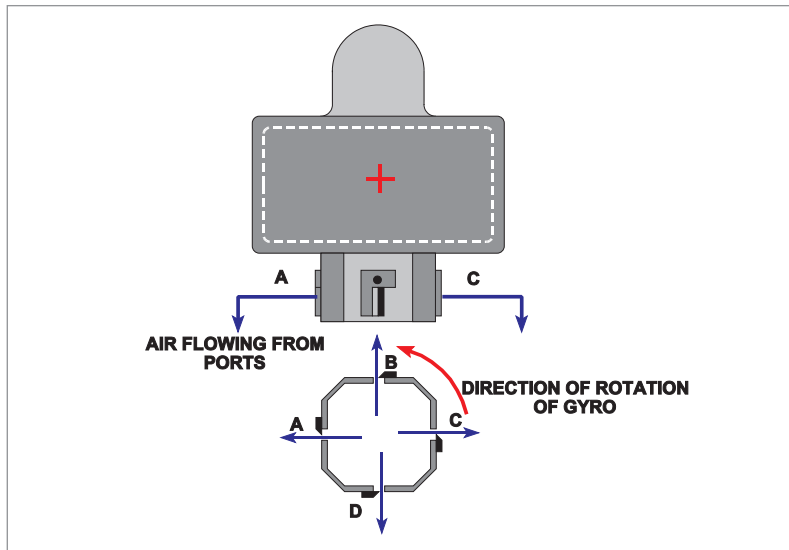


Figure 13.6 Equilibrium

However, the opposing vanes are fixed to a common spindle so that the four vanes operate as two pairs. The positioning of the vanes is such that if the rotor axis wanders from the vertical, one vane will hang clear of its slot, allowing unrestricted airflow, while the opposite slot is completely obstructed by its vane. The resulting unbalanced airflow precesses the gyro and corrects the tilt, returning the gyro axis to the vertical. Exactly how this correction is achieved is shown in [Figure 13.7](#).

The gyro has wandered from the vertical so that vanes A and C are not affected and remain half covering their slots. However, vanes B and D, on a common spindle, hang down so that slot B is now closed and D is wide open. A strong jet exits through D causing an equal and opposite reaction 'R' on the gyro. This reaction is precessed through 90° in the direction of rotor spin (anticlockwise when viewed from the top) and acts in the direction of 'P' which restores the gyro axis to the vertical.

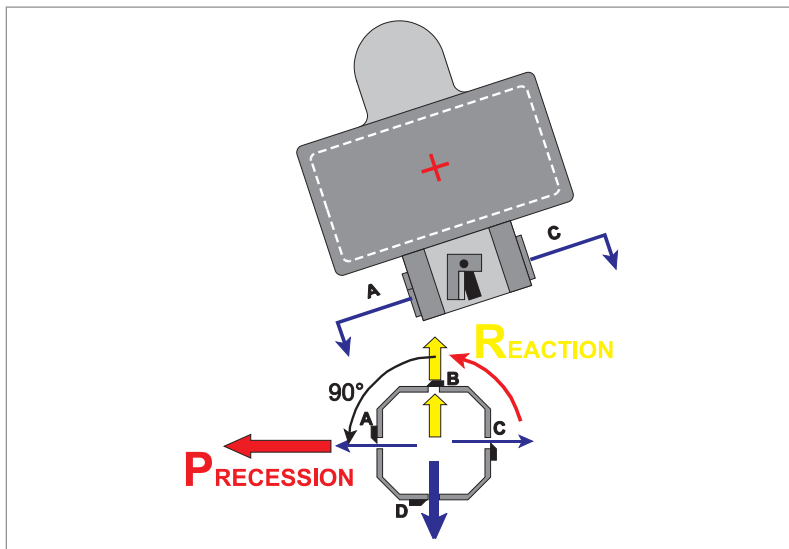


Figure 13.7 Rotor axis displaced from vertical

## Acceleration Error in the Air Driven Artificial Horizon

The control system of the air driven artificial horizon depends on the pendulous vanes being affected by the earth's gravity. However, the vanes will be affected by any acceleration, not just that due to gravity.

When an aircraft **accelerates** in a level attitude (such as during the take-off run) a false nose-up, right wing down, or **climbing right hand turn indication** will result. The **pitch error** is due to the effect of acceleration on the **lateral pendulous vanes**. The **roll error** is due to the inertia of the **bottom-heavy rotor housing**. These effects are now considered in more depth.

- Pitch Error.** During acceleration, the lateral vanes lag, swinging back towards the pilot, opening the starboard slot and closing the port slot. This results in a reaction 'R' which acts to port (see [Figure 13.8](#)). By the rule of precession the effect on the gyro is as if the direction of application of R had been moved 90° in the direction of rotor spin (anticlockwise). The gyro will now be precessed out of vertical with the base moving backwards towards the pilot. As shown in [Figure 13.8](#), this movement is transmitted via the guide pin and horizon bar arm to bring the horizon bar below the gull-wing giving a nose-up indication.

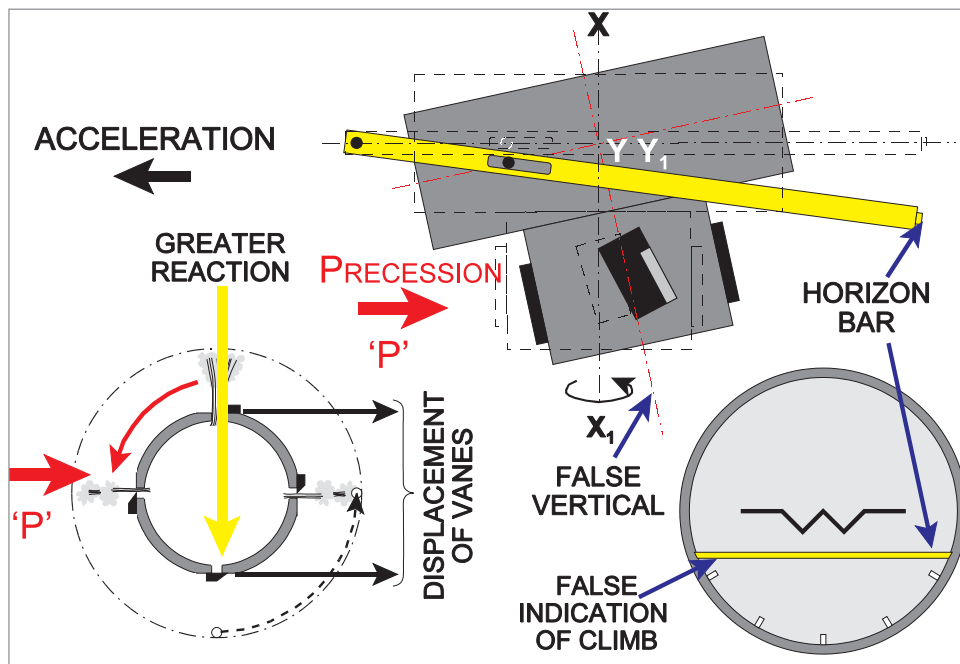
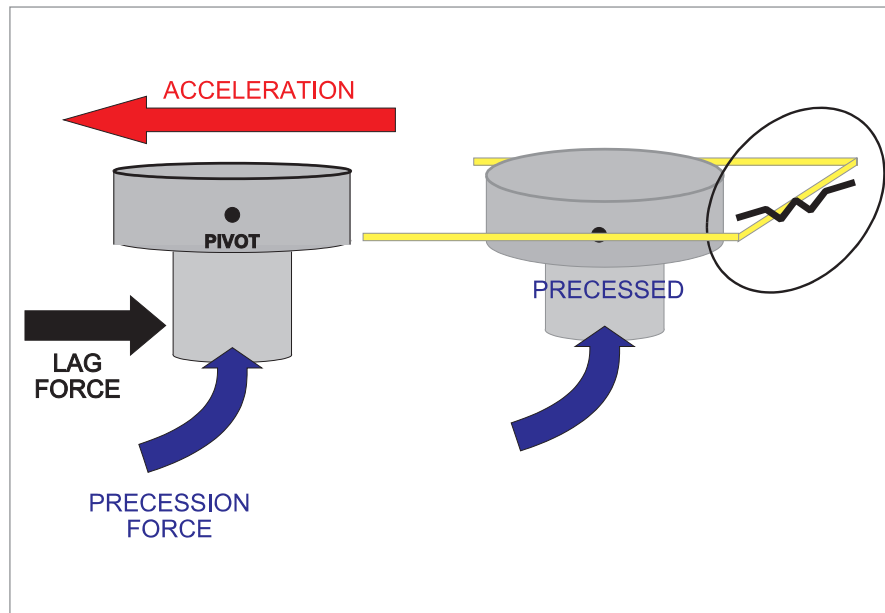


Figure 13.8 Pitch Error Due to Acceleration

- **Roll Error.** Due to inertia, the weighted base of the rotor housing tries to lag during acceleration. However, this force will be precessed, resulting in the base of the rotor housing moving to starboard and the gyro axis precessing out of the vertical (see [Figure 13.9](#)). This rotates the whole rotor / gimbal assembly about the longitudinal axis to give a right wing down indication.



*Figure 13.9 Roll Error Due to Acceleration*

Deceleration will cause a nose-down, left wing low error, the opposite of the acceleration error indication. These errors assume that the rotor is rotating anticlockwise when viewed from the top, which is the case for British air driven artificial horizons. Most electric horizons and some American air driven horizons have clockwise rotor spin, giving opposite errors.

## Turning Errors in the Air Driven Artificial Horizon

Whenever an **aircraft turns** there must be an acceleration towards the centre of the turn (centripetal force). Since the pendulous vanes are now affected by a horizontal acceleration as well as acceleration due to gravity, **errors in pitch and roll indications** will occur. During the turns the centrifugal force will act on the fore and aft pendulous vanes ('erection' error) and weighted base of the rotor housing ('pendulosity' error). The errors are complex and change as the turn progresses, cancelling out after a 360° turn. The magnitude of the errors varies with speed, rate of turn and type of horizon. For a chosen speed and rate of turn, the errors can be **compensated** for by **tilting** the top of the **rotor axis** slightly **forward** (for erection error) and slightly to the **left** (for pendulosity error).

However, in an uncorrected instrument the following errors will occur. (Assume a classic instrument - air driven with the gyro rotating anticlockwise when viewed from above).

Turning through 90°:	Under-reads bank angle	Pitch error – indicating a climb
Turning through 180°:	Bank angle correct	Pitch error – indicating a climb
Turning through 270°:	Over-reads bank angle	Pitch error – indicating a climb
Turning through 360°:	Bank angle correct	Pitch angle correct

The tilts are of the order of 2°. The setting of the horizon bar has to be similarly modified to indicate correctly in level flight. Small residual errors occur, particularly if the speed and rate of turn are not those for which compensation has been applied, but the errors are very much smaller than they would be had no compensation been made.

## Rigidity

High rotor speeds in suction horizons of up to 15 000 rpm result in high gyroscopic inertia. With electric horizons, speeds of 22 500 rpm are typical giving even greater rigidity. Because of the high inertia, precession rates are low and therefore once a horizon topples it will take a significant period for re-erection unless a rapid erection device is fitted.

## Serviceability Checks

**Before Flight.** Check that the horizon bar takes up a laterally level position with the correct pitch indication for the aircraft type, and that this indication is maintained when taxiing. If a caging device is fitted, the instrument should be uncaged at least five minutes before take-off to ensure that the rotor axis has had time to reach alignment with the true vertical.

**In Flight.** The artificial horizon should give an immediate and correct indication of any change in pitch or roll attitude.

## The Electric Artificial Horizon

The main advantage of an electrical artificial horizon over the air driven horizon is its greater rigidity due to its faster spin rate. This greater rigidity results in increased accuracy due to reduced errors. The basic principle of the instrument is the same as the air driven horizon. The vertical gyro is still tied by earth's gravity, but by **mercury / levelling switches** and **torque motors** rather than the pendulous vanes of the air driven horizon.

### Electric Artificial Horizon Control System

The gravity-operated control system consists of mercury / levelling switches (which are fixed to the base of the rotor) and electric torque motors. If a levelling switch is not level, the mercury liquid ball moves from its central position and closes the circuit to drive its torque motor. The torque motor provides the force which is precessed to return the gyro axis to the vertical. There are **two levelling switches**, one to sense **pitch** and one to sense **roll**.

They activate the pitch and roll torque motors respectively which precess the gyro back to the vertical as soon as it starts to wander.

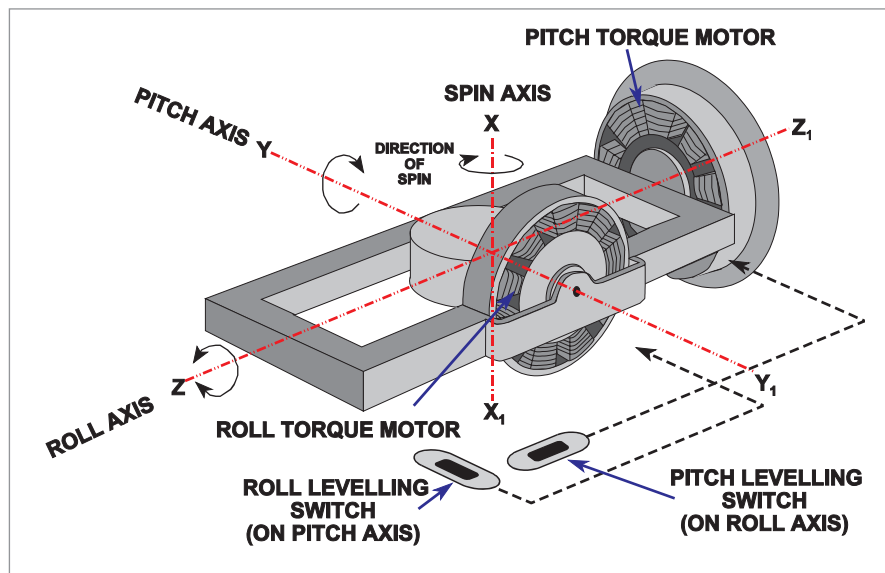


Figure 13.10 The Electric Horizon Control System

Because of the 90° precession rule, the torque motor on the side of the inner gimbal corrects wander in the rolling plane (applying torque round the lateral axis to produce rotation about the longitudinal axis). Likewise the pitch torque motor is on the outer (longitudinal) gimbal so that the precession is about the lateral axis to correct for pitch.

This control system, like that of the suction horizon, is designed to compensate for turning errors by maintaining the rotor axis slightly tilted away from the true vertical and having the horizon bar compensated by a similar amount. The amount and direction of this tilt depends on the particular model of instrument in use.

## Acceleration Errors in the Electric Horizon

Acceleration errors are minimal in the electric artificial horizon for the following reasons:

- The **high rotor speed** of an electric artificial horizon results in very high gyro rigidity and therefore very low precession rates. There is therefore less potential for the gyro to move out of the earth's vertical.
- The **rotor housing is less bottom heavy** in the electric artificial horizon and therefore roll error is reduced while accelerating.
- **Pitch and roll cut-out switches.** When an aircraft in a level attitude accelerates the pitch levelling switch will falsely complete the circuit as the mercury 'ball' moves back in its tube (due to inertia). As this would then result in the pitch torque motor falsely precessing the gyro out of the vertical, a pitch cut-out switch is included in the circuit which activates when an acceleration of 0.18g or greater is detected. Similarly in a turn the roll mercury switch would falsely activate the roll torque motor. A cut-out is therefore incorporated in the circuit which is activated at 10 degrees angle of bank.

## Fast Erection System

In many electric horizons a fast erect system is included to give rapid initial erection and quick re-erection should the instrument topple due to exceeding the operating limits. Quoting typical figures, the normal erection rate of 4° per minute is increased to 120° per minute by pushing the fast erection knob on the face of the instrument. This action **increases the voltage** to the erection torque motors. One of the advantages of having a fast erection system is that the pendulosity (bottom-heaviness) of the gyro can be reduced, so decreasing the turning and acceleration errors.

**Note:** When airborne, the fast erection knob can only be used successfully in level flight with no acceleration. During acceleration or a turn, the liquid level switches would be 'off-centre', and operation of the fast-erection system would align the rotor axis with a false vertical.

## Adjustable Aeroplane Datum

This is a refinement found on some American artificial horizons. The idea is that if when an aircraft is trimmed out to fly **straight and level** it has a pitch up attitude, the 'aeroplane' datum can be adjusted to lie on the horizon. However, there is a risk that such adjustment in flight could result in a misleading datum for flying approach procedures in IMC conditions. The Aeronautical Information Circular 14/1969 discusses this risk in depth, and strongly recommends that in **light aircraft the datum be set before flight and thereafter left well alone.** EASA require that such movable datums be removed or otherwise rendered inoperative on aircraft having a maximum all-up weight in excess of 6000 pounds (2727 kg).

## Vertical Gyro Unit

This unit performs the same functions as the gyro horizon, i.e. it establishes a stabilized reference about the pitch and roll axes of an aircraft. It is sometimes referred to as a remote vertical gyro, or occasionally a vertically axised data generation unit.



Instead of providing attitude displays directly to a dial, it is designed to operate an electrical transmission system to "a steering computer", which is then usually displayed onto a combined attitude indicator and flight director display.

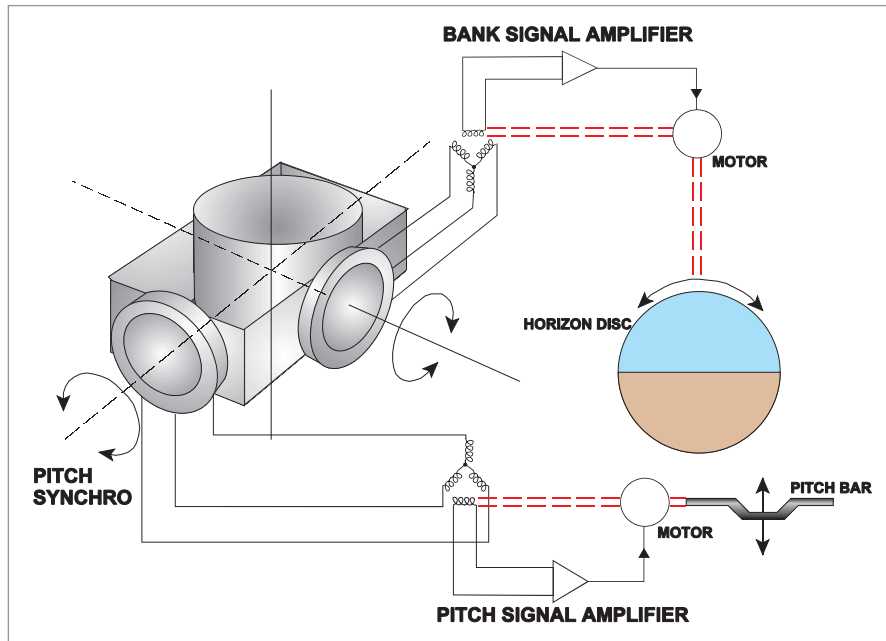


Figure 13.11 Vertical gyro unit

## Questions

1. An artificial horizon utilizes (i)..... to show (ii)..... in (iii)..... and (iv).....

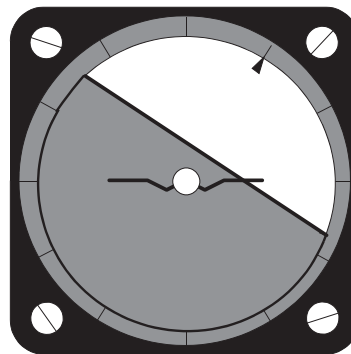
- |    |                   |               |                |                |
|----|-------------------|---------------|----------------|----------------|
| a. | (i) an earth gyro | (ii) position | (iii) latitude | (iv) longitude |
| b. | (i) a space gyro  | (ii) attitude | (iii) degrees  | (iv) minutes   |
| c. | (i) an earth gyro | (ii) latitude | (iii) pitch    | (iv) roll      |
| d. | (i) an earth gyro | (ii) attitude | (iii) pitch    | (iv) roll      |

2. During the take-off run an air driven artificial horizon will usually indicate:

- nose up and incorrect left bank
- a false descending turn to the right
- increased nose-up attitude and right wing low
- a false climbing turn to the left

3. The indication on the right shows:

- a climbing turn to the right
- nose up and left wing down
- 30° starboard bank, nose up
- 30° port bank, nose below horizon



4. False nose-up attitude displayed on air driven artificial horizon during the take-off run is caused by:

- the high pendulosity of the rotor
- the lag of the lateral pendulous vanes
- the linear acceleration cut-out
- incorrect rotor speed

5. The rotor axis of an electrical horizon is tied to the earth's vertical by:

- four pendulous vanes
- the roll cut-out
- the low centre of gravity of the rotor housing
- two mercury level switches and two torque motors

6. False right wing low attitude shown on an air driven artificial horizon during an acceleration is caused by:

- the lag of the base of the rotor housing
- the longitudinal pendulous vanes
- the roll cut-out
- high rotor speed

**7. Inside an artificial horizon:**

- a. the inner gimbal ring is pivoted laterally inside the outer gimbal ring and the outer gimbal ring is pivoted longitudinally inside the case
- b. the inner gimbal ring is tied to the vertical by a control system
- c. the rotor axis is kept level by a calibrated spring attached to the outer gimbal ring and the instrument case
- d. there is only one gimbal ring

**8. When an adjustable aircraft datum is fitted to an artificial horizon in light aircraft:**

- a. it should be checked at regular intervals
- b. it should be set to the central position and left there
- c. it should be rendered inoperative
- d. it should be set to 15°

**9. An electrically driven artificial horizon has less errors during the take-off run because:**

- a. it is less pendulous, has a higher rotor speed and a linear acceleration cut-out
- b. the mercury level switches are more sensitive than the pendulous vanes fitted to air driven types
- c. the roll cut-out speed is activated
- d. it is less aperiodic than the air driven types

## Answers

1	2	3	4	5	6	7	8	9
d	c	d	b	d	a	a	b	a

Chapter

**14**

The Turn and Slip Indicator

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## The Rate of Turn Indicator

This instrument incorporates two measuring devices, both indicating on the same instrument face. One of these, the rate of turn indicator, (commonly shortened to 'turn' indicator), uses a rate gyro to measure rate of turn about a vertical axis. The other, the slip indicator, is a very simple pendulous device which is used mainly to show whether or not a turn is balanced, (whether the angle of bank is correct for the TAS and rate of turn), and if not, to indicate the extent of slip or skid.

### The Rate Gyro

The turn indicator is based on a horizontal-axis rate gyro, which has only one gimbal and therefore only one degree of freedom. If the aircraft banks (without turning) the gyro axis has the freedom to remain horizontal. However, if the aircraft yaws, the frame, fixed to the airframe, applies a force (labelled primary torque in *Figure 14.1*), in a direction where the gyro is not gimbaled, and therefore has no freedom. This results in a precession which will cause the gyro to depart from the horizontal. A spring system prevents the gyro from turning all the way to the vertical, and the amount of spring stretch is a measure of the rate of turn.

### Operation

*Figure 14.1* and *Figure 14.2* illustrate the principal of operation. If the aircraft turns, the rotor is subjected to a primary torque acting about the ZZ axis. This produces a primary precession about the YY axis, the direction of this precession being as if the applied force were moved 90° in the direction of rotor spin. As the rotor tilts about the YY axis it causes a spring between gimbal and frame to be extended. The resultant spring tension subjects the rotor to a secondary torque acting about the YY axis. This secondary torque, with the precession, will continue until the gimbal has tilted just the right amount to give the spring tension required to generate a rate of secondary precession equal to the rate of turn of the aircraft. This gives equilibrium. It should be emphasized that the chain of events is virtually instantaneous - as the aircraft goes into a turn, the gimbal takes up the appropriate angle of tilt.

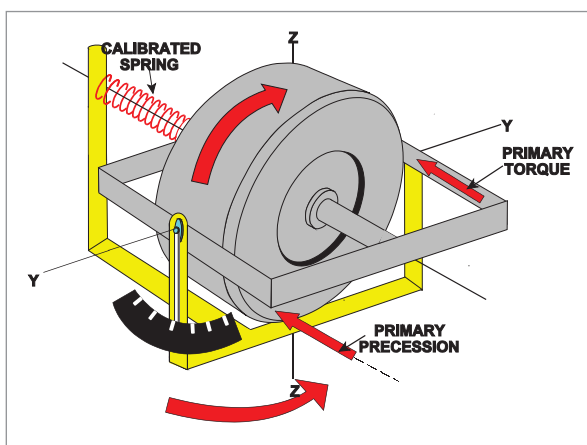


Figure 14.1

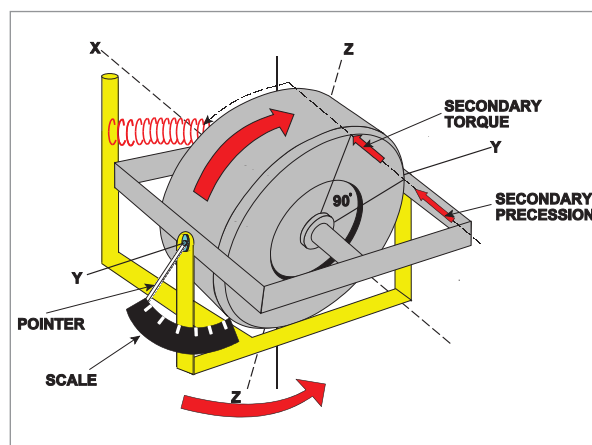


Figure 14.2

If the rate of turn changes, the tilt of the gimbal will also change, to re-establish the balance of torques on the gyro. The angle of tilt is thus a measure of the rate of turn. A pointer fixed to or linked with the gimbal indicates the tilt on a scale on the face of the instrument. The scale is calibrated to indicate rates of turn either side of the centre zero so that the first graduation corresponds to a Rate 1 turn with the aircraft turning 3° per second.

A second mark for rate 2 corresponds to 6° per second. There may be further graduations for higher rates of turn.

Calibration of correct rate of turn (the spring tension) is optimized for a design TAS.

However, only a small amount of error is introduced, even for quite large departures from design TAS.

In practice the errors produced by TAS deviations are not serious. One manufacturer quotes a maximum error of 5% over a speed range of 85 to 350 knots, the calibration value being 260 knots.

## Constructional Details

Suction and electrically-driven types are available. With the former, an engine-driven pump or venturi tube is used to apply suction to the case. Replacement air enters via a filter and is directed by a jet at the 'buckets' cut in the periphery of the rotor. The rotor rpm are low compared with those of the DGI and artificial horizon. This is because the gyroscopic property of precession is used to measure rate to turn, so that a high gyroscopic rigidity is undesirable. A damping system fitted to the gimbal reduces oscillation. This may be the piston-in-cylinder type or an electromagnetic device. Stops limit the movement of the gimbal to tilt corresponding to a turn of about 20° per second.

### Note

*As there is only one gimbal, the gyro will not 'topple' when it comes against the stops.*

## Effect of Varying Rotor Speed

If the suction is inadequate (at high altitude, or with a choked filter, or a leaking suction tube) with an air driven instrument, gyro rigidity will be lowered as the gyro is "**underspeeding**". Consequently the secondary precession needed to equal the aircraft turn can be generated by a smaller secondary torque. This reduced torque will be produced by a smaller angle of gimbal tilt, and this means that the instrument will **under-read** the turn rate.

Alternatively, if the gyro were to "**overspeed**", by the same token it will **over-read** the rate of turn that is being achieved by the angle of bank applied.

## Errors in the Looping Plane

In a gently banked turn, the aircraft is turning mainly in the yawing plane, but in a steep turn there is more movement in the looping plane. Normally movement in the looping plane means that the aircraft is rotating about the rotor axis, with no effect on the gyro. However, if the gimbal is tilted before movement in the looping plane commences, as happens with a yaw, the movement in the looping plane will cause additional precession of the rotor.

The usual positive movement in the looping plane in a steep turn will increase the gimbal tilt causing the indicator to **over-read**, sometimes coming against the stops.



## The Slip Indicator

It is desirable that turns should be properly balanced, with no side slip or skid. This implies that the angle of bank should be correct for the TAS and rate of turn. The correct bank angle could be accurately calculated, or rules of thumb may be used. For instance, with rate 1 turns, we can take one tenth of the TAS and add seven to give approximately the required bank angle. For example; rate 1, TAS 150 knots; bank angle should be  $15 + 7 = 22^\circ$ . This rule gives reasonable accuracy for rate 1 turns with TAS between 100 and 250 knots. During instrument flight however, the fewer the calculations that have to be made, the better. The slip indicator gives a direct indication of the state of balance of the turn.

## Construction

Early types of slip indicator employed a simple metal pendulum suspended in the instrument case, its oscillations being controlled by a piston-in-cylinder damping device. The modern version is usually a 'ball-in-tube inclinometer'. This comprises a solid ball in a curved tube containing liquid with damps out the unwanted oscillations. It is sketched in [Figure 14.4](#). and [Figure 14.6](#). The heavy ball behaves like a pendulum, with the centre of curvature of the tube acting as the effective point of suspension.

## Operating Principles

Consider first the aircraft in level flight with lift  $L$  balancing weight  $W$  viewed in [Figure 14.3](#). The weight  $W$  of the ball in the tube acts downwards and is exactly balanced by the equal and opposite reaction of the base of the tube on the ball, acting upwards towards the centre of curvature of the tube. If the wings are level, the ball will lie just between the two vertical lines etched on the tube, as indicated in [Figure 14.4](#).

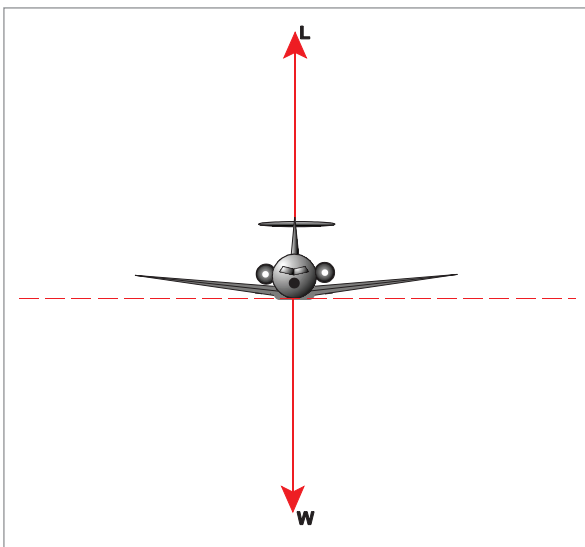


Figure 14.3 Aircraft in Level Flight

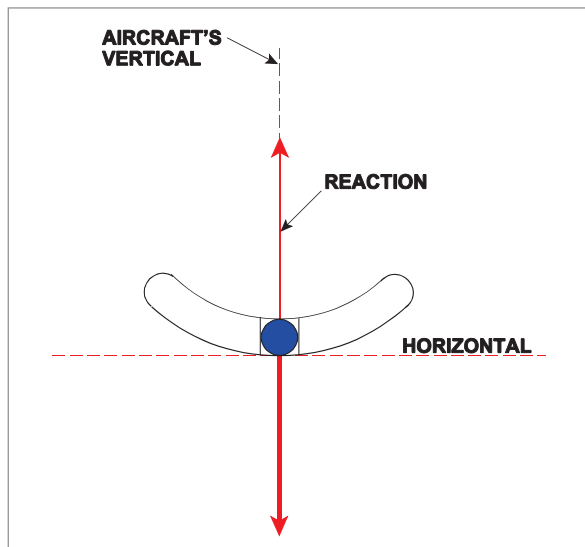
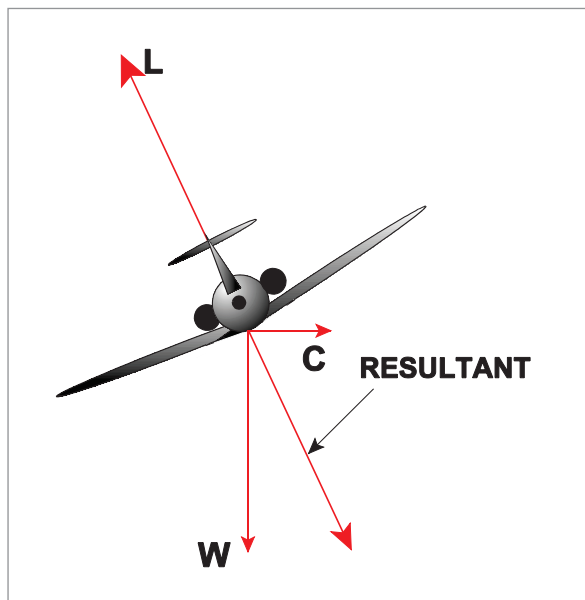


Figure 14.4 Ball-in-tube (Level Flight)

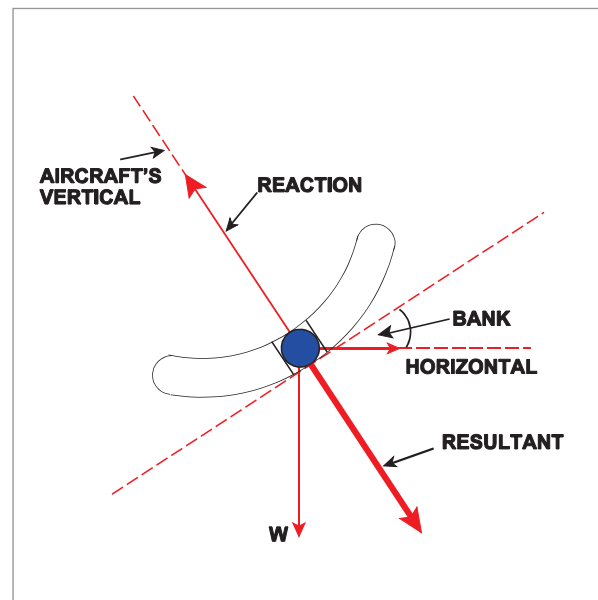
Now let us consider a balanced turn to the left. *Figure 14.5* shows the aircraft with lift  $L$  equal and opposite to the resultant of aircraft weight  $W$  and centrifugal force  $C$ , the latter being proportional to TAS and rate of turn.

The ball is also subject to a centrifugal force depending on TAS and rate of turn, so it rolls outwards, taking up a new equilibrium position such that the reaction of the base of the tube on the ball is again exactly balanced, this time by the resultant of ball weight  $W$  and centrifugal force  $C$  (*Figure 14.6*)

Because both aircraft and the ball are experiencing the same TAS and rate of turn (and so the same acceleration towards the centre of the turn) it can be proved that the resultant weight and centrifugal force for the aircraft will be parallel to the resultant of weight and centrifugal force for the ball. Now if the ball is laterally central in the tube, (between the two etched lines), the resultant and reaction forces of the ball must lie in the aircraft's vertical (see *Figure 14.6*). These forces are parallel, as stated above, to the resultant of aircraft weight and centrifugal force with must therefore also lie in the aircraft's vertical and will thus be in the same line as the lift  $L$  (*Figure 14.5*) - which means that the turn is balanced.



*Figure 14.5 Balanced Turn To Port*



*Figure 14.6 Balanced Turn To Port*

Unbalanced turns are most simply considered as follows. Let us assume that the TAS and rate of turn are the same as in *Figure 14.5* and *Figure 14.6*, so that the ball will not have moved laterally. Now if too much bank is applied, (for the TAS and rate of turn), the tube will have been rotated too far in the rolling plane so that the ball appears as in *Figure 14.7*, no longer central, but correctly indicating the aircraft to be '**slipping in**' to the turn, the radius of which will be less than it should be. If on the other hand insufficient bank has been applied, the instrument will be indicating that the aircraft is '**skidding out**' of the turn (see *Figure 14.8*), the radius of turn this time being greater than it should be.

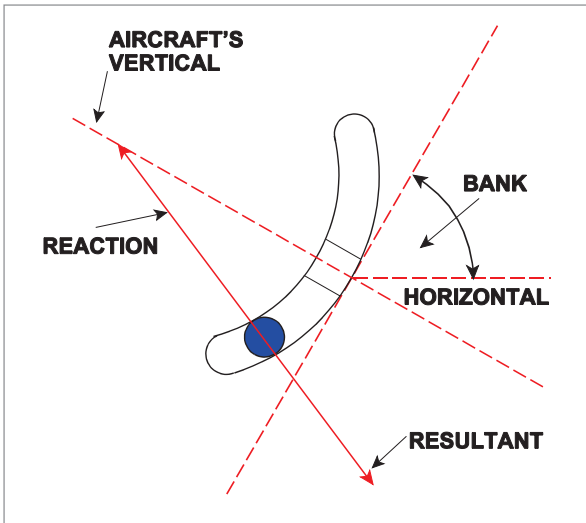


Figure 14.7 Unbalanced Turn Port (Slipping)

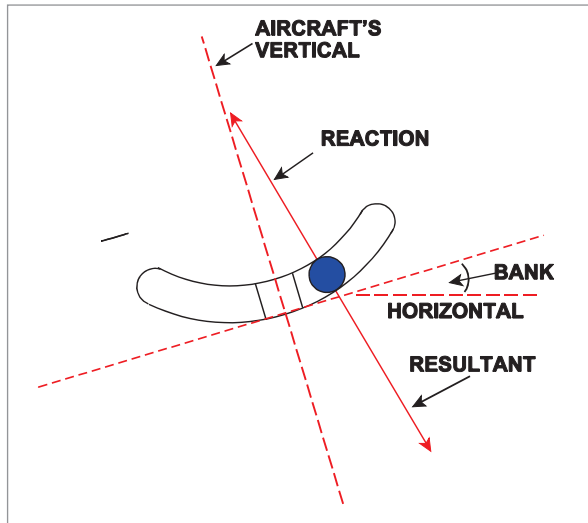


Figure 14.8 Unbalanced Turn Port (Skidding)

### Turn and Slip Displays

Several examples of turn and slip indications (needle and ball type) are drawn in Figure 14.9.

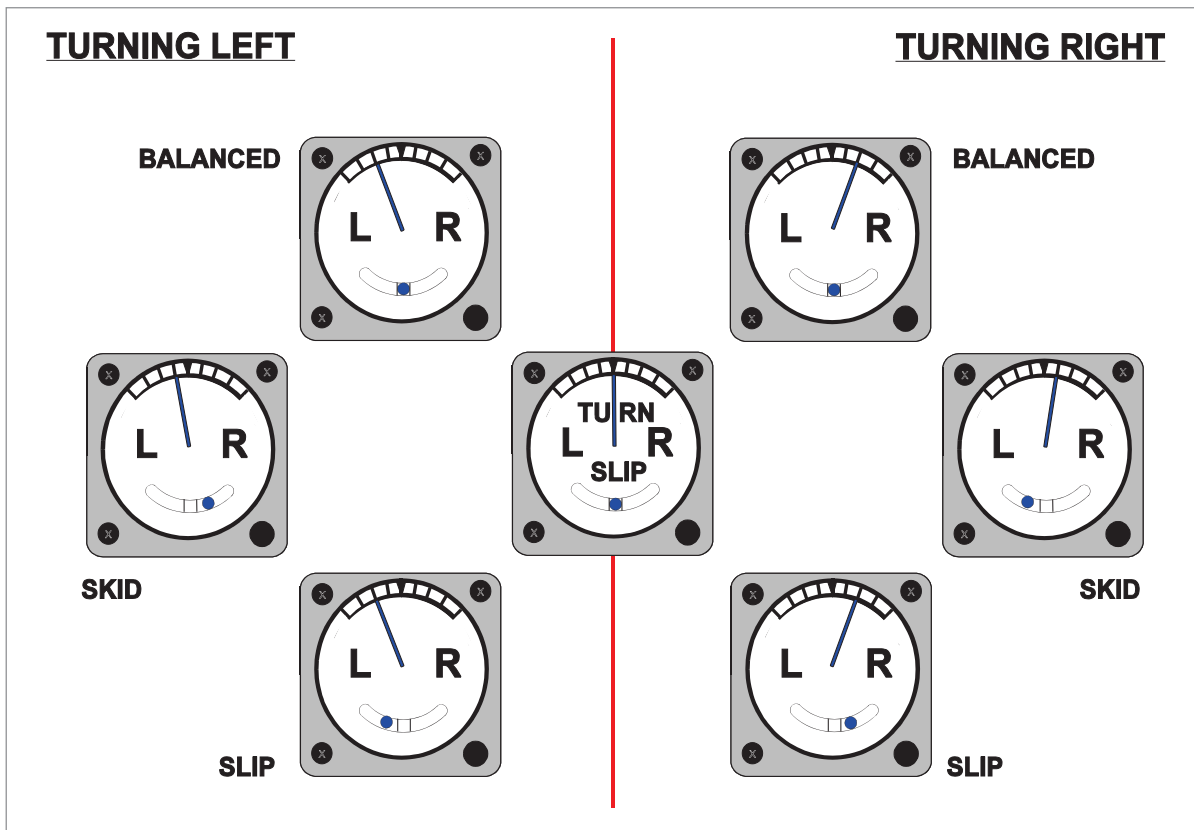


Figure 14.9 Needle and Ball Displays

**Example Question - Rate one turn**

Flying at 360 kt what is the turn diameter at rate 1?

1. A rate 1 turn takes 2 minutes to complete, therefore.....  
Flying at 360 kt each minute you will fly 6 NM and so..... in 2 minutes you will cover 12 NM.
2. The circumference of a circle =  $\pi \times d$  (diameter of the circle)
3. The circumference is 12 NM, therefore  
 $12 = 22/7 \times d$   
or  
 $12 \times 7$  divided by 22 = d  
 $3.8 = d$
4. To check we can ..... divide 360 by 100 = 3.6 — 4 "NM"  
similarly for 400 = "4 NM"  
and for 500 = "5 NM"

Questions

1. The rate of turn indicator uses (i) ..... which spins (ii).....

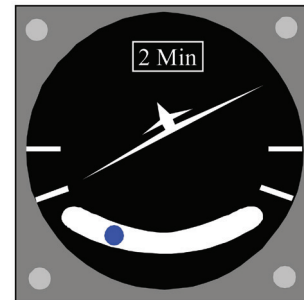
- |    |                   |                                      |
|----|-------------------|--------------------------------------|
|    | (i)               | (ii)                                 |
| a. | a space gyroscope | up and away from the pilot           |
| b. | a tied gyro       | anticlockwise when viewed from above |
| c. | a rate gyro       | up and away from the pilot           |
| d. | an earth gyro     | clockwise                            |

2. The gyro in a rate of turn indicator has (i) ..... operating speed than the gyros used in other instruments because (ii).....

- |    |          |   |
|----|----------|---|
|    | (i)      | (ii)  |
| a. | a lower  | a higher rigidity is not required                     |
| b. | the same | it uses the property of rigidity                      |
| c. | a higher | a low precession rate gives a greater operating range |
| d. | variable | more than one rate of turn is desired                 |

3. The TBI shown alongside indicates:

- a. a rate of turn to the left, slipping in
- b. an aircraft taxiing and turning starboard
- c. that the aircraft will complete a turn in one minute
- d. the aircraft is yawing to the right



4. When the pointer of a rate of turn indicator shows a steady rate of turn:

- a. the calibrated spring is exerting a force about the lateral axis equal to the rate of turn
- b. the force produced by the spring is producing a precession equal to but opposite to the rate of turn is correctly banked
- c. the spring is providing a force which produces a precession equal to the rate of turn (in the opposite direction)
- d. the spring is providing a force which produces a precession equal to the rate of turn (in the correct direction)

5. If the filter of the air driven rate of turn indicator becomes partially blocked:

- a. the aircraft will turn faster than indicated
- b. the instrument will over-read
- c. the rate of turn indicated will be unaffected
- d. the radius of the turn will decrease

6. The radius of a turn at rate 1, and TAS 360 kt is:

- a. 10 NM
- b. 5 NM
- c. 7.5 NM
- d. 2 NM

## Answers

1	2	3	4	5	6
c	a	a	d	a	d

Chapter

# 15

## The Turn Co-ordinator

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## Turn Co-ordinator

The **Turn Co-ordinator** is an interesting development of the **Turn** and **Bank** indicators.

The primary difference is in the setting of the precession axis of the rate gyroscope and the method of display.

The gyroscope is spring restrained and is mounted so that the axis is at about 30 degrees with respect to the aircraft longitudinal axis, thus making the gyroscope sensitive to banking of the aircraft as well as to turning.

A turn is normally initiated by banking the aircraft, the gyroscope will precess, and this in turn will move the aircraft symbol in indicate the direction of bank and enable the pilot to anticipate the resulting turn.

The pilot then controls the turn at the required rate by alignment of the aircraft with the graduations on the instrument dial. The rate of turn will depend on the instrument in use either as a rate one turn, 3 degrees per second, or any other rate dependent on instrument design. The ball still has to remain central for a balanced rate of turn.

The annotation "No Pitch Information" on the indicator scale is given to avoid any confusion in pitch control which might result with the similarity with the presentation of the gyro horizon.

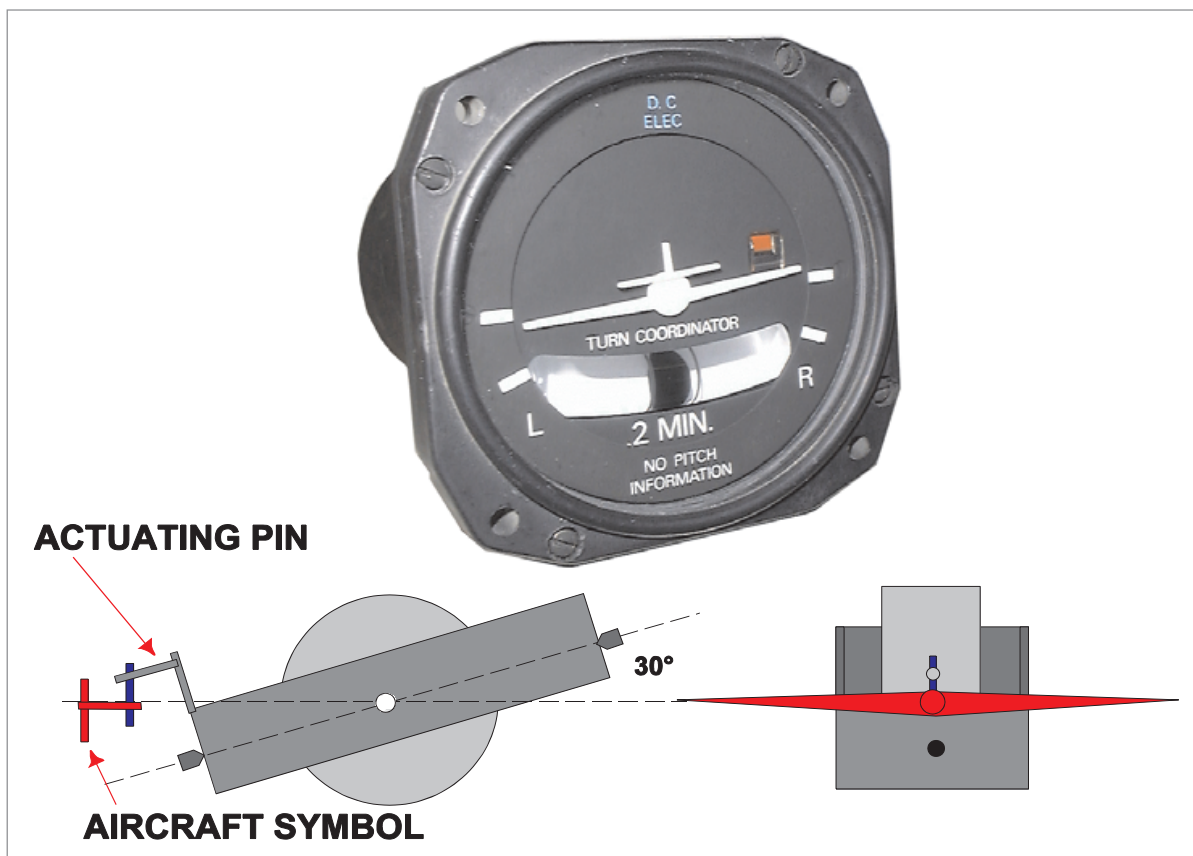


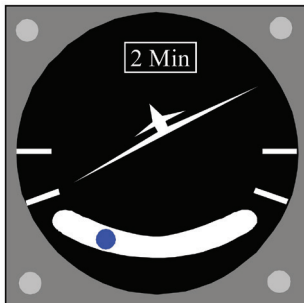
Figure 15.1

### Questions

1. The gimbal ring of a turn co-ordinator is inclined at about  $30^\circ$  with respect to the aircraft's longitudinal axis in order to:
  - a. make the rate of turn more accurate
  - b. make the gyro sensitive to banking of the aircraft as well as to turning
  - c. make the gyro more effective during inverted flight
  - d. have a higher rotor speed which will prolong the life of the instrument
  
2. If an aircraft turns as indicated in Figure 1, below:
  - a. the aircraft will turn through  $180^\circ$  in two minutes
  - b. it will take one minute to turn through  $90^\circ$
  - c. the aircraft is turning left at less than  $3^\circ$ / second
  - d. the aircraft is turning left at  $3^\circ$ / second
  
3. A turn co-ordinator has (i) ..... pivoted (ii) ..... in the case
 

	(i)	(ii)
a.	two gimbal rings	orthogonally
b.	a single gimbal ring	longitudinally
c.	one gimbal ring	laterally
d.	two gimbal rings	mutually perpendicular

Figure 1





## Answers

1	2	3
b	c	b

Chapter  
**16**

**Aircraft Magnetism**

---

Deviation . . . . .	.201
Compass Swing . . . . .	.201
Hard Iron Magnetism . . . . .	.202
Soft Iron Magnetism . . . . .	.202
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## Deviation

The compass needle would accurately define the magnetic meridian were it not for the aircraft's own internal magnetism deflecting it. Deviation is the angular difference measured between the direction taken up by a compass needle and the magnetic meridian. Deviation is named easterly or westerly depending on whether the north-seeking end of the compass needle lies to the east or west of the magnetic meridian.

	Compass Heading	Deviation	Magnetic Heading
Deviation West Compass Best	095	-5	090
Deviation East Compass Least	090	+5	095

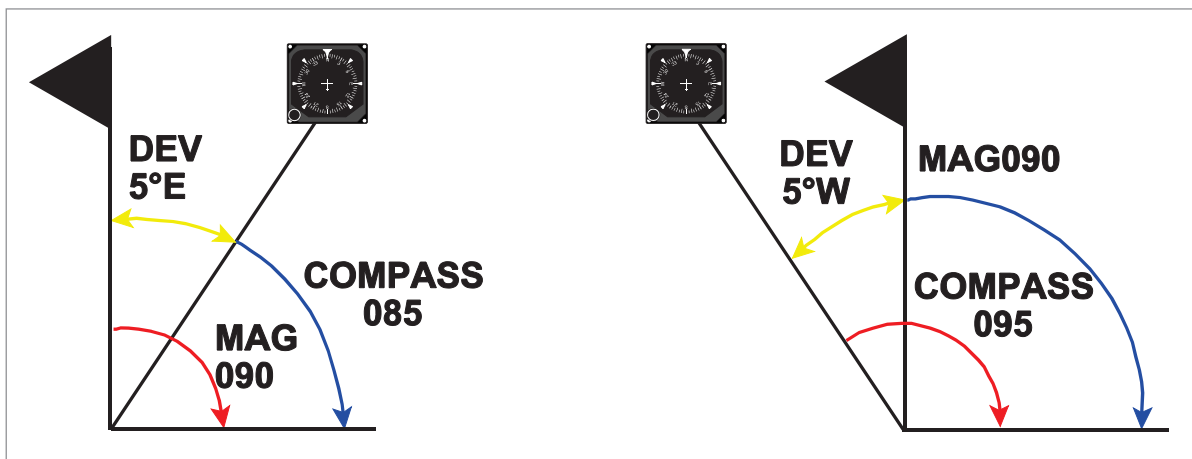


Figure 16.1

## Compass Swing

The basic method of determining deviation is to compare the aircraft's heading compass reading with magnetic heading as defined by a high quality 'land or datum' compass. This comparison of aircraft compass and magnetic datum readings is carried out in an area selected specifically for this purpose.

Therefore the aims of a compass swing are as follows:

- To observe / determine the deviations / differences between magnetic north (observed on a landing compass) and compass north (observed in the aircraft) on a series of headings
- To correct / remove as much deviation as possible
- To record the residual deviation which is left after the compass has been adjusted

The magnetic deviation observed during a compass swing can be said to be derived from hard iron and soft iron magnetism and this total field can in turn, for our purposes, be later resolved into two further combined components (coefficients B and C).

## Hard Iron Magnetism

The total force at the compass position produced by permanent hard iron magnetism can be resolved into three components. These components will be fixed for a given aircraft and will not change with change of heading.

## Soft Iron Magnetism

Soft iron magnetism is induced in parts of the aircraft structure by surrounding fields - the most important of these being the earth. The earth's field has a vertical as well as horizontal component. However, again for our purposes we will within the constraints of the syllabus only consider vertical soft iron (VSI) magnetism (Z is the vertical component of the earth's field and H is the horizontal component). The component Z has an increasing effect with latitude as the compass magnets try to follow the earth's flux lines therefore VSI magnetism must also vary with latitude. However, Z is zero at the equator, where the horizontal component H is greatest, so no VSI magnetism is induced there.

When we examine the effective positioning of the imaginary magnets found when completing a compass swing we must remember that we use a real system (the compass) to give us aircraft heading and that this read-out is affected by these magnetic forces which we have gone to some trouble to discover.

We can see from [Figure 16.3](#) that the positioning can vary, (even to the extent of having two imaginary magnets affecting our compass) but the effect will be easily resolved by the compass swing which can cater for any positioning as long as we follow the basic rules.

For example we may examine the case where the effect of the blue pole is said to be in the nose or forward of the aircraft compass.

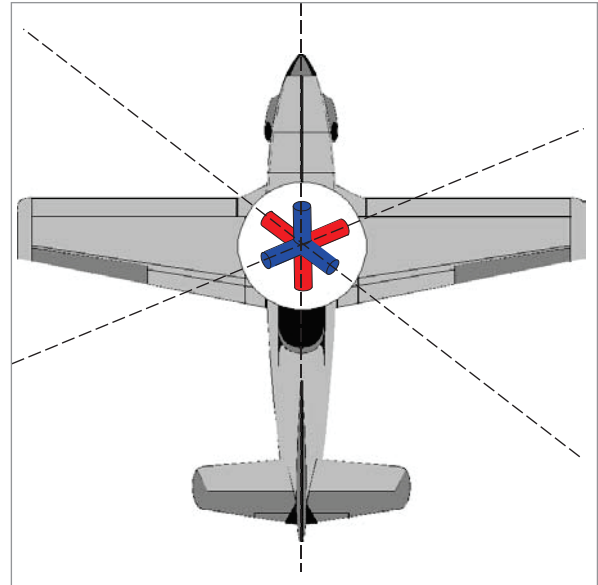


Figure 16.2

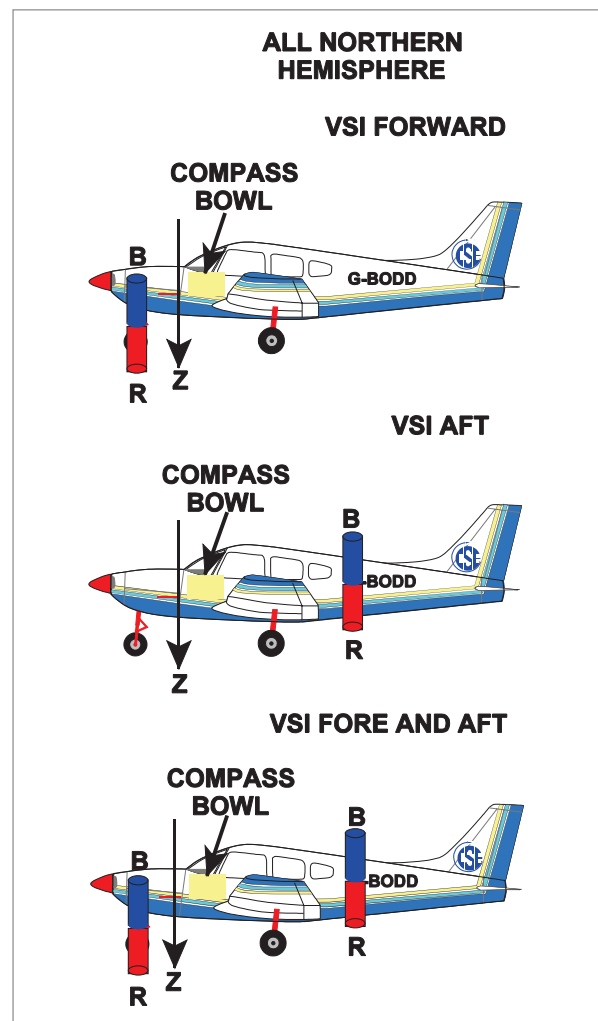
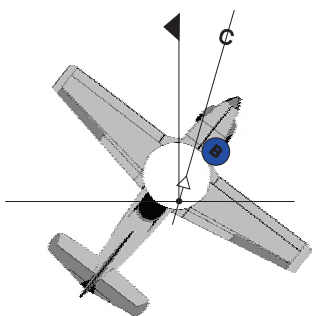
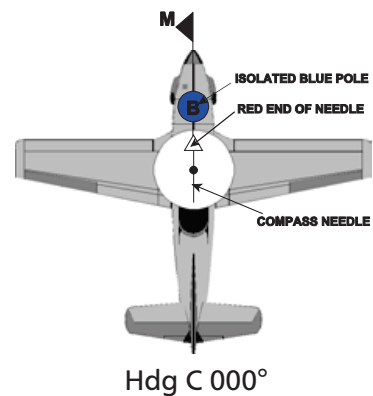


Figure 16.3

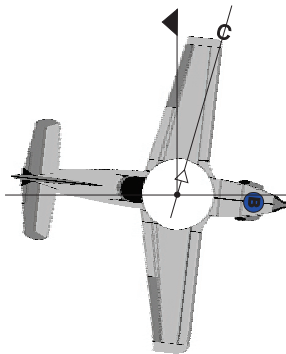


Heading north the isolated blue pole is in the same horizontal direction as the earth's blue pole and so the needle is not deviated. The directive force or alignment of the earth's field is being augmented by the blue pole, effectively they are pulling together.

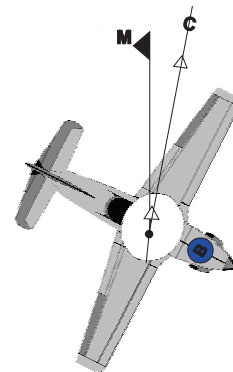
As the aircraft now turns right on to 045°, deviation begins to take place and by 090° this has become maximum. It then starts to become less as we approach 180°.



Hdg C 045°



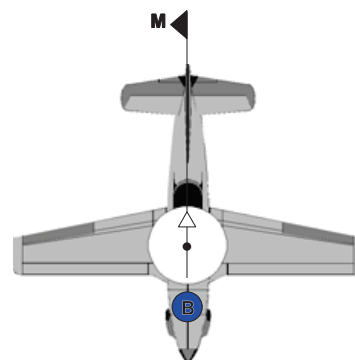
Hdg C 090°



Hdg C 135°

Remember that the blue pole represents a magnetic force which on this heading acts along the same line but in opposition to the stronger earth's field.

On the remaining headings 180° to 360° the effects of the blue pole in the nose are as expected i.e. the red end of the compass needle is being attracted to the west of magnetic north giving the maximum westerly deviation on 270°.



Hdg C 180°

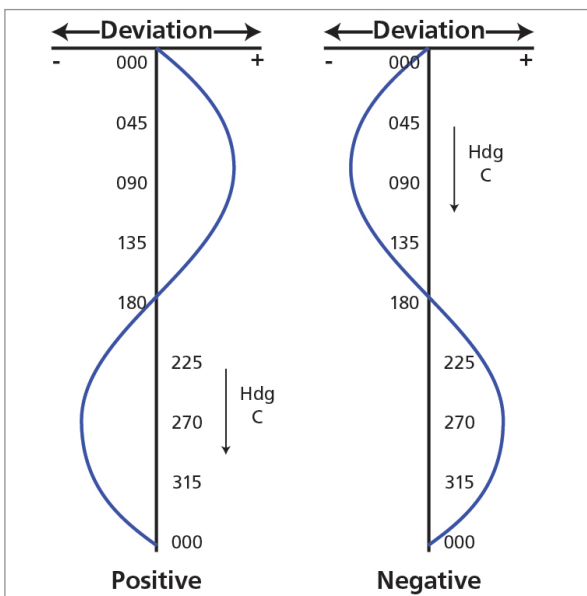


Figure 16.4

If the deviations caused by the blue pole in the nose are plotted against compass heading, a positive sine curve is obtained. Had the blue pole been aft of the compass a negative sine curve would have been obtained. This would mean that on a heading of 090° the deviation would reach a maximum westerly value instead of a maximum easterly value. The changes in directive force would also be revised, the maximum occurring on 180° and the minimum on 360°.

Equally we should be able to see that if a further magnetic source is resolved to the right wing we would achieve a positive cosine curve along the same lines.

## Correction of Coefficients

The principle for correcting coefficients is the same for any system and can be summed up as follows:

**Coefficient A** - a mechanical problem of a displaced lubber line corrected by loosening the bolts holding the compass body or in the case of the RIMC the detector unit and carefully turning it until the correct heading is in place.

**Coefficient B** - correction required because of magnetic deviating forces acting upon the DRMC or the detector unit and giving errors known as deviation. Firstly calculate the error to be removed or more correctly the heading you wish to make the compass read and this will be done on an easterly or westerly heading.

**Coefficient C** - correction required because of magnetic deviating forces acting upon the DRMC or the detector unit and giving errors known as deviation. Firstly calculate the error to be removed or more correctly the heading you wish to make the compass read and this will be done on a northerly or southerly heading.

We can see that the correction for B and C are very similar but that we must remember to apply the sign of the correction properly to ensure an accurate correction to our compass system. When the compass swing is completed we of course have to check our work and this 'check swing' is carried out using eight or perhaps twelve points of the compass to allow us to derive a compass card that will be placed in the aircraft. This compass card indicates to us the residual deviations that we have been unable to resolve within the essentially horizontal procedure. Alternatively, the Residual Deviations affecting the compass after the completion of a compass swing may be shown by the use of a graphical table or a curve constructed from the information obtained. Either set of calculations will allow for the placing of a Compass Deviation Card near to the compass in the aircraft.

## Accuracy Limits

In accordance with CS OPS-1 (European Regulations) the aircraft's compasses must, after correction, be within the following limits:

Direct Reading Magnetic Compass	+/- 10°
Remote Indicating Compass	+/- 1°

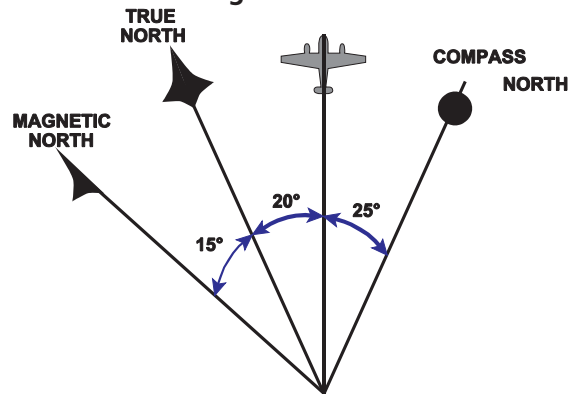
## Occasions for Swinging the Compass

- When compass components are installed or replaced.
- Whenever the accuracy of the compass is in doubt.
- After a maintenance inspection if required by the schedule.
- After a significant aircraft modification, repair or replacement involving magnetic material.
- When carrying unusual ferromagnetic payloads!
- When the compass has been subjected to significant shock.
- If the aircraft has been struck by lightning.
- After significant modification to aircraft radio/electrical systems.
- After the aircraft has been given a new theatre of operations if the move involves a large change of magnetic latitude.
- If the aircraft has been in long term storage standing on one heading.

## Questions

1. **European regulations (EU OPS-1) state that the maximum permissible deviations after compensation are:**
  - a. one degree for a remote indicating compass, and ten degrees for a direct reading magnetic compass
  - b. three degrees for a direct reading magnetic compass, and one degree for a remote indicating compass
  - c. ten degrees for a remote indicating compass, and one degree for a direct reading magnetic compass
  - d. one degree for a direct reading magnetic compass, and eleven degrees for a slaved compass
  
2. **Compass swings should be carried out:**
  - a. on the apron
  - b. only on the compass swinging base or site
  - c. at the holding point
  - d. on the active runway
  
3. **Aircraft magnetism caused by vertical soft iron:**
  - a. varies with magnetic heading but not with magnetic latitude
  - b. varies with magnetic latitude but not with heading
  - c. it is not affected magnetic latitude or heading
  - d. varies as the cosine of the compass heading
  
4. **Aircraft magnetism caused by hard iron:**
  - a. is not usually influenced by the earth's magnetic field
  - b. varies directly with magnetic latitude
  - c. varies indirectly with magnetic latitude
  - d. is maximum on east and west
  
5. **The aim of a compass swing is:**
  1. to find deviation on the cardinal headings and to calculate coefficients A, B and C
  2. to eliminate or reduce the coefficients found
  3. to record any residual deviation and to prepare a compass correction card
  - a. only answer 1 is correct
  - b. answers 1 and 3 are correct
  - c. answers 1, 2 and 3 are all correct
  - d. none of the above answers are correct
  
6. **Deviation due to coefficient A is mainly caused by:**
  - a. hard iron force acting along the longitudinal axis
  - b. hard and soft iron forces acting along the lateral axis
  - c. vertical soft iron forces
  - d. a misaligned lubber line

7. In the diagram below, the compass heading of the aircraft is ....., the magnetic heading ..... and the true heading .....



- a. 025° 015° 020°
- b. 335° 035° 020°
- c. 335° 340° 035°
- d. 025° 015° 340°

## Answers

1	2	3	4	5	6	7
a	b	b	a	c	d	b

# Chapter 17

## Remote Indicating Magnetic Compass

---

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## Limitations of the Direct Reading Compass

The Direct Reading Compass has three major limitations:

- **Turning and acceleration errors.** The compass cannot be read accurately during a turn.
- The **magnetic sensing element** (the magnets on the vertical card) is contained within the instrument and is therefore situated close to the pilot so that the card can be seen. The cockpit area is **close to sources of deviation**, such as electric lights, electric motors and ferrous metal.
- The instrument is self-contained. It is **not possible to take a magnetic heading and input it into other equipment.**

## Limitations of the Directional Gyro Indicator

The **Directional Gyro Indicator** attempts to solve some of these problems by use of an air driven or an electromechanical gyro. Turning and acceleration errors are eliminated and an output can be taken to other equipment. However, there is **no magnetic input**, so if the **gyro drifts with time there is no correction** except by the pilot manually synchronizing to the direct reading compass at regular intervals.

## Requirement for the Gyro-magnetic Compass

What is required is a system which **combines the best of both**. The **short-term rigidity of the gyro** overcomes turning and acceleration errors. This needs to be combined with the **longer-term monitoring of the earth's magnetic field** so that if the gyro starts to drift, a servo system slaves it to alignment with a magnetic input. Such a system is a **gyro-magnetic compass**.

The gyro-magnetic compass is known by several names. It can be referred to as the:

- **Gyro-magnetic Compass.**
- **Remote Indicating Compass.**
- **Slaved Gyro Compass.**

They all mean the same thing.

## Basic System Description

At its simplest, the system comprises the following elements:

- **Magnetic Detector Unit.** This is also often known as a **flux valve** or a **flux detector**.
- **Heading Indicator.** This is what most people refer to as 'the compass'.
- **Precession Amplifier.** This may also be known as a **slaving amplifier**.
- **Precession Motor.** This may also be known as a **slaving or synchronizing motor**.
- Horizontal Gyro.

In simple systems, the horizontal gyro is directly connected to the compass card of the heading indicator via a **bevel gear** and a **drive shaft**. This is assumed in the description which follows.

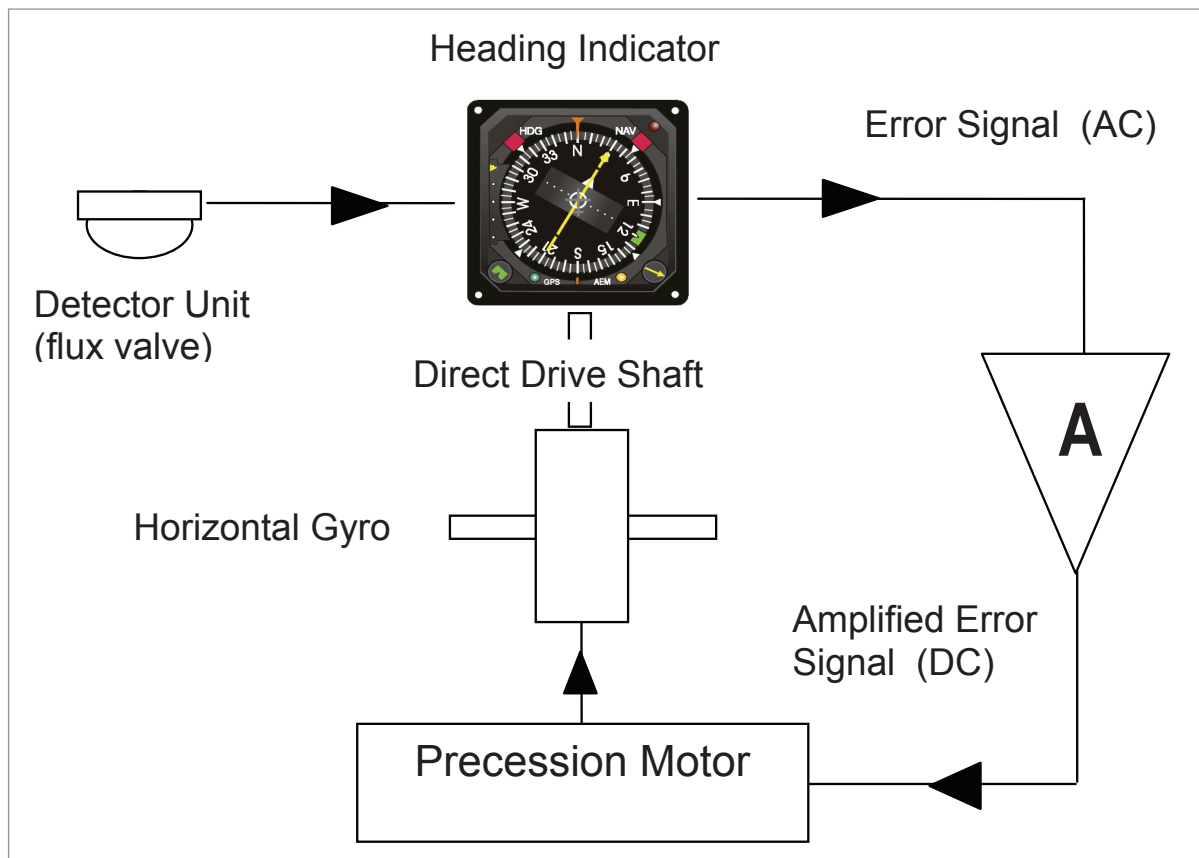


Figure 17.1 Simple signal routing

### Operation with Steady Heading – Correction for Gyro Drift

This description assumes a steady heading, which gives a steady input from the detector unit (flux valve), and assumes that the compass has already been synchronized. Any difference between the flux valve field and the gyro alignment would only arise if the gyro were to drift.

- The detector unit (flux valve) senses the earth's magnetic field and reproduces it within the compass unit, where it is compared with the position of the gyro drive shaft (which also positions the compass card indicator – the indication of heading to the pilot).
- If the two are aligned, no further action takes place. The compass card is reading the right heading. If, however, the gyro starts to drift, the drive shaft will not be in alignment with the flux valve field, and an AC error signal is generated and passed to the precession amplifier (marked with a big A in [Figure 17.1](#)), where it is amplified, phase detected, and rectified to DC.
- The DC signal drives the precession motor, which turns the gyro. This gyro output is fed via the direct drive shaft to the heading indicator for comparison with the flux valve signal.
- If the two are aligned, the compass is synchronized and no further action takes place. If not, the error correction continues until the compass is synchronized.

It would be possible to cut out the gyro, in theory. The flux valve field could be passed for comparison with the compass card and the error signal passed to a motor which would directly drive the compass card shaft. This would still give an electromagnetic compass, with a detector unit remote from the major source of deviations, and its output could be used to drive other systems. However, such a system would be overly responsive to any fluctuations in the field detected by the flux valve and would suffer significantly from turning and acceleration errors. The **use of the gyro gives stability and rigidity** because the precession motor applies corrections to the drive shaft at the rate of **approximately only 3° per minute**.

### Operation in a Turn – Gyro Drift Small Over Period of Turn

Now consider what happens in a turn. We will assume, initially, that the gyro does not drift during the turn, which is not unreasonable, because even during a full 360° orbit, the turn will only take 2 minutes. The aircraft turns, but the gyro, having rigidity, does not. This gives relative rotation between the horizontal gyro and the instrument case and so operation of the bevel gear causes the direct drive shaft to rotate, changing the heading indication on the compass card. However, at the same time, the heading sensed by the flux valve, which is being passed to the driveshaft for comparison, is changing at the same rate. Therefore no error signal is generated and the compass should remain synchronized during the turn.

If there is some gyro drift during the turn, on completion of the turn there will be a small error signal. This will be taken out as described in the previous paragraph.

### Rapid Synchronization

When the gyro is started up on initial switch-on, the alignment it adopts is random and is unlikely to be in synchronization with the earth's magnetic field. Therefore an error signal is detected at the gyro drive shaft. The problem is that the precession motor's normal correction rate is only 3° per minute and if the gyro happened to be 90° out, it would take 30 minutes to synchronize, which is obviously unacceptable.

The solution is to have a rapid synchronization facility, which can either be a mechanical clutch operated by the pilot (as in the DGI) or, in later compasses, a high gain mode for the precession amplifier (similar in principle to rapid erection in the electric artificial horizon). This is operated by a 2 position switch, spring-loaded to the normal position, which has to be held against the spring for rapid alignment. Operation of this switch increases the precession motor's correction rate so that synchronization takes only a few seconds.

### Detector Unit (Flux Valve)

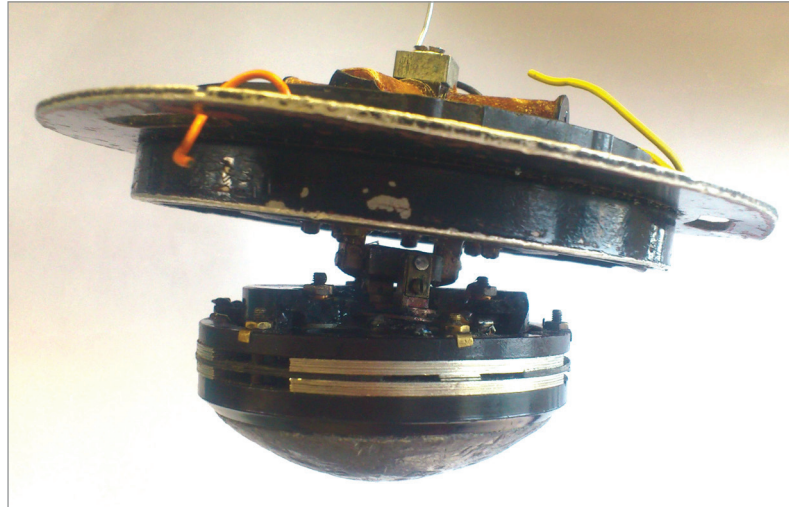


Figure 17.2 A magnetic detector unit

The **detector unit is positioned** in a part of the aircraft least affected by on-board electrical fields (usually the wing tip or tail fin, where any aircraft generated magnetic disturbances are at a minimum). Its function is to sense the direction of the earth's magnetic field. It contains a pendulous magnetic detecting element mounted on a **Hooke's Joint** which enables the detector to swing within limits of **25° about the pitch and roll axes**, but allows **no rotation in azimuth**. The unit itself is contained in a sealed case partially filled with oil to dampen any oscillations created during flight.

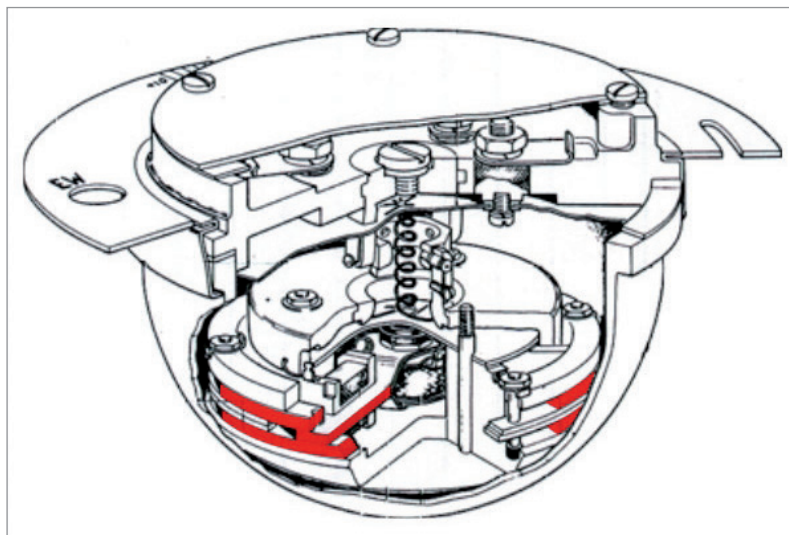
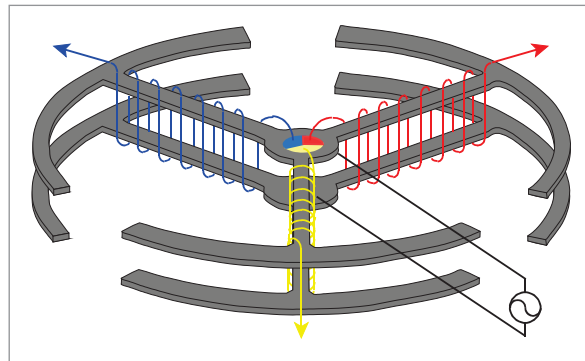


Figure 17.3 Flux valve components

The circular plate is screwed to the underside of the wing. The black hemisphere protrudes out into the airflow and is simply a protective cover for the flux valve inside. The cable carrying the signals passes along inside the structure of the wing.

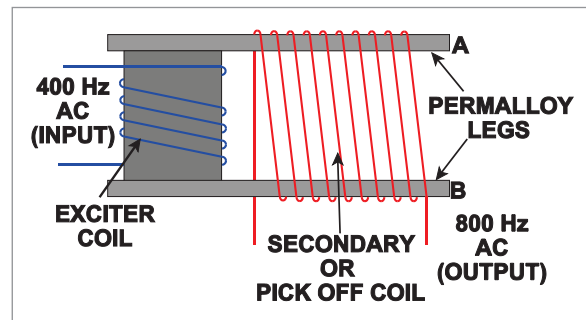
The primary component is the **flux valve**, a 3-spoked device, fixed in azimuth but with some freedom in the vertical to allow alignment with the plane of the earth's magnetic field. Parts of the flux valve are shown in [Figure 17.3](#).

The parts in red in *Figure 17.3* are shown in isolation in *Figure 17.4* below. All 3 legs are shown together, as in the actual unit:



*Figure 17.4 Three flux valve legs*

The curved 'rams' horns' at the end of each of the 3 legs are simply to improve magnetic flux gathering efficiency, but they do not affect the principle. The flux valve would detect even without them. To explain how the flux valve works, we will start by considering just a single leg (without the 'rams' horns').



*Figure 17.5 A single leg of a flux valve*

A simplified diagram of a flux valve leg is shown here. Alternating current is fed to the coil wound around the centre post which in turn produces fields of opposite sign in the top and bottom legs of the flux valve.

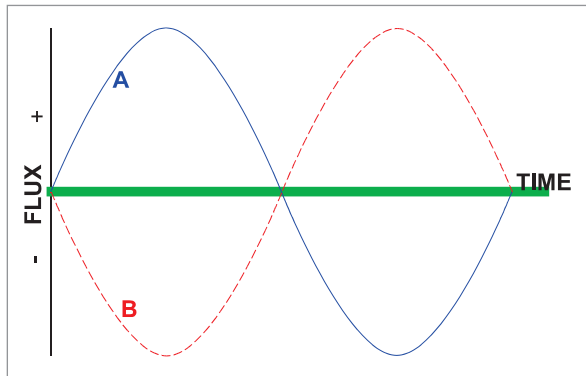


Figure 17.6 Flux fields at A and B and resultant

The magnetic flux (a measure of the density of the lines of force) in legs A and B is shown in *Figure 17.6* as red and blue lines. They are at the same frequency and amplitude, but are in anti-phase. If we add the components A and B together, the resultant flux (the green line) is therefore zero, and so no current is induced in the pick-off coil.

If the flux produced by the earth's magnetic field were present as a background, the positive and negative flux would start from a different baseline, which would not be zero. This is shown in *Figure 17.7*.

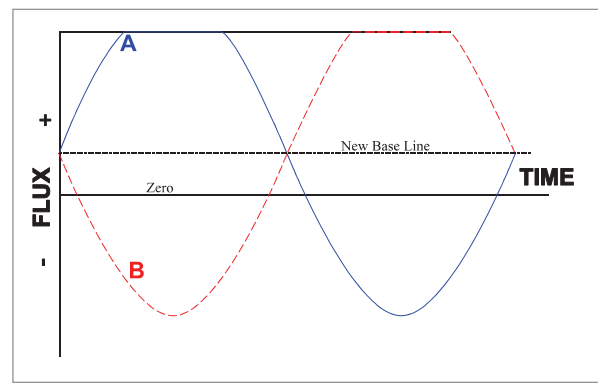


Figure 17.7 Effect of earth's background magnetism

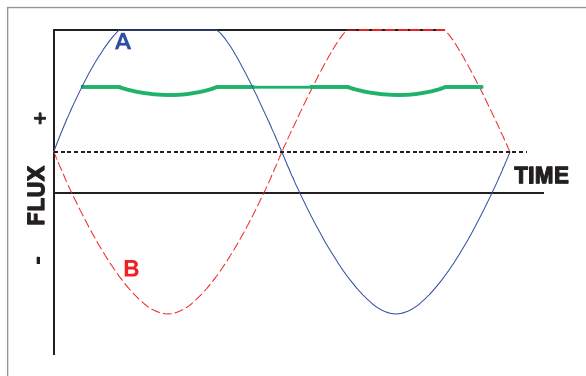


Figure 17.8 Flux density

However, the physical characteristics of the metal used in the flux valve legs are such that they magnetically saturate at a certain level. The metal will not magnetize further beyond a certain point. In this case, the saturation curve tops-out at a limiting saturation level, giving the response shown (the upper end of the blue and red lines).

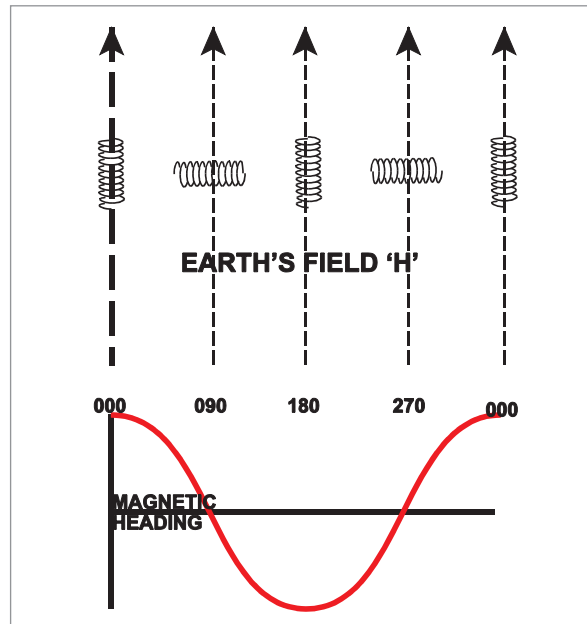
When we add the flux level together to see the resultant, the total flux follows the path of the green line.

The operation of the flux valve is in accordance with **Faraday's Law of Electromagnetic Induction**:

*'If the number of lines of force threading a circuit is changing, an induced electromotive force (EMF) will be set up in the circuit, the magnitude of the EMF being proportional to the **rate of change** in the number of lines of force threading the circuit'*

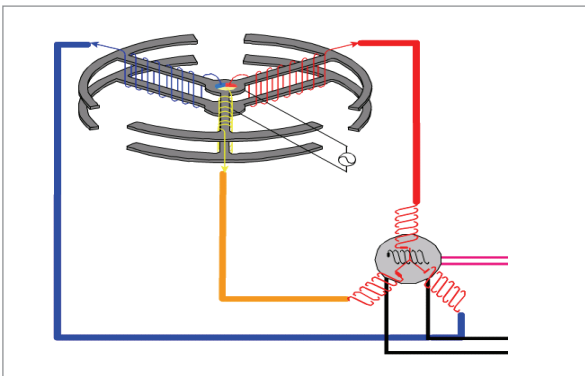
Therefore the **secondary** winding (the one coloured red in *Figure 17.4*) will pick up **change** in magnetic flux density (the dips in the green line) as an EMF. This will be detected as an AC signal.

*Figure 17.9* shows that if the flux valve leg is **in line** with the earth's field, then the EMF induced will be at a **maximum value**. The **secondary winding** (which is aligned with the leg) is shown in the diagram. If the flux valve leg is at **right angles** to the field, then the EMF induced will be **zero**.



*Figure 17.9* Effect of earth's field on different directions of flux valve leg

*Figure 17.9* shows that the EMF induced varies as the cosine of the magnetic direction of the flux valve leg. Unfortunately, this cannot be transformed directly into heading because, except for  $0^\circ$  and  $180^\circ$ , there are 2 possible values of heading for each value of voltage. Furthermore, any slight change of input voltage would give an altered value of output voltage, resulting in a different measured heading.



*Figure 17.10* Connection of flux valve to stator legs

Instead, the 3-leg system shown in *Figure 17.4* is used and the output from each leg is fed to one of the 3 legs of a stator. This re-creates the earth's field relative to the direction of the flux valve as shown in *Figure 17.10*, around the direct drive shaft from the gyro to the heading indicator compass card.

## Error Signal Comparison

Earlier we said that if there were any difference between the gyro shaft alignment and the magnetic field detected by the flux valve, an error signal would be generated which would be amplified to precess the gyro so that it takes up the alignment of the magnetic heading. This error signal detection is achieved by rotor-stator comparison.

A wound coil is mounted on the gyro drive shaft. This coil is known as a rotor. If the coil is in line with the AC field generated by the stators, a secondary AC voltage will be induced in the rotor (similar to the situation shown in *Figure 17.9*). If the rotor is at 90° to the AC field, no secondary voltage is induced. This is known as the **null position**. At any position other than the null, some secondary voltage is induced. This secondary induced voltage is passed to the **precession amplifier** where it is amplified, phase detected and rectified to DC.

**Amplified.** The reason for the amplification is that the un-amplified error signal is not powerful enough to drive the precession motor.

**Phase Detected.** It is important that the precession motor 'knows which way to turn'. Suppose that the gyro shaft is misaligned 2° clockwise from the null. The motor should rotate the shaft 2° anticlockwise, not all the way round 358° clockwise. Otherwise, the system would go into continuous rotation. The purpose of phase detection is to detect the sense of the error.

**Rectified.** The mechanism of the precession motor is an electromagnetic solenoid acting on a permanent magnet. This requires DC. The amplified AC is therefore rectified to DC which, depending on the phase detected, will either be in a positive or a negative direction, turning the shaft either clockwise or anticlockwise for the shortest route for error correction.

## Heading Indicator

The heading indicator dial (compass card) is directly driven by the shaft from the gyro. The compass card rotates as heading changes and the heading is read against the index line in the 12 o'clock position (the **lubber line**).

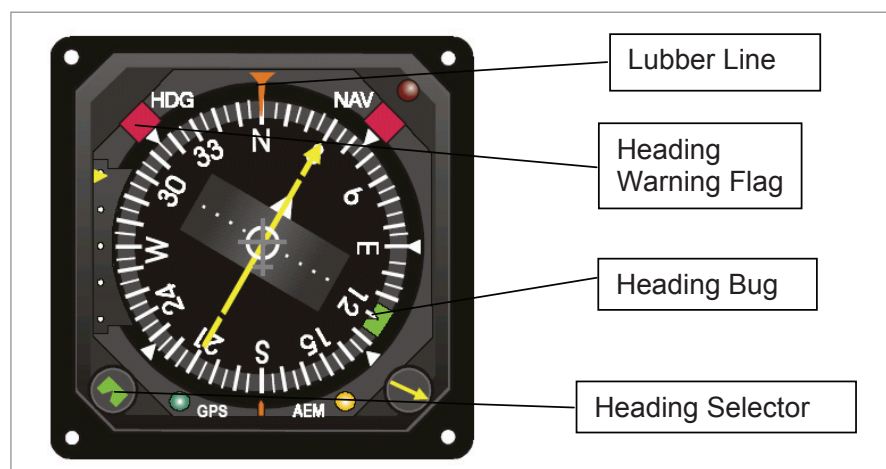


Figure 17.11 Typical Heading Indicator



A **desired heading** can be selected by the pilot by rotating the **heading selector control**. The **heading select marker** (usually called a 'bug') indicates the selected heading. If the magnetic input from the flux valve fails, a warning is given in the form of a **heading warning flag**.

## Operation as a DGI

If the magnetic input from the flux valve should fail or if it becomes unreliable due to proximity to one of the earth's magnetic poles, it is possible to operate the gyro-magnetic compass in gyro mode only, in which case it acts as a DGI and will need to be re-set periodically to a directional reference such as a standby compass or other source of aircraft heading. When it operates as a DGI, this is referred to as **FREE** mode, whilst its normal magnetically monitored operation is referred to as **SLAVED** mode.

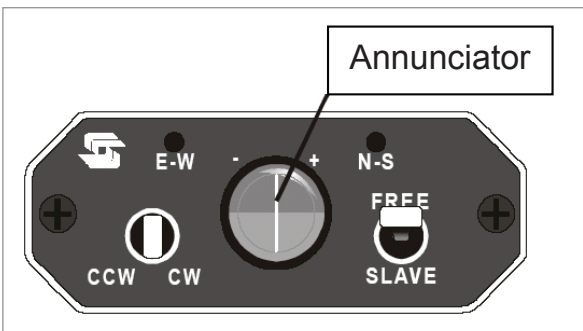


Figure 17.12 Compass Control Panel

Figure 17.12 shows a typical modern compass controller. With the **FREE/SLAVE** switch at **SLAVE**, the compass operates as previously described, with the gyro slaved (in the long term) to the input from the flux valve. If the switch is moved to **FREE**, the magnetic signal from the flux valve is disconnected, the rotor/stator comparison ceases, and the gyro is no longer tied in azimuth and so acts as a free gyro (DGI).

When the Heading Indicator in **FREE** the pilot adjusts the indicated heading in order to correct it to an external datum heading by use of the CCW/CW (counter-clockwise/clockwise) control switch, which is spring-loaded to the central position.

## Annunciator

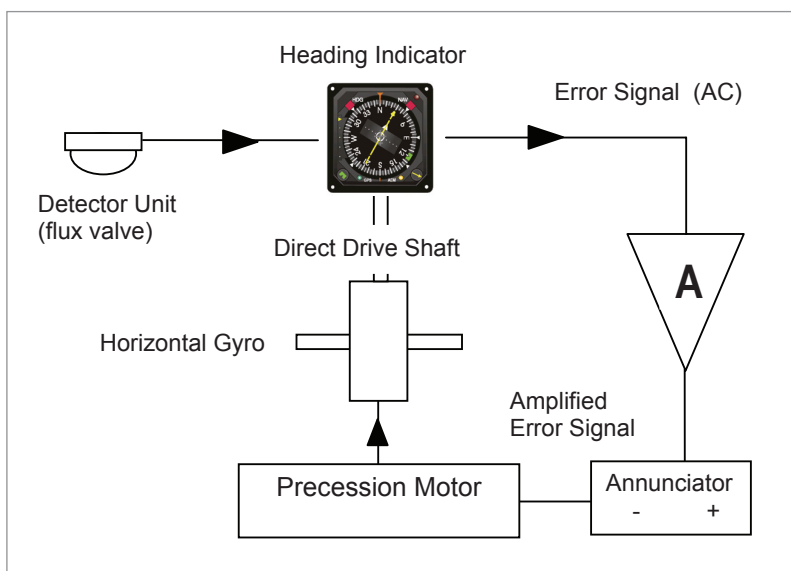


Figure 17.13 Annunciator in Circuit

During normal flight in **SLAVE** mode there is usually continuous slight motion due to oscillations of heading and to vibration which means that rotor/stator comparison of the magnetic flux valve signal against the gyro shaft position continuously generates very small error signals. The error signal and therefore the precession amplifier are continuously 'hunting'. This is normal, and is how the system is designed to work.

These error signals pass through an indicator on their way from the amplifier to the precession motor. This indicator is called an **annunciator** and an example is shown in Figure 17.12.

The annunciator is useful to the pilot for 2 main reasons:

- It is an indication that magnetic monitoring of the gyro is taking place. It shows that **the compass is 'synchronized'**.
- On systems where it is necessary for the pilot to synchronize manually, it indicates **which way to turn** the compass.

## Keeping the Gyro Axis Horizontal

Gyro wander can take 2 forms - drift and topple. The tendency to drift is overcome, as already described, by slaving the gyro to the flux valve output, thereby making it a tied gyro in azimuth.

However, the gyro would still topple, over a period of time, unless prevented from doing so. It therefore needs to be tied, either to the aircraft yaw axis, or to gravity in order to keep it erect. Both the yaw axis and the vertical as defined by gravity have been used as the datum in various models of compass. Both systems use a **levelling switch** and a **torque motor**.

To tie the gyro to the yaw axis, the inner and outer gimbals are maintained at 90° to each other by a system of commutators, insulating strips and brushes. To tie the gyro to the vertical, mercury gravity switches are used. Either way, the correcting signals are passed to a torque motor which applies a rotational force to the gyro in the yaw axis. The resulting precession causes the gyro to return to the horizontal, but at a slow precession rate, so that it does not react wildly to temporary departures from the horizontal such as turns, accelerations, climbs and descents.

## Transmitting Heading Output to Other Instruments

One of the advantages of the gyro-magnetic compass over the simple direct reading compass is the facility to electrically transmit heading information to use as an input into other instruments. The information is picked off from the drive shaft between the gyro and the compass card. The transmitting and receiving device is called a Selsyn Unit.

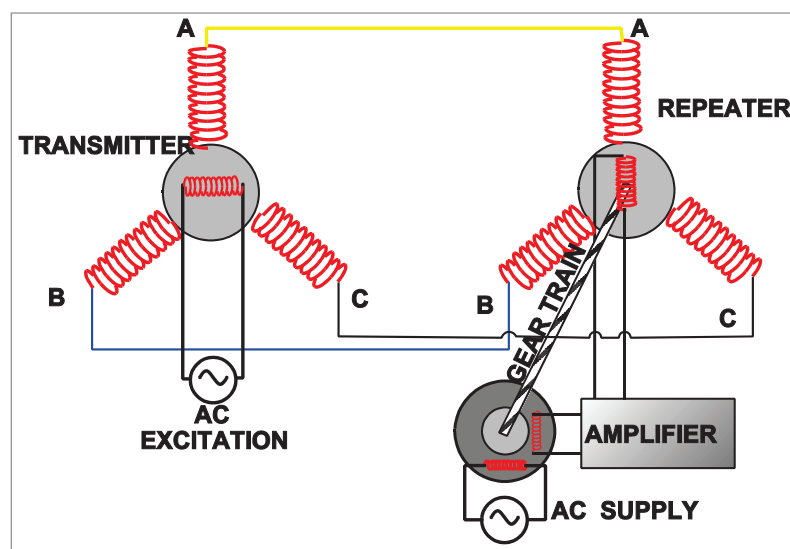


Figure 17.14 Selsyn unit

The rotor of the transmitter (shown in [Figure 17.14](#)) is attached to the heading drive shaft and rotates with it. The orientation of the rotor is therefore the heading which is to be transmitted. The rotor is supplied with a constant primary excitation AC voltage, which induces a field in the stators. The stators are directly connected by 3-strand wire to the 3 stator arms of the repeater, so an identical field is reproduced there.

If the rotor of the repeater is not perpendicular to the field in the repeater stators, an AC voltage will be induced in this repeater rotor. This is passed to an amplifier and then to a motor to turn a shaft on which the repeater rotor is mounted. The repeater shaft will turn until no further voltage is detected. The repeater shaft therefore follows any heading changes in the main gyro drive shaft.

## Summary

The gyro-magnetic compass system overcomes the weaknesses of the direct reading compass (turning and acceleration errors, magnetic element close to source of deviations, no feed to other equipments) and of the directional gyro (no magnetic monitoring).

The gyro-magnetic compass system combines the short term stability of a gyroscope with the long term directional stability of the earth's magnetism.

**Questions**

1. **A gyro-magnetic compass or magnetic heading reference unit is an assembly which always consists of:**

- 1 - a directional gyro
- 2 - a vertical axis gyro
- 3 - an earth's magnetic field detector
- 4 - an azimuth control
- 5 - a synchronizing control

The combination of correct statements is:

- a. 2 and 5
  - b. 1, 3 and 5
  - c. 2, 3 and 5
  - d. 1 and 4
2. **A slaved directional gyro derives its directional signal from:**
- a. a direct reading magnetic compass
  - b. the flight director
  - c. the flux valve
  - d. the air data computer
3. **The gyro-magnetic compass torque motor:**
- a. causes the directional gyro unit to precess
  - b. causes the heading indicator to precess
  - c. feeds the error detector system
  - d. is fed by the flux valve
4. **The heading information originating from the gyro-magnetic compass flux valve is sent to the:**
- a. error detector
  - b. erector system
  - c. heading indicator
  - d. amplifier
5. **The input signal of the amplifier of the gyro-magnetic compass resetting device originates from the:**
- a. directional gyro erection device
  - b. error detector
  - c. flux valve
  - d. directional gyro unit

6. A flux valve senses the changes in orientation of the horizontal component of the earth's magnetic field:

1 - the flux valve is made of a pair of soft iron bars

2 - the primary coils are fed AC voltage

3 - the information can be used by a "flux gate" compass or a directional gyro

4 - the flux gate valve casing is dependent on the aircraft three inertial axis

5 - the accuracy of the value of the magnetic field indication is less than 0.5%

The combination of correct statements is:

- a 2, 3 and 5
- b 1, 3, 4 and 5
- c 3 and 5
- d 1, 4 and 5

## Answers

1	2	3	4	5	6
b	c	a	a	b	a

Chapter  
**18**

Inertial Navigation Systems

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## Introduction

The fundamental element of this complex system is the Inertial Sensor System (ISS). To make up this system we have a stable platform consisting of high quality gyros and accelerometers and a computer.

The purpose of the computer is to integrate the accelerometer outputs with time to give velocity and then integrate velocity with time to give distance travelled. From this is available pitch and roll attitude, true heading, true track, drift, present position in latitude and longitude, ground speed and wind. To change all this information from ISS to Inertial Navigation System (INS) we have a further computer which allows us to inject and store waypoints and then compute track angle error, distance and time to go to reach them. This information can be used by the autopilot, flight director or for normal manual flying of the aircraft.

The modern INS was the first self-contained single source of all navigation data; now joined by the similar IRS, laser gyro system which will be discussed later. The current state-of-the-art engineering has enabled production of INS with performance, size and weight characteristics which far exceed other older navigation systems.

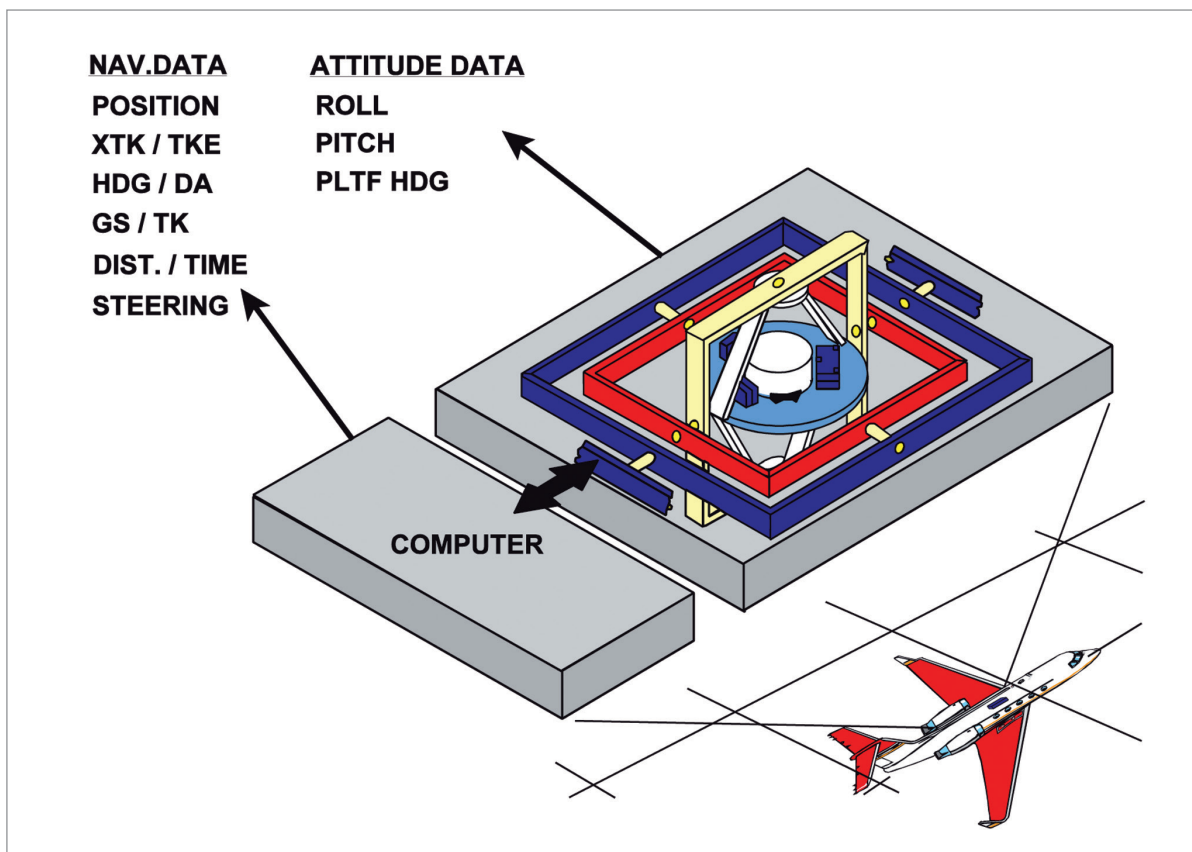


Figure 18.1 Data available from INS

## Basic Principles of INS

Newton's laws of motion state:

- A body continues in a state of rest, or uniform motion in a straight line, unless it is acted upon by an external force.
- The acceleration - rate of change of velocity - of a body is directly proportional to the force acting on the body and is inversely proportional to the mass of the body.
- To every action there is an equal and opposite reaction.

Einstein however, in 1905, totally destroyed the premise of absolute motion. The substance of his new theory was that nothing is at rest and that the term at rest meant merely that the object under observation was moving at the same velocity as some other object, its co-ordinate system and the observer.

The primary measuring device in an INS, the accelerometer, demonstrates this theory for it makes no distinction between at rest and any other fixed velocity. It does, however, make distinction between truly fixed velocities and those which we may regard as fixed, but are really fixed speeds along curved paths.

## Accelerometer and Integrators

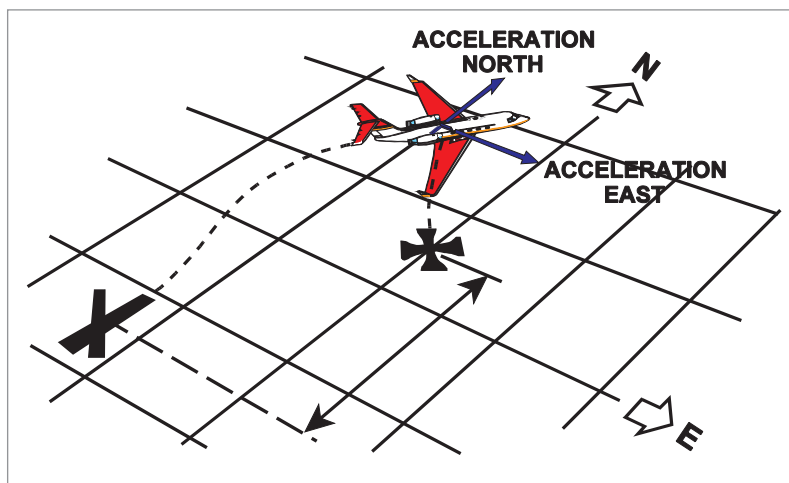


Figure 18.2 Accelerometer and Integrators

Two accelerometers are mounted at the heart of the inertial system. These acceleration measuring devices sense any change in the aircraft's velocity either as an acceleration or deceleration very accurately.

One of the accelerometers measures the aircraft's acceleration in the north-south direction and the second in the east-west direction.

The accelerometer is basically a pendulous device. When the aircraft accelerates, the pendulum, due to inertia, swings off the null position. A signal pick-off device tells how far the pendulum is off the null position. The signal from this pick-off device is sent to an amplifier and current from the amplifier is sent back into a torque motor located in the accelerometer. A torque is generated which will restore the pendulum to the null position. The amount of current that is going into the torquer is a function of the acceleration which the device is experiencing.

The accelerometers would be mounted on a platform; there would be two, one in the north-south direction, the other in the east-west direction (often a third accelerometer is fitted to measure vertical acceleration).

Accelerometers

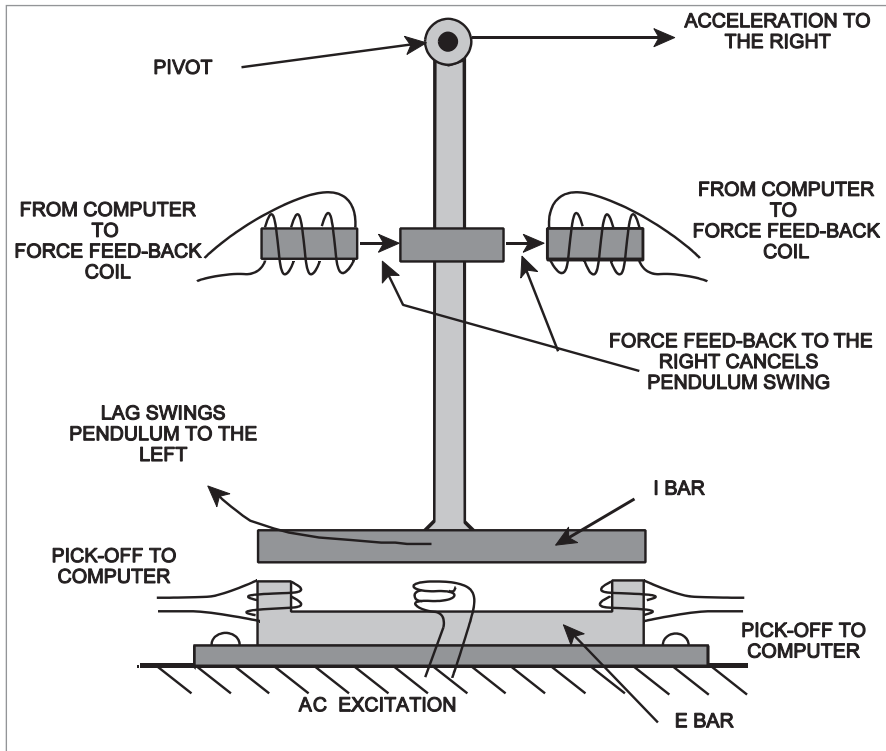


Figure 18.3 Accelerometer

ACCELEROMETERS

These are sensitive pendulous devices that produce an electrical output which is double integrated with time to produce a) velocity b) distance.

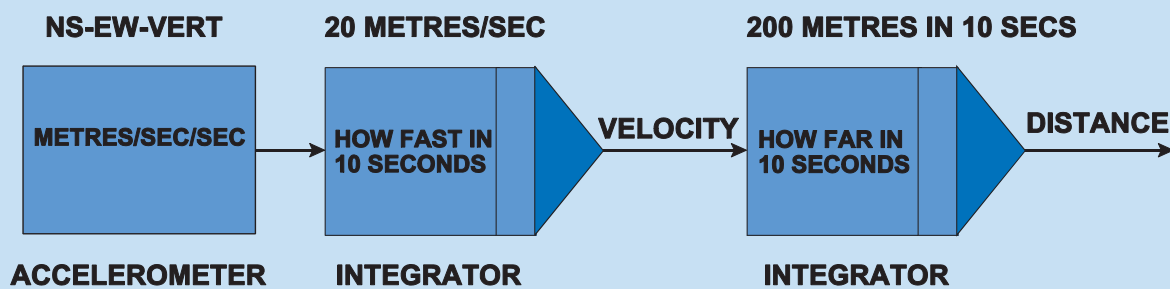


Figure 18.4 Accelerometers

The acceleration signal from the amplifier is also sent to an integrator which is a time multiplication device. It starts out with acceleration which is in feet per second squared. In the integrator, it is literally multiplied by time and the result is a velocity in feet per second.

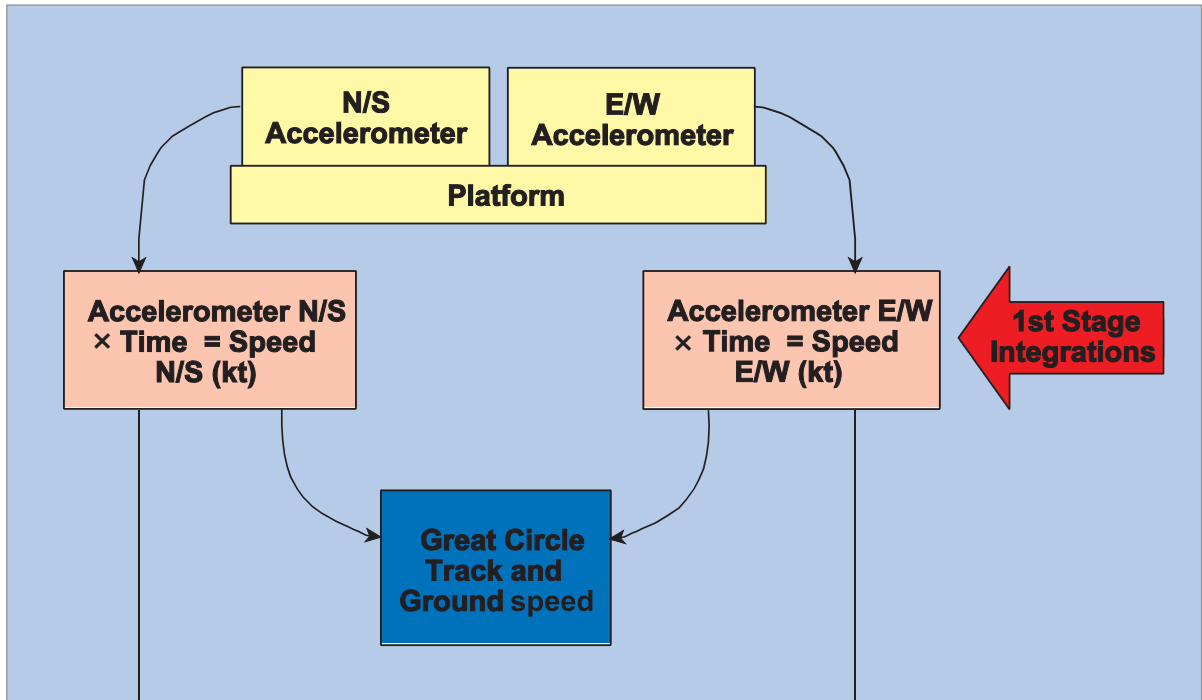


Figure 18.5 Accelerometers and integrators

It is then sent through a second integrator, and again, it is just a time multiplier. With an input of feet per second which is multiplied by time, the result is a distance in feet or nautical miles.

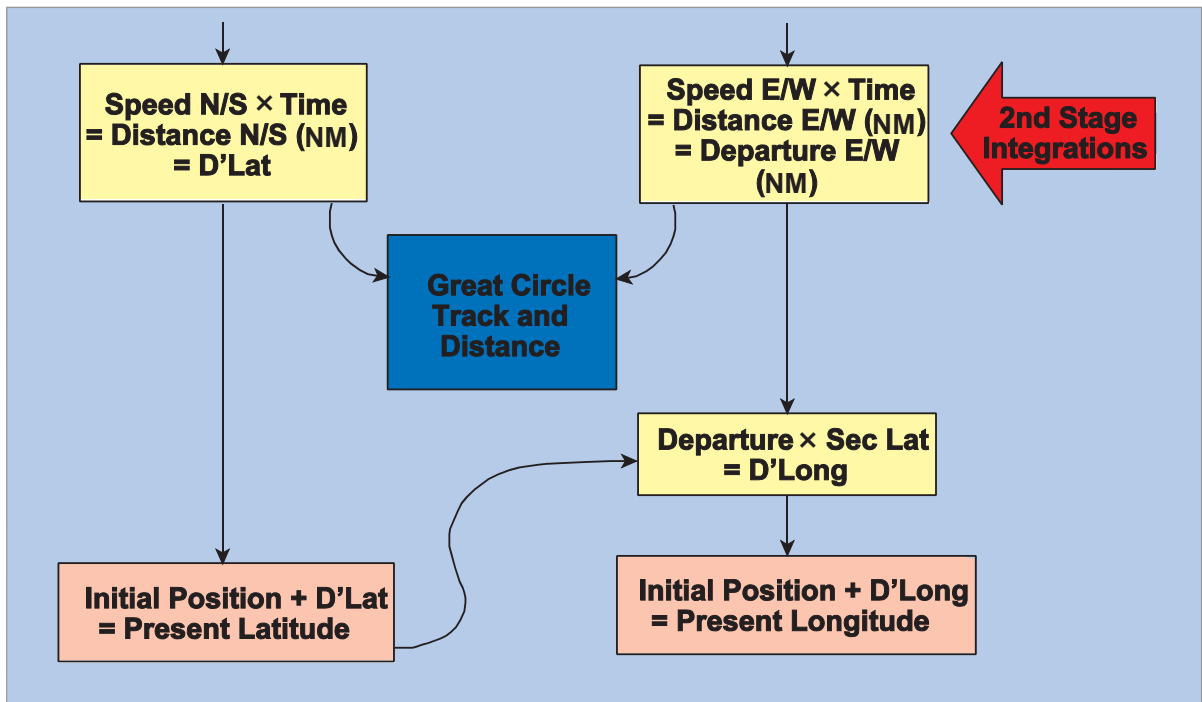


Figure 18.6 Accelerometers and integrators

The accelerometers would be mounted on a platform; there would be two, one in the north-south direction, the other in the east-west direction (often a third accelerometer is fitted to measure vertical acceleration).

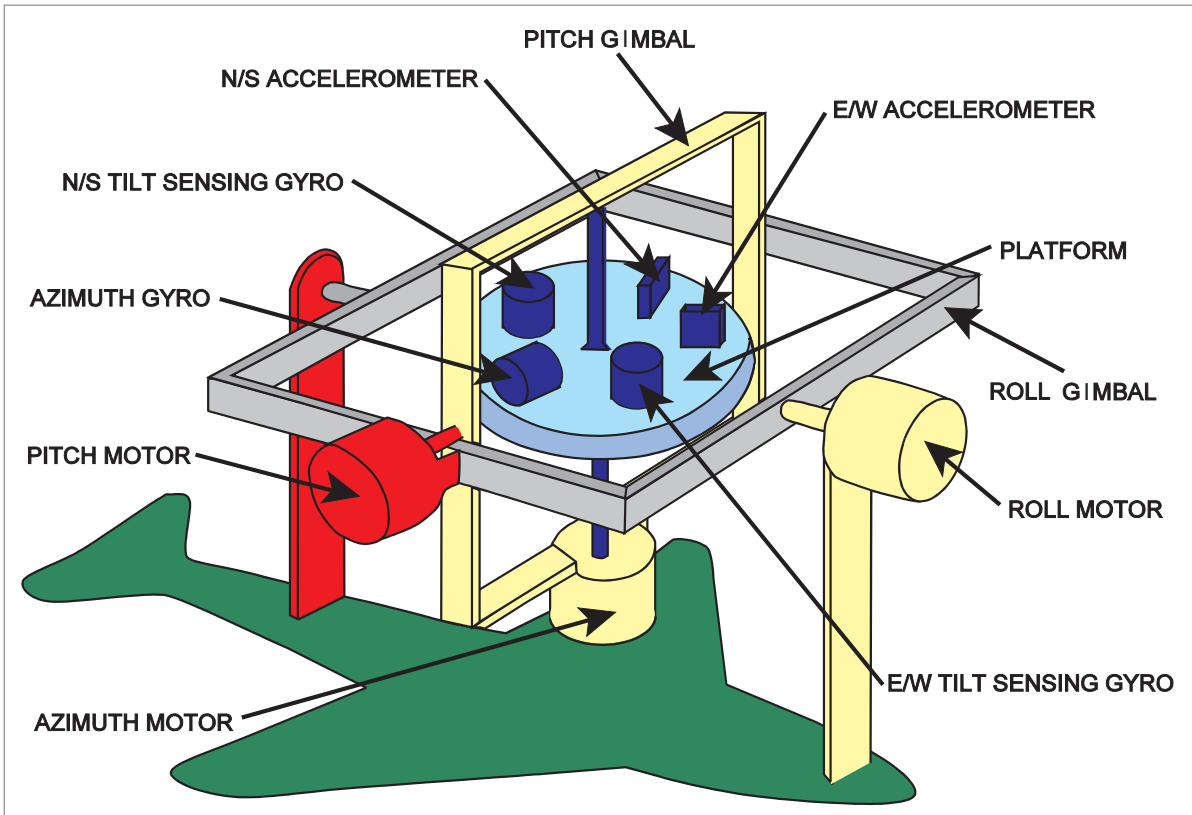


Figure 18.7

The computer associated with the inertial system knows the latitude and longitude of the take-off point and calculates that the aircraft has travelled so far in the north direction and so far in an east direction. The computer can then compute the new position of the aircraft and give a digital read-out which we should note is to tenths of a minute.

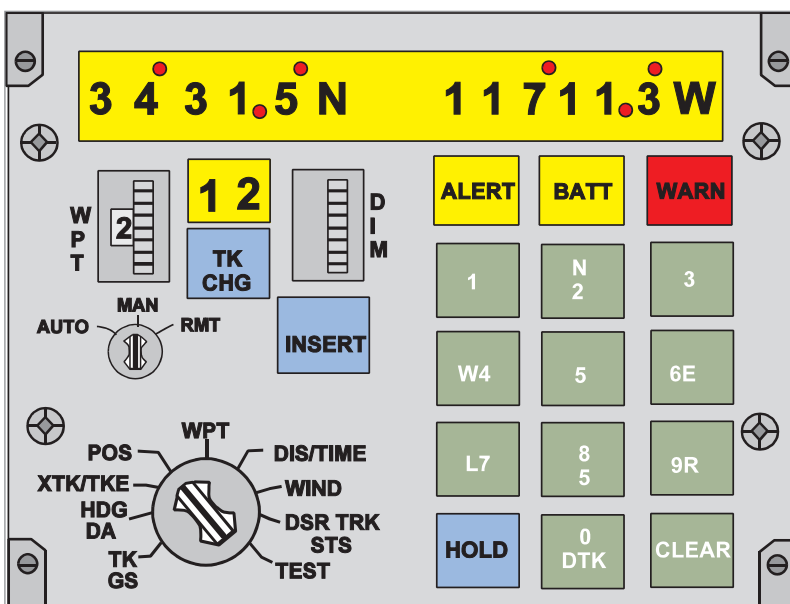


Figure 18.8 POS (present position)

In like manner using stored velocity and present positions the system is able to calculate additional navigation data and display it as requested by the operator. The information is chosen for display through the rotary switch at the bottom left of the control unit; the information that may be obtained will be described in full later.

## Gravity Effects on Accelerometer

Normally the accelerometer is part of the gyro stabilized platform, but if it was hard mounted to the aircraft it could suffer problems in the pitch and roll planes.

The fact that the device has been tilted makes the pendulum swing away from the null position through the effects of gravity.

If this were to take place it would obviously output an erroneous acceleration signal which would in turn result in an erroneous velocity and distance travelled. Therefore, if we allow this there will be a false acceleration problem caused by the pitch or roll angle. If the accelerometer was kept earth horizontal this would not happen and no error would occur.

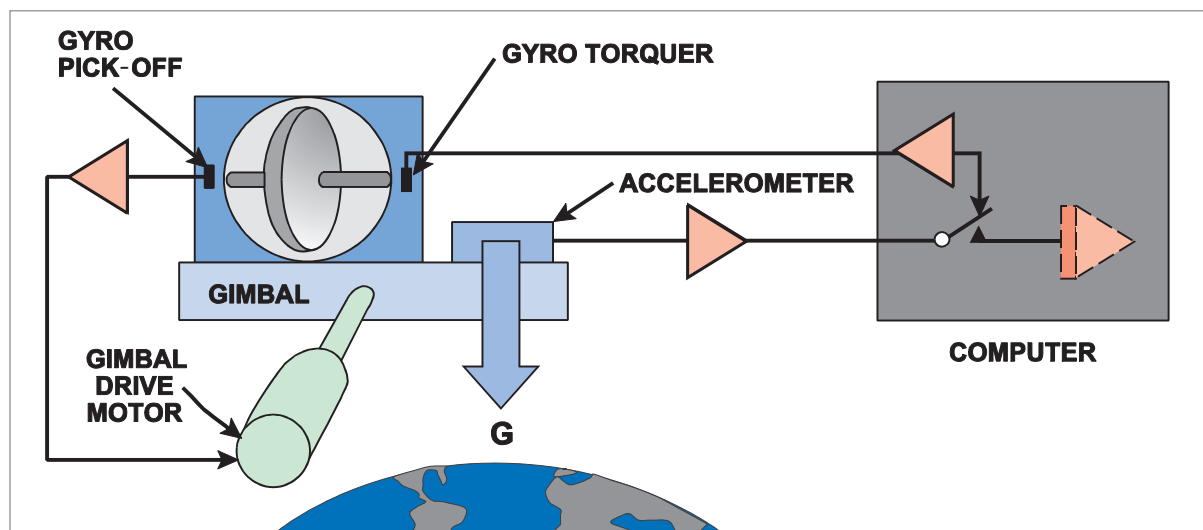


Figure 18.9 Gravity effects on the accelerometer

### The Integrating Gyroscope

An integrating gyroscope used in INSs is a one degree of freedom gyro using viscous rather than mechanical (spring) restraint as seen in the more commonly found rate gyroscope

Figure 18.10 shows a simple rate-integrating gyro. It is basically a can within which another can (the inner gimbal) is pivoted about its vertical axis. The outer can (frame) is filled with a viscous fluid which supports the weight of the inner gimbal so reducing bearing torques.

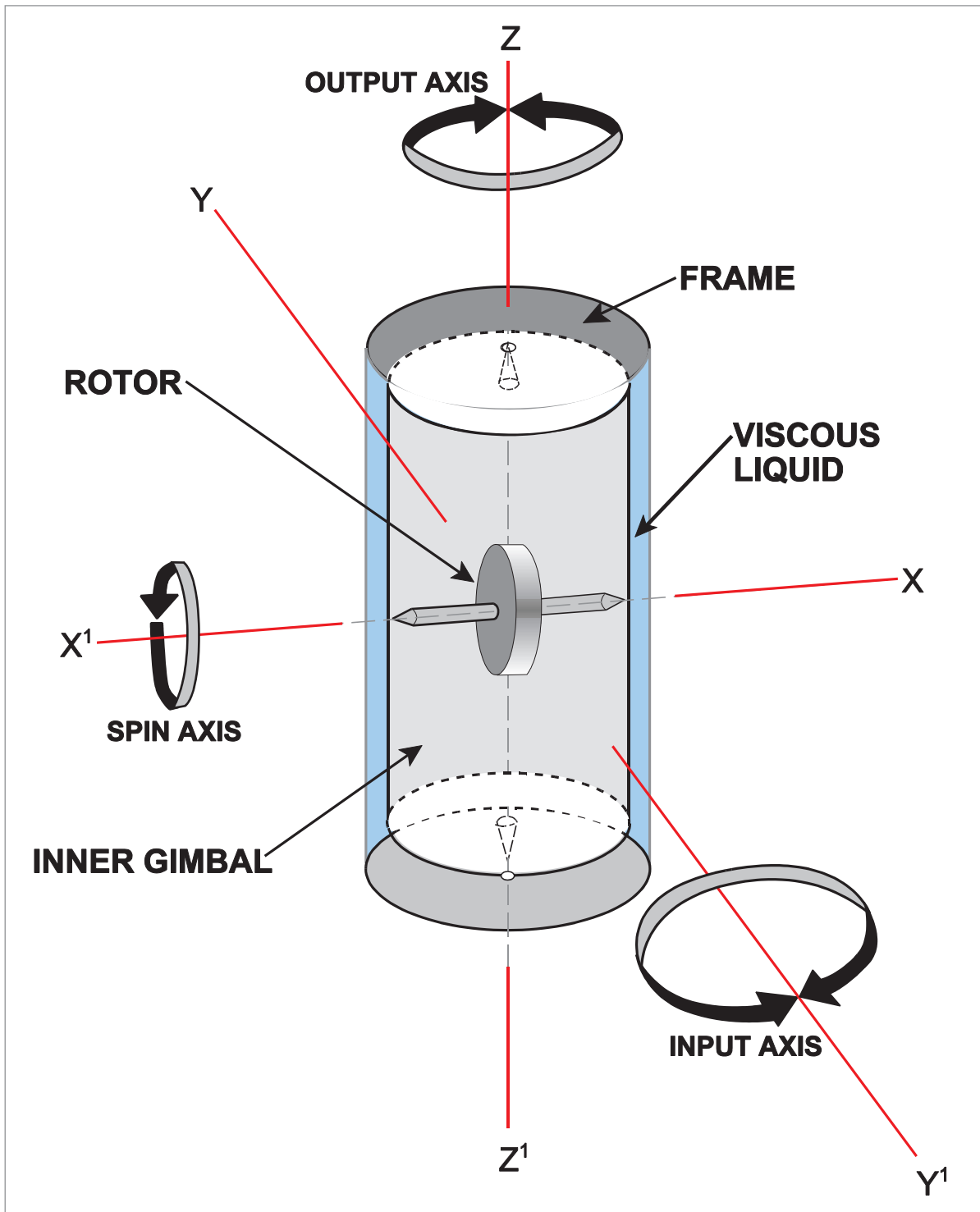


Figure 18.10 Rate-integrating gyroscope

## The Platform

To keep the accelerometer level, it is mounted on a gimbal assembly, commonly called the platform. The platform is nothing more than a mechanical device which allows the aircraft to go through any attitude change and yet the very inner element of the platform on which the accelerometers are mounted is able to stay earth level. Gyroscopes which are used to stabilize the platform are also mounted on the inner-most element of the platform. They provide inputs to amplifiers and motors which control the gimbals and keep the accelerometers level.

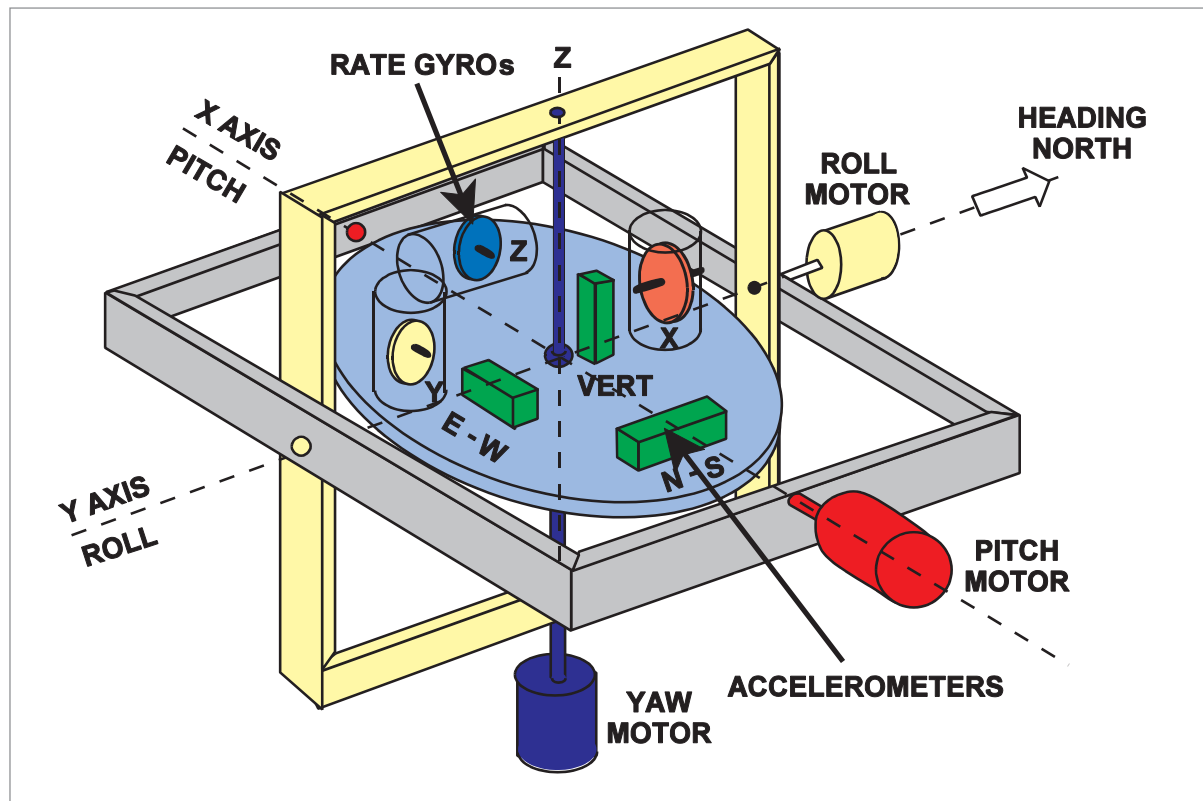


Figure 18.11 The platform

The gyro and the accelerometer are mounted on a common gimbal. When this gimbal tips off the level position, the spin axis of the gyro will remain fixed. The case of the gyro, then, is moved off level and the amount that the case is tipped will be detected by the signal pick-off in the gyro. That signal is then amplified and sent to a gimbal drive motor which restores the gimbal to the level position again. Since the accelerometer is always kept level, it does not sense a component of gravity and is able to sense only the horizontal accelerations of the aircraft as it travels across the surface of the earth.

In reality, three integrating gyros are mounted on the inertial platform, with their input axis mutually perpendicular. Three gimbal motors drive the platform gimbal rings about the pitch, roll and vertical axes respectively. The gyros sense incipient displacement of the platform and activate the appropriate motors to provide for the relative movement of the gimbal rings, as the aircraft moves about the stable platform.



### Earth Orientation

The previously described gyro stabilized platform would remain fixed in space, but the aircraft is not operating in space. It is operating on an earth which is rotating and an earth which is assumed to be round. In order to keep the accelerometers level with respect to the earth so that they sense acceleration of the aircraft in a horizontal direction only, some compensation must be made for the earth rotating and the earth being assumed to be round.

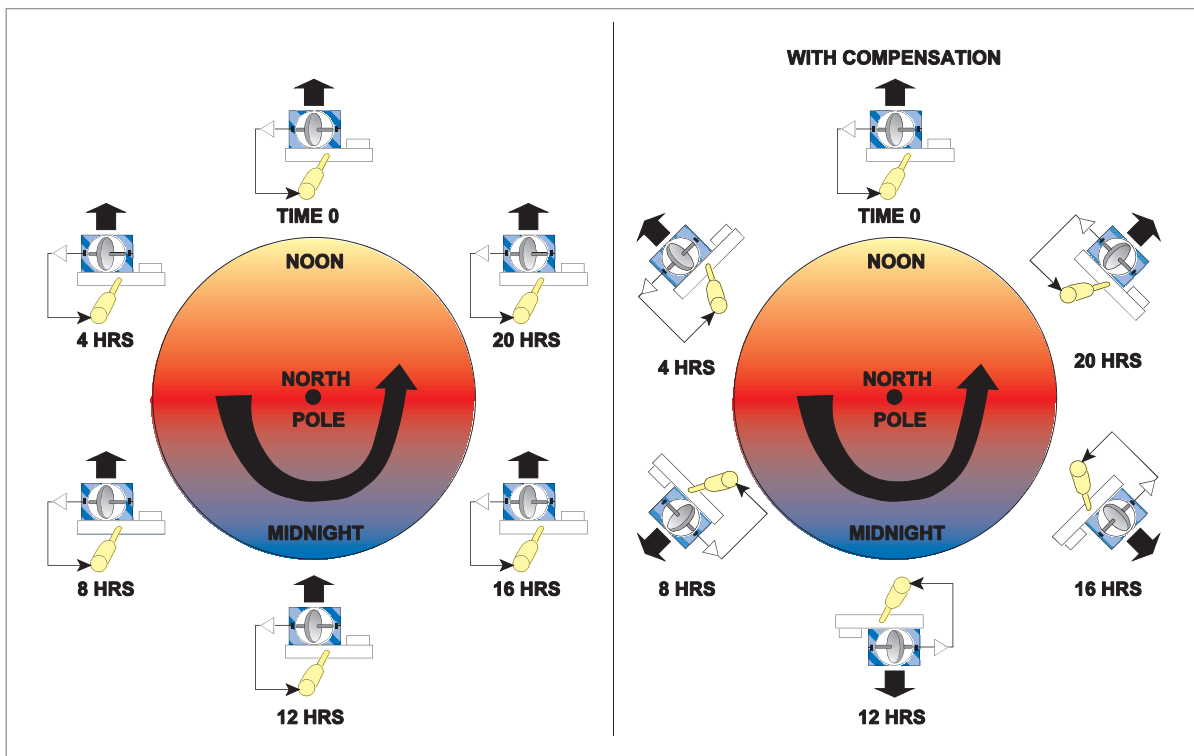


Figure 18.12 Earth orientation

### Apparent Wander

Corrections must be made to gyroscopically stabilized platforms to allow for apparent wander due to earth rotation and aircraft movement over the earth. The required earth rate compensation is a function of latitude since what is being compensated for is the horizontal component of the earth rate felt by the gyros, and that varies with latitude. At the equator, this value is zero degrees per hour and with travel either further north or south, it increases until it becomes a maximum of +/- 15.04 at the poles.

Transport rate compensation is developed using the velocity signal. The electronics through which it is sent contain a term proportional to the earth's radius. So, in reality, the transport rate signal torquing the gyro is the velocity of the aircraft divided by the earth's radius.

Both the earth rate and transport rate compensations are compensated by torquing the gyro. The following diagram should be used to follow the system as explained.

There are a number of compensations generated within the system.

Coriolis and centrifugal effects must be compensated for within the system. Other compensations are necessary because the earth is not a perfect sphere.

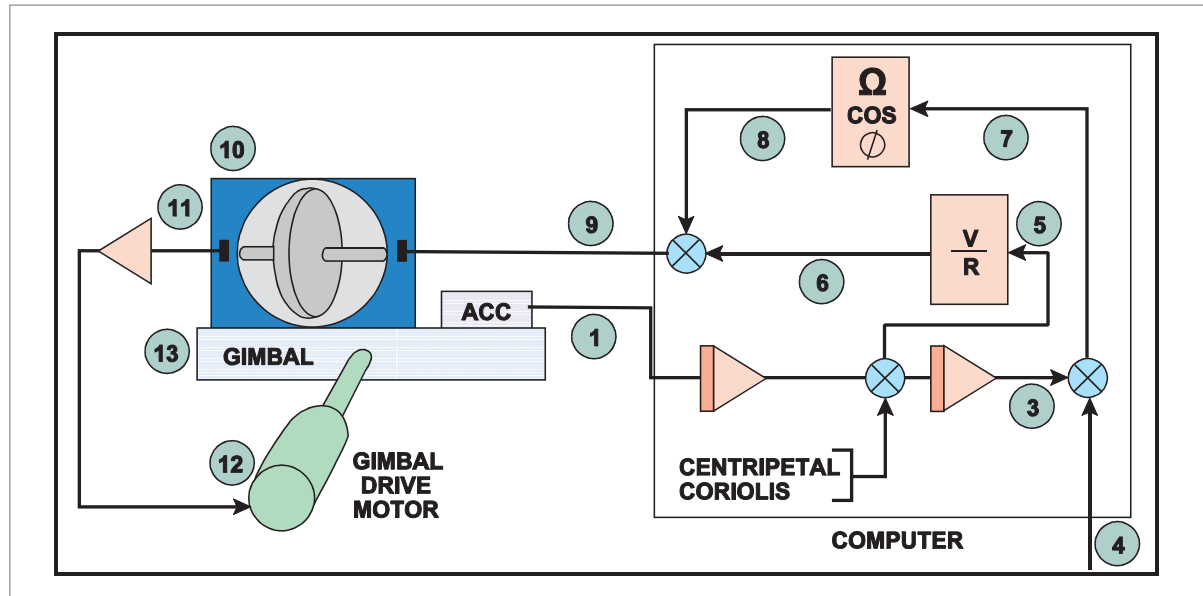


Figure 18.13

- Centrifugal accelerations caused by platform rotation to maintain the local earth vertical.
- Coriolis accelerations caused by the aircraft following a curved path in space when flying normal earth referenced flights.

## Alignment of the System

### ● ACCELEROMETERS MUST BE LEVELLED (VELOCITY SET TO ZERO)

**Initial levelling.** If the platform is not Earth horizontal one or both of the accelerometers will sense an output which is due solely to gravity (since the aircraft is stationary). These tilt-induced outputs from the accelerometers are used to drive the appropriate torque motors (pitch and / or roll) to level the platform.

### ● PLATFORM MUST BE ORIENTED TO TRUE NORTH (GYROCOMPASSING) (POSITION VERIFIED)

**Initial alignment (Gyro-compassing).** Once the platform is levelled, the alignment processed is automatically commenced, using a technique which is known as gyro-compassing. The platform is now earth horizontal, but will not remain so because of the earth's own rotation about its spin axis.

Figure 18.14 Accelerometers and integrators

The stable element in an INS must be accurately aligned in both azimuth and attitude to allow the accelerometers to measure accelerations along their chosen axes.

- **Warm up period** - the first stage in any alignment sequence is to bring the fluid-filled components to the correct operating temperature. This phase normally takes between 3 to 4 minutes.
- **Coarse alignment** - the platform is roughly levelled and aligned in azimuth; this removes gyro alignment errors and cuts the time to a minimum.
- **Coarse levelling** - pitch and roll driven until they are at 90° to each other. The platform is then roughly levelled using either the aircraft frame as reference, or using the outputs from gravity switches or the horizontal accelerometers.
  - Coarse azimuth alignment - is achieved by turning the platform until the heading output agrees with the aircraft's best known True Heading.
  - Coarse alignment level and aligns the platform within 1° - 2° in a few seconds.
- Fine levelling - with zero output from the accelerometers fine levelling is achieved. The process takes anything up to 1 to 1½ minutes, levelling the platform to within 6 seconds of arc.
- Gyro compassing - the platform can be aligned in azimuth by connecting the gyro normally used to stabilize the platform about an east-west axis, to the azimuth gimbal motor. With the platform correctly aligned in azimuth the east gyro should not be subject to rotation of its input axis due to earth rotation; when the platform is out of alignment the east gyro will detect a component of earth rotation and the resultant output signal can be used to torque the azimuth gyro until the table is aligned.
- Accelerometers must be levelled (velocity set to zero).
- Platform must be orientated to true north - gyro compassing (position verified).

## Schuler Period

Schuler postulated an earth pendulum with length equal to the radius of the earth, its bob at the earth's centre and point of suspension at the earth's surface. If the suspension point were accelerated around the earth, the bob would remain vertically below the suspension point because it is at the earth's centre of gravity.

A platform mounted on the suspension point tangential to the earth's surface, i.e. horizontal would therefore remain horizontal irrespective of the acceleration experienced.

The vertical defined by the normal to the platform is therefore unaffected by acceleration. If, for any reason the bob on the earth pendulum became displaced from the earth's centre, the pendulum would start to oscillate. The oscillation period would be 84.4 minutes.

The INS stable element is maintained normal to the local vertical by feeding back the aircraft's radial velocity as levelling gyro signals, and in this way the north and east accelerometers are prevented from detecting components of the gravity acceleration.

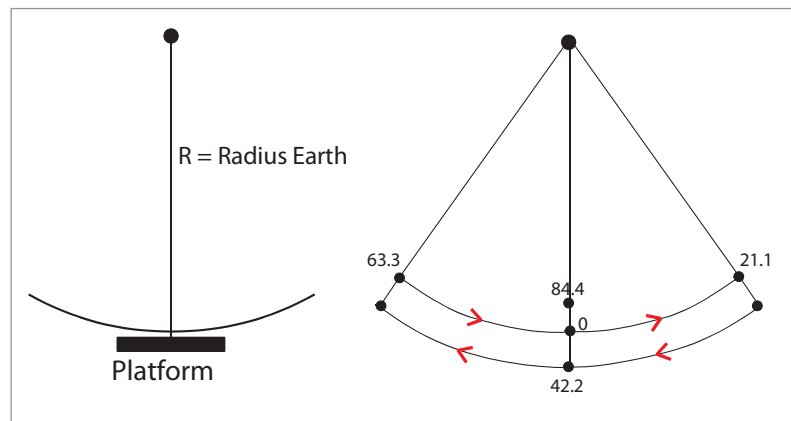


Figure 18.15 The Schuler Period

The control signals are the V/R and U/R terms for vehicle movement. By mechanizing the platform to remain horizontal, an analogue of the earth pendulum of period 84.4 minutes is produced. Should the platform be displaced from the horizontal it would oscillate with a period of 84.4 minutes, which is known as the Schuler Period.

## Errors of INS

Errors can be conveniently considered under the following headings:

- Bounded errors
- Unbounded errors
- Inherent errors

### Bounded Errors

Errors which build up to a maximum and return to zero within 84.4 minutes Schuler cycle are termed bounded errors. The main causes of these errors are:

- Platform tilt due to initial misalignment
- Inaccurate measurement of acceleration by accelerometers
- Integrator errors in the first stage of integration

### Unbounded Errors

Unbounded errors - are either cumulative track errors or distance errors:

- Initial azimuth misalignment of the platform
- Wander of the azimuth gyro

Errors which give rise to cumulative errors in the recording of distance run:

- Wander in the levelling gyros. This causes a Schuler oscillation of the platform but the mean recorded value of distance run is increasingly divergent from the true distance run.
- Integrator errors in the second stage of integration.

## Inherent Errors

The irregular shape and composition of the earth, the movement of the earth through space and other factors provide further possible sources of error. Such errors vary from system to system depending upon the balance achieved between accuracy on one hand and simplicity of design, reliability, ease of construction and cost of production on the other.

## INS Control and Display Panels

There are many makes and models of INS currently on the market. The “state-of-the-art” trend is towards a single control/display unit with a standard keyboard, but with a single small video screen (rather than the various individual LED windows which are shown in the following illustrations). With the modern video screen presentation, the loading and extraction of information is achieved by selecting a “page number”, with each page (which is displayed on the screen) dealing with associated functions. One major advantage of this type of system is that hundreds or even thousands of waypoints can be “stored” in the machine memory. These waypoints (normally airway reporting points such as VORs and intersections) are automatically loaded from a master disc, which is supplied and regularly updated by specialist agencies.

Because of the high capital investment which was involved in the last generation of INS systems, and because they are proving to be extremely reliable, you are perhaps more likely to encounter the traditional type of control/display units described below. Another good reason for considering this system, rather than the modern one, is that the JAA examination questions are based on the older type of INS.

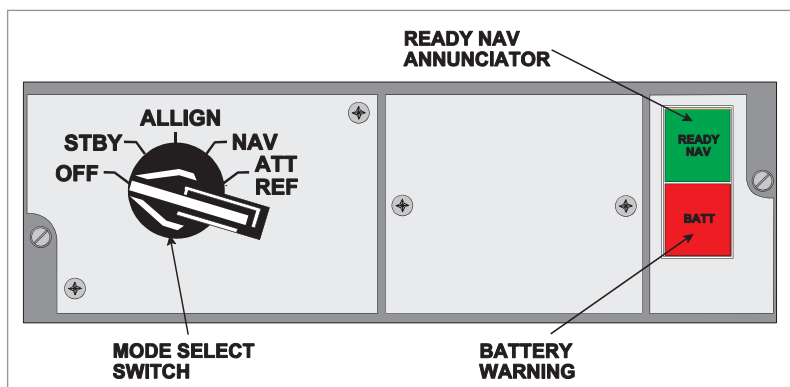


Figure 18.16 Mode Selector Unit

In any event, please appreciate that the following paragraphs are intended only as a general guide and not as a definitive operating instruction for any particular model of INS.

The traditional INS system employs two panels for control and display. The simpler of the two, the mode selector panel, is shown at [Figure 18.16](#).

The function of the mode selector panel is straightforward:

- In the standby mode the power is supplied to all parts of the system. It is normal to insert the start position (the aircraft’s ramp position in lat/long to the nearest tenth of a minute of arc) whilst the equipment is in this mode.

- In the alignment mode the platform is levelled and aligned (gyro compassed), and when these processes are complete and **READY NAV** illuminated. The equipment can now be switched into the Nav mode, and the aircraft is free to taxi without degrading the accuracy of the INS.

There may be occasions when full navigation computing is not available but the gyros are serviceable. This could be in the event of a computing malfunction so that earth rate and transport wander corrections cannot be calculated or it could be after an alignment failure in flight.

However, on many aircraft the gyros are used as primary attitude information as well as for inertial navigation and it may be possible to retain gyro information. This is done by selecting ATT REF on the MSU.

Selecting ATT REF disconnects computing and loses alignment, if this has not already happened anyway. The accelerometers now act as gravity switches, as they do during the levelling phase of alignment and the gyros become gravity-tied in the long term - earth gyros. The system now gives attitude information and a limited form of heading. The gyros are normally very accurate, but there is no correction for earth rate and transport wander and the heading needs to be reset periodically to an independent (usually magnetic) source. In effect, the gyros are acting as a super-accurate form of DGI and as an attitude indicator.

Should the aircraft electrical supply to the INS cease for any reason the INS will automatically switch to its own battery pack. For as long as a satisfactory level of power is being supplied by the internal battery, the INS BATT light will be illuminated on the Control and Display Unit. As the power from the battery starts to fail, the BATT warning light on the Mode Selector Unit will illuminate, indicating that the INS is about to fail. If you are half way across the Pacific Ocean at this time, this could spoil your whole day, since of course the INS cannot be re-levelled and/or re-aligned in flight (for this the aircraft must be stationary, and the exact position known).

The control/display unit (CDU) is shown at [Figure 18.17](#).

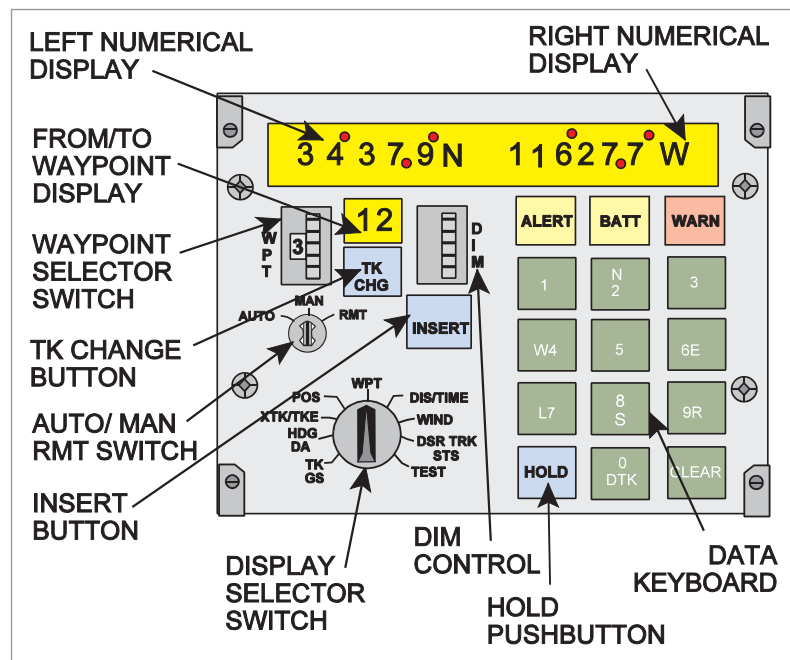


Figure 18.17 Control and display unit (CDU)

The reader who has completed his or her studies of the radio syllabus will undoubtedly notice the similarity between this CDU and the control/display panel of a similar vintage VLF/Omega receiver. Although the inputs for the two equipments are vastly different, the presentation of navigational information to the pilot is more or less identical in both cases.

As already mentioned, these notes are not intended as an operator's brief, but rather to help you pass an examination. Please appreciate that, although other INS panels may look dissimilar to the one shown at [Figure 18.17](#) the information given by the system will be basically the same.

Using the panel shown at [Figure 18.17](#) let us start at the top left hand corner and look at the function of each of the displays and controls.

The two large windows at the top of the panel (labelled left and right numerical displays) comprise the principal outputs of the system. Glance now at the function selector (bottom left hand corner), we will start with the selector in the seven o'clock position (TK/GS) and work anticlockwise through the functions considering the values shown in the two LED windows as we go.

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 lat/longs before flight, reloading new waypoints in flight, or checking that waypoints are correctly loaded.

The **FROM/TO** waypoint display shows the two waypoints between which the INS assumes that it is flying. All digital read-outs 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 lat/long to begin with.

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).

The DIM control governs the brightness of the LED displays and the panel lighting.

The ALERT annunciator warns the operator that the aircraft is approaching the next waypoint. In AUTO mode the alert light will come on, steady, 2 minutes to run to the waypoint, and will extinguish as the track changes overhead the waypoint. In MANUAL mode the alert light will come on, steady, 2 minutes to run to the waypoint; the light will then flash 30 seconds before the waypoint, and will continue to flash until the track is changed. The annunciator will not illuminate below a set speed (typically either 100 kt or 250 kt).

The battery annunciator will be illuminated when the INS is operating on internal power.

The warning annunciator illuminates when a system malfunction occurs.

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 operator is required to update the waypoint from/to read-out as each waypoint is overflown. The exact function of the remote position will depend on the complexity of the INS computer programme, and is outside the scope of this syllabus but in general terms it allows for simultaneous insertion of waypoints into more than one INS from one CDU.

The INSERT push-button is used in conjunction with the data input keyboard to enter information into the system.

Finally, the HOLD push-button is used primarily for updating the INS position when overflying a reliable fix, such as a VOR overhead. The HOLD button is depressed as the fix is overflown,

the function switch is placed in the POS (position) mode, the exact lat/long of the radio fix (in this case the lat/long of the VOR) 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.

### Summary INS Warning Lights

	LIGHT INDICATION	ACTION REQUIRED
READY NAV (MSU)	Green light, indicates alignment complete.	Select 'NAV'
BATT (MSU)	Red light, indicates battery power too low for operation.	Check power supplies
ALERT (CDU)	Amber light, indicates approaching (overflying in a MAN) a waypoint.	None, unless in MAN mode when TK CHG is initiated
BATT (CDU)	Amber light, indicates INS operating on back-up power.	Check power supplies.
WARN (CDU)	Flash red light, indicates system malfunction.	Set selector to DSR TK/STS note action code and consult user's guide for appropriate action.

### LED Display

**Note:** All of the following descriptions are based upon a Desired Track between waypoints of 060°.

#### Track & Ground Speed

The INS derived aircraft track (°T) is shown to the nearest tenth of a degree in the LH (left hand) window.

The INS derived ground speed is shown to the nearest knot in the RH window.

The track is 040°(T) and the ground speed 502 kt at [Figure 18.18](#).

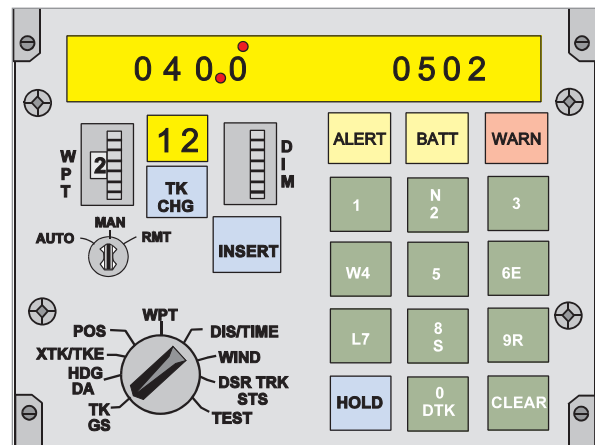


Figure 18.18 TK / GS (Track and ground speed)



**Heading & Drift**

The INS derived true heading (the angle between the north-south axis of the platform and the aircraft fore and aft axis in a north aligned system) 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 in the RH window, and is preceded by an L (left/port drift) or an R (right/starboard drift).

The heading is 050°(T) and the drift 10°.

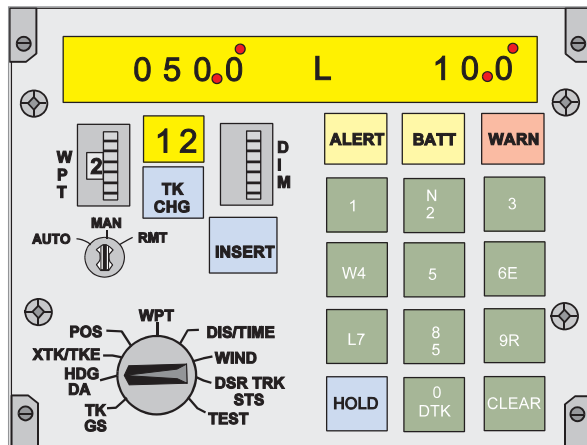


Figure 18.19 HDG / DA (Heading and drift angle)

**Cross Track Error & Track Keeping Error**

**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.

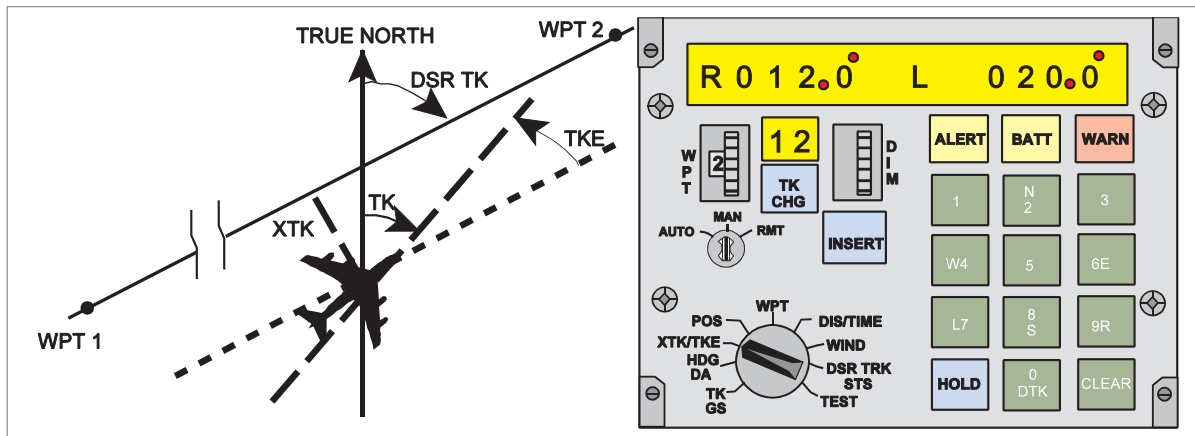


Figure 18.20 XTK / TKE (Cross track distance and track error angle)

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. In this case, remembering that we desire a track of 060°, we must be “making good” a track of 040° because we have an indication that we are tracking 20° to the left of our desired track between consecutive waypoints.

In summary the cross track error is 12 NM to the right and the track angle error is 20° to the left in the situation also shown at [Figure 18.20](#)

### Present Position

The aircraft's present latitude is shown to the nearest tenth of a minute of arc in the LH window.

The aircraft's present longitude is shown to the nearest tenth of a minute of arc in the RH window.

The aircraft's position is therefore shown as 34°31.5'N 117°11.3'W.

### 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 placed 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.

Let us consider briefly how this could be useful. Suppose that you are half way between, say, waypoints 3 and 4 and air traffic control clear you direct to waypoint 6. By selecting a track change from waypoint 0 (the aircraft's present position) to waypoint 6 and inserting it, the aircraft will fly you directly to Wpt 6 if coupled to the flight director/autopilot.

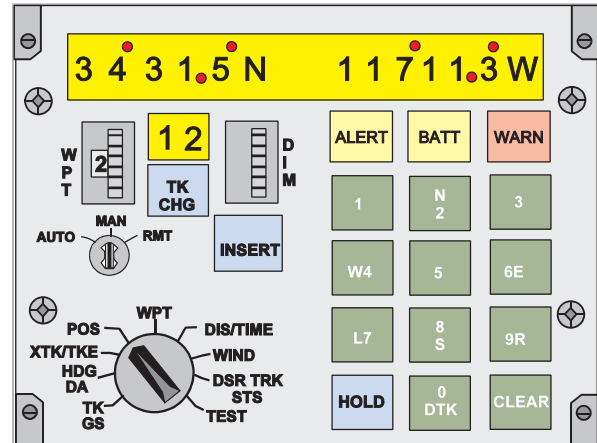


Figure 18.21 POS (Present position)

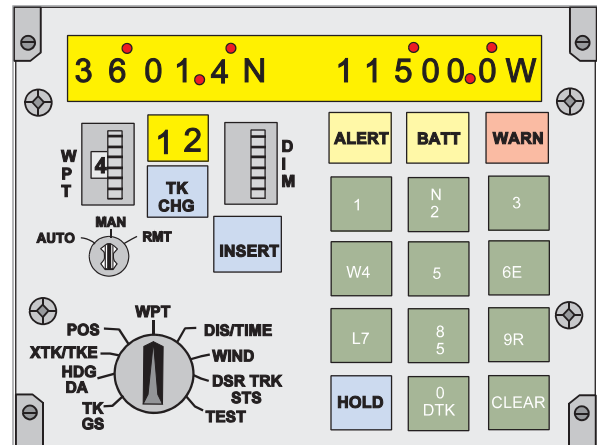


Figure 18.22 WPT (Waypoint position)

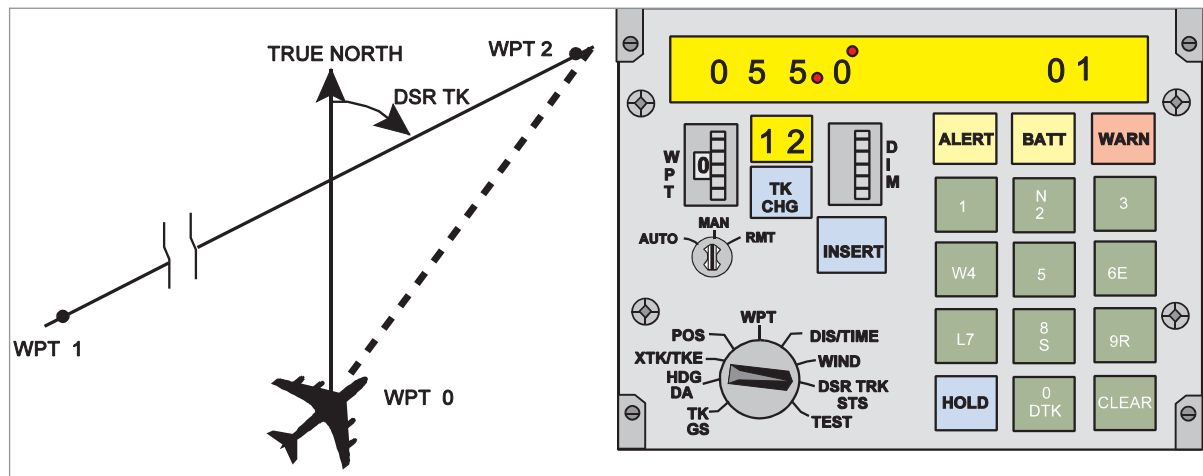


Figure 18.23 Waypoint Zero

Waypoint zero is reserved for the computer to establish a track from the aircraft's present position and will not accept operator entered waypoint co-ordinates.

## Distance and Time

The distance to go from the aircraft's present position direct 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 here is shown as 140 NM and the time as 16.7 minutes.

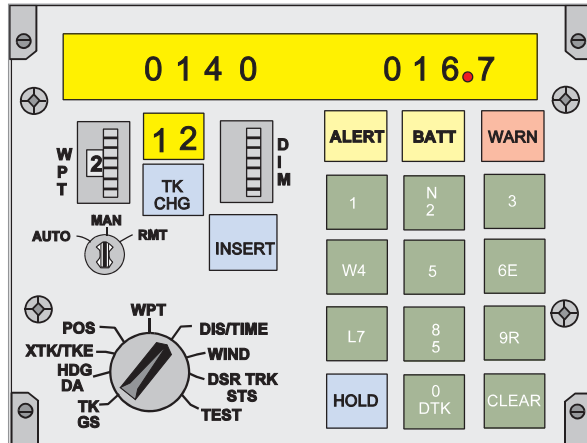


Figure 18.24 DIS / TIME

## Wind Speed and Direction

The INS derived wind direction ( $^{\circ}$ T) 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  $155^{\circ}$ (T)/85 kt.

## Desired Track and Status

The desired track (assuming that the aircraft is on the direct great circle track between the two selected waypoints) is shown in degrees true to the nearest tenth of a degree in the LH window.

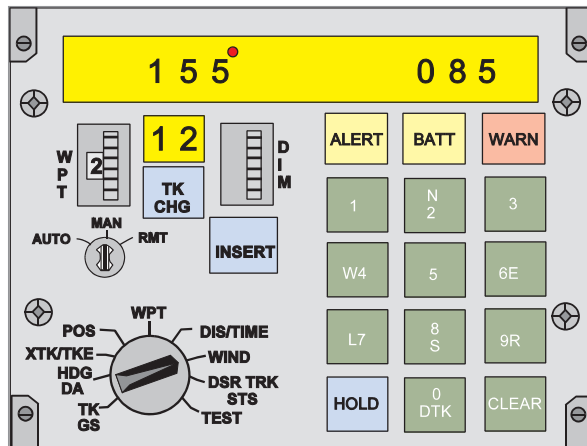


Figure 18.25 WIND (Wind velocity)

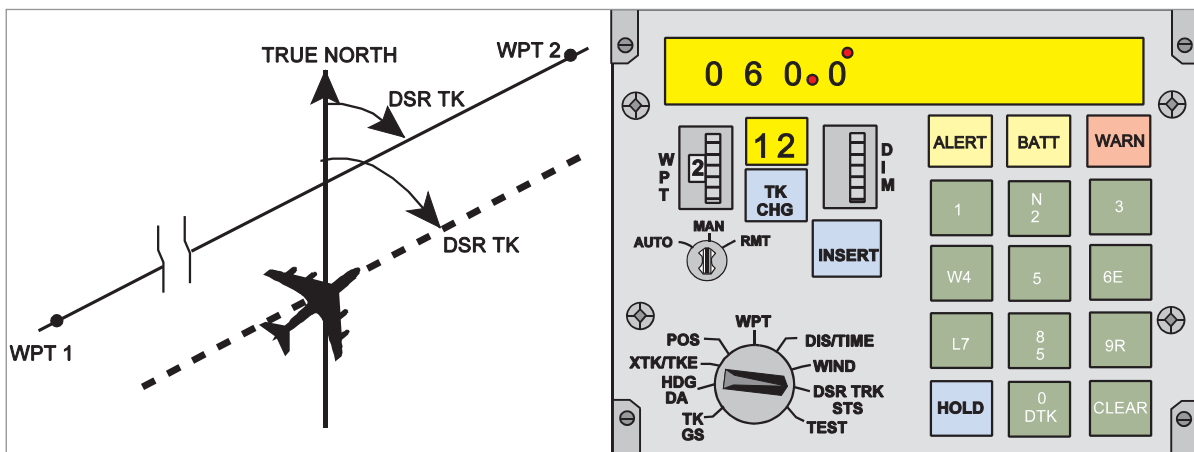


Figure 18.26 DSR TK / STS (Desired track and status)

The RH window will normally be blank, since the status check is generally only available with the equipment in the alignment mode.

The desired track is  $060.0^\circ(T)$  which allows us to see the relationship between the aircraft's current position and track and the direct track between the two selected waypoints. 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.

### Test

The diagram at [Figure 18.27](#) shows the function switch in the test position resulting in all of the digits on the various displays being illuminated either showing a figure or letters. This enables the operator to check that all of the functions are operating.

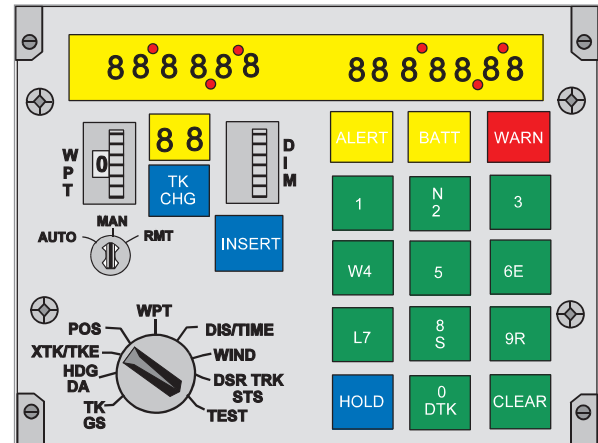


Figure 18.27 TEST (Light emitting diode test)

## Manual and Automatic System Checks

At the initial setting up stage the start position must be fed into the INS computer with a high degree of accuracy. 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 incorrect latitude. Likewise, and for the same reasons, the platform will not remain directionally aligned with respect to north.

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.

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.

An incorrect input of the lat/long 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 (finger trouble)! 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.

A second check is to call up the initial great circle track (TK/GS) and distances (DIS/TIME) between consecutive waypoints, and to compare these values against those shown on the flight log/flight progress log/flight plan.

## Questions

1. **INS errors are classified as "bounded errors" and "unbounded errors".**
  - a. An "unbounded error" is an error that increases with time, an example being the distance gone error due to a ground speed error
  - b. An "unbounded error" is an error that increases with time, an example being an increasing ground speed error due to the platform not being levelled correctly
  - c. A "bounded error" is an error that is subject to sudden unpredictable random changes. Most notable during pitching manoeuvres and when raising or lowering flap and U/C
  - d. A "bounded error" is an error that is "tied" to the real wander rates of the gyros on the platform
  
2. **Two checks that can be carried out to check that two selected sequential waypoints have been entered correctly are:**
  - a. select DSR.TK/STS and check that the status is less than 4; select DIS/TIME and check that the time agrees with the flight plan time
  - b. select DIS/TIME and check that the distance agrees with the distance on the flight plan; then check that the time agrees with the flight plan time for the leg
  - c. select DIS/TIME and check that the distance agrees with the distance on the flight plan; select DSR.TK/STS and check that the track agrees with the flight plan track for the leg
  - d. select DIS/TIME and check that the distance agrees with the distance on the flight plan; select HDG/DA and check that the heading agrees with the flight plan heading for the leg
  
3. **In an INS the E/W accelerations are converted into an E/W speed (kt) at the first stage of integration and into E/W distance gone (NM) at the second stage of integration. This gives:**
  - a. departure which is multiplied by cosine of the present latitude to obtain d'long (min) which is used to automatically update the present longitude
  - b. d'long (min) which is used to automatically update the present longitude
  - c. departure which is multiplied by secant of the present latitude to obtain d'long (min) which is used to automatically update the present longitude
  - d. departure which is multiplied by sine of the present latitude to obtain d'long (min) which is used to automatically update the present longitude
  
4. **At the second stage of integration E/W speed is converted into E/W distance gone. To convert this departure into change of longitude it has to:**
  - a. be divided by secant of the latitude
  - b. be multiplied by secant of the latitude
  - c. be divided by tangent of the latitude
  - d. be multiplied by cosine of the latitude

5. **The amber ALERT light on an INS control and display unit:**
- illuminates steadily 2 minutes, in AUTO mode, before reaching the next waypoint
  - start flashing 2 minutes before reaching the next waypoint and goes out at 30 seconds to run
  - illuminates if power from the aircraft bus bar has been lost and the system is operating on standby battery
  - illuminates steadily after passing a waypoint in manual mode, until the next leg is programmed in
6. **With reference to Inertial Navigation Systems, the functions of the integrators are:**
- at the second stage of integration to suppress unbounded errors (when in the NAV mode)
  - at the first stage of integration to convert acceleration, with respect to time, into speed, (when in NAV mode)
  - at the second stage of integration to convert speed, with respect to time, into distance gone, (when in the NAV mode)
  - to align the platform (when in the level and align modes)
- all the above statements are true
  - only (ii), (iii) and (iv) of the above statements are true
  - only (i), (ii) and (iii) of the above statements are true
  - only (ii) and (iii) of the above statements are true
7. **The computer of a north referenced Inertial Navigation System (INS) in flight, provides compensation for:**
- aircraft manoeuvres, real wander, apparent wander, transport wander
  - coriolis, real wander, apparent wander, transport wander
  - earth rotation, transport wander, coriolis
  - transport wander, apparent wander, coriolis, magnetic variation
8. **During initialization of an INS the aircraft must not be moved until:**
- the ramp position has been inserted and checked
  - the platform is levelled
  - the gyros and accelerometers are in the "null" position
  - the green "ready NAV" light has been illuminated and the mode selector switch has been set to the "NAV" position



## Answers

1	2	3	4	5	6	7	8
a	c	c	b	a	d	c	d



Chapter  
**19**

**Inertial Reference System**

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## Inertial Reference System

The **laser gyro** has caused a technological revolution in the design of **inertial reference and navigation systems**. This solid state high precision, angular rate sensor is ideally suited for highly reliable strap down system configuration. It eliminates the need for gimbals, bearings, torque motors, and other moving parts, and consequently changes the system operation considerably from conventional **inertial navigation systems**.

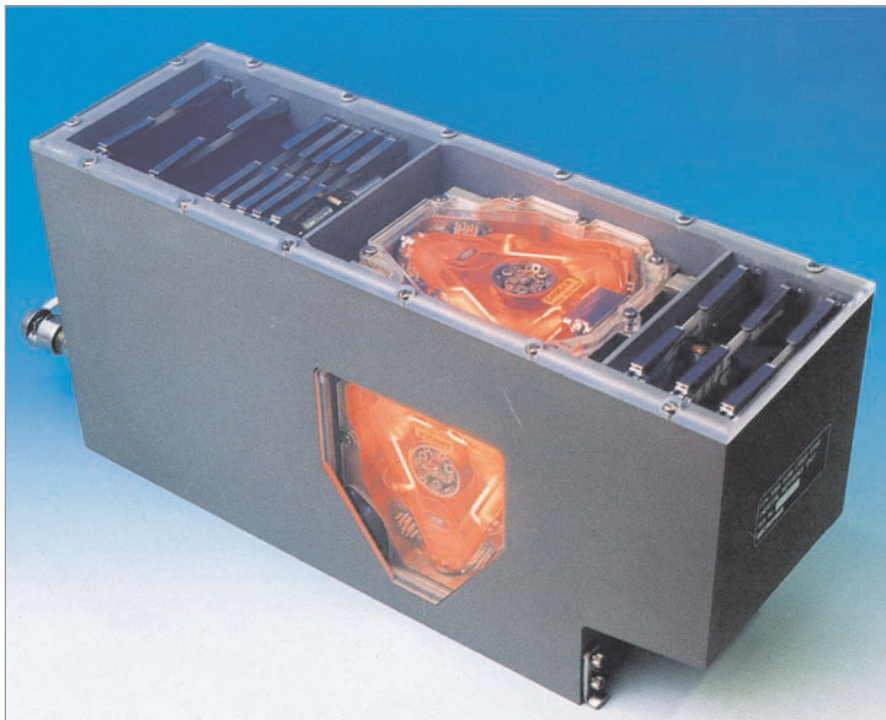


Figure 19.1 GEC-Marconi FIN3060 Commercial Aircraft Inertial Reference

## Inertial Navigation

**Inertial Navigation** means the determination of a vehicle's location without the aid of external references. **Strap down inertial navigation** goes a step further by enabling navigation without the use of a mechanically stabilized platform. This has been achieved through the advent of **laser gyros / rate sensors** and powerful, high speed microprocessors. The **laser gyros** allow a microprocessor to maintain a stable platform mathematically, rather than mechanically.

## Inertial Reference Unit

The **Inertial Reference Unit** (IRU) is the heart of the **Inertial Reference System** (IRS). It provides all required **inertial reference** outputs for the aircraft's avionics.

Outputs are:

Primary attitude	Pitch and roll
Heading	True, Magnetic
Accelerations	Lateral, Longitude, Normal
Angular rates	Pitch, Roll, Yaw
Inertial velocity	N/S, E/W, GS, TA, Vertical rate
Position	Latitude, longitude, inertial altitude
Wind data	Wind speed, wind angle, drift angle
Calculated data	Flight path angle and acceleration Along and across track acceleration Inertial pitch and roll rate Vertical acceleration Potential vertical speed.

## Inertial Information

**Inertial information** is used by:

- Flight management computer
- Flight control computer
- Thrust management computer
- Stability augmentation system
- Weather radar
- Anti-skid auto brake systems
- Attitude direction indicator
- Horizontal situation indicator
- Vertical speed indicator
- Radio direction magnetic indicator
- Flight data recorder

## The Primary Sources of Information

The primary sources of information for the IRU are its own internal sensors, three **laser gyros**, and three **inertial accelerometers**. The only other inputs required are initial position, barometric altitude, and True Airspeed (TAS).

Initial position is required because present position is calculated from the distance and direction travelled from the initial start position entered.

Barometric altitude stabilizes the vertical navigation, and thereby stabilizes the vertical velocity and inertial altitude outputs.

The TAS input allows the IRU to calculate wind speed and wind direction.

## The Laser Gyro

The **laser gyro** is an example of the application that uses the characteristics of light to measure motion. This device operates based on the SAGNAC effect. One beam rotates in one direction and the other beam in the opposite direction. One of the conditions that must be satisfied to maintain **lasing** is that the number of wavelengths in the beams path length must equal a whole number. When the wavelengths change there is a concurrent change in the light's frequency. This means that in a rotating gyro, one laser beam will exhibit an increase in frequency, whereas the other beam will exhibit a frequency decrease. The frequency difference between the two beams is easily and accurately measured along optical paths.

## Principles of Laser Gyros and IRS

Laser (Light Amplification and Stimulated Emission of Radiation) gyros measure rotation by comparing two laser beams created and directed to rotate in opposite directions within a very narrow tunnel. Photons are emitted within the laser cavity in **all directions** but only the light that radiates backwards and forwards between the mirrors is reinforced by repeated trips through the **gain medium**: continued passage amplification soon reaches saturation and a steady state oscillation ensues, a **laser beam**.

## Construction and Operation

**Construction.** The laser gyro contains three mirrors to achieve a **rotational** path for two beams that are generated and sent around in a triangular path in opposite directions. The **lasers** are sent around small tunnels drilled parallel to the perimeter of a triangular block of temperature stable glass with reflecting mirrors placed in each corner.

Lasing is achieved by running high voltages through **helium neon** gas between the anodes and the cathode transforming many of the atoms of the gas into light in the **pinkish orange** part of the visible spectrum (this action is helped by the tuned cavity effect of the tunnel in the glass block).

**Operation.** The laser beam that is created can be described as a high energy beam of coherent light which is said to be of a **pure frequency**. The light will be reflected by the mirrors but light of **unwanted frequencies** ( i.e. not at the design frequency) will be absorbed by the mirrors and their coatings. Because the frequency of the light is known it can be measured and modified by adjustment of the path length i.e. "If the path length is decreased, the light is compressed and the frequency will increase - if the path length is expanded, the frequency decreases".

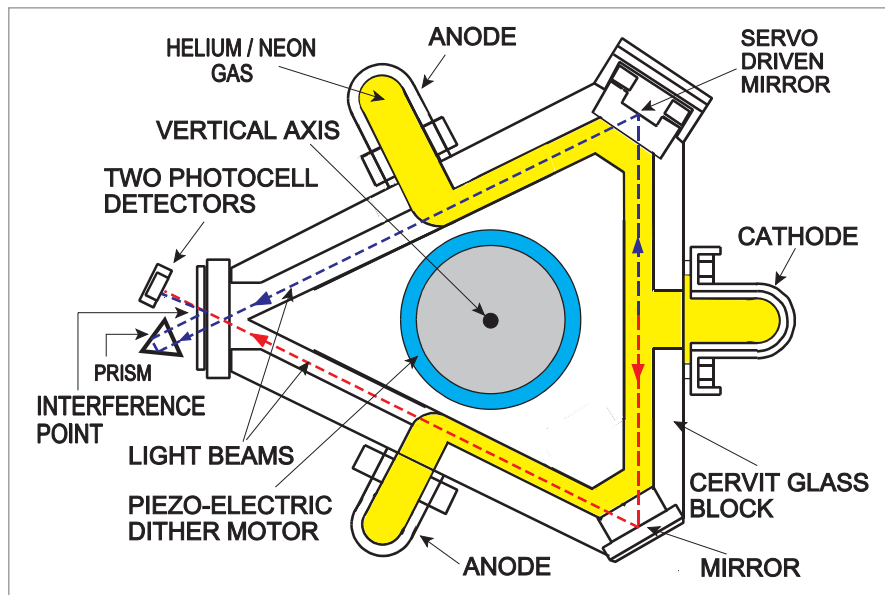


Figure 19.2

The triangular path of the device does not rotate but the two beams of light are caused to travel in opposite directions and will of course travel at the same speed - **the speed of light**.

If there is no movement of the device the beams cancel each other out but when movement is induced one of the beams will take longer to complete its path and the other, in opposition, a measurably shorter length of time to complete its journey. This whole process is measured by devices known as **gain elements** and the rate of rotation can be calculated.

A laser gyro is used in an IRS as an **Angular Rate Sensor**.

The change in frequency, caused by the change in path length due to rotation of the gyro, is known as the **SAGNAC** effect.

The three mirrors involved are not identical - one makes micro adjustments to keep the physical light path accurately aligned and another is partially transparent to allow the laser light to be detected on the photo cell detectors.

Included with the second mirror is a prism which **flips / redirects** the light beam around causing it to meet and **interfere** with the light beam that is aimed directly at the photo cell. The beams alternately cancel and reinforce each other (known as interference) **thus generating a fringe pattern**.

The photoelectric cell detects the direction and speed at which the **fringe pattern** moves. The change in the pattern, moving in one direction or other, depends upon which way the **laser gyro** is being rotated. The faster the rotation the faster the **fringe pattern** moves across the photoelectric cell - this is then converted to signals used within the aircraft systems.

### Limitations and Accuracy

**Drift.** The principle source of error with this form of device, as with the conventional **gyro stabilized platform INS** device, is associated with random drift. In a conventional gyro this is caused by imperfections of gyro bearings and mass imbalances but with the **laser** system noise is the cause and this is derived almost entirely from imperfections in the mirrors and their coatings.

**Accuracy.** The accuracy of the **laser** system is directly influenced by the length of its optical path - the longer the path available the greater the accuracy with a small percentage increase in length leading to a substantial increase in accuracy.

**Lock in.** The most significant potential problem is **lock in**, also known as **laser lock**, which occurs at very low rotation rates.

At very low rotation rates the output frequency can drop to zero as a result of **back scattering** between the two beams which causes the beams to synchronize, that is, no longer indicate the rotation correctly and indeed introduce undesirable errors. This phenomena is overcome by the introduction of a vibration device known as a **piezo electric dither motor** which breaks the lock in. The motor is mounted in such a way that it vibrates the **laser ring** about its **input axis** through the lock in region, thereby unlocking the beams and enabling the optical sensor to detect the smaller movement of the **fringe pattern**. The motions caused by the dither motor are decoupled from the output of the **ring laser gyro / rate sensor**.

## Platform / Strap Down Principles

**Platform.** The INS (**platform set in gimbals**) requires three rate integrating gyros and accelerometers to achieve an output that we can use and this process is started by ensuring that the platform is horizontal at the correct latitude. **IRS (strap down)** attaches the three laser gyro / rate sensors and accelerometers directly to the aircraft chassis.

**High speed microprocessors** then achieve a stable platform **mathematically** rather than mechanically (as per the INS) - this results in greatly improved accuracy and reliability.

**Integration.** Integration principles are used as per the older INS system.

**Gravity.** Gravity - the microprocessor subtracts the effect of local gravity from any vertical acceleration to compensate for local effects.

**Earth rotation.** Earth rotation rate - compensated for at the rate of 15.04 degrees as with a gyro (INS) system.

**Transport compensation.** Transport rate - **Schuler tuning** is again required to compensate for oscillation errors as the system is transported over the earth (this in relation to pendulum theory which results in an 84.4 minute error cycle as described in the older INS).

**Calibration.** Automatic calibration - completed automatically by computer to enhance the overall accuracy of the system.

## Platform Alignment

**True North.** The system, as with the INS, requires to find **true north** to achieve an alignment and this is achieved when the aircraft is stationary on the ground and the only rate of change is that associated with the movement of the earth. True north is then found.

**Latitude.** Initial latitude must be put into the system by the operator, the computer then, after assessing the rotational vectors that it is experiencing, compares the latitude it finds with that entered by the operator during initialization. However, it should be noted that with this system the inbuilt memory function remembers its position at landing and will indicate to the crew any errors of initial position input (lat' or long') upon startup.

**Alignment.** The computer, after confirming the latitude, completes a full mathematical levelling process - initial latitude and longitude must be entered manually as a **present position** to assist this align - **THE AIRCRAFT MUST NOT BE MOVED DURING THIS PROCESS.**

This process is called **Establishing the Trihedron.**

## Advantages

**Activation.** Almost no spin up time, one second activation for the rate sensor.

**Manoeuvring.** Insensitive to "g" attitude, rolling, pitching manoeuvres.

**Construction.** Mechanically simple and highly reliable.

**Range.** Wide dynamic range.

**Drift.** Very small drift rates - greatest errors induced by the operator.



## Question

1. **Dither is used in a laser gyro in order to:**
  - a. enhance the accuracy of the gyro at all rotational rates
  - b. increase the maximum rotational rate that can be sensed by the gyro
  - c. stabilize the laser frequencies at peak power output
  - d. break the frequency lock which would prevent small rotational rates from being sensed by the gyro

## Answer

1
d

# Chapter **20** Radio Altimeter

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## Introduction

The Radio Altimeter is a device capable of measuring the height of an aircraft above ground with a high degree of accuracy. Apart from providing a flight deck display of height above ground level (AGL), the radio altimeter has two other important functions:

- It supplies the automatic flight system with data to affect automatic landings when used in association with the ILS / MLS.
- It furnishes height information and rate of change of height to the Ground Proximity Warning System (GPWS), and is a crucial component of this system.

The instrument makes use of primary radio principles transmitting a Frequency Modulated Continuous Wave (FMCW) in an elliptical pattern vertically below the aircraft.

The radio altimeter determines the time taken for a radio wave to travel from the aircraft to the ground directly beneath the aircraft and back again. During this time the transmitted frequency changes at a known rate from its start level to +50 MHz and back again to complete a "cycle".

The carrier frequency cannot be increased indefinitely and so after half a wavelength the change is reversed, the frequency then being decreased at a constant rate down to a specified value before being increased again. The complete "modulation cycle/frequency sweep" is illustrated in *Figure 20.1*.

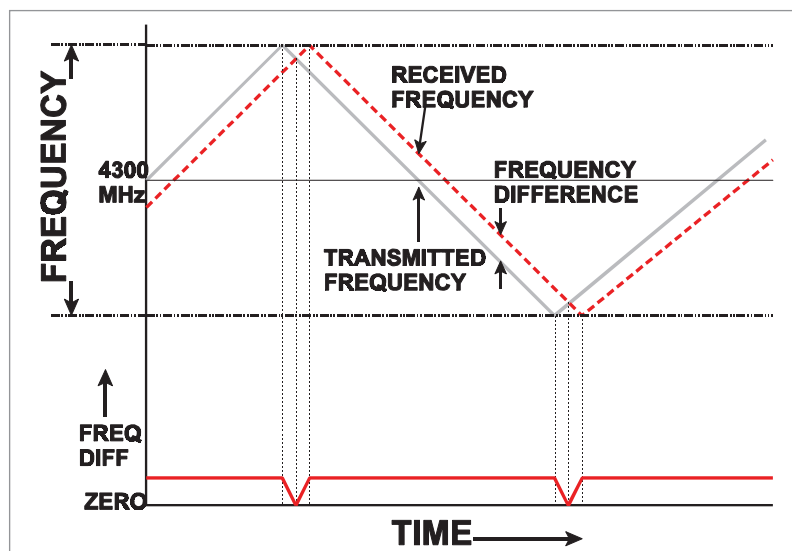


Figure 20.1

The equipment compares the frequencies of the transmitted and received signals and since the rate of frequency change is known, the frequency difference becomes a measure of the time taken for the radio wave to travel to and from the surface. From the information gained aircraft height may be determined.

The breakdown of frequency difference, which occurs when the transmitter changes the direction of its frequency sweep, is overcome by relating aircraft height to the average beat frequency (the difference between transmitted and received frequency) observed over a short sampling period. The frequency changeover points are thereby ignored.

## Frequencies

Two frequency bands have been used in the past, but only the SHF band is used at present:

**4200 MHz to 4400 MHz - SHF band**

**1600 MHz to 1660 MHz - UHF band**

The total sweep of the carrier frequency is automatically varied  $\pm 50$  MHz 300 times a second. At very low altitude with the reflection returning almost instantaneously, a wide sweep is necessary to give a measurable frequency difference. The signal is transmitted downwards from a flush mounted horn antenna. The conical / elliptical beam generated is wide enough to always allow some portion of the beam to travel vertically even with pitch angles of  $\pm 30^\circ$  and roll angles of  $\pm 60^\circ$ . The height will be determined from the shortest path to the ground which, of course, will always be vertically below the aircraft.

Transmission being continuous, it is necessary to use a separate antenna, similar to the transmitting antenna for reception. The receiver antenna needs to be positioned far enough away to avoid interference with the transmitted signal.

Radiated power generated is of the order of one watt.

## Basic Indicator

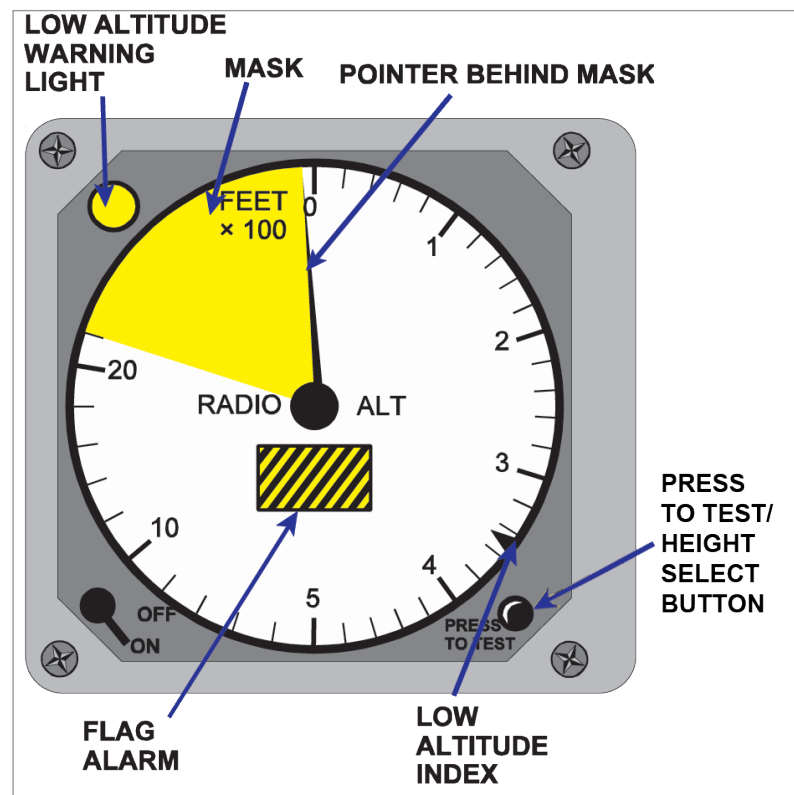


Figure 20.2

### Height Scale

The scale is logarithmic being expanded from zero to 500 feet and at a reduced non-linear scale from 500 to 2500 feet.

### Mask

The height pointer disappears behind a mask:

- when altitude exceeds 2500 feet
- when there is any fault in the transmitted signal
- when the altimeter is switched off

### Failure Warning Flag

The flag appears when there is too much radio noise which will corrupt the returning signal, or if local reflections are received from the airframe itself, or in the event of a loss of power to the equipment.

### Press to Test Button/Height Selector

When this button is pressed, the height pointer swings round to a known pre-set altitude. This provides a confidence check for the user indicating that the equipment is likely to operate satisfactorily. This button is a dual-purpose device. When turned, it selects the reference Low Altitude Index to the desired height.

### Low Height Warning

The warning light illuminates if the aircraft is flown below any pre-selected height set by the pilot. This occurrence is also audibly marked by the sudden cessation of an alert tone which will sound with increasing loudness from approximately 100 feet above the decision height setting.

## EFIS Indicator - Boeing Style

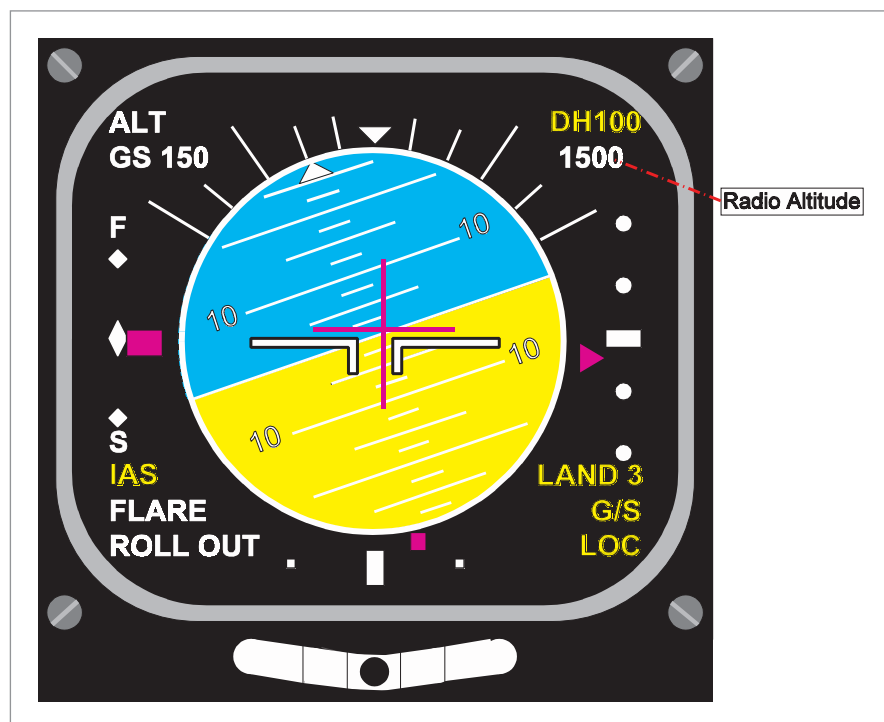


Figure 20.3

### Digital Read-out - Boeing Style

A digital read-out, and when below 1000 ft a pictorial image of an altimeter dial, is drawn on some EFIS displays. The colour of this icon (which also shrinks in size below 1000 ft indicating height change) changes from white to flashing amber as decision height is approached. One further function of the radio altimeter is to desensitize the autopilot and flight director response to the ILS glidepath in the latter stage of an approach.

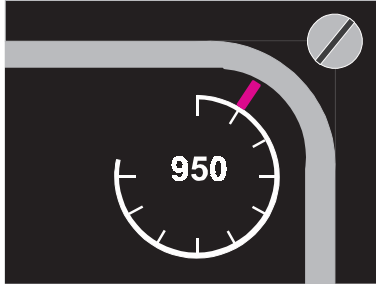


Figure 20.4

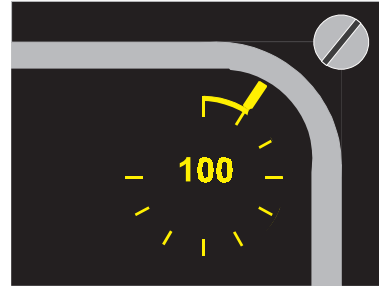
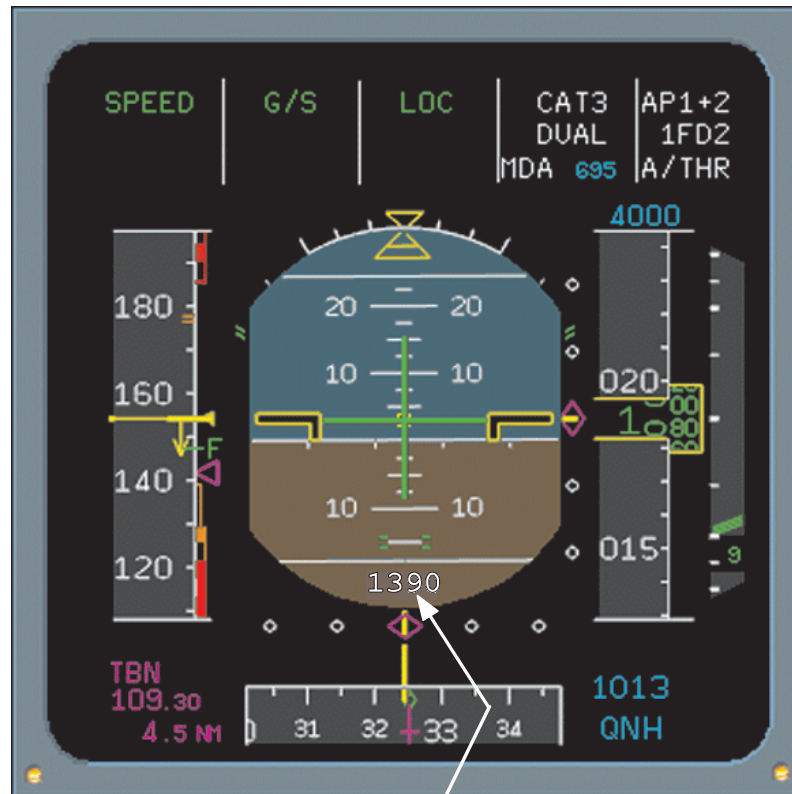


Figure 20.5

### EFIS Indicator - Airbus Style



**RADIO ALTIMETER - green above DH, amber below DH and bigger numbers**

Figure 20.6

On some more modern systems, such as that used by the A300, the indication of height is given at the base of the EADI / PFD attitude display. This height indication changes colour from Green to Amber and the numbers also grow in size as Decision Height is passed. It is also important to note that the Radio Altimeter is a major component of the Ground Proximity Warning System (GPWS).



## Range and Accuracy

The instrument can be used between zero and 2500 feet above the surface with an overall expected accuracy of  $\pm 3\%$  of indicated height or  $\pm 1\text{ft}$  whichever is the greater. The figures include various error contributors, principally Doppler shift, step error in the digital counting circuits, and height lag.

## Aircraft Installation Delay

### (Cable Length Compensation)

The radio altimeter is required to indicate zero height AGL as the main wheels touch down on the runway; because of this it has to be extremely accurate and in practice is designed to perform to an accuracy of  $\pm$  one foot. However, in practice a single manufacturer's product may be found in multiple aircraft types from the very large Boeing 747 to a much smaller corporate commuter jet and this must be catered for. At either extreme the aircraft weight and oleo compression will vary and this leads to the need for compensation to cater for this variable. The height difference between the antennas on the fuselage and the bottom of the trailing wheels on the main landing gear bogeys, on the approach to a touchdown, is known as the Residual Height. In addition, the different physical sizes of the aircraft concerned will create differences in cable run length between the avionics bay and the position of the antennas on the underside of the aircraft fuselage. When investigating the larger aircraft this distance may be as much as 100 ft or, in the smaller jet, as little as 6 ft. If compensation for cable length is not catered for in the larger aircraft an error would be generated and this would be seen as an error of height (perhaps up to 100 ft in the example above).

The aircraft installation delay is therefore adjusted to compensate for residual height and cable length (times two - Tx antennas/Rx antennas to avionics bay). This is done to ensure that at touchdown with the main bogeys trailing the radio altimeter reads zero.

Additionally, it should be noted that when on the ground, the radio altimeter may show a small negative altitude. The reason for this is that the equipment has been adjusted to indicate zero when the main wheels first touch the runway surface on landing and therefore when the aircraft is level on the ground the antenna will be below its calibrated position relative to the aircraft landing attitude. The effect is particularly noticeable with aircraft such as the B747 (which actually indicates -8 ft) which have multi-wheel assemblies which are inclined at an upward angle when deployed in flight and thereby create a larger difference between antenna position and wheels at the point of touchdown.



Figure 20.7 Undercarriage residual height



Chapter  
**21**

**Flight Management System**

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## Principle of Operation

Flight management systems are designed to improve navigation, aid fuel efficiency and to reduce crew workload. Computers are used to fly aircraft along complex routes using Lateral Guidance (**LNAV**).

Vertical Guidance (**VNAV**) enables the system to calculate optimum cruise altitudes and to determine the best combination of autothrottle control and speed during climb and descent.

At all times when the crew are not actually controlling the aircraft by hand, they use the FMS controls to “fly” the aircraft. The controls of an FMS are, in effect, a miniature flight deck with fingertip control.

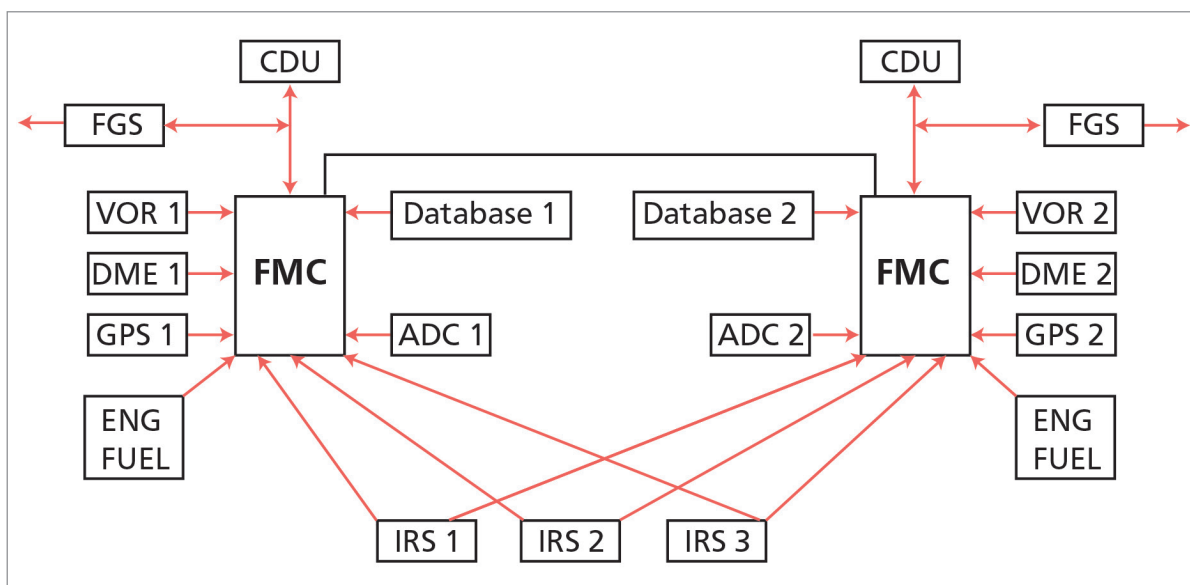


Figure 21.1 Schematic layout of a typical system

Legend:

- CDU** - Control and Display Unit
- FGS** - Flight Guidance System
- FMC** - Flight Management Computer
- ADC** - Air Data Computer
- IRS** - Inertial Reference System
- GPS** - Global Positioning System

## Control and Display Unit (CDU)

The primary function of the CDU is to act as the interface between the aircraft and the crew. The CDU can be used to command completely automatic control of the aircraft or semi-automatic with varying degrees of pilot involvement including full manual control.

Two CDUs are usually fitted either side of the centre console with the left CDU normally being the master (in the B747-400 they are joined by a third CDU placed upon the centre console for use primarily by engineering staff). They comprise a monochrome or coloured cathode ray tube (CRT) display on which different “pages” of selected data can be shown, and a selector key panel.

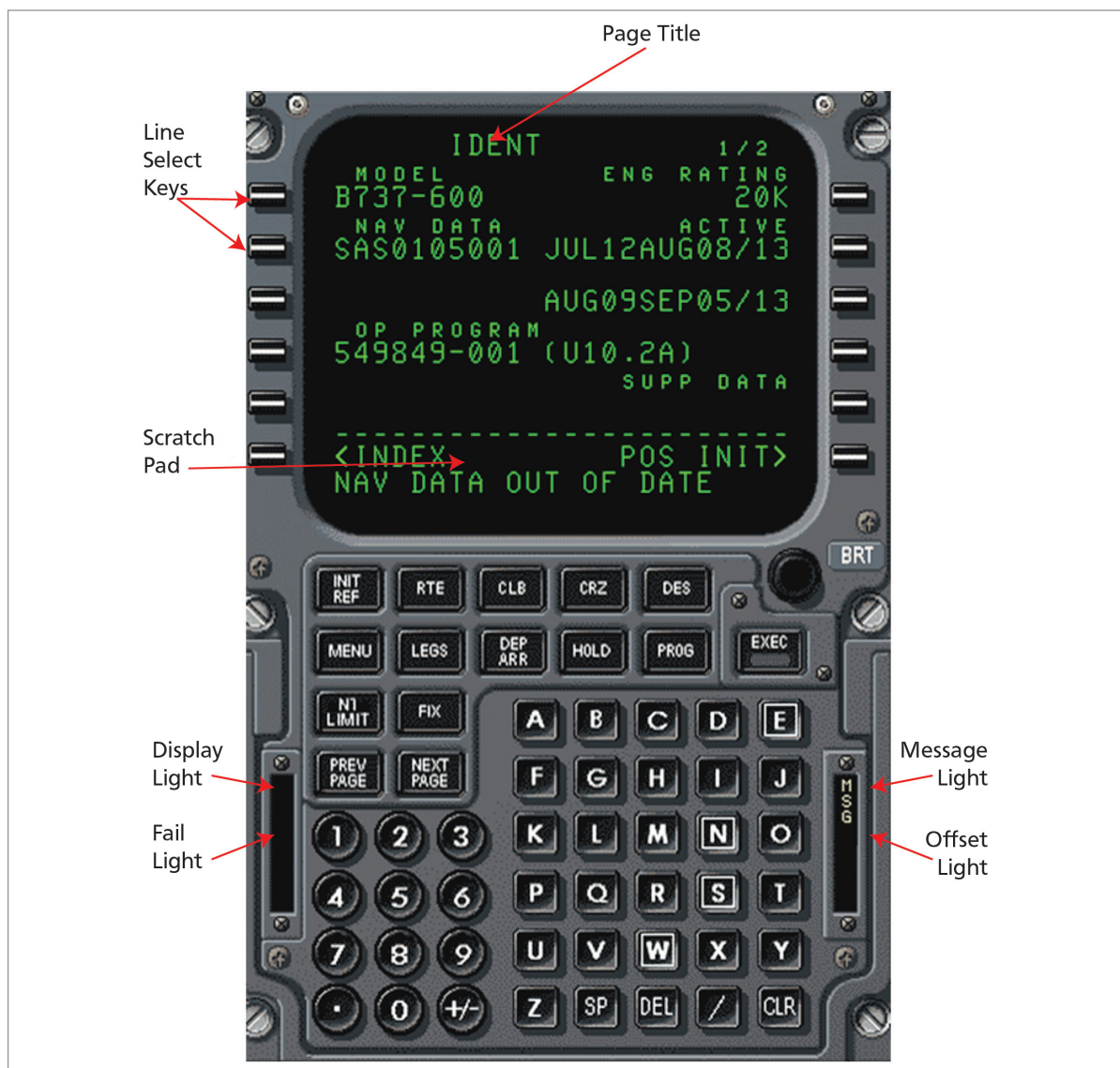


Figure 21.2

The FMCs may be decoupled to provide fully **Independent Mode** operation. This is not usual in that there will be no safety/cross check between the two FMCs.

When operating in **Dual Mode** (the norm for ordinary flight profiles) both FMCs independently process pilot entries on both MCDUs and **compare** the results to ensure that crucial information is consistent on both systems. The same output is then passed to both FMCs.

If there is a failure of an FMC, the second system can be expected to operate the aircraft successfully on its own - this is known as **Single Mode**. The failed FMC may be selected out of the system to allow single mode operation of the "surviving" FMC if required by the crew.

## Data Base

The information stored in the FMC is called its data base. The data base is divided into two major sections. One contains **performance** related information and the other contains information dealing with **navigation**.

The purpose of the performance data base is to reduce the need for the flight crew to refer to the Cruise Control Manual during flight and to provide the FMC with information required to calculate pitch and thrust commands. All reference data normally required can be displayed on the FMS-CDU. The data stored in the data base includes aircraft drag and engine characteristics, maximum and minimum speeds. Maintenance can refine the data base for each aircraft by entering factors for drag and fuel flow.

The FMC navigation data base includes most information that would normally be determined by referring to navigation charts. This information may be displayed on the FMS-CDU / AMD and eliminates most of the cockpit chart reading in aircraft without an FMC. The geographic area covered includes all areas where the aircraft is normally flown. The stored information includes the location of navigation aids, airports, runways and other airline selected information such as SIDs, STARs, approaches and company routes.

**The FMC contains two sets of navigation data, each valid for 28 days.** Each set corresponds to the normal revision cycle for navigation charts. During preflight the flight crew can select which set is active. The FMC uses the active set for navigation calculations. The contents of the navigation data base are **updated by maintenance** every 28 days. When the navigation chart revision date arrives, the new data is already in the FMC and ready for selection.

## Operational Procedures - Initial Actions

**IDENT Page:** Upon application of power to the aircraft the FMS immediately executes a self-test procedure and upon its successful self completion the IDENT page of the system is displayed. The IDENT page of the FMC allows the crew to confirm aircraft details on start-up and this in turn initiates a series of prompts to direct the crew through the route procedures that need to be generated for their flight. Importantly on this page we have confirmation of the navigation data base in operation and an indication of the date of changeover to the next programme - if the data base is out of date it will tell us to change to the in date programme.

**POS INIT Page:** When we have checked the IDENT page we will be directed to the POS INIT page where we must check the FMS clock time against the aircraft clock to ensure synchronization between the systems: data is saved on the FDR against time and of course ETAs are generated by the FMC and need to be in line with times indicated by the aircraft clock. As we complete this task we must also ensure that the airfield datum or gate position, if available, is entered accurately to allow for IRS alignment (this position will be suitable for alignment but is often updated at the take-off point to obtain the best possible initial position for use in flight).

**RTE Page:** After completing our tasks upon the POS INIT page we will be directed to the RTE page where we will enter our starting and destination airport ICAO designators. We may then expect to enter our flight number details and identify a "standard" company route to take us to our destination: if a standard route is not available, other actions will have to be taken to input the information into the system.

**PERF INIT Page:** We may now move on to the PERF INIT page to update the aircraft to its current performance / weight configuration for the route to be flown. On this page we may input details of fuel weight, fuel reserves required, cruise altitude and even, in the case of the B747, if we are carrying a fifth engine to our destination upon its suspension point on the wing. At this stage we may also enter Cost Index requirements related to our flight as discussed later in the chapter.

## Summary

The following is a summary of the initial pages that you may expect to see on the Boeing series of aircraft currently in use but of course this information may vary from company to company as they modify the system for their own use:

### IDENT page -

- Aeroplane Model / Configuration
- Operational Programme Identifier
- Drag / Fuel Flow Factors
- Navigation Data Base Identifier / Cycle

### POS INIT page -

- IRS Position Reference
- IRS Heading Reference
- GMT / UTC / Time Zone Display

### RTE page -

- Origin Airport
- Destination Airport
- Flight Number
- Route Activation

### PERF INIT page -

- Gross Weight
- Fuel Quantity
- Zero Fuel Weight
- Fuel Reserves
- Cost Index
- Cruise Altitude
- Spare (fifth) Engine Carriage (B747 specific)
- Altitude Step Size

## Operational Procedures - Climb Vertical Navigation (VNAV)

Entering a cost index of "ZERO" provides economy speeds representing a maximum range cruise. The **VNAV** profile that the FMC commands is a climb with climb thrust to remain within all airspeed and altitude constraints that are part of the SID entered into the active route, then climb at economy speed to the entered cruise altitude.

If when flying the climb speed profile it would cause a violation of an altitude constraint, the UNABLE NEXT ALT message is displayed. The pilot must then select a different speed on the FMS-CDU that provides a steeper climb angle. Reaching cruise altitude, the FMC commands cruise at economy speed until the top of descent point.

A number of cost index modifications are allowed until passing Top of Descent point (TOD) for example Long Range Cruise (LRC) and "selected speed" cruise may also be entered.



**Notes:** *Time Related Cost* is a function of speed selected; the higher the speed in climb, cruise or descent the higher the "trip fuel cost" but the lower the "trip time cost".

**Economy Cruise Mode** will yield the lowest operating cost based on the cost index.

**Cost Index** is determined by dividing aeroplane operating cost (\$ / £ per hour) by fuel cost (\$ / £ per pound or litre). A cost index of zero results in "minimum trip fuel operation" and so will include cruise at "maximum range" cruise and a low speed descent.

## Operational Procedures - Cruise Lateral Navigation (LNAV)

LNAV guidance outputs from the FMC are normally great circle tracks between the waypoints making up the active route. However, when a procedure stored in the FMS data base is entered into the active route the FMC can supply commands to fly a constant heading, track or follow a DME arc, as required to comply with the procedure.

FMC determines present position by using inputs from the IRS / INS, DME, VOR and other navigation systems fitted. It uses its calculated present position to generate lateral steering commands along the active leg to the active waypoint. To function, the FMC requires position information from at least one IRS / INS. While the aircraft is on the ground, the FMC calculates present position based only on information received from the IRS / INSs.

The FMC present position is normally the combination of all IRS / INS positions and since inertial systems accumulate position errors as a function of time, the position information being used by the FMC is slowly accumulating errors. These position errors can be detected by observing the various positions of the individual IRS / INSs on the CDU. If an extended ground delay occurs and a significant map error is noticed, the IRS / INS should be realigned and present position re-entered.

## Operational Procedures - Descent

When a programmed "arrival" is entered, the FMC calculates a descent path based on the procedure's airspeed and altitude constraints and the End of Descent (E/D). The E/D is a waypoint with an altitude and airspeed constraint that coincides with a final approach fix or runway threshold.

For VFR and non-precision approaches, the FMC computed path is built to a point that is 50 feet over the approach end of the runway. It is the flight crew's responsibility to not descend below "DH" until adequate visual contact has been achieved. During a missed approach, LNAV guidance is available to the missed approach point and altitude.

## Operational Procedures - Accuracy

Radial error rates of less than 0.05 NM/hour are not uncommon. Introduction of Ground Positioning by Satellite (GPS) as a navigation input will improve overall performance. It must be stressed, however, that the skill of the operator and the need for constant and careful monitoring will always be a deciding factor.

## Operational Procedures - Control and Display Unit

**CDU Key Groups.** The keys on the lighted switch panel of the CDU perform various functions and may be broken down into three major groups:

- Alphanumeric
- Function and Mode
- Line Select Keys (LSK)

The function of each of the keys is briefly described below:

**Alphanumeric Keys** - Pressing any alphanumeric key results in that character appearing in the scratch pad.

**Function and Mode Keys** - Used for initializing the system, access to flight planning functions and status, and modifying the flight plan. Select climb, cruise or descent information for preplanning or modification.

<b>CLB</b>	CLB (climb) - displays current or alternate climb mode for assessment and selection. Cruise altitude is enterable, as is a speed / altitude restriction.
<b>CRZ</b>	CRZ (cruise) - displays current or alternate cruise mode for assessment and selection. Information about optimum altitude, step-climb savings, and turbulence penetration N1 targets is also available.
<b>DES</b>	DES (descent) - displays current or alternate descent mode for assessment and selection. Target speed is enterable, as is a speed / altitude restriction. Flight Path Angle (FPA), Vertical Speed (V/S), and Vertical Bearing (V/B). Information is provided for crew reference
<b>INIT REF</b>	INIT/REF (initialization / reference) - allows access to data pages required for start-up of the FMCS and IRS. Also, the operator may select various reference data and maintenance pages.
<b>N1 LIMIT</b>	N1 Limit - permits manual command of the active N1 limit, and selection of any Reduced Climb N1 Limit that may apply. (Allows the crew to select an engine, "LP Turbine", rpm.)
<b>MENU</b>	The N1 Limit key may be shown as a menu key on the master (left hand) CDU and can be used to find data within the system.
<b>DEP ARR</b>	DEP/ARR (departures / arrivals) - used for selection of the procedures and runways at the origin and destination airports.
<b>RTE</b>	RTE (route) - permits flight plan data entries. A primary means for lateral flight plan alteration.
<b>LEGS</b>	LEGS (route legs) - displays and accepts entries of detailed data concerning each leg of the flight plan, for both the lateral and vertical paths
<b>HOLD</b>	HOLD - permits planning or initiation of holding at a designated waypoint.

DIR INTC	DIR/INTC (direct / intercept) - provides data capability to proceed direct to any point desired, or to intercept any leg presently in the flight plan.
FIX	FIX (fix information) - displays range and bearing data from the present position to an entered fix. Facilitates creation of fixes for use in flight planning.
PROG	PROG (flight progress) - displays current flight status information such as ETA, fuel remaining at waypoint, navigation radio tuning status, wind, and path errors.
----- -----	Line Select Keys (LSK) - entry of data from the scratch pad into the selected line and field is accomplished by using the LSKs. There are twelve LSKs on the CDU panel, six each to the left and right of the CRT display. Data entries are permitted only on lines adjacent to the LSKs. Data can also be duplicated into the blank scratch pad by pressing the LSK adjacent to the desired data line.
EXEC	EXEC (execute) - used to incorporate data displayed on the CDU as part of the active flight plan. The EXEC key is operable when its annunciator bar is illuminated. The key is used for activating the flight plan, changing the active flight plan, changing the active guidance mode, or inserting data which will affect the active flight plan, guidance mode, or data base. Illumination of the white annunciator bar indicates that a valid set of data is on display and may be made active for guidance of the aircraft.
+ / -	Change Sign Key - changes data in the scratch pad from positive to negative and back again. May also be used to insert a hyphen for specialized data entries.
NEXT PAGE	Page Select Keys - when multiple-page displays are shown, pressing the NEXT PAGE key advances the display to the next higher page number.
PREV PAGE	Pressing PREV PAGE backs up the display to the next lower page number. Page access wraps around.
CLR	Clear Key - the scratch pad's contents can be edited or cleared by pressing the CLR key. When an entry is present in the scratch pad, a brief depression of the CLR key will clear the last character in the entry. If the CLR key is held down for more than one second, the entire entry in the scratch pad will be cleared. The CLR key is also used to clear advisory and alerting messages from the scratch pad.
DEL	Delete Key - the delete (DEL) key is used to remove data from a display (and thus a flight plan) after it has been line selected and accepted into a data field. If the scratch pad is empty, depression of the DEL key writes ("DELETE") into the scratch pad. The delete process is then completed by line-selecting (LSK) the data item to be removed. If the deletion is a valid one, the data field reverts to its default value (box prompts, dashes, or a system-generated value). The system prevents invalid use of the DEL key.

## *Illuminated Annunciators*

There are four annunciators on the front of the CDU as shown in the illustration:

<b>MSG</b>	Illuminates white in colour; indicates an alerting or advisory message or pending messages.
<b>FAIL</b>	Illuminates amber in colour; lit if FMC failure is detected.
<b>DISPLAY</b>	Illuminates white in colour if the page displayed is not related to the active flight plan leg or to the current operational performance mode.
<b>OFFSET</b>	Illuminates white in colour when a parallel offset is in use (i.e. the aeroplane is flying parallel to, but a fixed distance from, the FMS preprogrammed track).

# Chapter 22

## Electronic Flight Information Systems

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## The Electronic Flight Instrument System (EFIS)

The Electronic Flight Instrument System presents attitude and navigation information to the pilot on two electronic display units in a format that is easier to read and less likely to be misinterpreted than some older mechanical instruments.

As far as the pure basic functions and number of display units are concerned, this system, which is generally referred to as 'EFIS' is fully integrated with digital computer-based navigation systems, and utilizes colour Cathode Ray Tube (CRT) or Liquid Crystal Display (LCD) types of **Attitude Director Indicator (ADI)** and **Horizontal Situation Indicator (HSI)**.

The system is therefore extremely advanced, not only in terms of physical construction, but also in the extent to which it can present attitude and navigational data to the flight crew of an aircraft.

## The Units of a System

As in the case of a multi-crew conventional flight director system, a complete EFIS installation is made up of left (Captain), and right (First Officer), systems.

Each system in turn is comprised of:

- Electronic Attitude Director Indicator (EADI) or Primary Flight Display (PFD)
- Electronic Horizontal Situation Indicator (EHSI) or Navigation Display (ND)
- Control Panel
- Symbol Generator (SG)
- Remote Light Sensor Unit

A third (centre) symbol generator is also incorporated so that its drive signals may be switched to either the left or right display units in the event of failure of their corresponding symbol generators.

The signal switching is accomplished within the left and right symbol generators, using electromechanical relays powered from an aircraft's DC power supply, via pilot-controlled switches.

The interface between EFIS units, data busses and other systems is shown in [Figure 22.1](#).

## Symbol Generators (SGs)

Symbol generators provide the analogue, discrete, and digital signal interfaces between an aircraft's systems, the display units and the control panel, and they also perform symbol generation monitoring, power control and the main control functions of the 'EFIS' overall.

The interfacing between the card modules of an SG is shown in [Figure 22.1](#).

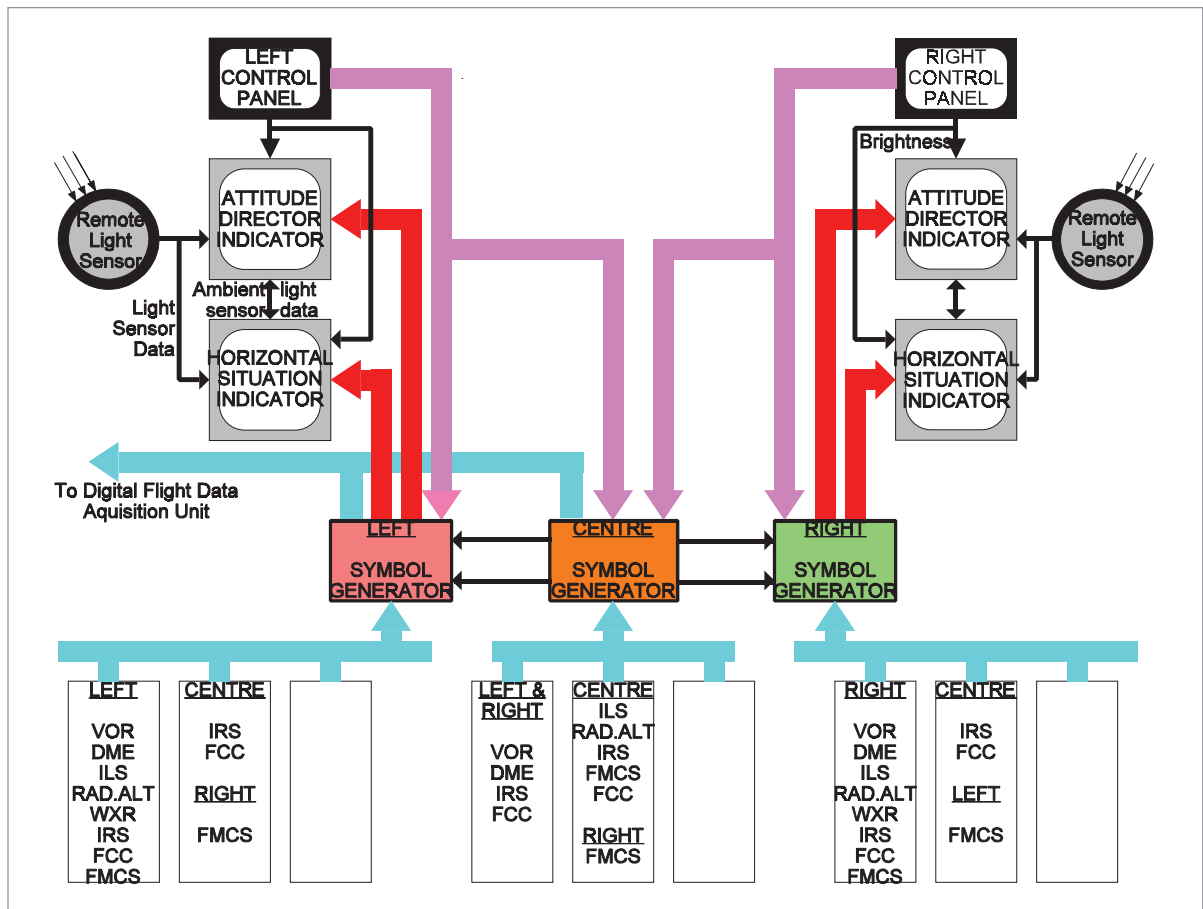


Figure 22.1 Multi-crew EFIS units and signal interfacing

### Display Units

The display units may be Cathode Ray Tubes (CRTs) or Liquid Crystal Displays (LCDs). LCDs have the advantage of being smaller, and they generate less heat therefore need less cooling. The PFD and ND are usually identical units to facilitate spares commonality and are often interchangeable with the systems display units (EICAS or ECAM).

### The Colour Display System

There is no set colour standard and so colour displays may vary slightly.

In a typical display system, 5 colours are usually assigned for the display of the many symbols, failure annunciators, messages and other alphanumeric information, with a sixth colour (RED) for weather (WXR):

- WHITE** Display of present situation information.
- GREEN** Display of present situation information where contrast with white symbols is required, or for data having lower priority than white symbols. Engaged autoflight modes.
- MAGENTA** All 'fly to' information such as flight director commands, deviation pointers, active flight path lines.



- CYAN** Sky shading on an EADI and for low-priority information such as non-active flight plan map data.
- YELLOW** Ground shading on an EADI, caution information display such as failure warning flags, limit and alert annunciators and fault messages.
- RED** For display of heaviest precipitation levels as detected by the weather radar (WXR).

## The Remote Light Sensor

The Remote Light Sensor is a photodiode device which responds to ambient light conditions on the flight deck, and automatically adjusts the brightness of the CRT displays to an acceptable level.

## The Control Panel

A control panel is provided for each system, and typically, as shown in *Figure 22.2*, the switches are grouped for the purpose of controlling the displays of their respective EADI and EHSI units.



*Figure 22.2 An EFIS control panel*

## The 'EADI' Section of the Control Panel

Brightness Control ( <b>BRT</b> ).	Used to adjust the brightness of the <b>ADI</b> display to the desired level.
Decision Height Selector ( <b>DH SEL</b> ).	Used to select desired decision height for <b>DH</b> alerting.
Decision Height Reset Switch ( <b>DH RST</b> ).	When pressed it resets a <b>DH</b> alert on the associated <b>ADI</b> . It changes the <b>RA</b> display from yellow to white.
Decision Height Reference Indicator ( <b>DH REF</b> ).	This displays the selected decision height on the controller, and on the <b>EADI</b> .

## The 'EADI' Display Presentation

The 'EADI' (*Figure 22.3*) displays normal pitch and roll attitude indications plus

- Attitude data from an Inertial Reference System (IRS)
- Flight director commands
- Localizer and glide slope deviation
- Ground speed
- Radio altitude
- Decision height
- Automatic Flight Control System (AFCS) and Autothrottle modes
- Speed error scale (Difference between commanded and actual)

**Note:** *The autoland status, pitch, roll-armed and engage modes are selected on the AFCS control panel.*

## Decision Height (DH)

Decision height is the wheel height above the runway elevation by which a go-around must be initiated unless adequate visual reference has been established and the aircraft position and approach path have been assessed as satisfactory to continue the approach and landing in safety.

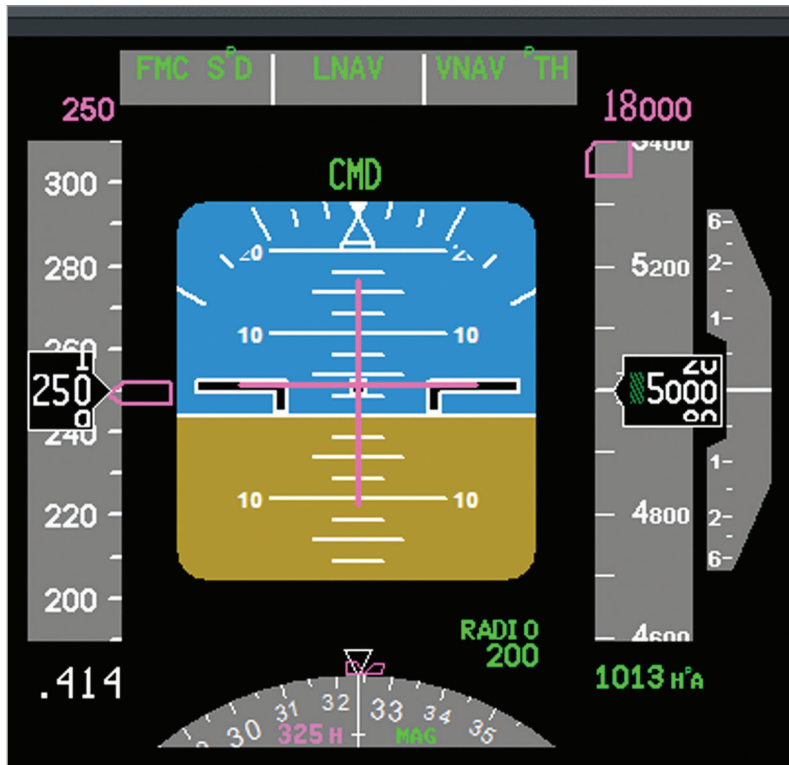
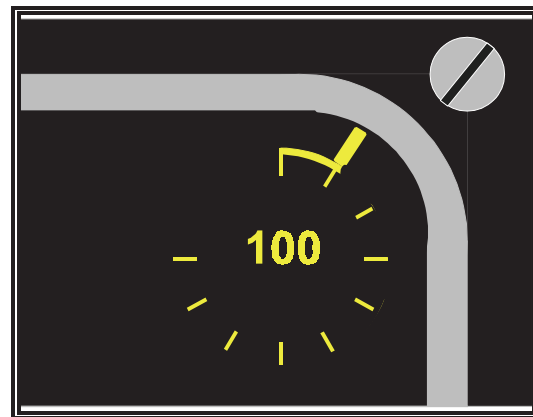
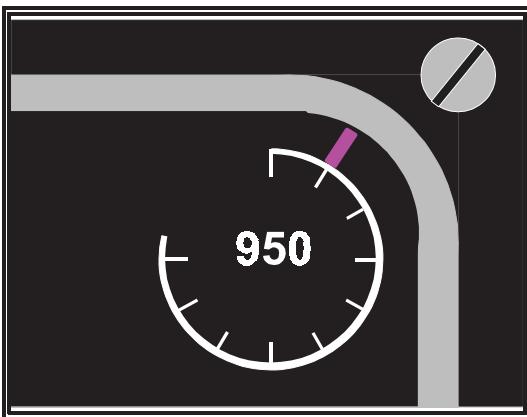


Figure 22.3 A Boeing EADI

Decision height (DH) is selected on the ADI control panel and shown on both the ADI and on the control panel.



Decision height and radio altimeter presentation below 1000 ft

At Decision Height

Figure 22.4

**Radio altitude** on Boeing-based systems is digitally displayed during when the aircraft is between 2500 ft and 1000 ft above ground level.

Below 1000 ft the display automatically changes to a white circular scale calibrated in increments of 100 ft, and the selected decision height is then displayed as a magenta-coloured marker on the outer scale. The radio altitude also appears within the scale as a digital read-out. As the aircraft descends, segments of the altitude scale are simultaneously erased so that the scale continuously diminishes in length in an anticlockwise direction.

On the descent, at decision height plus about 50 ft, an **aural alert chime** sounds at an increasing rate until decision height is reached.

On reaching **DH** the marker and scale flash and change from magenta to amber or yellow. Pressing the **EADI** control panel **DH RST** button will cancel the alert chime, stop the marker flashing and change the **DH** displays back to their normal colour.

Deviation beyond normal localizer and glide slope limits cause the scales to change colour to amber and the pointer to flash, which ceases if the aircraft returns to within limits.








**Note:** The **A300** system currently in use differs from the above by using digital read-out only. This system, which is also found on some other aircraft types, displays Rad Alt at the base of the PFD centre display using **green** numbers above decision height and **amber**, slightly bigger numbers below (see [Figure 22.3](#)). The digital read-out is enhanced by a voice warning system which will give aural indications of height passing to the pilot. The aural warnings will be given at significant heights as decided by the manufacturer working with the airline company at time of system build.

The **Speed Error Display** consists of a pointer which moves relative to a deviation scale to show difference between actual speed and commanded speed.

### The 'EHSI' Section of the Control Panel

<b>RANGE</b>	Selects range for displayed navigation data and <b>WXR</b> .
<b>MODE SELECTOR</b>	Selects display appropriate to mode required: <b>VOR, ILS, MAP, PLAN</b> .
<b>BRT (BRIGHTNESS)</b>	Outer knob sets main display brightness. Inner knob sets <b>WXR</b> display brightness.
<b>MAP switches</b>	Used in <b>MAP</b> mode. When selected, they cause their placarded data to be displayed on the <b>EHSI</b> :- <b>NAV AID</b> (NAVIGATION AID), <b>ARPT</b> (AIRPORT), <b>RTE DATA</b> (ROUTE DATA), <b>WPT</b> (WAYPOINTS).
<b>EXP ARC</b>	Selects <b>VOR</b> or <b>ILS</b> mode to show an <b>Expanded Arc</b>
<b>WXR</b>	When pushed in, <b>WXR</b> data is displayed during all modes except <b>PLAN</b> , or when the <b>VOR / ILS</b> mode is selected to present the <b>FULL</b> compass rose.

System Symbols

	
<p>Active Waypoint (Magenta) the waypoint the aircraft is currently navigating to.</p>	<p>Inactive Waypoint (White) a navigation point making up the selected active route.</p>
	
<p>Airports (Cyan).</p>	<p>Off Route Waypoint (Cyan)</p>
	
<p>Tuned Nav aids (Green).</p>	<p>Unused Nav aid (Cyan)</p>
	<p>Wind Direction (White) with respect to map display orientation and compass reference. Symbol shown in white against the black of the screen.</p>

## The 'EHSI' Display Presentation

The EHSI presents a selectable, dynamic colour display of flight progress and a plan view orientation. Four principal display modes may be selected on the EFIS control panel;-

- VOR
- ILS
- MAP
- PLAN

Of these VOR and ILS can be displayed as full or expanded compass displays.



Figure 22.5

The orientation of the displays may be **Magnetic or True, Track or Heading** as selected.

## Full VOR Mode

With a VOR frequency selected, the EHSI displays a full compass rose with the VOR source in the lower left and the frequency in the lower right.

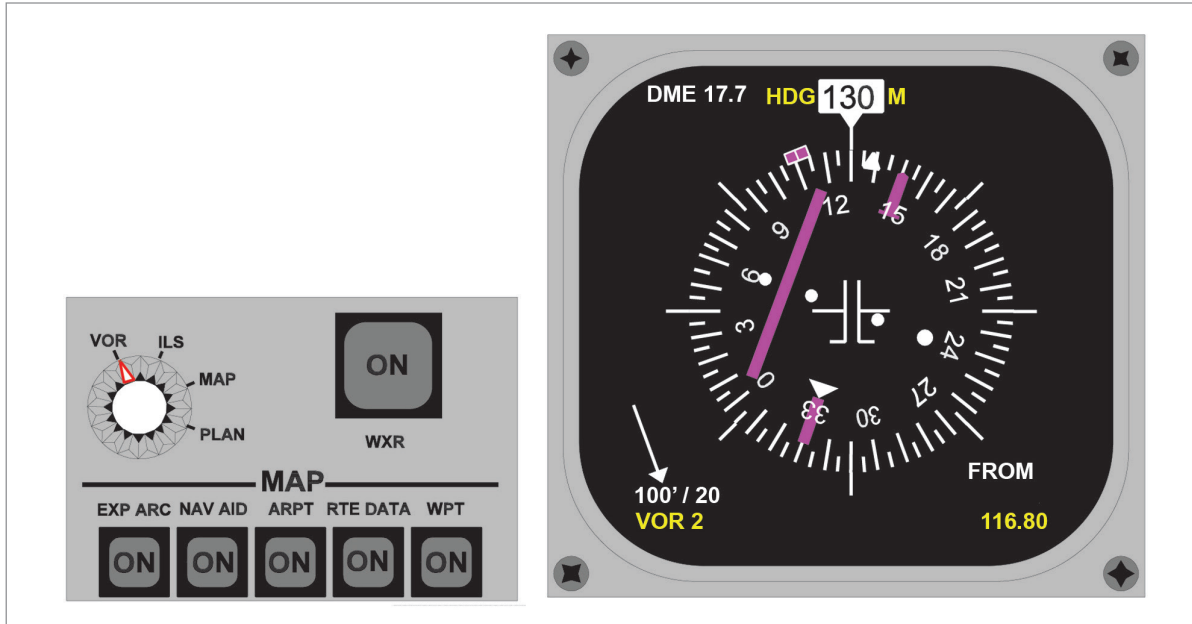


Figure 22.6

Course selection is displayed by the magenta course needle, the tip pointing to the selected course (150). Course deviation is shown by the traditional deviation bar moving across a two dot left and two dot right scale.

A **TO / FROM** pointer is shown in addition to the **TO /FROM** annunciation.

DME distance displayed in the top left corner.

Current heading is shown in the window and by the lubber line at the top of the compass rose (130), the current selection is Magnetic Heading as shown either side of the window.

Current track is shown by the white triangle on the inside edge of the compass rose.

Selected heading is shown by the magenta heading "bug" on the outer scale of the compass rose.

Wind speed and direction are shown in the lower left corner orientated to the display selection (Heading or Track, Magnetic or True).

Weather radar displays are not available.

## Expanded VOR

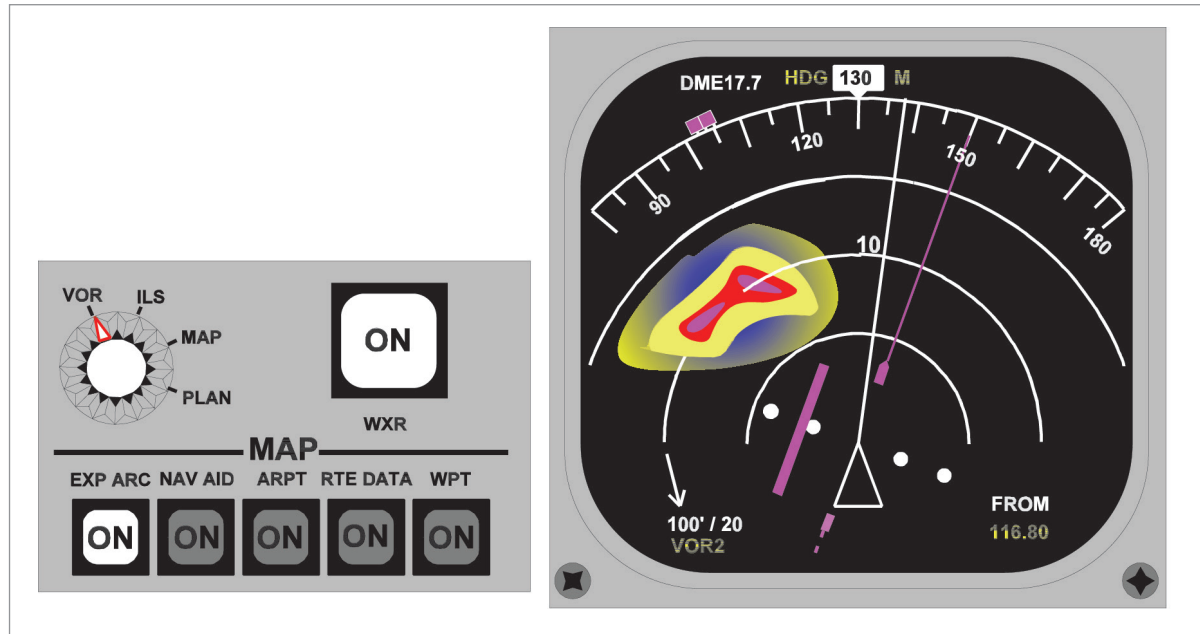


Figure 22.7

With a VOR frequency selected, the EHSI displays about 90° of compass rose with the VOR source in the lower left and the frequency in the lower right.

The white triangle at the bottom of the display is the aircraft symbol.

Selected course (track) is displayed by the magenta course needle, the tip pointing to the selected course (150). The course selectors are usually on either side of the autoflight main control panel (one for the Captain and one for the First Officer). Course deviation is shown by the traditional deviation bar moving across a two dot left and two dot right scale.

A TO /FROM annunciation is shown.

DME distance is displayed in the top left corner.

Current heading is shown in the window and by the lubber line at the top of the compass rose (130), the current selection is Magnetic Heading as shown either side of the track window. Current track is shown by the white line from the tip of the aircraft symbol to the compass arc. Selected heading is shown by the magenta heading "bug" on the outer scale of the compass rose. Wind speed and direction are shown in the lower left corner orientated to the display selection (Heading or Track, Magnetic or True).

Weather radar displays are available; when selected "on", range arcs are also visible. Weather radar shows three colours: green, yellow and red, green being the least turbulence, red being the worst. If turbulence mode is available, it is shown as magenta, the area of greatest activity in the cloud. The range of the display can be selected on the control panel, half scale range is displayed (10 NM) so this display is selected to 20 NM. The outer arc of the compass rose is the furthest range from the aircraft.



## Full ILS Mode

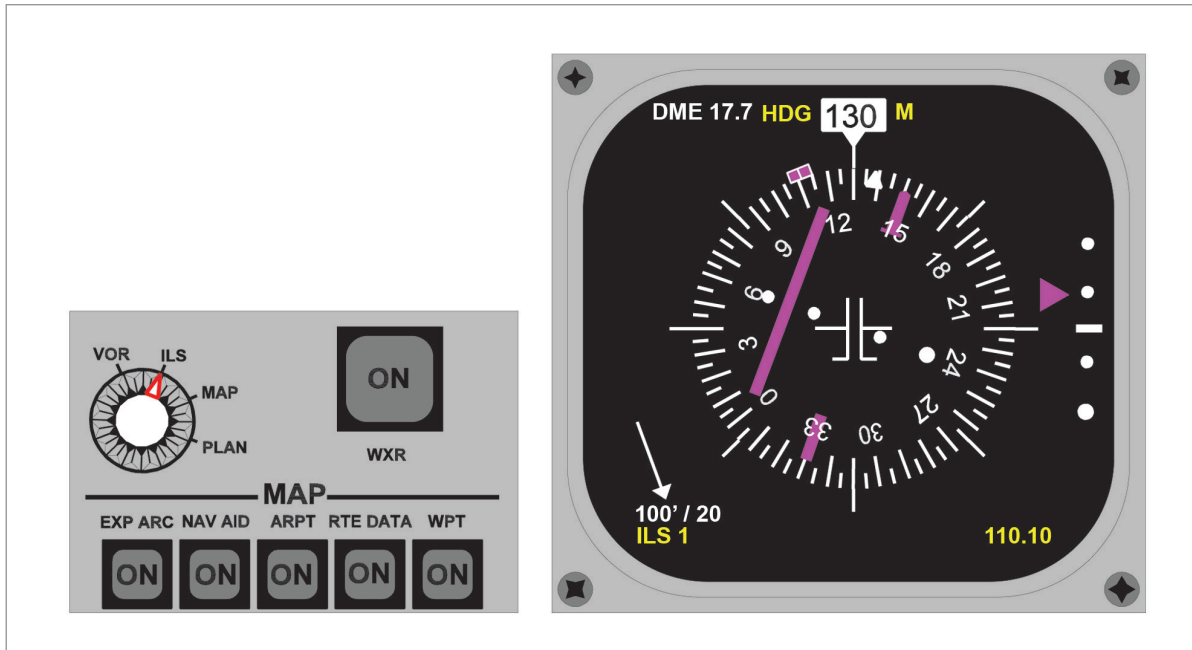


Figure 22.8

With an ILS frequency selected, the EHSI displays a full compass rose with the ILS source in the lower left and the frequency in the lower right.

Course selection (localizer) is displayed by the magenta course needle, the tip pointing to the selected course (150). Localizer deviation is shown by the traditional deviation bar moving across a two dot left and two dot right scale. This scale is exponential.

Glide slope deviation is shown by a magenta coloured triangle moving up and down the traditional scale on the right hand side.

DME distance is displayed in the top left corner.

Current heading is shown in the window and by the lubber line at the top of the compass rose (130), the current selection is Magnetic Heading as shown either side of the window.

Current track is shown by the white triangle on the inside edge of the compass rose.

Selected heading is shown by the magenta heading "bug" on the outer scale of the compass rose.

Wind speed and direction are shown in the lower left corner orientated to the display selection (Heading or Track, Magnetic or True).

Weather radar displays are not available.

## Expanded ILS

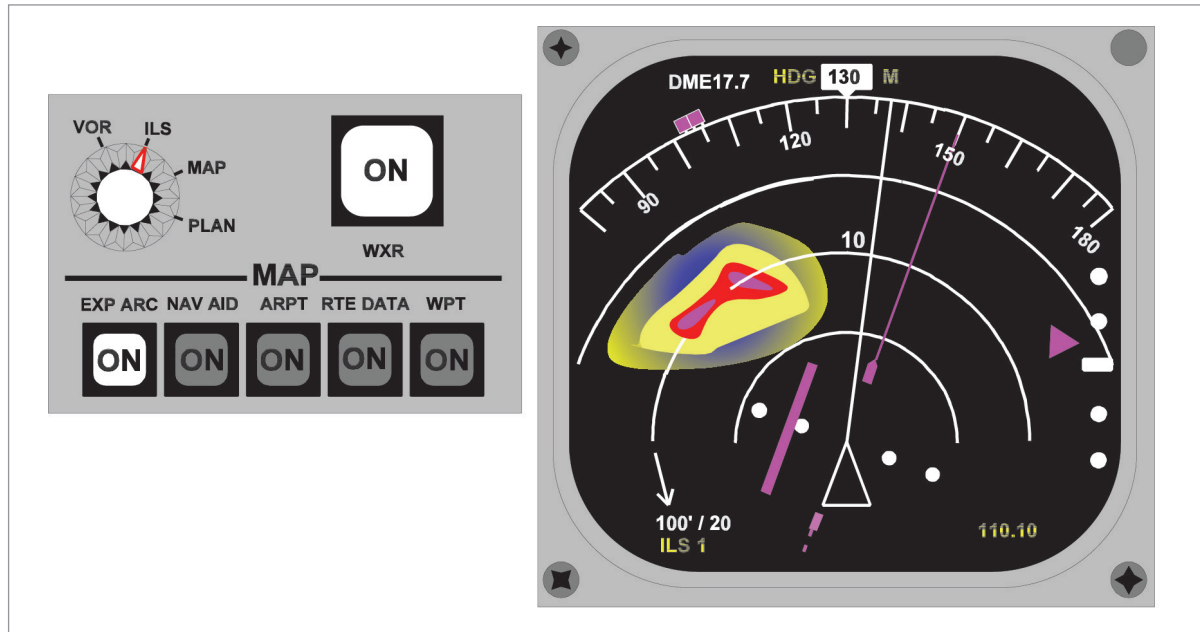


Figure 22.9

With an ILS frequency selected, the EHSI displays about 90° of compass rose with the ILS source in the lower left and the frequency in the lower right.

The white triangle at the bottom of the display is the aircraft symbol.

Selected course (track) is displayed by the magenta course needle, the tip pointing to the selected course (150). The course selectors are usually on either side of the autoflight main control panel (one for the Captain and one for the First Officer). Localizer deviation is shown by the traditional deviation bar moving across a two dot left and two dot right scale. Glide slope deviation shown on the right again in the traditional fashion.

DME distance is displayed in the top left corner.

Current heading is shown in the window and by the lubber line at the top of the compass rose. In this case the heading is 130° Magnetic, as indicated by markings either side of the window. Current track is shown by the white line from the tip of the aircraft symbol to the inside edge of the compass rose.

Selected heading is shown by the magenta heading "bug" on the outer scale of the compass rose. Wind speed and direction are shown in the lower left corner orientated to the display selection (Heading or Track, Magnetic or True).

Weather radar displays are available, when selected "on", range arcs are also visible. Weather radar is shown in three colours: green, yellow and red, green being the least turbulence, red being the worst. If **TURBULENCE MODE** is available, it is shown as magenta, the area of greatest activity in the cloud.

The range of the display can be selected on the control panel; half scale range is displayed (10 NM) so this display is selected to 20 NM. The outer arc of the compass rose is the furthest range from the aircraft.

## Map Mode

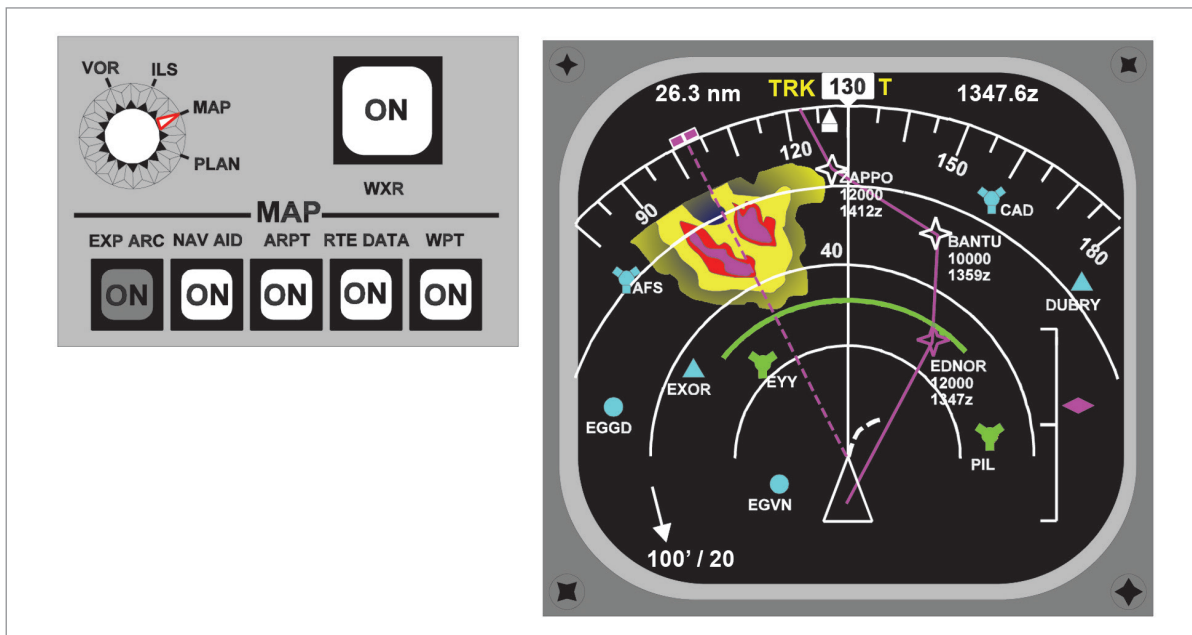


Figure 22.10

The mode used normally is the **MAP** display, which, in conjunction with the flight plan data programmed into a flight management computer, displays information against a moving map background with all elements to a common scale.

The symbol representing the aircraft is at the lower part of the display, and an arc of the compass scale, or rose, covering 45 degrees on either side of the instantaneous track, is at the upper part of the display.

Heading information is supplied by the appropriate inertial reference system and the compass rose is automatically referenced to magnetic north (via a crew-operated **MAG/TRUE** selector switch) when between latitudes 73°N and 65°S, and to true north when above these latitudes. When the selector switch is set at **TRUE** the compass rose is referenced to true north regardless of latitude.

The aircraft active route as derived from the FMC is shown as a magenta coloured line joining the waypoints. The **active waypoint** (the one the aircraft is currently navigating towards) is shown as a magenta coloured star. The other waypoints making up the active route are called **inactive waypoints** and are shown as a white star. Both are identified by name.

Distance to next waypoint and time at next waypoint are shown at the top of the display. Weather radar (WXR) return data and range arcs are displayed when the WXR switch is on. Turbulence mode (+T) may be available as previously described.

Indications of other data such as wind speed and direction, lateral and vertical deviations from the selected flight path are also displayed.

The flight management computer (FMC) can predict events by combining current ground speed and lateral acceleration to show a display of either a curved trend vector, white, (during turns) or a range to altitude arc, green, (during climb or descent). Off route waypoints, airports, nav aids can all be shown in their relative position to the aircraft's progress and selected range. Additional waypoint information can be displayed when selected: altitude, time etc.

## Plan Mode

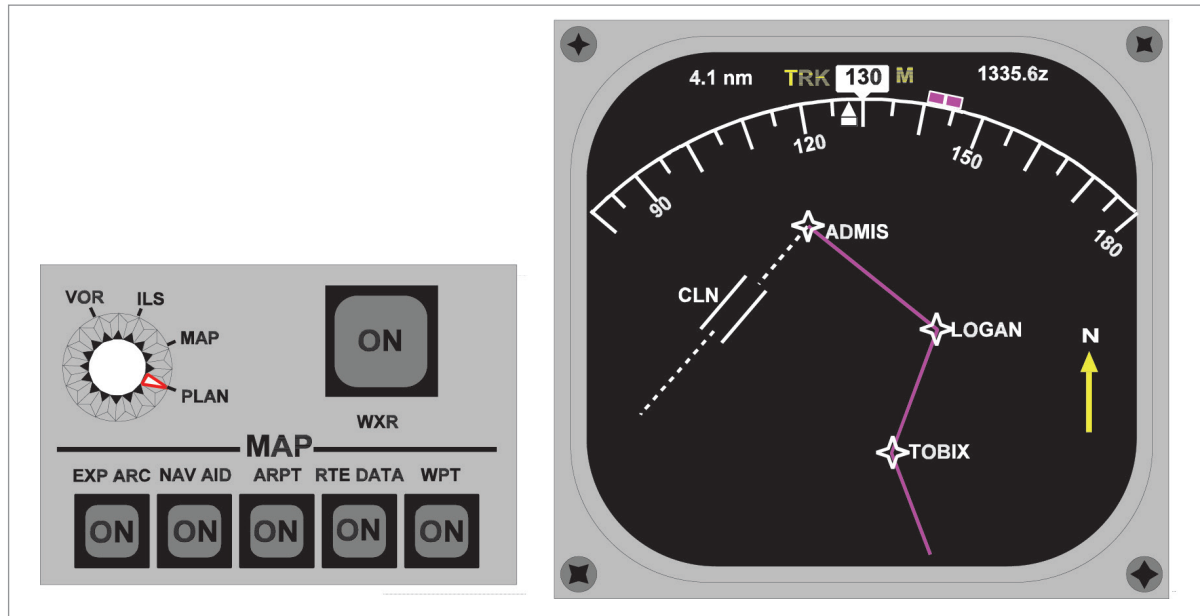


Figure 22.11

In 'PLAN' mode a **static** map background is used with active route data orientated to **true north**. Any changes to the route may be selected at the keyboard of the flight management computer, and the display shown on the **EHSI**, so they can be checked before they are entered into the FMC.

The top portion of the EHSI remains the same as in the map mode.

This mode allows the pilot to review the planned route by using the FMC / CDU LEGS page.

Weather radar display data is inhibited.

No wind speed or direction information is shown

Students should be aware that the arc of compass shown is "real time". The planning section below this is north orientated. The heading from TOBIX to LOGAN is approximately 020°, NOT 155°.

## Data Source Selection

In the type of system described earlier, means are provided whereby the pilots can independently of each other, connect their respective display units to alternate sources of input data through a data source switch panel, e.g. the left or right Air Data Computer (ADC), the Flight Management Computers (FMC), the Flight Control Computers (FCC), and the Standby Inertial Reference Systems (IRS).

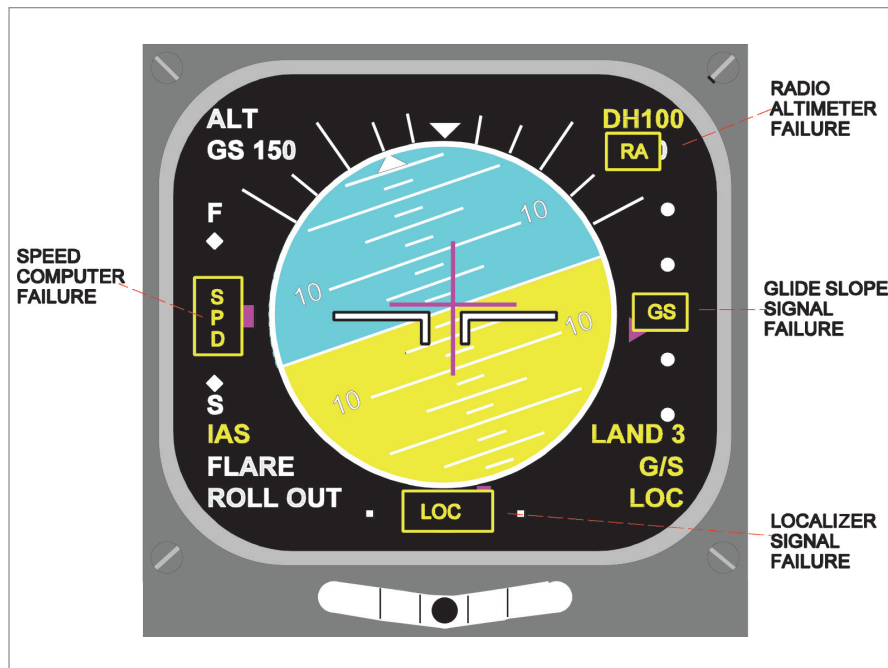


Figure 22.12 Failure Indications

### Failure Annunciation


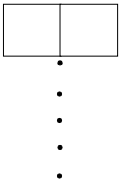

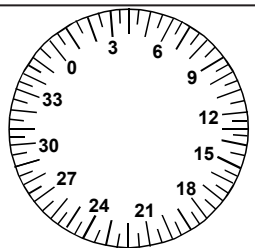
The failure of data signals from such systems as the ILS and radio altimeter is displayed on each EADI and EHSI in the form of yellow flags ‘painted’ at specific matrix locations on their CRT screens.

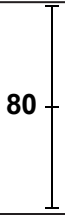

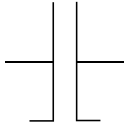
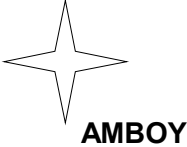


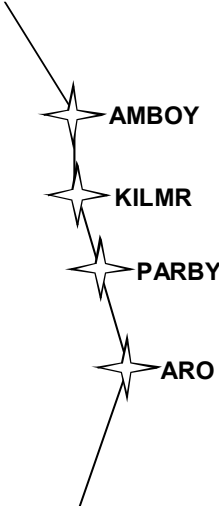
In addition, fault messages may also be displayed, for example, if the associated flight management computer and weather radar range disagree with the control panel range data, the discrepancy message ‘WXR/MAP RANGE DISAGREE’ appears on the EHSI.

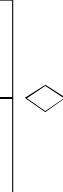



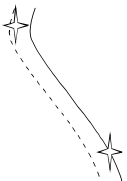


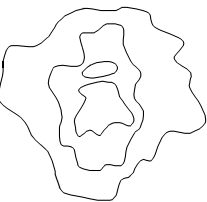
### Navigation Displays

The following symbols can be displayed on each HSI depending on EFI Control Panel switch selection. Symbols can be displayed with different colours but the general colour presentation is as follows:

- GREEN (G)** engaged flight mode displays, dynamic conditions
- White (W)** present status situation, scales, armed flight mode displays
- MAGENTA (M)(pink)** command information, pointers, symbols, fly-to condition
- CYAN (C)(blue)** non-active and background information
- RED (R)** warning
- AMBER (A)** cautionary information, faults, flags
- BLACK (B)** blank areas, display “off”

Symbol	Name	Applicable Modes	Remarks
<p>200 NM / 4.4 NM</p> <p>or</p> <p>DME 124 / DME 24.6</p>	Distance Display(W)	PLAN, MAP or VOR, ILS	Distance is displayed to next FMC Waypoint (NM) or tuned Navaid (DME).
<p><b>HDG</b>  <b>M</b></p>	HEADING Orientation(G) Indicator(W) Reference(G)	PLAN, MAP VOR, ILS	Indicates number under pointer is a heading - box indicates actual heading. Referenced to magnetic north between 60° South and 73° North if selected and true north when above those latitudes.
0835.4z	ETA Display (M) & (W)	PLAN, MAP	Indicates ETA at active waypoint based on current ground speed.
	Selected Heading Marker(M)	PLAN, MAP VOR, ILS	Indicates the heading set in the MCP. A dotted line(M) extends from the marker to the aeroplane symbol for ease in tracking the marker when it is out of view (except plan mode).
	Expanded Compass Rose(W)	PLAN, MAP VOR, ILS	Compass data is provided by the selected IRS (360° available but approximately 70° is displayed)
	Full Compass Rose(W)	Full VOR, Full ILS	Compass data is provided by the selected IRS.

	Present Track Line and Range Scale	MAP VOR, ILS	Predicts ground track which will result with present heading and winds. Displayed range mark is one-half the actual selected range.
	Aeroplane Symbol(W)	MAP VOR, ILS	Represents the aeroplane and indicates its position at the apex of the triangle.
	Aeroplane Symbol(W)	Full VOR, Full ILS	Represents the aeroplane and indicates its position at the centre of the symbol.
	Waypoint Active(M) Inactive(W)	MAP, PLAN	Active - Represents the waypoint the aircraft is currently navigating to. Inactive - Represents a navigation point making up the selected active route.
	Altitude Range Arc(G)	MAP	When intersected with the track line, it predicts the point where the reference altitude will be reached.
	Trend Vector	MAP	Predicts aeroplane directional trend at the end of 30, 60 and 90 second intervals. Based on bank angle and ground speed. Three segments are displayed when selected range is greater than 30 NM, two on the 20 NM and one segment when on the 10 NM scale.
	Active Route(M) Active Route Mods(W) Inactive Route(C)	MAP, PLAN	The active route is displayed with continuous lines(M) between waypoints. Active route modifications are displayed with short dashes(W) between waypoints. When a change is activated in the FMC, the short dashes are replaced by a continuous line. Inactive routes are displayed with long dashes(C) between waypoints.

	Vertical Pointer(M) and Deviation Scale(W)	MAP	Displays vertical deviation from selected vertical profile (pointer) in MAP mode during descent only. Scale indicates +/- 400 ft deviation.
	Glide Slope Pointer(M) and Deviation Scale(W)	ILS	Displays glide slope position and deviation in ILS mode.
	Drift Angle Pointer(W)	Full VOR, Full ILS	Displays difference between FMC track angle and IRS heading.
	Wind Speed and Direction(W)	MAP, VOR, ILS	Indicates wind speed in knots and wind direction with respect to the map display orientation and compass reference.
	Offset Path and Identifier(M)	MAP, PLAN	Presents a dot-dash line parallel to and offset from the active route after selection on the FMC CDU.
	North Pointer(G)	PLAN	Indicates map background is orientated and referenced to true north.
	Altitude 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), "T/D" (top of descent) and "S/C" (step climb).
	Weather Radar Returns Mapping Radar Returns (both G,A,R)	MAP, VOR, ILS	Multicoloured returns are presented when either "WXR ON" switch is pushed. Most intense regions are displayed in red, lesser amber, lowest intensity green.



Example Displays

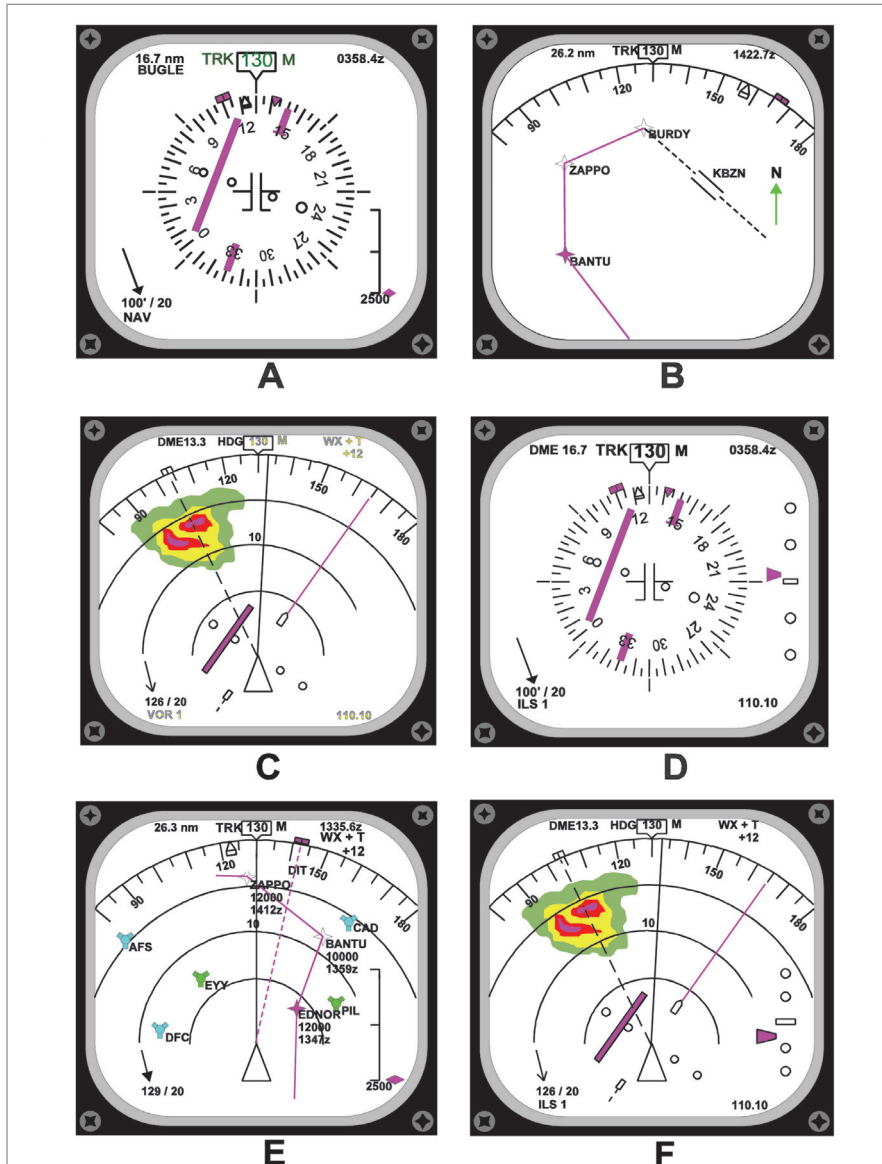
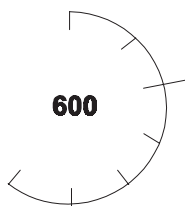


Figure 22.13

## Questions

Refer to Annex A, showing various EHSI displays and answer the following questions:

1. The displays marked A, B, C, and D are respectively:
  - a. Plan                      Map                      ILS                      VOR
  - b. VOR                      ILS                      Expanded ILS                      Plan
  - c. Map                      VOR                      ILS                      Plan
  - d. Map                      ILS                      Expanded VOR                      Plan
  
2. Refer to the display marked E and identify the correct statement:
  - a. The aircraft is closing the localizer from the right, heading 130°M and is approaching the glide path from above
  - b. When established on the localizer the inbound heading will be 165°M
  - c. The aircraft's track is 165°M
  - d. The localizer centre line is 133°M
  
3. On display D the track from ZAPPO to BANTU is:
  - a. 310°M
  - b. 130°T
  - c. 360°M
  - d. 180°T
  
4. On display C the centre of the weather returns is:
  - a. 106° relative, 18 NM
  - b. 332° relative, 13 NM
  - c. 100°M, 130 NM
  - d. 30 NM left of track, 15 NM ahead
  
5. The drawing below is shown on the (i)..... is displaying (ii)..... and (iii).....



- |    | (i)                    | (ii)                        | (iii)                         |
|----|------------------------|-----------------------------|-------------------------------|
| a. | Primary Flight Display | 600 kt TAS                  | 200 ft RA                     |
| b. | Navigation Display     | 600 ft RA                   | 200 ft DH                     |
| c. | EADI                   | 600 ft                      | 200 ft                        |
| d. | EHSI                   | Radio Altitude<br>600 kt GS | Decision Height<br>200 ft AGL |

6. The following yellow symbol appears in place of the normal radio altitude display when:



- a. the selected radio altitude has been reached
- b. the radio altitude needs re-setting on the EHSI
- c. there is a failure of the radio altimeter
- d. the aircraft descends below 1000 ft AGL.

7. The following symbols A, C, and E are best described respectively as:



A



B



C



D



E

- a. off route waypoint, airport, navigation aid
- b. next waypoint, navigation aid, airport
- c. off route waypoint, navigation aid, a navigation point making up selected route
- d. active waypoint aircraft currently navigating to, navigation aid, off route waypoint

8. When using EHSI, weather radar may be displayed on the following settings:

- a. map, VOR/ILS
- b. VOR/ILS, map, expanded plan
- c. expanded map, VOR/ILS, plan
- d. map, expanded VOR/ILS

9. WXR display is controlled from:

- a. captain's EHSI control only
- b. co-pilot's EHSI control only
- c. a special control panel
- d. both captain's and co-pilot's EHSI control panels

10. Decision height is adjusted and set on the:

- a. flight management computer
- b. HSI section of the EFIS control panel
- c. ADI section of the EFIS control panel
- d. ADI or HSI

11. Weather may be displayed, with modern EFIS fitted, on:

- a. the captain's CRT only
- b. the co-pilot's CRT only
- c. a special screen
- d. on both the captain's and co-pilot's CRTs

12. **Airspeed is shown:**

- a. only on the captain's EHSI
- b. on both EADIs
- c. on both EHSIs
- d. only on the flight management CRT


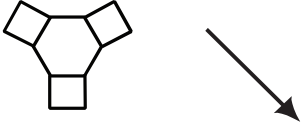
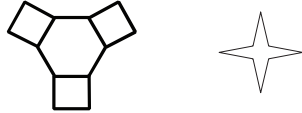
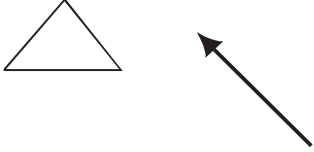
13. **With an EFIS flight director using IRS guidance, reference north can be:**

- a. magnetic north only.
- b. magnetic north between 73°N and 65°S and true north above these latitudes
- c. magnetic north between 65°N and 73°S and true north above these latitudes
- d. magnetic north between 75°N and 75°S and true north above these latitudes

14. **Modes available for (EFIS) HSI on some units are:**

- a. airspeed and Mach
- b. MAP and PLAN
- c. VOR, ILS, MAP and AUTO TRIM
- d. only from manometric sources

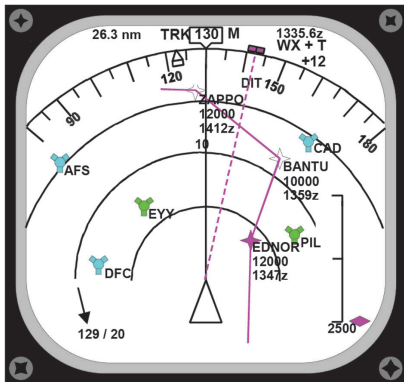
15. **The EFIS symbols for a navaid and en route waypoint are:**

- a. 
- b. 
- c. 
- d. 

16. **An EFIS as well as having a control panel, symbol generators and a remote light sensor also has:**

- a. EADIS and EHSIs
- b. EHSIs and altitude indicator
- c. EADIs and EICAs
- d. EADI and WXR display tubes

Annex A



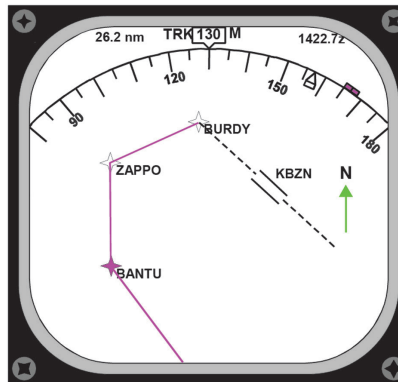
A



B



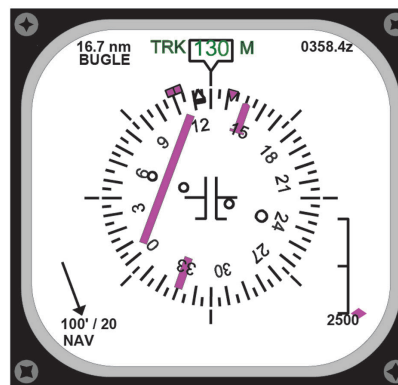
C



D



E



F

### Answers

1	2	3	4	5	6	7	8	9	10	11	12
d	a	d	b	c	c	d	d	d	c	d	b
13	14	15	16								
b	b	c	a								

Chapter  
**23**  
Basic Computers

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## Computers

A computer may be defined as: **A device or set of devices that can store data and a program that operates on the data. A general purpose computer can be programmed to solve any reasonable problem expressed in logical and arithmetical terms.**

The first fully operational general purpose computer, electromechanical and using binary digits, was the Z3, built in Germany in 1941 by Konrad Zuse.

Basically there are two types of computer:

- Analogue
- Digital

By far the most common is the digital computer or microprocessor which now plays a part in most aspects of everyday life.

## Analogue Computers

An analogue computer uses continuous physical variables such as voltage or pressure to represent and manipulate the measurements it handles.

Analogue computers are used as electronic models or **analogues** of mechanical or other systems in cases where conducting experiments on the system itself would be costly, time consuming or dangerous. For example when designing a bridge or aircraft wing or any structure where motion can occur, the engineer must know beforehand how it will react to various physical variables such as wind speed and temperature.

In recent years analogue computers have become less popular because it is now possible to program digital computers to simulate moving physical systems.

The remainder of this chapter will deal with digital computers and their use in aircraft.

## Digital Computers

Digital computers use digital data (binary data) in their operations. This form of data has only two levels of voltage as opposed to the analogue system's continuous variables. The two levels correspond to **ON** or **OFF** i.e. switching circuits. Digital circuits are **two state** circuits. Normally when working on paper we count from zero to nine - the decimal number system. When the digital computer works it has to use the **ON - OFF**, two state or **BINARY** number system.

## Digital Computer Components (Hardware)

Having discussed the language in which computers work, and remembering that **binary** is the basic language in which calculations are carried out and information is stored in memory, we shall now look at the construction of a basic computer.

All computers have the basic components shown in the diagram below:

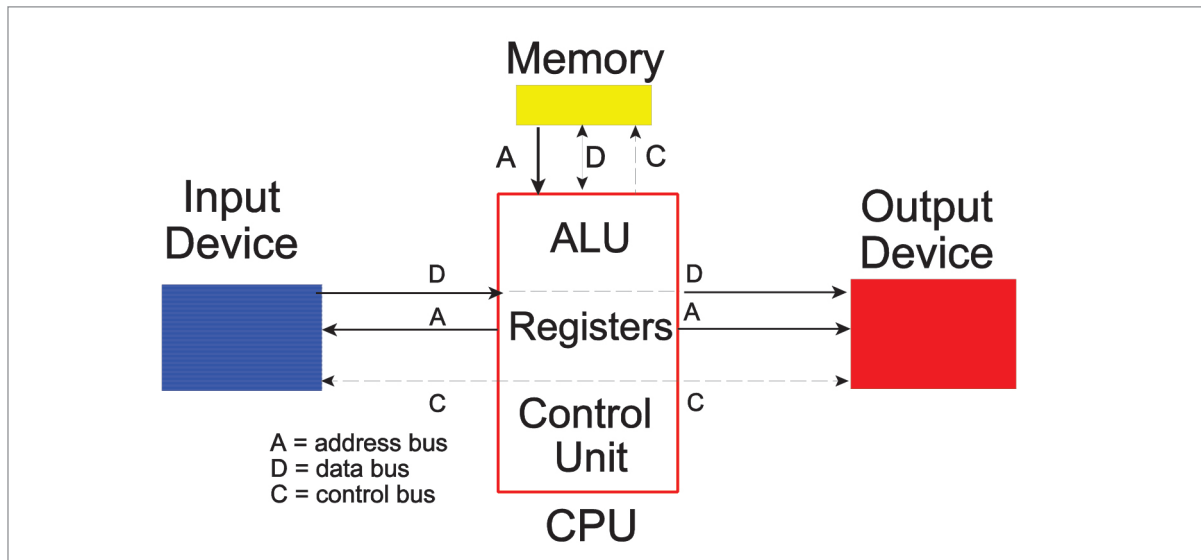


Figure 23.1 A basic digital computer Central Processing Unit (CPU)

The CPU performs, organizes and controls all the operations the computer can carry out. It is really the **brain** of the computer. What the CPU can do is controlled by an **instruction set**.

The CPU itself consists of:

- **Arithmetic Logic Unit:** The ALU performs arithmetic calculations and logical operations in the binary number system.
- **Shift Registers:** The shift registers are temporary **stores**; one of them, called the **accumulator** contains the data actually being processed.
- **Control Unit:** The control unit contains the computer's **clock**. This is a crystal controlled oscillator which generates timing pulses at a fixed frequency, typically between 120 and 45 MHz. This synchronizes computer operations.

## Input Output Devices

The CPU accepts digital signals from the input devices: keyboard, mouse or modem, in the case of a PC, via its input port. In an aircraft these could be various sensors: Rad Alt, Baro Alt, TAS, fuel flow, etc.

After processing these are fed out via its output port to a Visual Display Unit (Monitor) or printer. In an aircraft the output may be fed to an EFIS Symbol Generator or the FMS Control and Display Unit (CDU).

**BIOS (Basic Input Output System)** converts the input signals to a form the computer can work with and converts the outputs to a form the operator or another aircraft system can understand.

## Memory

**Working Memory:** A computer needs a working memory to run the programme of instructions (**software**) it has to execute. If the programme is fixed as in a computer controlled piece of equipment, the memory only has to be **read**. To do this a **Read Only Memory (ROM)** is used. This ROM is programmed by the manufacturer. A **Programmable Read Only Memory (PROM)**, **Erasable Programmable Read Only Memory (EPROM)**, or **Electrically Erasable Programmable Read Only Memory (EEPROM)** would be used if the user wanted to construct or modify the programme himself and keep it permanently in the memory.

Memory that retains data when the power is switched off is called **NON-VOLATILE MEMORY**. Memory that loses data in the event of a power failure or switch off is called **VOLATILE MEMORY**.

If the programme has to be changed during operation, then the memory must be able to be **written to** as well as **read**. To do this **Random Access Memory (RAM)** is used. This allows instructions to be written in, read out and altered at will. A RAM is also required to store the data for processing as this also will change continually. **RAM is normally volatile memory**.

**Permanent Memory:** As stated above RAM is volatile and work is lost when power is removed. Permanent storage for computer programs, and the work they generate, may be stored in 3 different ways.

- **Electronic chip.** Solid state memory, often known as "Flash Memory". Very fast access and re-writable. They may be re-written indefinitely.
- **Optical Disk.** Compact discs (CDs) read with a laser. These may be either read only or re-writable, but there will only be a very limited number of times they can be re-written.
- **Magnetic Disk.** These may be internal, external or removable. They may be re-written indefinitely. The internal "hard drive" on a computer is traditionally a magnetic disk. Removable "floppy disks" are now virtually obsolete.

## Software

### Assembly Language

A low-level programming language for computers, and other programmable devices. It is a symbolic representation of the machine codes, and other constants, needed to program a given CPU.

The code is usually defined by the hardware manufacturer.

### High Level Language

An advanced computer programming language that is not limited by the computer, or for one specific job, and is more easily understood by the author than an assembly language. Examples of high level languages are:

- FORTRAN
- ALGOL
- BASIC
- C++

### Scripting Language

A programming language that allows control of one or more applications. "Scripts" are distinct from the core code of the application, and apply to a specific program only. They are usually written in a different language (similar to Basic) and are often created or at least modified by the end-user.

### Specification for Software Used in Aircraft

To enable a flight to be conducted safely it is vital that any computer software must be safe to use and free of any errors that could endanger the flight. To this end a specification has been agreed to which all aircraft related software must comply. This is known as **EUROCAE ED12B**. There are 5 levels of software defined. The level required will depend upon the application. For example the galley oven controller could accept a major failure without endangering the aircraft, whereas any failure within the Flight Control Computer could be dangerous to the flight.

<i>Safety Criticality (EUROCAE-ED12B)</i>		
Software Level	Failure Condition	Failure Condition Interpretation in the Aviation Context
<b>A</b>	<b>Catastrophic</b>	<b>Prevent continued safe flight or landing</b>
<b>B</b>	<b>Hazardous/Severe-Major</b>	<b>Potential fatal injuries to a small number of occupants</b>
<b>C</b>	<b>Major</b>	<b>Impairs crew efficiency, discomfort or possible injury to occupants</b>
<b>D</b>	<b>Minor</b>	<b>Reduced aircraft safety margins but well within crew capabilities</b>
<b>E</b>	<b>No Effect</b>	<b>Does not affect the safety of the aircraft at all</b>

## Aircraft Systems

Systems which are computer controlled include:

- Flight Management System (FMS)
- Digital Flight Guidance System (DFGS)
- Ground Proximity Warning System (GPWS)
- Traffic Alert Collision Avoidance System (TCAS)

Of course fly-by-wire aircraft take computer control very much further, when the whole flight envelope is controlled by computer process with inputs from the crew when necessary.

Current design favours the use of dedicated computers for each separate system. In the future, however, we may see sharing of computer power in the form of an **Integrated Hazard Warning System (IHWS)**. Here a powerful central processor, with appropriate back-up, handles inputs from the stall warning system, windshear detection, GPWS, TCAS and even the Weather Radar, processes the information and prioritizes warnings to the crew.

### Analogue to Digital Conversion (A to D)

Many aircraft sensors produce analogue information in the form of varying voltages, pressures, temperatures, etc. Of course digital computers use digital (binary) information and a device called an **Analogue to Digital Converter** is required in the interface between the sensor and the computer input device.

### Digital to Analogue Conversion (D to A)

When a digital computer has to pass information to an analogue device the process is reversed and a **Digital to Analogue Converter** is used.

**Questions**

1. A basic digital computer consists of:
  1. input peripherals
  2. central processing unit
  3. inertial unit
  4. memory
  5. auto brightness control
  6. output peripherals
  - a. 1, 2, 3, 4 and 6
  - b. 1, 2, 4 and 6
  - c. 1, 4, 6 only
  - d. 2, 3, 4 and 6
  
2. The Central Processing Unit (CPU) consists of:
  1. input device
  2. output device
  3. Arithmetic Logic Unit (ALU)
  4. shift registers
  5. control unit
  6. hard disk
  - a. 1, 2, 3 and 5
  - b. 3, 4, and 6
  - c. 1, 2, 5, and 6
  - d. 3, 4 and 5
  
3. In computer terminology an input peripheral device would be:
  - a. a hard disk
  - b. a floppy disk
  - c. a keyboard
  - d. a screen display unit
  
4. In computer terminology an output peripheral device would be:
  - a. a floppy disk
  - b. a hard disk
  - c. a screen display unit
  - d. a keyboard
  
5. In computer terminology a memory which loses its data when power is removed is called:
  - a. non-volatile
  - b. non-permanent
  - c. non-retentive
  - d. volatile

6. In computer terminology a memory which retains its data when power is removed is called:
- non-volatile
  - volatile
  - RAM
  - ROM
7. Examples of input peripheral devices are:
- mouse
  - modem
  - printer
  - screen display unit
  - keyboard
- 2, 3, 4 and 5
  - 1, 2 and 5
  - 1 and 5
  - 1, 2, 3
8. In computer terminology "software" refers to:
- the memory system floppy disks, hard disks, etc
  - the RAM and ROM capacity
  - the programme of instructions
  - the BIOS
9. In computer terminology "hardware" refers to:
- the digital computer components, keyboard, monitor, CPU, etc
  - the permanent memory system and its capacity
  - the RAM capacity
  - the programme of instructions
10. The permanent memory of a digital computer usually takes the form of:
- integrated circuits rated in megabytes
  - shift registers whose capacity is rated in mega or gigabytes
  - floppy or hard disks whose capacity is measured in mega or gigabytes
  - Central Processing Unit
11. The purpose of the Arithmetic Logic Unit within the Central Processing Unit is to:
- act as a temporary store for information being processed
  - perform calculations in the binary number system
  - perform calculations in the binary, octal or hexadecimal system
  - perform all clock functions based on the computer clock frequency (clock time)
12. Within the Central Processing Unit, the temporary stores and accumulator which handle the data during processing are called:
- Arithmetic Logic Unit (ALU)
  - shift registers
  - control unit
  - BIOS

## Answers

1	2	3	4	5	6	7	8	9	10	11	12
b	d	c	c	d	a	b	c	a	c	b	b



Chapter  
**24**

**Future Air Navigation Systems (FANS)**

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## Introduction

Currently aircraft are controlled using voice communications. Over and close to populated landmasses, ATC use radar to provide positive control of aircraft with VHF communications. However, over oceans, deserts and polar regions VHF, and radar, may not be available and ATC will have to provide a procedural control service, which generally requires HF communications, resulting in high vertical, lateral and longitudinal separation distances, resulting in a low traffic density.

Position reports are passed by aircraft crossing the North Atlantic every 10° of longitude up to 70°N and every 20° north thereof, which means ATC receive a position update every 30 - 60 minutes. With the increase in international air traffic this is posing major limitations, compounded by the difficulties associated with HF communications which means that in these remote areas the pilot is rarely communicating directly with the ATC controller but rather through a third party who relays messages between the two.

Over and around populated areas radar allows positive identification and control of aircraft. The need to increase capacity over the rest of the world requires that ATC have real time information on the aircraft positions, which must have improved accuracy, and better communications between the air traffic controller and the pilot.

The advances in technology mean we now have global satellite navigation systems (GNSS) providing position accuracy to better than one NM. Although there are still some issues on the service provided by GNSS, in general this high level of position accuracy is now globally achievable. Satellite communication systems (SATCOM) are also available giving the potential for global communications through a single medium.

## Communications Systems

Until the late 1980s communications with ATC and the air operators was only achievable using VHF and HF voice communications. VHF communications are line of sight and therefore only available over and in the near vicinity of habitable areas and are limited to about 200 NM from the transmitter for aircraft at FL300. Outside these areas communications have to be effected using HF. HF is notoriously difficult to use, having high levels of static interference with fading of signals which leads to messages being repeated and/or relayed through other aircraft. This means that a third party communicates with the aircraft and relays messages between the ATC controller and the pilot. Because of the stressful nature of these communications, the pilot does not maintain a continuous listening watch but is alerted by a selective calling system (SELCAL) which is activated by the communicator when there is a message for the aircraft.

In the 1990s voice communications were extended to SATCOM (using UHF frequencies) via geostationary satellites. Because geostationary satellites can only be positioned above the equator, coverage is limited to about 80° of latitude. To provide a service in polar regions it is intended to use satellites in lower altitude orbits with the orbits inclined such that polar regions will have a full SATCOM service.

### Disadvantages of Voice Communications

- Many a/c on one frequency
- Synchronous transmission
- Language confusion
- Limited channels
- VHF - line of sight only
- HF - Interference, tiring to listen to

### Data Link

To overcome many of the problems with voice communications, data link services have been developed.

A data link is a means of connecting one location to another, for the purpose of transmitting and receiving information. Data links may be established on any frequency, but will require additional equipment on both the ground and the aircraft.

HF	VHF	SATCOM
Long Range	Short Range	Global (except over Poles)
Interference		
Sky Waves (long range)	Line of Sight	Line of Sight
Poor Quality	Good Quality	Good Quality
Slow Speed	Medium Speed	High Speed

Figure 24.1 Advantages/disadvantages of data link communication

The transmission of signals from an earth station to a satellite or airborne platform is known as an **Uplink**.

The transmission of signals from a satellite or airborne platform to an earth station is known as a **Downlink**.

Currently there are two organizations providing data link services to aviation:

**SITA**

Société Internationale de Télécommunications Aéronautiques (France)

and

**ARINC**

Air Radio Incorporated (USA)

The additional equipment normally fitted to an aircraft to operate a data link may be:

**CMU:** Communications Management Unit  
Used to select the frequencies required for all radio equipment

**DCDU:** Data Communications Display Unit  
Used to display any messages received, or sent, via the data link

**MCDU:** Multi Control and Display Unit  
This could be a combination of CMU and DCDU

**VISUAL:** "Attention Getters" - Light & Sound  
To alert the pilot of an incoming data link message.

**PRINTER:** Hard Copy Output

## Aircraft Communication Addressing and Reporting System (ACARS)

ACARS uses data link format to pass messages between the aircraft and ATC or aircraft operating companies using VHF, and the messages can be printed out in the cockpit. In the early 1990s this service was extended to SATCOM for flights outside VHF coverage using geostationary satellites. The gap in polar regions was closed in 2001 by extending the service to HF.

In the 1990s interfaces with the FMS were developed which allowed operational data, for example meteorological information and alternate flight plan routes, to be evaluated using the FMS. Maintenance functions were also added to allow independent monitoring of the aircraft systems. This is implemented through ATSU/DCDU (Air Traffic Service Unit/Data link Cockpit Display Unit).

## Air Traffic Service Unit (ATSU)

The ATCSU is able to provide the following services to aircraft:

- Flight Information Service
- Alerting Service
- Air Traffic Advisory Service
- Air Traffic Control Service
  - Area
  - Approach
  - Aerodrome

## Communications, Navigation and Surveillance Systems/Air Traffic Management (CNS/ATM)

CNS/ATM is designed to use various levels of automation, digital technology and satellite systems to give a **Seamless Global Air Traffic Management System**.

Communications, Navigation and Surveillance Systems refer to the facilities offered by an ATSU.

Air Traffic Management refers to the facilities offered by an Aircraft Operational Centre (AOC). This is basically the airline, or company, operating the aircraft.

## ATM Data Link Messages

There are many different types of message that can be sent by data link to or from an ATSU. Examples of the more important messages are:

### *Departure Clearance*

```
PDC 130044
CPA065 B742 VHHH 0100
CLEARED TO OMDB VIA 07R
LAKE1A DEPARTURE V1
ROUTE: FLIGHT PLAN ROUTE
MAINTAIN: 5000FT
EXPECT: F330
SQUAWK A5156
-END
```

### *Oceanic Clearance*

```
CLRNCE 123
SAS911 CLRD TO KEWR VIA
57N010W
RANDOM ROUTE
58N020W 58N030W
57N040W 55N050W OYSTR
STEAM
FM 57N010W/1034 MNTN
F310 M082
ATC/ENTRY POINT CHANGE
ROUTE CHANGE AT 58N 20W
END OF MESSAGE
```

## Digital ATIS

```

/HKGATYA.TI2/VHHH ARR ATIS Z
2205Z
HONG KONG ATIS
RWY IN USE 07R
EXPECT ILS DME APCH
WIND LIGHT AND VARIABLE
VIS 10 KM
CLOUD FEW 1600FT
TEMP 28 DP 22 QNH 1007
ACK Z ON FREQ 119.1 FOR ARR AND 129.9 FOR DEP
    
```

## AOC Data Link Messages

The aircraft operator (airline or company) needs to know where their aircraft are at any time to ensure an efficient and cost effective operation. To help them a data link can be set up to pass messages to and from the aircraft.

The important types of message are:

**O O O I**                      **Out of Gate, Off Ground, On Ground, In the gate**

**Out of Gate:**    OUTRP 1065/31 EGLL/KJFK .G-BOAC/OUT 0850

**Off Ground:**    OFFRP 1247/31 EGLL/KJFK .G-BOAC/OUT 0850/OFF 0900/ETA 1525

**On Ground:**     ONRP 1499/31 KJFK .G-BOAC/ON 1540

**In the Gate:**    INRP 1803/31 KJFK .G-BOAC/ON 1540/IN 1550

**Load Sheet**

LOADSHEET FINAL 9999 EDN 01  
SK9999/01 31SEP01  
ARN OSL ACREG 9901S 2/4  
ZFW 49764 MAX 59874 L  
TOF 6100  
TOW 55864 MAX 70760  
TIF 2100  
LAW 53764 MAX 64410  
UNDLD 10110  
PAX CM 60/17 TTL 77  
DOI 59  
DLI 56  
LIZFW 31  
MAC-ZFW 7  
MAC-TOW 10  
TRIM BY CABIN AREA - SECTION SI  
BOARDING PAX WEIGHTS USED:  
-OSL ADULT 84 CHILD 35 INF 00  
BOARDING BAG WEIGHTS USED:  
-OSL 13/PIECE  
TOTAL EET: 00:28  
AIRB TIME: 00:43  
ENDURANCE: 02:20  
ALTN: ENTO



## Passenger Information (Used by the cabin crew to help transferring passengers)

PASSENGERINFO - NEW

ATT SK0864 ETA:1235 GATE: 03A TERM: T5 BC:

REBOOKED

DEST	FLT	ETD	N-ETD	N-FLT	GATE	TM	CITY
-OER	SK2092	1215	1440	SK0094		T4	OERNSKOELDSVIK
-VXO	SK1151	1240	1520	SK1157		T4	VAEXJOE

CONNECTIONS

DEST	FLT	ETD	GATE	TM	CITY
-HEL	KF440	1300	04A	T5	HELSINKI
-FRA	LH3003	1300	06A	T5	FRANKFURT
-RIX	SK2759	1315	F63	T5	RIGA
-CPH	SK405	1315	08	T5	KOEBENHAVN
-SVO	SK730	1330	F58		MOSCOW-SHEREMETYEVO
-JKG	JZ1015	1345	53		JOENKOEPIG
-TLL	SK748	1405	F61	T5	TALLINN

SIGNED: PSJ

LN-BRQ NEXT FLIGHT SK0861 ARN OSL STD 09:55 PLANNED ?T/A 00:30

END

## Weather Reports

/HKGVOYA.TI2/VHHH ENR ATIS

HONG KONG VOLMET

VHHK SIGMET A2 VALID 090530/090930 VHHHHONG

KONG CTA EMBD TS FCST S OF N19 E OF E114 TOP FL 350 STNR NC =

METAR VHHH 090730Z 16011KT 130V190 9999 SCT028 BKN300 33/23 Q1007  
NOSIG=

METAR ZGGG 090700Z 13004MPS 9999 FEW040TCU FEW040 34/23 Q1007  
NOSIG=

METAR ROAH 090730Z 19013KT 9999 FEW015 32/25 Q1010=

METAR RCTP 090730Z 28012KT 9999 SCT012 FEW020CB BKN022 BKN050  
30/27 Q1009

TEMPO 3000 -SHRA=

METAR RCKH 090730Z 18017KT 9999 FEW020 SCT300 33/23 Q1007 NOSIG=

METAR RPLL 090700Z 14008KT 9999 SCT025 SCT100 31/18 Q1008 TCU W=

METAR RPVM 090700Z 24014KT 9999 FEW020 SCT300 31/26 Q1007 A2976=

TAF VHHH 090530Z 090615 18010KT 9999

**Maintenance Reports**

3E03 SNAG 0999/31 KJFK/EGLL .G-BOAC  
/FAL 35310102  
35. OXYGEN  
310L OXYGEN BTL AT STN6  
HAS BEEN USED FOR A PAX  
IN NEED.

**Free Text**

MSG ID - 123456  
INFO - xxxxxxxxxxx  
CONFIRM - Y  
RTG - EIDW/EKCH/020  
CTOT -  
REROUTING MESSAGE  
SK 9999 DUB/CPH STD: 1000 UTC (TIMETABLE)  
RE-ROUTED DUE FLOW REGULATIONS  
NEW ROUTING EIDW/EKCH/SAS852- ATS PLAN FILED ACC.  
NO RESTRICTION ON THIS ROUTING AT PRESENT TIME.  
/ CDR MUST CONFIRM RECEIPT OF THE REROUTING MESSAGE TO CPHOW  
/ VIA: DATALINK USING FLOW CONFIRM OR OTHER REPORT TO CPHOFSK/  
PHONE +XXXXXXXXXX / CUT XXX.X KKMSG CPHOWFLOW OR SITA CPHOCSK.  
NOTE/NEW FLT-TIME 0149  
BRGDS SAS CENTRAL FLOW MANAGEMENT  
FLOW MSG END

## Logon

To gain access to the data link service the pilot must perform a manual logon. This requires the pilot to input the four digit ICAO address for the ATSU into the FMS which then sends the logon message to the ATSU. The logon message includes the aircraft address and information on the services the aircraft equipment supports. The ATSU acknowledges the logon message then sends a connection request message to the aircraft. The aircraft responds with a connection confirm message and the process is complete. This known as the ATS facilities notification (AFN).

This initial logon is performed on first contact with the ATSU on the ground, when entering an area with CPDLC from a non-CPDLC area and if there has been any interruption to the service (that is if the link is broken). Once the service is established the equipment will perform an automatic transfer to subsequent CPDLC capable ATSUs.

The message from the aircraft to the ATSU is always referred to as the downlink (DL) message, and from the ATSU to the aircraft as the uplink (UL) message even if the aircraft is on the ground.

## Future Air Navigation Systems (FANS)

The aim of FANS is to provide an integrated air traffic control system in areas where radar is not available by using GNSS to define aircraft position with data link and voice communications through geostationary and lower orbit satellites providing global coverage to ATC centres. This will provide ATC with continuous information on the aircraft positions and direct pilot to controller communications. When in operation this will allow separation distances to be significantly reduced, and if position accuracy is degraded for any reason, the aircraft still have TCAS to warn of any potential conflict.

There are two systems established to give the required global coverage: FANS A and FANS B.

### FANS A

FANS A provides a communications, navigation and surveillance (CNS) system and an automatic dependent surveillance system (ADS). Communications utilize current frequency allocations in HF, VHF and GNSS provide the navigation input for the surveillance. Currently it is used by some AOCs to monitor the progress of aircraft at all stages of flight. So, for example, the aircraft system will automatically inform the AOC of gate departure, take-off, landing and gate arrival. In-flight progress and the operation of on-board systems can also be monitored and, where necessary, messages can be passed to alert/assist the crew when potential problems are detected. FANS A is used over oceanic and remote airspace, and is transmitted over the ACARS network, operated by ARINC.

The components of the FANS A system are:

**AFN:** ATS Facility Notification

A contact message initiated by aircrew, or an automatic trigger within the aircraft. If acknowledgement has not been received within a pre-set time or there is an erroneous reply, then an error message must be displayed to the aircrew.

**ADS:** Automatic Dependent Surveillance

Controller set-up contract (ADS-C) with an aircraft's FMS, without any pilot input, using automated and customized reports.

The flight crew have **no workload** associated with this set-up.

The types of contract available are:

- Periodic
- On Demand
- On Event
- Emergency Mode

Only the flight crew can declare and cancel ADS-C emergency reporting.

**Note:** Flight crew can initiate the emergency mode. But the aircraft cannot initiate a contract

**CPDLC:** CPDLC permits data link messages to be generated for all stages of flight. The messages have a fixed format defined within the FMS and ATSU computers and are activated by the ATC controller or pilot either as an instruction or response to a request. The messages are annotated according to whether a response is required or not. For example if the pilot is instructed to report levelling at FL310 then the message will remain open until the aircraft reaches FL310 when the FMS will send the response. The confirmation that the aircraft is at FL310 does not require a response from the controller, so the message will automatically close once received at the ground station.

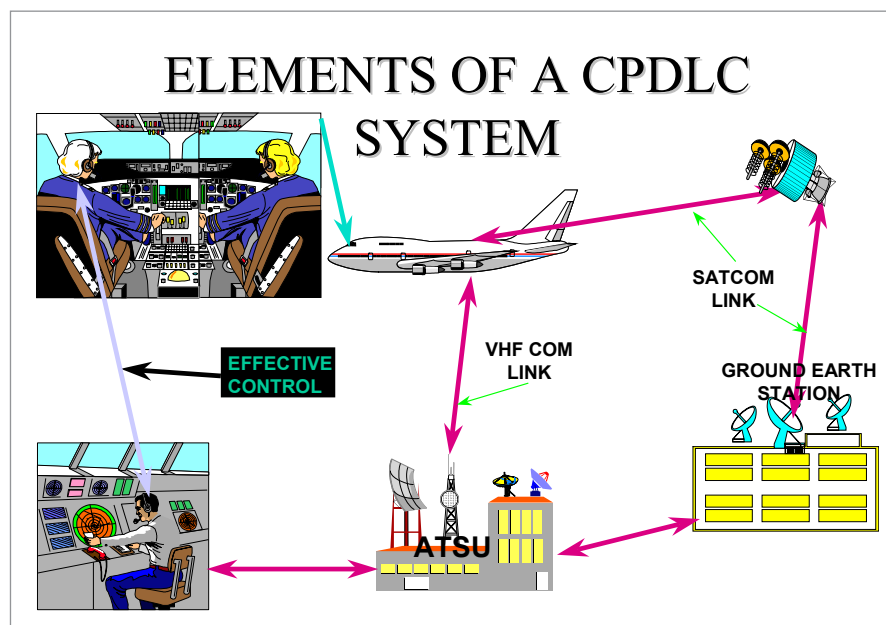


Figure 24.2

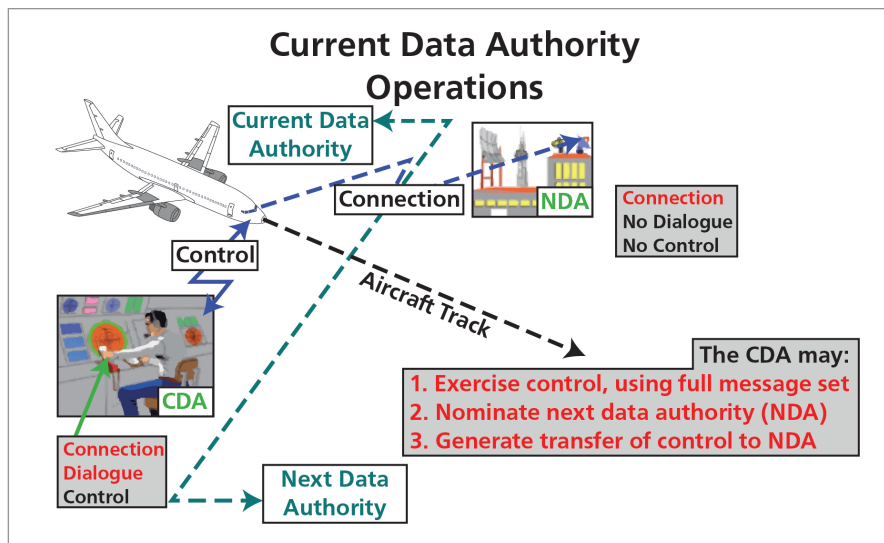


Figure 24.3

An ATSU may use:

- AFN Only or ADS Only or CPDLC Only
- or
- AFN and CPDLC
- or
- ADS and CPDLC
- but not**
- AFN and ADS

The typical architecture is shown in the next diagram:

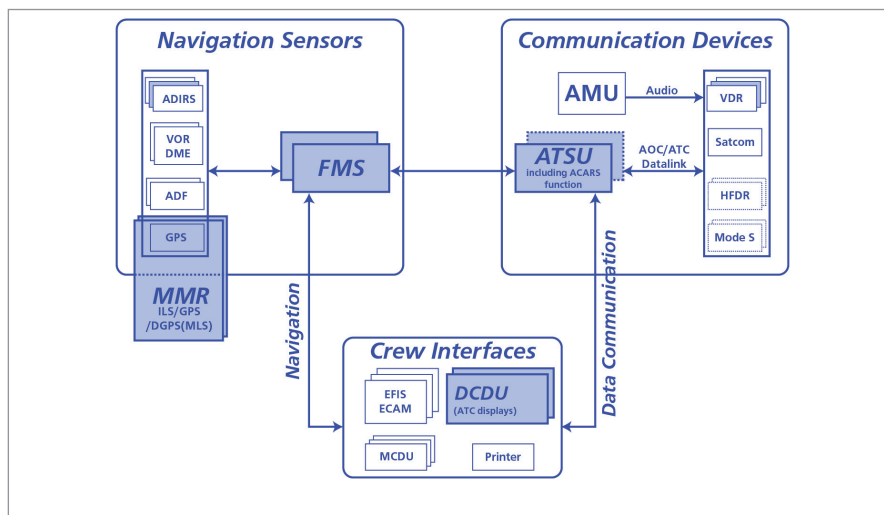


Figure 24.4



Figure 24.5 The diagram shows a typical DCDU message display

### FANS B

FANS B is very similar to FANS A but operates within High Density airspace having good VHF coverage.

FANS B is operated over the Aeronautical Telecommunications Network (ATN), which is operated by SITA.

The ATN is an internetwork architecture that allows ground/ground, ground/air and avionic data subnetworks to inter-operate.

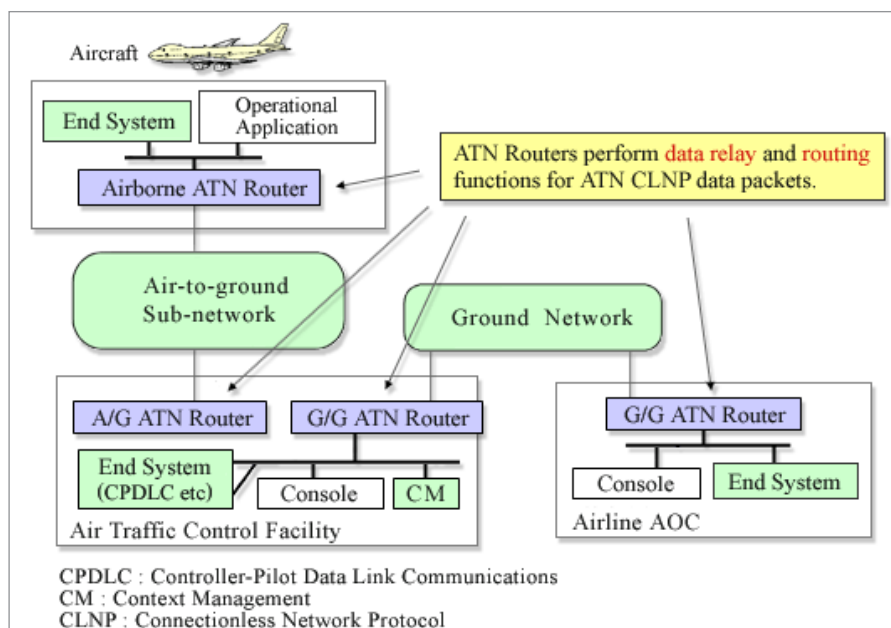


Figure 24.6 A Typical ATN Organization

## Useful Abbreviations

### Communications

<b>ATIS</b>	Automatic Terminal Information Service
<b>D-ATIS</b>	Data Link Air Terminal Information Service
<b>HF</b>	High Frequency. A portion of the RF spectrum
<b>SATCOM</b>	Satellite Communications
<b>VHF</b>	Very High Frequency. A portion of the RF spectrum
<b>FANS</b>	Future Air Navigation System
<b>ACARS</b>	Aircraft Communications Addressing and Reporting System
<b>ACMS</b>	Aircraft Condition Monitoring System
<b>ADS</b>	Automatic Dependent Surveillance
<b>ADS-C</b>	Automatic Dependent Surveillance - Contract FANS 1/A
<b>AFN</b>	Air Traffic Facilities Notification (FANS)
<b>AMS</b>	ACARS Message Security, as specified in ARINC 823
<b>AOA</b>	ACARS Over AVLK
<b>AOC</b>	Airline Operational Communication/Airline Operational Centre
<b>ARINC</b>	Air Radio Incorporated (USA)
<b>ATM</b>	Air Traffic Management
<b>ATN</b>	Aeronautical Telecommunications Network (France)
<b>ATSU</b>	Air Traffic Service Unit
<b>AVLC</b>	Aviation VHF Link Control
<b>CDU</b>	Control and Display Unit
<b>CMF</b>	Communications Management Function
<b>CMU</b>	Communications Management Unit
<b>CNS</b>	Communication, Navigation and Surveillance
<b>CNS/ATM</b>	Communication, Navigation and Surveillance / Air Traffic Management for North America & Europe
<b>CPDLC</b>	Controller Pilot Data Link Communications

<b>CVR</b>	Cockpit Voice Recorder
<b>DCDU</b>	Data Link Control and Display Unit
<b>DCL</b>	Departure Clearance
<b>FANS A</b>	Data Link Package for Oceanic/Remote airspace/Accommodation
<b>FANS B</b>	Data Link Package for High Density airspace
<b>FDAMS</b>	Flight Data Acquisition and Management System
<b>FMS</b>	Flight Management System
<b>FWS</b>	Flight Warning System
<b>HF</b>	High Frequency. A portion of the RF spectrum, defined as 3 MHz to 30 MHz
<b>HFDL</b>	High Frequency Data Link - an ACARS communications media
<b>HMI</b>	Human Machine Interface
<b>LRU</b>	Line Replaceable Unit
<b>MCDU</b>	Multi-function Control and Display Unit
<b>MIDU</b>	Multi-input Interactive Display Unit (often used as a third cockpit CDU)
<b>MU</b>	Management Unit. Often referred to as the ACARS MU
<b>OCL</b>	Oceanic Clearance
<b>OOOI</b>	Out of the gate, Off the ground, On the ground, In the gate
<b>POA</b>	Plain Old ACARS ACARS communications protocols in effect before the introduction of VDL
<b>RMP</b>	Radio Management Panel
<b>SDU</b>	Satellite Data Unit
<b>SITA</b>	Société Internationale de Télécommunications Aéronautiques (France) Airline Telecommunications and Information Service
<b>STDMA</b>	Self-organizing Time Division Multiple Access
<b>VDL</b>	VHF Data Link
<b>VDL2</b>	VHF Data Link Mode 2
<b>VDR</b>	VHF Data Radio
<b>VHF</b>	Very High Frequency. A portion of the RF spectrum, defined as 30 MHz to 300 MHz





# AUTOMATIC FLIGHT AND CONTROL SYSTEMS

## ATPL GROUND TRAINING SERIES

10

10



Chapter  
**25**

Flight Director Systems

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## Glossary

A/P	Autopilot
A/T	Autothrottle
A/L	Autoland
ADC	Air Data Computer
ADI (EADI)	Attitude Director Indicator (Electronic)
AFCP	Auto-flight Control Panel (see also MCP)
AFCS	Auto-flight Control System
AFDS	Auto-flight Director System
AFS	Auto-flight System
ALT ACQ	Altitude Acquire (mode)
AoA	Angle of Attack
APP	Approach (mode)
CADC	Central Air Data Computer
CDI (bar)	Course Deviation Indicator (bar)
CDU	Control and Display Unit
CMD	Command (or Autopilot Engage)
CWS	Control Wheel Steering
DG / DI	Directional Gyro / Direction Indicator
ECAM	Electronic Centralized Aircraft Monitoring
EFIS	Electronic Flight Instrumentation System
EICAS	Engine Instruments and Crew Alerting System
EPR	Engine Pressure Ratio
FADEC	Full Authority Digital Engine Control
FCC	Flight Control Computer
FD	Flight Director
FDC	Flight Director Computer
FDS	Flight Director System
FMA	Flight Mode Annunciator
FMC	Flight Management Computer
FMGS	Flight Management and Guidance System
G/S	Glide slope or ground speed dependent on context
GA	Go around or Gear Altitude dependent on context
HDG	Heading (mode)
HSI (EHSI)	Horizontal Situation Indicator (Electronic)
INS	Inertial Navigation System
IRS	Inertial Reference System
LAND 2	Fail-passive
LAND 3	Fail-operational
LNAV	Lateral NAVigation (mode)
LOC	Localizer (mode)
LVL CHG	Level Change (mode)

MCP	Mode Control Panel (see also AFCP)
MI	Magnetic Indicator
N1	rpm of the first stage of compression (normally the fan in a high bypass engine) expressed as a percentage
ND	Navigational Display
PFCU	Powered Flying Control Unit
PFD	Primary Flight Display (EADI including speed, altitude, VSI tapes and commonly a compass and annunciator panel)
PMC	Power Management Computer
QDM	"Q" code for a magnetic heading to fly assuming zero wind
RA	Radio Altitude
SPD	Speed (mode)
TCS	Touch Control Steering
TLA	Throttle Lever Angle
TMA	Thrust Mode Annunciator
TMS	Thrust Management System
TO/GA	Take-off / Go-around
TRK	Track
V/S	Vertical Speed (mode)
VG	Vertical Gyro
VNAV (SPD/PTH)	Vertical NAVigation /Path or Speed (mode)
VOR	VOR tracking (mode) or a VHF omni-range beacon dependent on context

## Introduction

The Flight Director System (FDS) was originally developed as an aid used by the pilot during landing. The FDS provides a pilot with steering and attitude signals on one instrument which helps reduce his workload. As autopilots became more advanced the signals produced by the FDS could be coupled to the autopilot allowing it to perform more complex tasks.

With an FDS, information about the attitude, heading and flight path of an aircraft, can be integrated with navigation information to produce either easy to interpret visual instructions for the pilot and/or input to the autopilot, or both.

To bring the terminology of FDS and autopilot together it is usual to describe the FDS as having 2 "channels". The first channel is the **roll channel**, the second is the **pitch channel**. You will learn more about channels in the autopilot section.

Information for the FDS can come from several possible sources:

- Pitot-Static system or Air Data Computer (ADC).
- VHF Nav receiver allowing input from VOR beacons or Instrument Landing System.
- Flight Management System, Inertial Navigation/Reference System.

The FDS also requires attitude and directional information. On older, electromechanical systems this would come from the gyro-magnetic compass and a vertical gyro system. More modern aircraft use Inertial Navigation/Reference System (INS/IRS) information in place of a vertical gyro and will be able to feed the navigation data from these systems into the FDS/Autopilot combination.

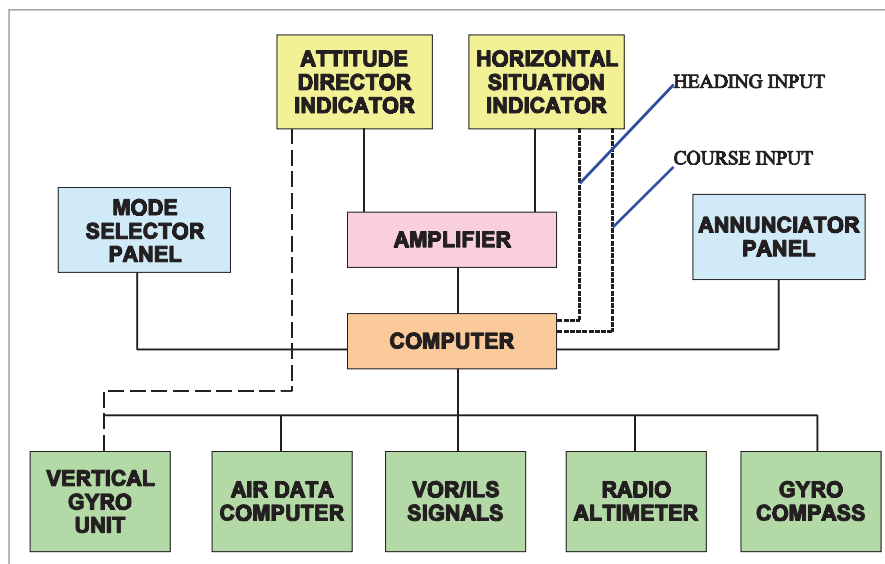


Figure 25.1 A "typical" electromechanical flight director system

### Flight Director System Components

There follows a description of a "typical modern" FDS:

#### Electronic Attitude Director Indicator (EADI)

This is a fairly standard artificial horizon providing pitch and roll information and gives the Attitude to the name of the instrument. The Director part comes from the instruments ability to display demand information from the flight director system using Flight Director Command Bars. These come in 2 main forms as shown below.

Both of the indications for these apparently different displays are intuitive and essentially the same in that the pilot is required to "fly to" either the point where the "wires" cross, or the point between the wedges, in order to satisfy the demand from the FDS.

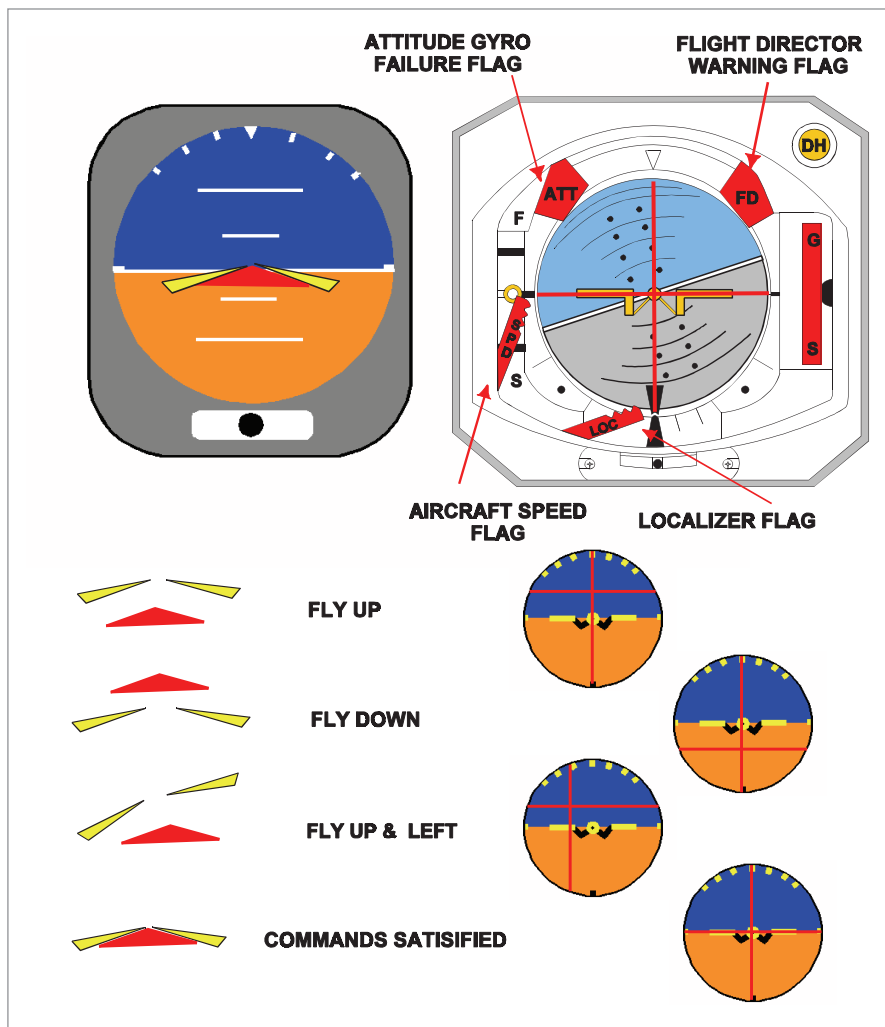


Figure 25.2



**Primary Flight Display (PFD)**

The PFD is part of an Electronic Flight Instrument System (EFIS) display and brings all of the information required to fly the aircraft onto one display. It has an EADI normally surrounded by speed, altitude and vertical speed tapes and often a compass display incorporating some minimal navigation information. It also has an area which is used for annunciating flight director, autopilot and autothrottle modes and status, known as the Flight Mode Annunciator (FMA).



Figure 25.3

### Electronic Horizontal Situation Indicator (EHSI)

You may already be familiar with the gyro-magnetic compass. An HSI is a gyro-magnetic compass display with a Course Deviation Indicator (CDI) bar, a series of dots representing deviation in degrees (the scale varies with the type of display), a from/to pointer, a selected course window and a DME display of range. A heading bug is also included. In older systems the course selection is done directly on the HSI using an attached knob. The system we will refer to uses a remote centralized FD mode control and AP panel called the Autoflight Mode Control Panel (AMCP or simply MCP). The system we will be referring to also uses a Navigational Display (ND). This, like the PFD, is a more flexible display but is able to show "classical" representations of an HSI.

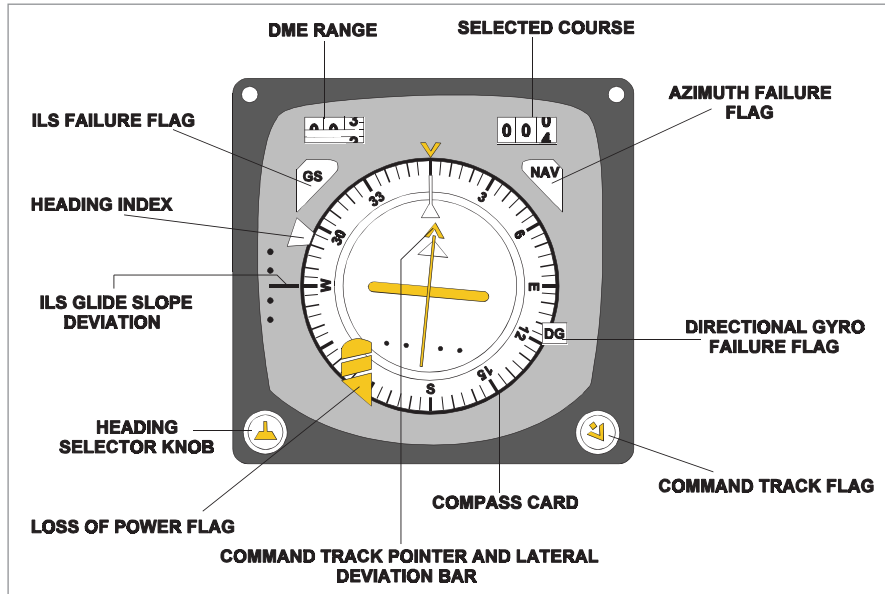


Figure 25.4 HSI Display



Figure 25.5 EHSI

### Flight Director Computer (FDC)

This is where all the information is gathered and processed. On older aircraft this information could be in the form of analogue outputs from the ADC and VG. Current systems will be purely digital. Older aircraft that have been refitted may have analogue inputs to a digital converter so that modern displays and autopilot systems can be used. Outputs from here are sent to the symbol generators for the EADI/EHSI and/or the autopilot as required.

### Optional Components

FDS use other components depending on what generation they belong to:

#### Instrument Amplifier or Symbol Generator

Where information is required to be displayed on electromechanical instruments the signals require amplification to drive the associated motors. On EFIS fitted aircraft the FDC output can be fed directly to the symbol generators for the Primary Flight Display (PFD) and Navigational Display (ND) units.

#### Vertical Gyro or INS/IRS

In older or smaller aircraft types, normally without any INS/IRS, reference to the vertical for the Artificial Horizon part of the ADI is provided by a remote vertical gyro system. This simply means that the gyro that acts as the artificial horizon is not contained in the instrument in the panel in front of the pilot but can be anywhere on the aircraft (normally near the C of G). The information derived from it is fed to motors that drive the display of the ADI. The benefits of this system are that the gyro can be near the aircraft's C of G and therefore provide a more accurate display of the attitude of the aircraft, and this data in electrical form can be fed to any other items requiring attitude information e.g. the autopilot.

More modern and larger aircraft may use data from their INS or IRS to replace that normally provided by a VG as the INS is simply a more modern and much more sensitive version of a gyro.

#### Mode Controller or Mode Control Panel (MCP)

The mode controller allows the pilot to change the mode of the FDS, alter the pitch trim and switch the FDS display on or off as required. The modes available depend on the fit of the aircraft. On modern aircraft the MCP, on which most of the information for the autopilot is programmed, is used to replace the separate FD mode controller. Typical modes and the pitch trim will be discussed a little later. The MCP will also be described in more detail later.

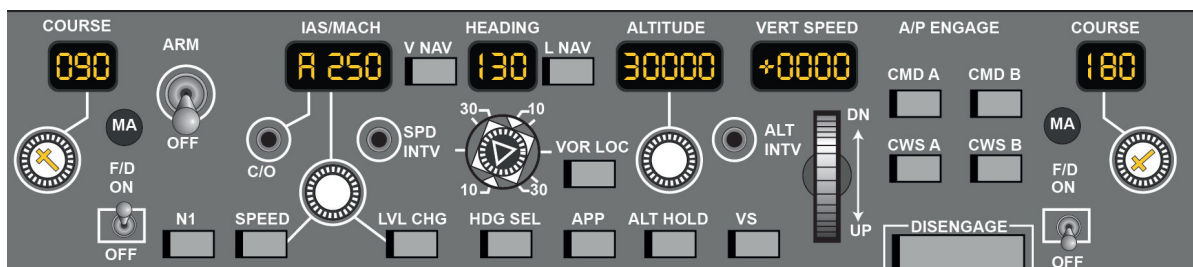


Figure 25.6

#### Mode Annunciators / Indicators or PFD Flight Mode Annunciators (FMA)

These are a series of simple lights, magnetic indicators or a small panel of illuminated indicators that show the pilot what mode or phase of a particular mode the FDS / autopilot is in. In more basic systems the most common indications are during an ILS approach. These panels are usually only powered with the FD switch on.

When the aircraft is awaiting capture of the localizer, the LOC (localizer) light will typically be amber. When the localizer has been captured the light will change to a green LOC light. Approaching the glide slope the GS light will be amber and again will change to green to indicate the fact that the FDS has locked on to the GS.

More capable systems will have more functions and consequently more lights, such as those indicating the state of the autothrottle and flare or go-around modes.

EFIS equipped aircraft display the appropriate information on the PFD/EADI in an area called the Flight Mode Annunciator. As the same space can be used for all the different messages, it can be kept small. Here all the modes for the FD autothrottle and autopilot are displayed. Newly changed information is often emphasized on the FMA type display by surrounding it in a box. Engaged modes are shown on the top line in green while armed modes are on the line below in white.

## Flight Director Modes

As has been said the modes for each system vary. The modes described now are the most common but not all may be available depending on aircraft fit.

First it must be appreciated that there is a distinction in the way data is displayed to the pilot. The glide slope indicator to the side of the ADI and the CDI bar are what is termed **raw information**. That is the information is not processed in any way and simply indicates that the aircraft is left or right of track / localizer, or above or below the glide slope and knowledge of the system will tell you by how many degrees.

In order to anticipate turns, climbs and descents for smooth flying the FDS can use the raw data, compute and signal commands to either the pilot via the flight director bars or directly to the autopilot. This is called **computed information** and is derived from the **rate of change of deviation** of the incoming signal. As the rate of change of deviation from the desired track increases, the FDS computes that in order to intercept the track correctly, rather than fly straight through, it must indicate a turn onto track.

This becomes important if the FDS fails as, although the computed information will no longer be reliable / available, the raw information may still be available and used.

Many of the FD modes are common with the autopilot modes. We have detailed the FD modes in the following sections. Where they are similar to the autopilot mode the description is fairly basic with the full description being found in the autopilot chapter. Where the FD mode differs from the autopilot mode the description will detail the differences.

### FD Fail Indications

Before going on to discuss the modes individually we need to know how the FDS informs us if the information it is giving is reliable. On electromechanical displays warning flags are used. If the vertical gyro or other vertical referencing system fails or its power supply fails then a "Gyro" warning flag will pop into view, normally on the ADI. Failures of the FDC, the instrument amplifier, or the ADI itself, are indicated by a flag labelled either "ATT" for attitude or "FD". This will normally again be on the ADI display.

If glide slope information is unreliable, due to poor signal strength or failure of the system, a "GS" flag will appear in front of the raw glide slope scale where it appears, either to the side of the ADI or the HSI.

Poor reception, unreliable or loss of VOR, LNAV or LOC information is indicated by a "NAV" flag, normally located on the HSI.

The HSI will have a power failure flag to indicate loss of power to the instrument or the compass gyro. It will also indicate if the compass system is operating in Directional Gyro (DG) mode i.e. the magnetic reference has been lost and the compass is now essentially a DI.

### *Command attitude changes*

When flying level the pilot selects an attitude that, for the given airspeed, will achieve level flight. With the aircraft at low airspeed the pitch angle will be several degrees nose-up to avoid descending. As an aid to keeping a constant pitch angle the FD command bars can be offset vertically to provide an intuitive aiming point for the pilot.

This can also be used for keeping constant pitch angles for climbing and descending. The movement of the bars is achieved by moving the Pitch Trim knob or wheel (dependent on age of design) until the command bars are in the desired position. The pitch trim system is inhibited whenever any other pitch mode is active.

### *Flight Director Take-off Mode*

Initially both flight director systems should be switched on prior to starting the take-off roll. The FD take-off mode is engaged by pressing either of the TO/GA switches on the throttles. The AFDS annunciation is "TO/GA". The initial FD commands are for 10 degrees nose-down pitch and wings level. At 60 kt IAS, the FD command changes to 15 degrees nose-up and wings level.

To engage the FD system during the take-off even if the FD switches are off press the TO/GA button after 80 kt IAS but before 2000' or 150 seconds after lift-off and the command bars will automatically appear.

The FD provides pitch commands after lift-off. It continues to command 15 degrees nose-up pitch until sufficient climb rate is achieved. It then commands pitch to maintain the MCP speed plus 20 kt IAS; this speed is calculated and set during the pre flight. Next, when either autopilot is engaged or when the MCP speed selector is rotated, 20 kt IAS is added automatically to the MCP IAS display.

FD roll commands wings level from take-off mode engagement through to the take-off climb-out. To terminate the take-off mode below 400' RA, both of the FD switches must be turned off. Above 400' RA, the take-off mode can be terminated by selecting other FD pitch modes or by engaging an autopilot.

Engaging an autopilot after a FD take-off automatically engages the autopilot and FD in LVL CHG (level change) for pitch and HDG SEL (heading select) for roll. If the FD mode had been changed from TO/GA to LNAV, HDG SEL or VOR/LOC, the autopilot initially engages in the same roll mode as the FDs. When LVL CHG engages, the MCP IAS/MACH display and airspeed cursors change to  $V_2 + 20$  kt.

If an engine fails during take-off before reaching  $V_2$  speed, the FD pitch commands are referenced to  $V_2$ . If the engine failure occurs after reaching  $V_2$ , but less than  $V_2 + 20$  kt, the reference speed is that at which the engine failure occurred. If the failure occurs at or above  $V_2 + 20$  kt,  $V_2 + 20$  kt is the commanded speed. Reference speed is never less than  $V_2$  for the current flap setting. Roll control remains the same as for all engines operating.

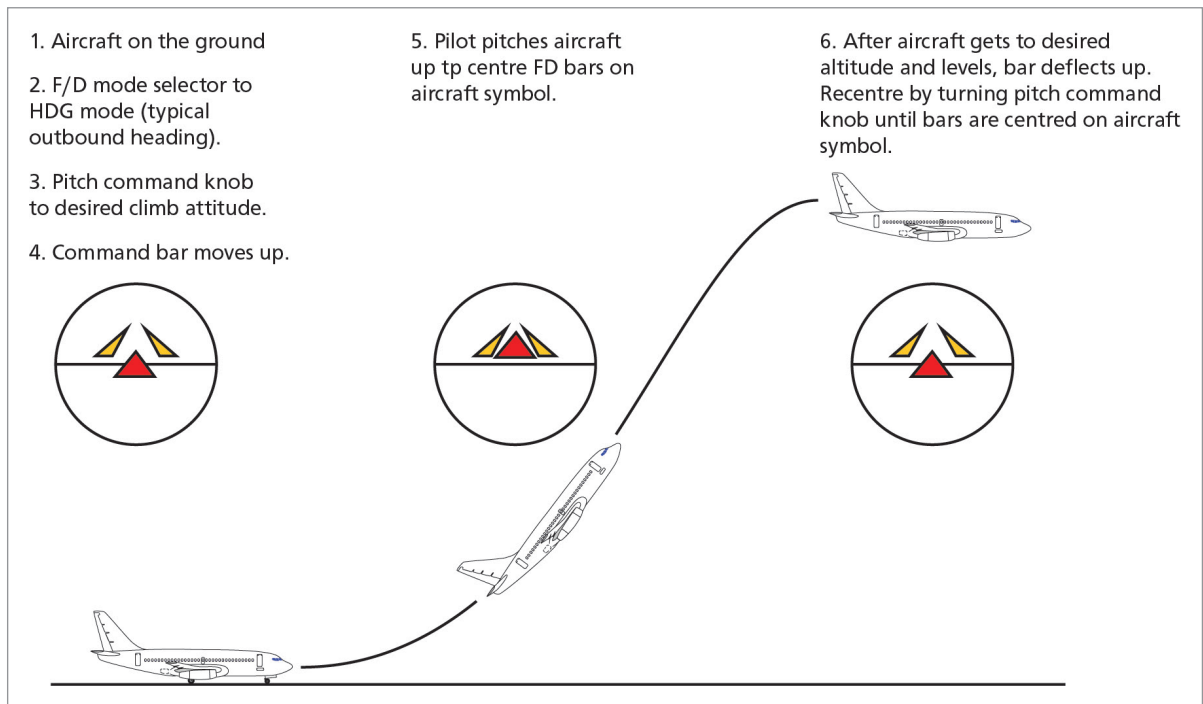


Figure 25.7

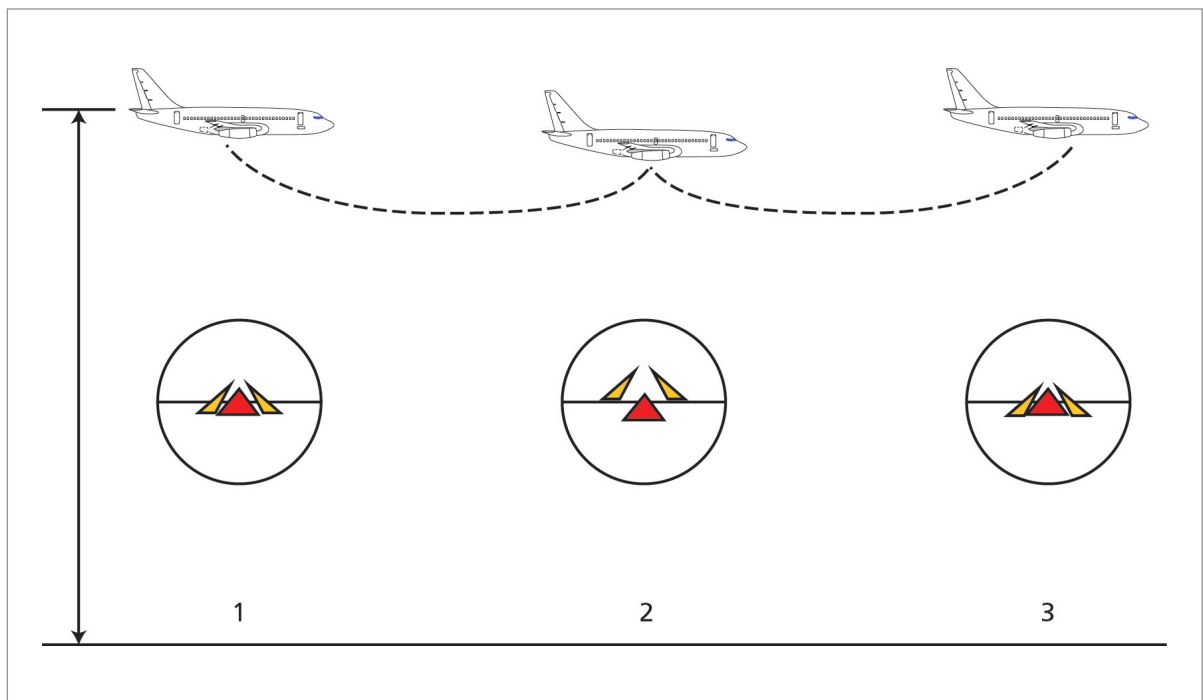


Figure 25.8

### Altitude Acquire / Altitude Hold

In the cruise we require to fly at a constant level. The FDS can be used to help achieve this. Signals from the ADC through the FDS mean that when we are at the selected level, the FD command bars indicate a neutral position. If the aircraft were to deviate above the desired altitude the command bars would indicate a pitch down command and vice versa.

## Navigation Modes

### Heading Mode

In this mode VOR beacons and various internal navigation systems can be linked to either display information using the FDS or linked to the autopilot or both.

The simplest nav mode is to have the aircraft keep flying in the direction you select. This is called **Heading Mode**. In this mode the pilot selects a desired heading, either using a knob on the HSI, using the central FDS/autopilot panel or on the MCP. The FDS will now indicate a fly to command to bring the aircraft on to the desired heading.

### LOC/VOR (LNAV)

This selection allows VOR, Localizer or INS/IRS/FMS(GPS) nav information to be fed and displayed on the FDS. In the case of the VOR, after tuning and correctly identifying the station to be used, with VOR/LOC selected on the mode selector and the desired track to (or from) the beacon set in the course selection window (using either on the HSI, MCP or the CDU), the FDS will give appropriate steering command to intercept and then maintain track.

Intercepting the LOC is very similar but will be discussed in more detail later.

With aircraft being fitted with more accurate navigation systems it is only logical that this information should be made available to the pilot and autopilot. INS/IRS/FMS/GPS information can be selected for display in very much the same way as the VOR information and is represented in the same way.

Note that because the actual track of the aircraft is being compared with the desired track, and the flight director commands are given to fly the desired track, the flight director system will effectively compensate for drift. There is no need for wind velocity information (although the INS could provide it), it is purely an effect of the flight director giving commands to fly the desired course/track selected.

### Flight Director Approaches (FDA)

The ILS frequency is tuned into the VHF nav radios and identified and the QDM for the approach is set up in the course window. The mode selector should be set to AUTO/APP. The annunciator will show that the FDS is looking to capture the LOC. After beam capture, in a similar manner to VOR tracking the FDS will (given enough distance) arrange the intercept to establish on the localizer with a turn demand and the annunciator panel will indicate LOC capture.

The FDS is now looking for the glide slope signal and so the annunciator GS light or magnetic indicator will indicate such. As soon as the glide slope is captured the annunciator changes again and the FDS will indicate a pitch nose down demand to fly the ILS glide slope.

If the interception of the localizer has been misjudged it is possible to end up established on the localizer past the point of GS intercept i.e. above the glideslope. Modern systems may automatically adjust and capture from above by increasing the rate of descent. On older aircraft it was sometime possible to temporarily switch the FDS mode to MAN/GS. This forces the FDS into accepting a capture from above. It can also be used to establish a fixed intercept angle of the LOC beam and to force a LOC or GS capture condition if it is known that the beam-sensing circuits of the computer are inoperative.

### FD Go-around (GA)

For the 737-400 series of a/c 2 criteria must be met before the FD can engage in the GA mode. The FD switches can be either on or off and:

In-flight below 2000' RA and not in the TO mode

TO/GA switch pressed

After engaging in GA, command bars appear for both pilots, TO/GA is annunciated for the FD pitch mode, the MCP IAS/MACH display blanks and the Airspeed Cursors display manoeuvring speed for the existing flap setting.

Below 400' RA, both of the FD switches must be turned from ON to OFF to exit the FD GA mode. Above 400' RA, other pitch and roll modes can be selected. If the roll mode is changed first, the FD pitch mode remains in the GA mode. If the pitch mode is changed first, the FD roll mode automatically changes to HDG SEL.

Engaging an A/P following a FD GA automatically engages both the A/P and FD in LVL CHG and HDG SEL for pitch and roll respectively.

For a 2-engine GA the FD commands a 15 degrees nose-up pitch and roll to hold the approach ground track at the time of engagement. After reaching a programmed rate of climb, pitch commands hold the manoeuvring speed for each flap setting.

During a single-engine GA the FD pitch command is initially to 13 degrees nose-up but as climb rate increases, FD pitch commands maintain a target speed. Roll commands are the same as for the 2-engine case. If engine failure occurs prior to GA engagement, then MCP selected speed becomes the target speed. If the engine failure occurs after GA engagement, then FD target speed depends on whether 10 seconds have elapsed since GA engagement:

- If prior to 10 seconds, the MCP selected approach speed becomes the target speed.
- If after 10 seconds and the airspeed at engine failure is within 5 kt of the GA engagement speed, the airspeed that existed at the GA engagement becomes the target speed.
- If after 10 seconds and the airspeed at engine failure is more than 5 kt above GA engagement speed, then the current airspeed becomes the target airspeed.

In all cases, the GA target speed is not less than  $V_2$  speed based on flap position unless in windshear conditions.

The FD target speed is displayed on the MCP and by the airspeed cursors. No commanded acceleration can occur until a higher speed is selected on the MCP.

### FD Manoeuvre Protection

Because the modern FDC is configured for each particular aircraft type it will have the aircraft performance parameters stored in its memory. As it has inputs from the ADC and other systems it can ensure that it **never commands a manoeuvre which will overstress the aircraft**. This is the beginning of the systems used for protection in aircraft with fly-by-wire controls (discussed fully in another chapter).



### Flight Director Gain Scheduling

Gain scheduling is the varying of the gain of the pitch and roll demands of the FDC in relation to the task. This has many parallels with autopilot gain scheduling or gain adaption. The FD mode of operation using gain scheduling is the FDA. As the approach progresses the glide slope beam converges with the runway (which is how it works and what we want). The ILS works as a beam set at a certain angle diverging from the runway, normally about 3 degrees. ILS equipment displays that received signal as an error in degrees from the ideal.

At 6 NM one degree of error equates to about 608 ft of vertical distance. At  $\frac{1}{2}$  NM that same one degree of error equates to about 54 ft of vertical distance. It should be obvious that although the indications on the raw ILS glide slope will be the same, a less forceful correction is required as the aircraft nears the ILS transmitter. **So the FD computed pitch information must be modified as the approach progresses to reduce the commanded corrections.**

As the aircraft approaches touch-down the magnitude of the pitch changes required to follow GS reduce. Gain scheduling reduces the magnitude of the commands as the aircraft proceeds on the FDA. So initially the FDS can demand manoeuvres almost to the full authority of the system. As the threshold approaches, however, the gain is reduced to perhaps a  $\frac{1}{2}$  or  $\frac{1}{3}$  of the original value.

The initiation of this scheduling can come in many forms depending on the age of the system:

Early systems simply used time e.g. 45 seconds after GS capture the gain is reduced.

The next systems used the marker beacons to try to actually match the scheduling required to the approach being flown. This system has fallen into disuse however because of the loss of the marker beacon systems at many airfields.

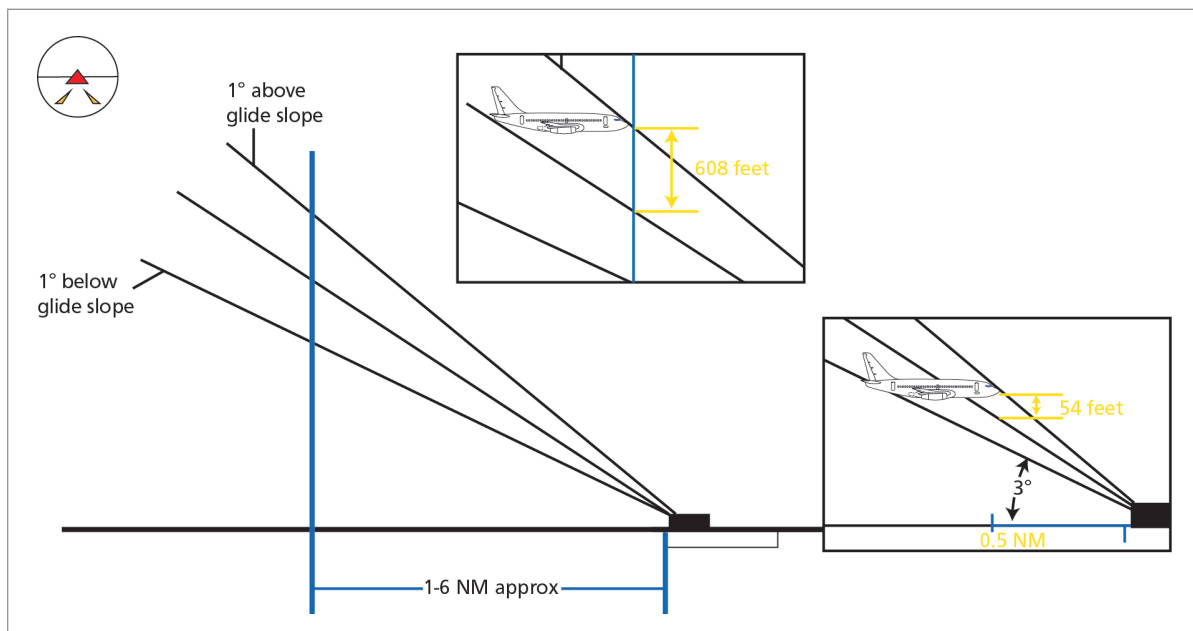


Figure 25.9

### *Radio Altimeter*

This relies on no ground signals and gives accurate scheduling in relation to actual aircraft height. This also means that the actual scheduling can be phased in gradually as opposed to the stepped method of the timing or marker systems.

Gain scheduling or adaption can be made to occur for any change in the flight regime to reduce the demands from the flight director to ensure adequate safety.

### **Dual FDS**

When, as on large aircraft, 2 FDS are fitted (one for each pilot), each system can be used to monitor the indications of the other. This system is the FD Comparator or Monitor.

The FD Comparator monitors command bar positions. The command bars are removed when a difference is sensed between the 2 FDS of approximately 1 to 4 degrees of pitch and/or 3 to 9 degrees of roll. FD command bars reappear when the difference returns to within limits.

FD comparison is only active during certain modes of FD operation. First, both FD switches must be on and neither autopilot engaged. Second, it only operates in either the TOGA or APP mode below 800 ft RA.

FD comparison is inhibited for several reasons. It is inhibited on the ground or when either FD is affected by electrical bus transfer. Also, it is inhibited by failure of either a FD sensor or an FD itself.

Having 2 FDS also means a certain amount of redundancy in the system. It also means that if one system should fail, as long as the associated display equipment is serviceable, both sets of display can if necessary be run from one FDC. This on electromechanical instruments is often simply a switch which, when operated, splits the FDC output from the serviceable FDS instrument amplifier and feeds it into both sets of ADIs and HSIs. The flight instrument and FD power (and that of navigational information sources) may also be separate to aid redundancy.

EFIS equipped aircraft have a spare symbol generator also to retain extra redundancy.

# Chapter 26 Autopilot

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## Introduction

The main purpose of the autopilot is to relieve the pilot of the physical and mental fatigue of flying the aircraft, especially during long flights. This will result in the pilot being more alert during the critical phase of landing.

Autopilot systems also enable the aircraft to fly a prescribed route accurately due to the autopilot's ability to react quicker than a human pilot to disturbances.

Many different autopilot systems exist offering many different modes of operation and facilities. Generally, however, today's modern airliner, when fitted with an autothrottle system, will have the facility to fly the aircraft automatically for almost the entire route. In each such system, the autopilot flies the aircraft as it responds to commands from attitude sensors, navigation systems, and pitot-static systems. Power is controlled through the engine throttles moved by autothrottle servos responding to commands from the thrust management computer.

It should be noted that, as yet, the autopilot does not carry out the take-off which has to be done by the pilot. The autopilot, though, can be engaged shortly after take-off at about 400 feet or possibly even a lower height.

## The Autopilot

The basic autopilot has been in existence for about 50 years. It was introduced as an aid to the pilot flying the aircraft for 2 major reasons:

### *Reduction in Workload*

Allowing automatic systems to fly the aircraft means that the crew not only are more rested for the more demanding phases of flight, but it also allows the pilots to concentrate on other tasks such as navigation.

### *The Response Time*

An autopilot is much quicker than a human and as a result it can fly an aircraft more accurately. A human pilot takes approximately 1/5 of a second (200 milliseconds) in detecting a change in the aircraft's attitude and then suffers a further delay while deciding which control to apply to oppose the disturbance. An autopilot will detect a disturbance and put on the required control to correct the disturbance in approximately 50 milliseconds.

## Fail Safe Autopilot

With any automatic system it is necessary to protect against malfunctions - in particular, runaways. This is achieved either by limiting the authority of the actuator or the rate at which the actuator can travel. In this way the pilot should always be able to override the effects of a malfunction and retain control of the aircraft in the event of autopilot failure. Such a system would be called a fail-safe system and the term applies to any single autopilot.

### *The Basic Autopilot*

The basic autopilot is a very simple system. Understanding of the basic autopilot is essential for understanding and explaining what any autopilot is doing throughout the flight.

### *Aircraft Stabilization*

This is the key function for an autopilot (and this is all that some basic autopilots may achieve). All the modes such as VOR tracking and altitude hold etc. are "extras".

Consider an early aircraft design. All that was originally required was a system that would keep the aircraft flying in the same attitude once the pilot was happy. He could then concentrate on navigation, disengaging the autopilot as necessary to correct headings and so on. To design such a system we need to consider the sequence of events that must occur to keep the aircraft's attitude constant. Consider yourself flying a light aircraft, e.g. a Warrior, in slightly turbulent conditions. Now consider that the aircraft experiences a disturbance in pitch:

A human pilot controls the aircraft's stability by sensing a change in aircraft attitude, computing the necessary corrective action required and using his muscles to move the flight controls. The aircraft then manoeuvres about its centre of gravity back towards its original position. He will then sense that corrective action has taken place and move the flight controls back to remove the correcting input.

The autopilot is capable of producing the same actions as the pilot to maintain aircraft stability in a shorter reaction time. It must detect the disturbance and then work out appropriate corrective action. It must then apply that correction using its "muscles" which will cause the aircraft to manoeuvre about its centre of gravity back towards its original position. Finally it must detect that the correction has taken place and re-centre the controls. This is known as inner loop control (or sometimes closed loop or auto-stabilization). In list form it can:

- Sense changes in attitude.
- Compute the amount and direction of control required.
- Provide the muscle to move the control surfaces using servomotors.
- Detect that the control has been applied and that the aircraft has responded.
- Return control surfaces back to the neutral condition when the disturbance has been corrected.

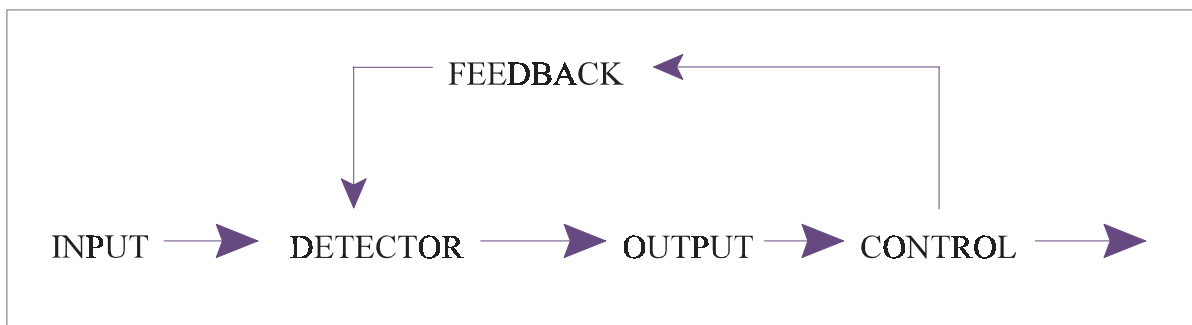
As you will learn in the principles of flight, aircraft will naturally tend to be stable in any case so you may wonder why go to the bother of installing this basic autopilot. Yes, aircraft are naturally stable but you will learn that there are different type of stability and that stability can change with respect to the ambient conditions. The basic autopilot then augments and fine-tunes this stability to provide enhanced stability over a greater range of ambient conditions.

It must be emphasized that the most basic autopilot available will be of a type that will only provide auto-stabilization.

## Control Loops

An autopilot is a control system which uses control loops. The inner loop is a classic example of a closed loop control system (hence one of its common titles). An outer loop interacts with the inner loop to give the aeroplane guidance so as to achieve the required flight path. The basic elements of a closed loop control system are shown in *Figure 26.1* and comprise:

- Input
- Error detector / Signal Processor
- Output
- Control element
- Feedback



*Figure 26.1 Closed Loop Control*



*Figure 26.2 Open Loop Control*

*Figure 26.2* shows an open loop control system which does not have feedback. The controller may consist of a pre-determined programme or a human operator. However, if a human operator is used then the system, in effect, becomes a closed loop system with the human closing the loop and feeding back the output signals.

The difference between open loop and closed loop systems can be illustrated by the domestic central heating system. A system with a timing controller but no thermostat would be an open loop system i.e. the pump would continue to send hot water round the house regardless of the room temperature for the duration of the period set on the timer control. On the other hand, a system with a thermostat would interrupt the circulation when the room temperature reaches the preselected level i.e. it has feedback control.

Feedback control systems used for positional control e.g. aircraft flying controls, are usually referred to as servo systems or servomechanisms. A servomechanism can be defined as a closed loop control system where a small input is converted into a larger output in a strictly proportionate manner.

## Aircraft Inner Loop Control System

The components of an inner (closed) loop control system in an aircraft shown in *Figure 26.3* are:

### *Attitude Sensor*

A rate gyro senses disturbance of the aircraft in one axis only.

### *Transducer*

Converts mechanical movement of the gyro into an electrical signal.

### *Signal Processor*

The error detector. Compares the signals from the transducer with the input signals, determines the required corrective action (the error) and transmits a signal to the servomotor. Receives and compares position and rate of movement feedback signals from the servomotor.

### *Servomotor*

Converts processed signal into movement of the aircraft flight controls proportional to rate and direction of signal. Uses hydraulic, electrical or pneumatic power.

### *Aerodynamic Feedback*

The attitude reached by the aircraft is sensed by the rate gyro which gives a measure of the output.

A disturbance to the selected flight path produces an error signal; the autopilot operates to move the aircraft back towards its stabilized condition. This causes the error signal from the transducer to be progressively reduced and therefore removes the control surface deflection after the disturbance has been corrected.

### *Inner Loop Systems*

Inner loop systems are those that provide the auto-stability only. It is the innermost control loop.

### *Outer Loop Systems*

Outer loop systems are those extra facilities offered e.g. Altitude Hold, Heading Hold, LNAV, VNAV. They are still essentially loops that act from an external position on the inner loop and make the inner loop manoeuvre the aircraft into the position required by the outer loop control.



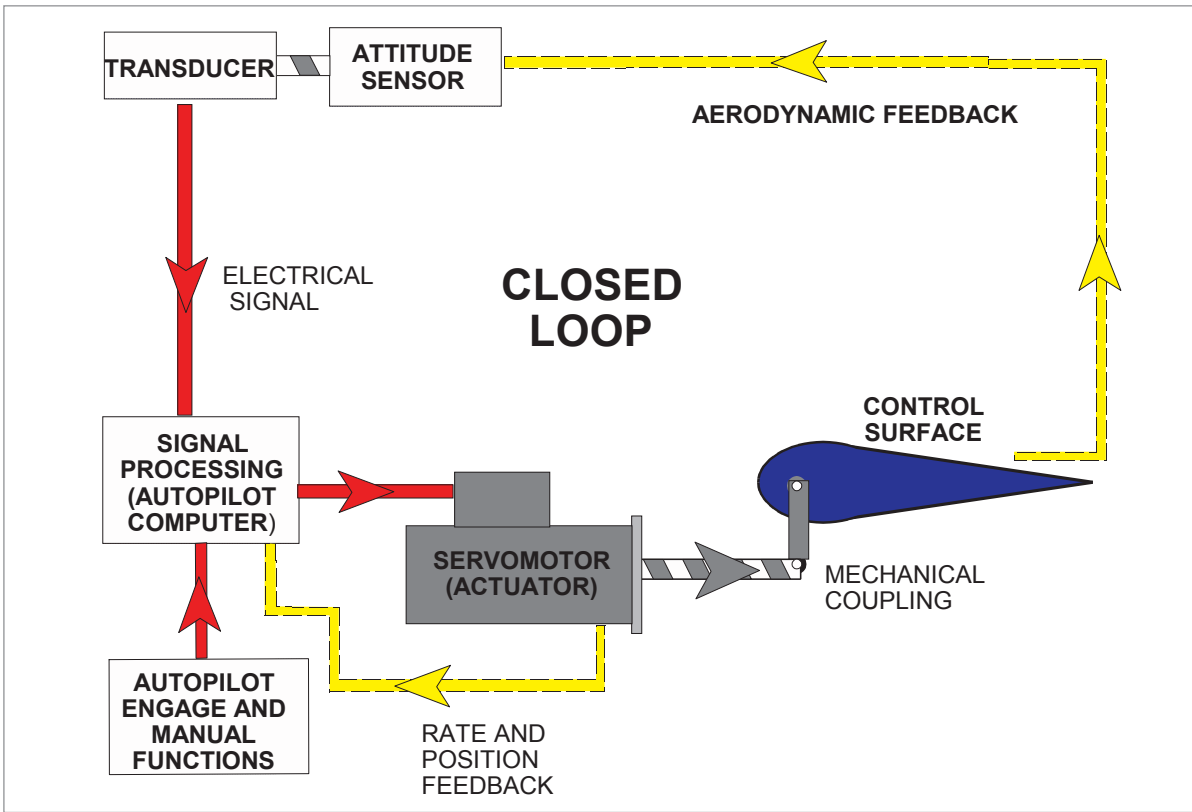


Figure 26.3 Closed loop or inner loop

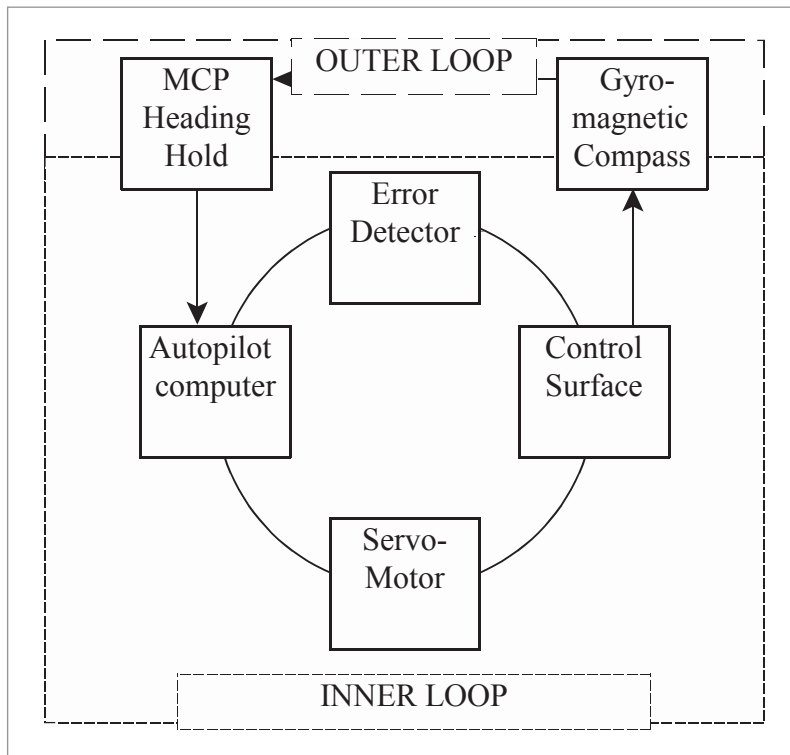


Figure 26.4 Diagram showing the relationship of an outer loop function to the inner loop

## Types of Autopilot

An aircraft can be subjected to disturbances about its three control axes i.e. longitudinal (roll), lateral (pitch) and vertical or normal (yaw). Stabilization must therefore be controlled about the same three axes. Autopilot systems are broken down into three basic control channels:

- Roll           to control the Ailerons
- Pitch         to control the Elevators
- Yaw          to control the Rudder

It is therefore possible to get an autopilot that is classed as single axis, twin axis or triple axis. There will be a separate inner loop for each axis of autopilot control. If an aircraft has more than one autopilot there will be one inner loop for each axis of control for each autopilot.

### *A Single Axis System*

A single axis attitude control system would normally be limited to the roll axis only, i.e. a single autopilot channel controlling the ailerons. At its most basic, the single axis system will only give lateral stability or level the wings. The roll axis is known as the primary axis. This system is sometimes simply called a Wing Leveller.

### *A Two Axis System*

A two axis control system would control the aircraft attitude in the roll and pitch axes. The pitch axis is known as the secondary axis. There are two autopilot channels that control the ailerons and the elevators.

### *A Three Axis System*

A three axis system would give attitude control in all three axes, roll, pitch and yaw. The yaw axis is the third or tertiary axis.

The roll and pitch channels are used as the primary control channels. It is these two channels to which outer loop signals are fed to control the various modes.

The rudder channel is basically a stability channel.

It is common to have interaction between the roll channel and rudder channel to assist in coordinated turns and to give faster stability response.

Three axis control is required for autoland.

## EU-OPS Requirements

Single pilot operation under IFR or at night.

An operator shall not conduct single pilot IFR operations unless the aeroplane is equipped with an autopilot with at least ALTITUDE HOLD and HEADING MODE. This means that the aircraft must have at least a two-axis autopilot.

Installation of automatic pilot system

Each automatic pilot system must be approved and must be designed so that the autopilot can be quickly and positively disengaged to prevent it from interfering with the control of the aeroplane.

Unless there is automatic synchronizing, 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.

Each manually operated control for the system must be readily accessible to the pilots.

Quick release (emergency) controls must be on both control wheels, on the side of each wheel opposite the throttles.

Attitude controls must operate in the plane and sense of motion specified for cockpit controls. The direction of motion must be plainly indicated on, or adjacent to, each control.

The system must be designed and adjusted so that it cannot produce hazardous loads on the aeroplane, or create hazardous deviations in the flight path, either during normal operation or in the event of a malfunction.

If the autopilot 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 is also required.

Means must be provided to indicate to the pilots the current mode of operation and any modes armed by the pilot.

## Types of Actuator

Actuators produce the physical movement of the control surfaces and can be of different types depending on their principle of operation which can be:

- **Electromechanical**
- **Electrohydraulic**
- **Pneumatic**

There are two types of configuration in which actuators are connected to the flying controls:

### Parallel

The actuator produces the movement of the control surface as well as providing feedback to the control stick i.e. the stick will move when the autopilot is controlling the control surfaces.

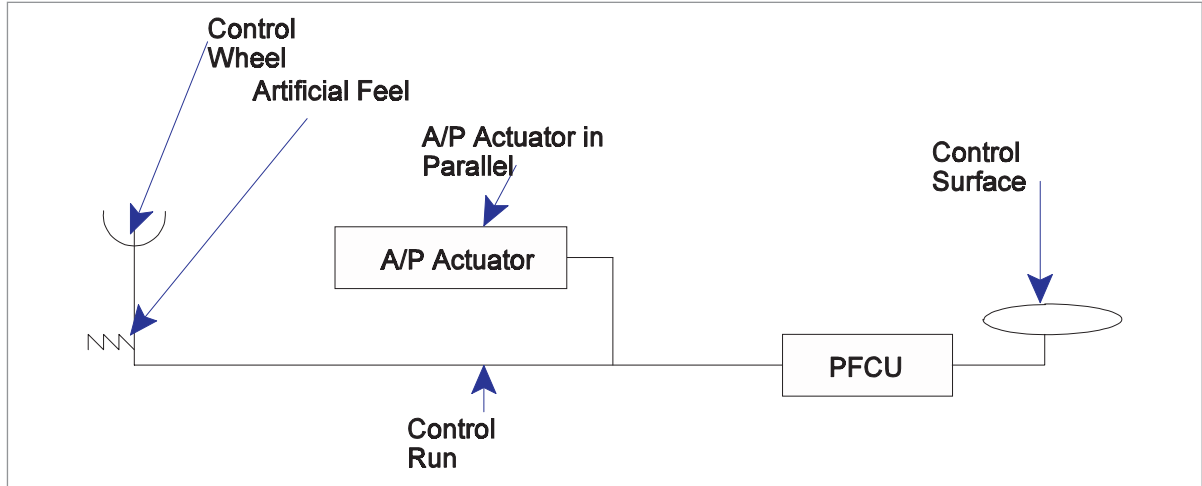


Figure 26.5 A/P Actuator in parallel

### Series

The actuator produces movement of the control surface but not the control stick.

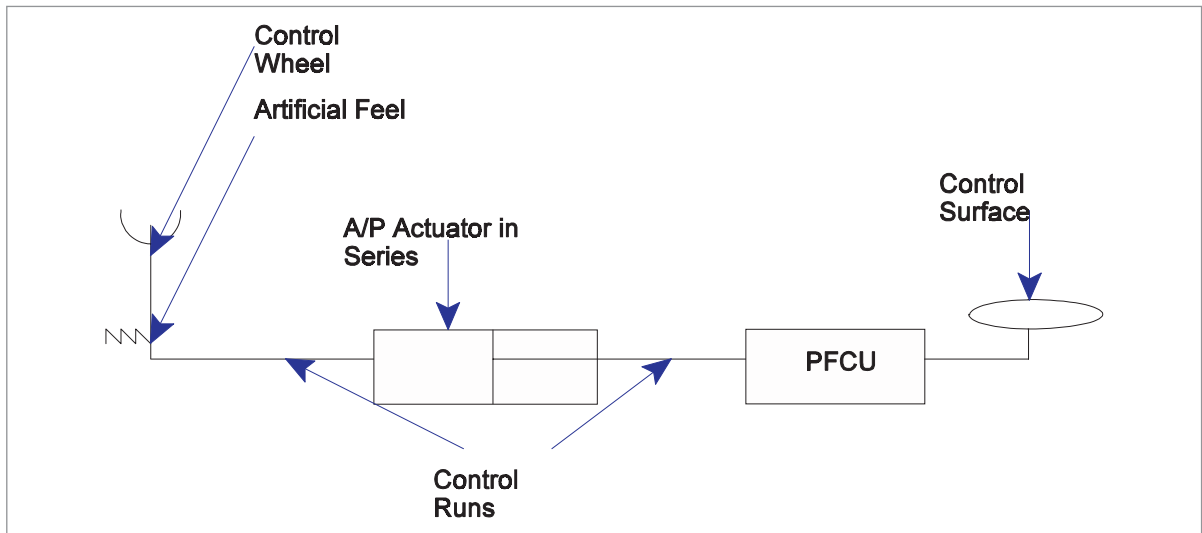


Figure 26.6 A/P Actuator in series

It is also possible to have a combined series/parallel configuration.

### Torque Limiter

In flight, particularly where high rates of control are to be produced, the movement of the flight control surfaces can result in loads which may impose excessive stresses on the aircraft structure. It is necessary therefore, under automatically-controlled flight conditions, to safeguard against such stresses, and furthermore to safeguard against a servomotor 'runaway' condition which would cause control surfaces to be displaced to their maximum hard-over positions.

Such safeguards are implemented by limiting the torque applied to the servomotors, and also by allowing them either to slip, or to be completely disengaged, in the event that present torque limits are exceeded. The methods adopted usually depend on either mechanical, electrical or electromechanical principles.

## Engagement Criteria

### Autopilot Interlocks

Before coupling an autopilot with the aircraft's control system the integrity of the Autopilot Inner Loop must be established to ensure that it may safely take control of the aircraft. To monitor the performance of the inner loop components a system of interlocks is provided which close to allow autopilot engagement and hold it engaged if the correct valid signals have been received. The function of the interlocks can be represented by a number of relays in series (see [Figure 26.7](#)), although in modern aircraft the actual switching is more likely to be accomplished by solid state logic switching. Failure of a circuit monitored by a relay will cause the autopilot to disengage accompanied by the associated aural and visual warning indications. Operation of the disengage switch will have the same effect.

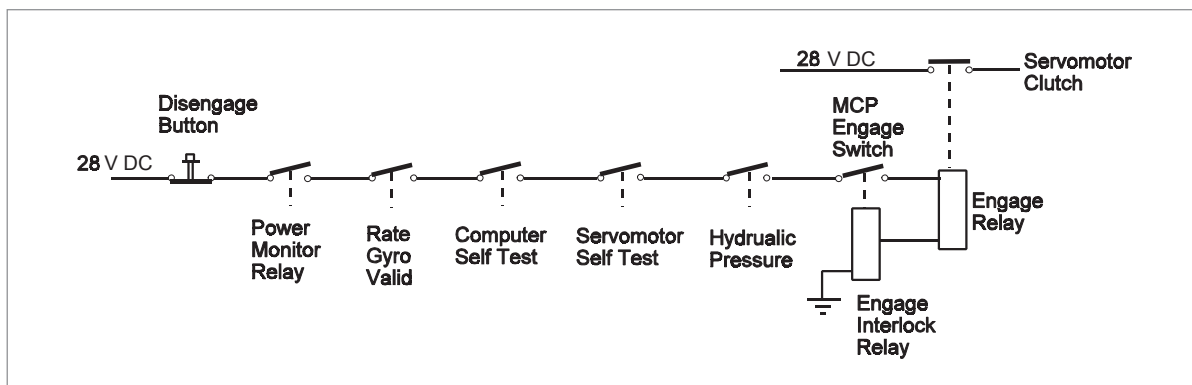


Figure 26.7 Autopilot interlocks

### Conditions of Engagement

Before the autopilot can be engaged, certain conditions must be met. These conditions vary with aircraft type. For the 737-400 these conditions are as follows.

Each autopilot can be engaged by pressing a separate CMD or CWS engage switch. A/P engagement in CMD or CWS is inhibited unless both of the following pilot-controlled conditions are met:

- No force is being applied to the control wheel.
- The Stabilizer Trim Autopilot Cut-out Switch is at NORMAL.

Once the above conditions are satisfied and no failures exist, either A/P can be engaged in CMD or CWS by pressing the respective engage switch. Control pressure applied after an A/P is engaged in CMD overrides the A/P into CWS pitch and/or roll. The light remains illuminated in the CMD engage switch.

The A/P automatically disengages when any of the following occur:

- Pressing either A/P disengage switch.
- Pressing either TOGA switch with a single A/P engaged in CWS or CMD below 2000 ft RA.
- Pressing either TOGA switch after touchdown with both A/Ps engaged in CMD.
- Pressing a lighted A/P engage switch.
- Pushing the A/P disengage bar down.
- Activating either pilot's control wheel trim switch.
- Moving the Stabilizer Trim Autopilot Cut-out Switch to CUT-OUT.
- Loss of respective hydraulic system pressure.
- Repositioning the EFI transfer switch.
- Either left or right IRS system failure or FAULT light illuminated.
- Loss of electrical power or a sensor input which prevents proper operation of the engaged A/P and mode.

Only one A/P can be engaged at a given time unless the approach (APP) mode is engaged. Pressing an engage switch for the second A/P, while not in the APP mode, engages the second autopilot as selected and disengages the first A/P. The second A/P then operates in CWS or CMD without interrupting CWS or command operation.

If an A/P is engaged with the CMD engage switch during FD only operation while pitch or roll commands are more than ½ scale from centred, the A/P automatically engages in CWS for pitch and/or roll and the FD command bars retract.

### **Automatic Synchronization**

In addition to the pre-engage requirements that the autopilot circuits are electrically complete, it must also be ensured that on engagement the 'take-over' is effected smoothly and without 'snatching' of the aircraft's control system. In other words the aircraft must be trimmed for the desired flight attitude before engagement and the automatic control system must be synchronized to maintain that attitude on engagement.

In the majority of autopilot systems synchronization is effected by specifically designed synchronizing circuits which automatically sense any existing 'standing signals' in the pitch and roll channels and automatically reduce or 'wash out' these signals to zero. This stops the servo actuator in a position which is synchronized with the datum attitude detected by the sensing element, such position being indicated by the return of the trim indicator pointer to its central position.

## Trim

### *Manual Systems*

The purpose of the trim system is to relieve the pilot of forces on the aircraft controls while maintaining aircraft attitude. In manual control, trim on light aircraft is provided in all three axes through mechanical linkages to trim tabs on the control surfaces. On larger aircraft this is usually achieved by electrical actuators that bias the Powered Flying Control Unit (PFCU), particularly for pitch trim. Trim steering signals would be provided to the trim tab actuators for elevator, aileron and rudder as well as to the horizontal stabilizer.

Manual operation of the pitch trim will, in most systems, automatically disconnect the autopilot as it cannot co-ordinate manual trim movement with movement of the C of G or aerodynamic movements.

### *Automatic Trim (Auto-trim)*

As the aircraft uses fuel, or changes speed, thrust or configuration, the pitch attitude will change. If we are flying the aircraft manually we would trim these forces out manually to eliminate stick forces. Currently as our basic autopilot stands if it is flying the aircraft and the pitch requirement changes it will simply hold the stick forces using the brute strength of its servomotor outputs. This will not pose much of a problem unless the forces overwhelm the servomotor or until such time as we wish to disconnect the autopilot. Not only does this mean that the aircraft is producing more drag than necessary but if there is a standing load on the controls when we disconnect the autopilot we will not know of its existence or which way the aircraft will pitch. The aircraft will “snatch” or lurch in response to the out of trim condition. This is not a very satisfactory situation so a system of Automatic Pitch Trim was included in most autopilot systems.

When the autopilot is engaged automatic trim is available only in pitch. This is called Automatic Pitch Trim or simply Auto-trim. Auto-trim is active only when the autopilot is engaged. Fly-by-wire arrangements will be explained later. It is typically achieved by a separate trim servo actuator operating either the normal trim tab or, more commonly on modern jet transports, the variable incidence horizontal stabilizer. The latter permits the elevator to always be in neutral position with respect to the horizontal stabilizer, therefore allowing the autopilot full elevator control authority both sides of the trimmed position. Another important consideration is that in the event of autopilot disconnect the aircraft will be in a trimmed condition and thus will not suddenly pitch up or down.

The out of trim condition is sensed most commonly by using one of two methods:

- A standing load being sensed on an electrical actuator. Due to the out of trim situation the autopilot is having to hold a force against the out of trim condition in exactly the same way as a pilot would have to. The magnitude of the load on the actuator is going to be directly proportional to the force being held, and of course the direction is known from which way the actuator is having to apply that load. That information can be used to move the normal trimming system of the elevators to reduce the standing load to zero. The aircraft is now in trim.

- The actual position of the actuator. If there is an out of trim force to be coped with the actuator will be positioned to input the required control displacement to hold that force. The displacement of the actuator will again give the direction and magnitude of the force. Large modern aircraft tend to use Trim Tabs or all moving tail-planes. Both of these systems, due to the way they trim, reset the control inputs to give full elevator movement up and down from the trimmed position. So now, as the auto-trim moves the normal elevator trim, the displacement of the normal control input is reset, removing the actuator displacement.

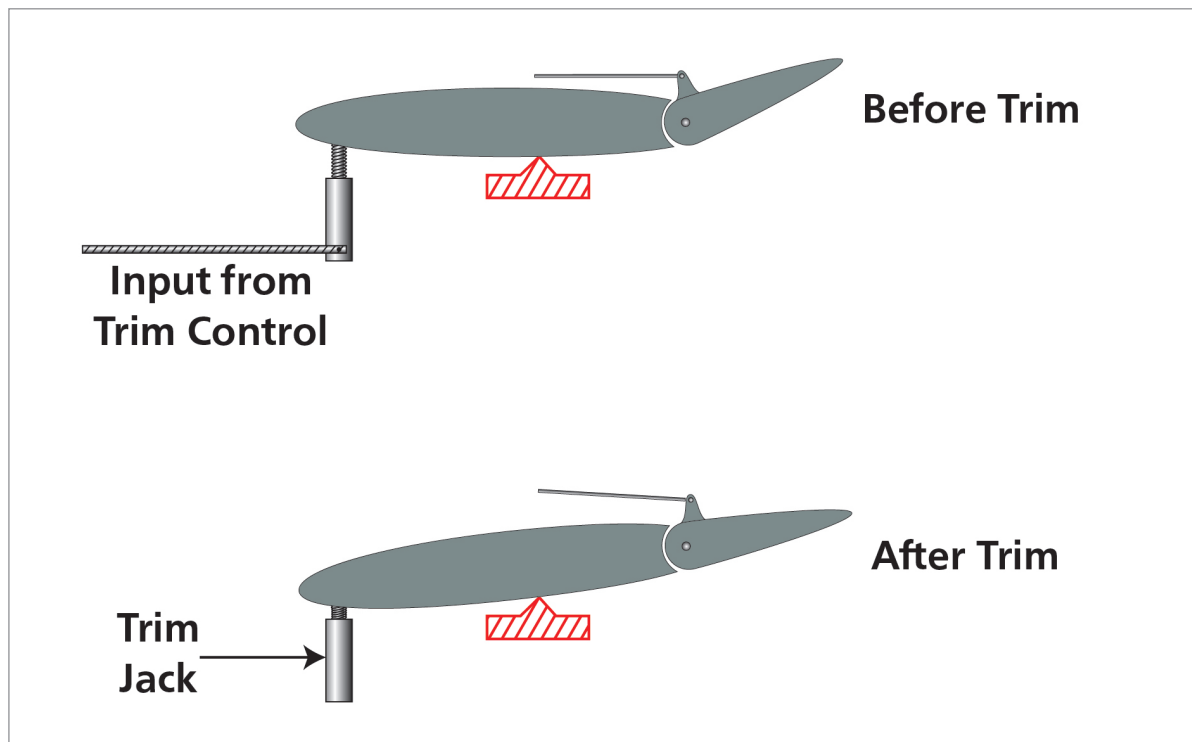


Figure 26.8 Trimming by variable incidence tailplane

The output is applied to the trim tab actuator or the horizontal stabilizer.

In the event of a failure of the trim system the pilots would be alerted by warning lights and/or suitable indications on the electronic display unit. At the same time the autopilot would disengage, giving both visual and aural alerts. The latter can also sound when there is an excessive trim input as, for example, in an actuator runaway situation. If the auto trim system is not available, then the autopilot may become inoperative (will become inoperative for the B737-400). If another autopilot is available, it would be common practice to use the fully operative system.

If there is only one autopilot or the aircraft is not fitted with auto-trim, the aircraft must be correctly in trim before the engagement of the autopilot to minimize the control loading expected on disengagement. The standard operating procedure for the aircraft will stipulate a time period after which the autopilot must be disengaged, the aircraft re-trimmed and then the autopilot re-engaged. This will minimize the control snatch on autopilot disengagement.

The pilots may have some indication of the trim controls but in the case of auto-trim there is always a stabilizer trim indicator and an auto-trim failure warning so that the system can be monitored during autopilot operation. The autopilot may not engage if there is too great a standing load, i.e out of trim condition, already present.



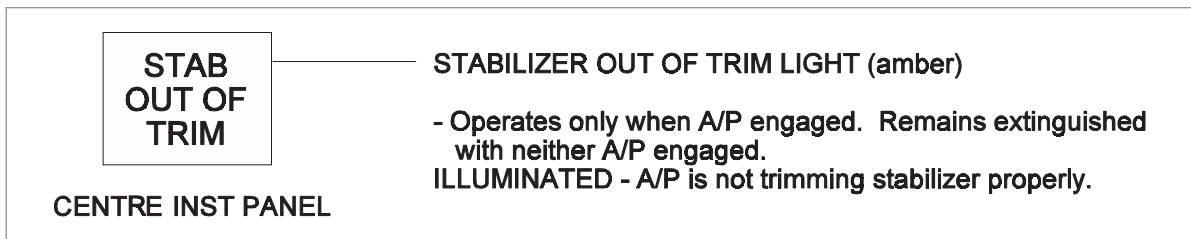


Figure 26.9 Auto-trim Failure Light

### Independent Systems

Mach trim (if required) will operate independently of the autopilot. A Mach trim system is provided in aeroplanes that fly at high subsonic speeds and are susceptible to Mach tuck. At these speeds as the aeroplane approaches its critical Mach No. the centre of pressure moves aft resulting in a nose-down attitude known as Mach tuck. This condition is automatically trimmed out by a Mach trim system. The Mach trim system will be armed at all stages of flight but will only activate at high subsonic speeds.

Yaw dampers will be covered in detail in another chapter. They are, however, another example of an inner loop system.

### Fly-by-Wire

Fly-by-wire aeroplanes function differently to conventional aeroplanes in that the pilot or autopilot input is via a flight control computer. The computer translates this input into an aircraft attitude using all three axes, which it will then maintain until another input is made. Even if configuration, thrust or speed changes are made, which would affect the pitch attitude of the aeroplane, the flight control computer will maintain the aircraft attitude selected.

Any load being maintained by the flight control computer will be trimmed on a regular basis so the aeroplane is always in trim. For example in pitch, the flight control computer will deflect the elevators, using an electric servomotor to reach the correct aircraft attitude as selected by the pilot or autopilot. After a short period of time this would be trimmed using the horizontal stabilizer allowing the elevator to return to its neutral position ready for any future use.

There are mechanical trim backups in pitch and yaw which will override the automatic flight control computer trim. The pitch trim is operated by moving the pitch trim wheel which will cause the horizontal stabilizer to move, the autopilot to disconnect and override the flight control computers pitch attitude.

### Outer Loop Control (also Known as Flight Path Modes)

In addition to performing the primary function of stabilization, an Automatic Flight Control System (AFCS) can also be developed to control the path of the aircraft horizontally or vertically to predetermined conditions. For example to hold a selected airspeed, altitude, magnetic heading or intercept and track radio beams from ground-based aids, etc. The number of inputs available serve as an indication of the progressive development of automatic flight from the basic single-axis wing-levelling type of autopilot, to the highly sophisticated flight guidance systems now used in many present-day transport aircraft.

The outer loop inputs are applied to the inner loop in such a way as to fool the auto-stability control into believing that the aircraft is being disturbed. The inner-loop systems reaction to

the input is calculated to produce the result required, such as altitude holding or turning to follow a heading.

Such data inputs constitute **Outer Loop Control** and can be referred to as **Command Modes** or **Flight Path (Referenced) Modes**. Outer loop command modes are coupled to the relevant autopilot channel by selection on the Mode Control Panel (MCP) which is located on the glare-shield and provides the pilot's interface with the auto-flight system. A typical example of modern twin jet MCP is shown in *Figure 26.12*. It allows the pilot to engage an autopilot and select any of the pitch and roll outer loop inputs. Incorporated in the selector switches are lights to indicate which autopilot or command modes are selected.

Only one command mode may be engaged in a single channel at any one time (i.e. one in roll and one in pitch). For example it is impossible for the autopilot to hold speed and maintain altitude at the same time by pitching the aeroplane. If the aeroplane had an autothrottle the autopilot could hold altitude while the autothrottle maintained speed.

The provision of raw data inputs relevant to a particular flight path is referred to as 'coupling' or as a 'mode of operation'. Other terms commonly used in connection with operating modes are 'hold' and 'capture'. For example, an aircraft flying automatically at a selected altitude is said to be in the 'altitude hold' mode. The term 'capture' relates principally to modes associated with the selection and interception of beams from ground-based radio navigation aids; for example, 'glide slope capture'.

In some cases, mode switching is automatic; thus, to switch from intercepting a beam or a heading, to tracking the beam on reaching it, a **Beam Sensor** is installed. This device senses beam deviation and switches modes automatically when the aircraft flies into the beam. Glide slope capture can also take place automatically, in this case the pitch control channel is switched from 'altitude hold' mode to glide slope track when the aircraft flies into the glide slope beam.

The raw data is supplied from aircraft sensors (attitude, air data, heading, radio etc.) to the relevant auto-flight computer which compares the data with the selected values on the MCP and computes control inputs to achieve those selected values.

In a modern transport aircraft which is using a flight guidance system with an automatic landing capability, the outer loop inputs could comprise some or all of the modes listed in the following table.

### *Roll Channel*

- Heading hold
- Heading select
- VOR intercept and track
- LOC intercept and track
- Inertial Nav or L NAV

### *Pitch Channel*

- Altitude hold
- Speed hold
- Mach hold
- Vertical Speed
- V NAV

## Autoland

Roll Channel	Pitch Channel	Yaw Channel
Localizer	Glide slope	Runway align
Roll-out	Flare	Roll-out

## Mode Annunciator

The Mode Annunciator will indicate the current auto-flight system status and can be a separate indicator or an integrated part of the EFIS primary flight display (*Figure 26.10*). The electronic display indicates armed and engaged modes of the auto-flight system in different colours. It can also indicate autopilot, autothrottle, autoland and flight director status.

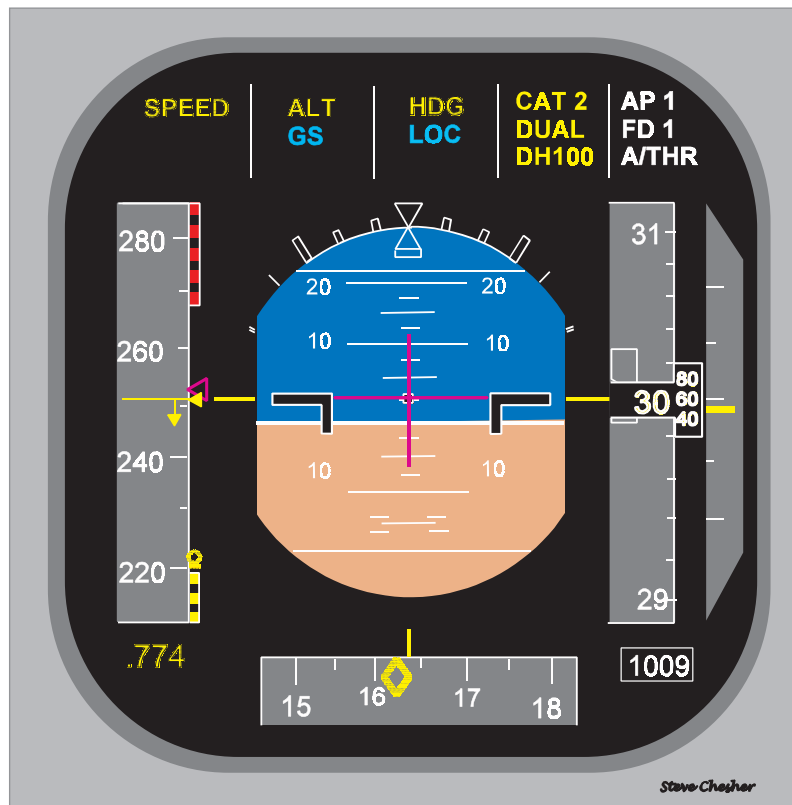


Figure 26.10 Primary Flight Display

There will also be illuminated switches or simply warning lights that will indicate whether an autopilot is engaged, warning of disengagement, warning of autothrottle disengagement, failure to achieve target speed and of auto-trim failure.

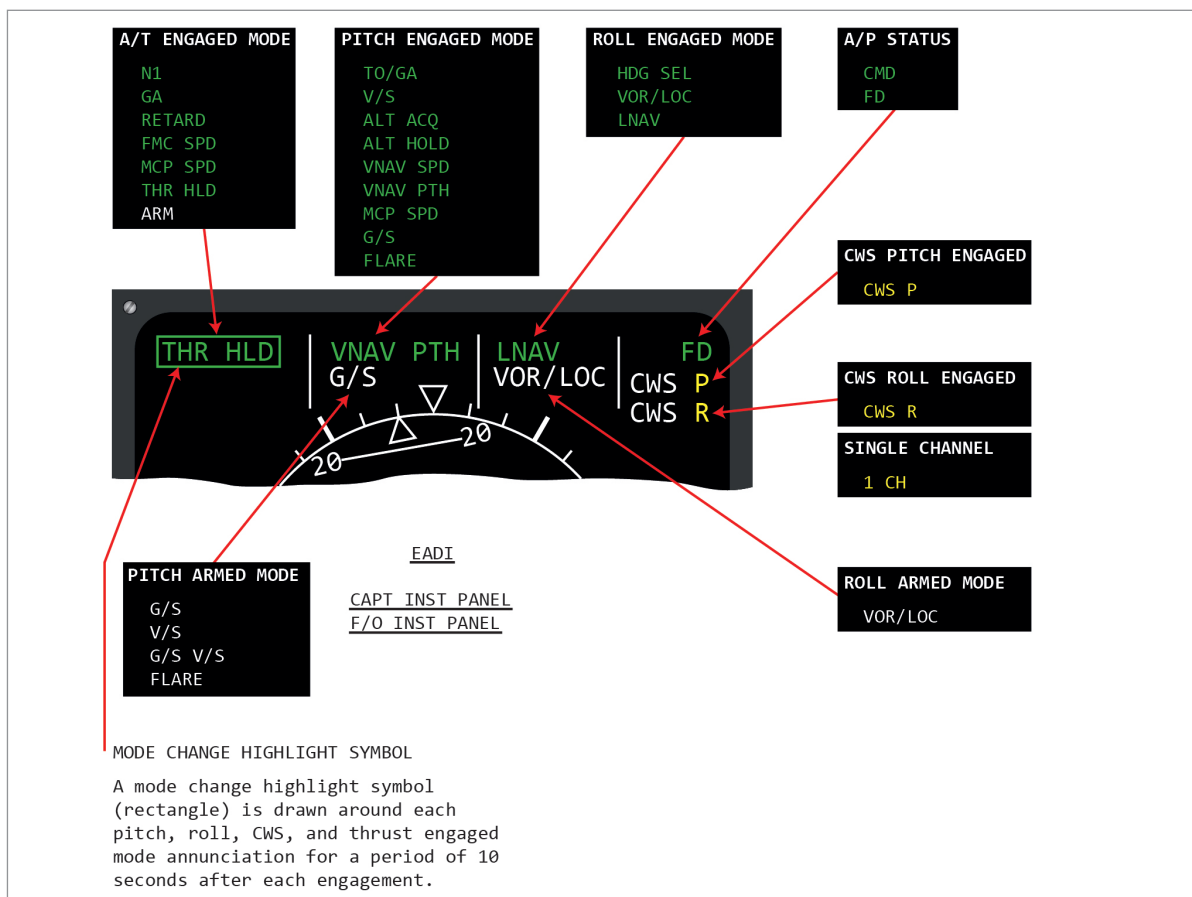


Figure 26.11 Boeing 737-400 Flight Mode Annunciations

### Aircraft Sensor Inputs to the Autopilot Computer

#### Manometric (or Air) Data

Raw data inputs which come under this heading are those associated with altitude, airspeed/Mach No. and vertical speed. Each of these provides current aircraft status for outer loop control in the pitch channel of the autopilot.

Sensing may be carried out either by independent sensor units, or by a Central Air Data Computer (CADC). The sensors operate on the same fundamental principles as the basic pitot-static instruments, the measuring elements being coupled to appropriate types of electrical transducers instead of instruments.

#### Attitude Reference

Attitude reference data (roll, pitch, yaw) is fed into the auto-flight computer from the primary attitude sensors which could be a Vertical Gyro and Directional Gyro combination, an Inertial Navigation System or an Inertial Reference System depending on the age of the aircraft. These sensors may also transmit data to the ADI and HSI.

#### Magnetic Heading Reference

A Magnetic Heading Reference System (MHRS or gyro-magnetic compass) combines inertial heading with magnetic compass heading providing magnetic heading signals to the HSI and reference data to the auto-flight computer.

### Radio Navigation

To allow the auto-flight system to be able to capture and track a radio beam data signals are transmitted to the relevant auto-flight computer from the VOR and ILS receivers.

### Computer Generated Data

Modern aircraft can follow a computer generated flight profile in both roll (lateral) and pitch (vertical) from a Flight Management System (FMS). Steering signals from the Flight Management Computer (FMC) are connected to the auto-flight computer to control the attitude of the aircraft.

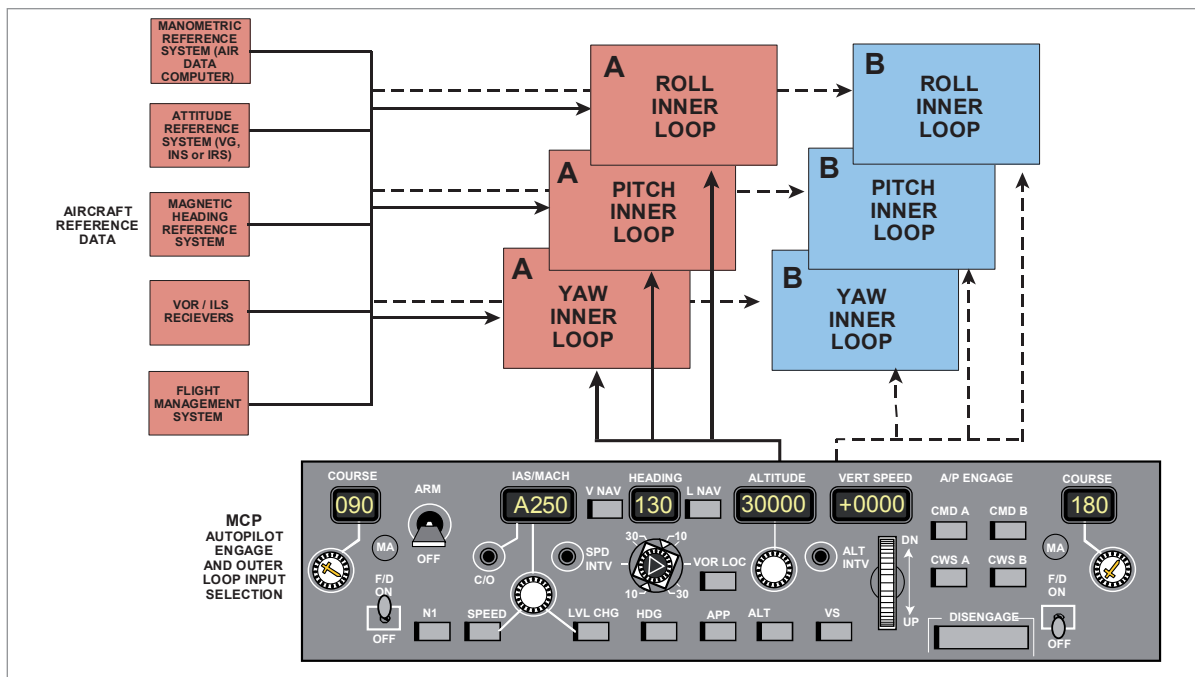


Figure 26.12 Inputs to the MCP

### An Example of an AFDS (737-400)

For these notes the Boeing 737-400 is used as an example. It is also officially the AFDS that the JAR objectives and exams are currently based around. Therefore a description of the AFDS as fitted to this aircraft is given.

#### General

The Automatic Flight System (AFS) for the 737-400 consists of the Autopilot Flight Director System (AFDS) and the Auto throttle (A/T). The Flight Management Computer (FMC) provides N1 limits and target N1 for the A/T and command airspeeds for the A/T and AFDS.

The AFDS and A/T are operated from the AFDS Mode Control Panel (MCP) and the FMC from the Control and Display Unit (CDU).

The AFDS MCP provides co-ordinated control of the autopilot (A/P), Flight Director (FD), A/T and altitude alert functions.

AFS mode status is displayed on the Flight Mode Annunciators (FMA) on each pilot's PFD.

Normally, the AFDS and A/T are used to maintain airspeed and thrust settings calculated by the FMC.

### Autopilot Flight Director System (AFDS)

The AFDS is a dual system consisting of 2 individual Flight Control Computers (FCCs) and a single MCP.

The 2 FCCs are identified as A and B. For A/P operation, they send control commands to their respective pitch and roll hydraulic servos, which operate the flight controls through 2 separate hydraulic systems.

For FD operation, each FCC positions the FD command bars on the respective PFD.

### MCP Mode Selector Switches

The mode selector switches are pressed to select desired command modes for the AFDS and A/T. The switch illuminates to indicate mode selection and that the mode can be deselected by pressing the switch again. While a mode is active, de-selection can be automatically inhibited and this is indicated by the switch light being extinguished.

When engagement of a mode would conflict with current AFS operation, pressing the mode selector switch has no effect. All AFDS modes can be disengaged by selecting another command mode or by disengaging the A/P and turning the FDs off.

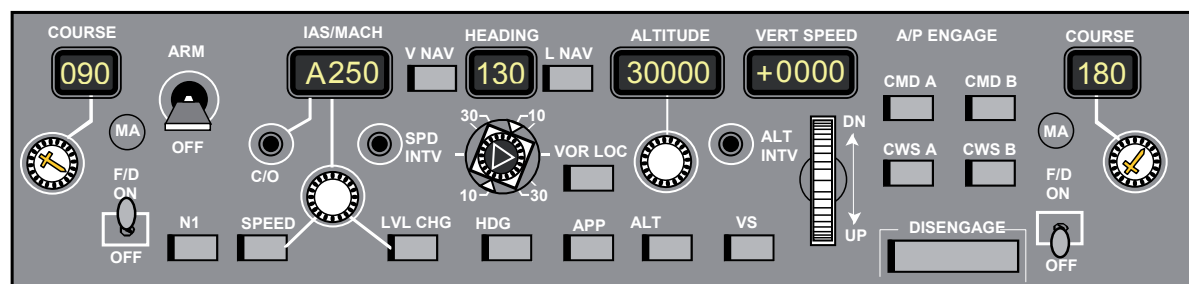


Figure 26.13 Mode control panel

### MCP Parameter Selection

The following information is in addition to that contained in the controls and indicators section of this chapter.

Parameter selections common to both FCCs for speed, heading, altitude and vertical speed are made from the MCP.

Two course selectors and course displays are located on the MCP. The Captain's course selector provides selected course information to the A FCC, the No. 1 VHF Nav receiver and to the Captain's HSI course pointer and course deviation bar. The First Officer's course selector provides selected course information to the B FCC, the No. 2 VHF Nav receiver and to the First Officer's HSI course pointer and deviation bar.

## Examples of Outer Loop Inputs in Roll

### Heading Select and Hold

The heading select mode sends roll commands to turn and maintain the heading shown in the MCP heading display. After mode engagement, roll commands are given to turn in the same direction as the rotation of the heading selector only. The bank angle limit is established by the Bank Angle Limit Selector on the MCP. Pressing the Heading Select Switch on the MCP engages the heading select mode. HDG SEL is annunciated for the AFDS.

The HDG SEL mode automatically disengages upon capture of the selected radio course in the VOR LOC and APP modes.

### ***VOR Localizer Tracking (VOR LOC) Mode***

The VOR mode gives roll commands to capture and track the selected VOR course. The LOC mode gives roll commands to capture and track the selected localizer along the inbound front course bearing. Back-course tracking is not available.

Pressing the VOR LOC switch selects the VOR mode if a VOR frequency is tuned, or selects a LOC mode if a localizer frequency is tuned. The VOR LOC switch illuminates and VOR LOC armed is annunciated.

The selected course can be intercepted while engaged in L NAV, HDG SEL or CWS ROLL, with an autopilot engaged in CMD. The capture point is variable and depends on intercept angle and closure rate. Localizer capture occurs not later than ½ dot deviation. When within the course capture area, the VOR LOC annunciation changes from armed to captured and roll commands track the VOR or localizer course.

### ***Tracking Through VOR "Cone of Confusion"***

The 'cone of confusion' is an area overhead a VOR navigation beacon where the signals are unusable. Thus an aircraft transiting the VOR will receive no usable signals for a period depending upon its ground speed and altitude.

As the aircraft approaches the VOR the radials are converging and the course deviation indicator becomes more sensitive. At some point, before it enters the cone of confusion, the information from the selected inbound radial becomes unusable due to the convergence. At this point the VOR signals are 'cut off' by the 'over station sensing' circuits i.e. the roll channel automatically de-couples from the radio beam and controls the aircraft through the cone of confusion on the drift-corrected heading existing when the radio signals are de-coupled. In other words the autopilot goes into Heading Hold for a set period after which it reverts to the VOR Mode. Note that the autopilot does not go into Heading Mode.

### ***Inertial Navigation System (INS)/Inertial Referencing System (IRS)***

Older aircraft without a Flight Management System may be able to couple the Inertial Navigation/Reference System to the autopilot to allow the aircraft to be steered sequentially through a series of waypoints that have been loaded into the INS/IRS before flight.

### ***Lateral Navigation (L NAV)***

In the L NAV mode, the FMC controls AFDS roll to intercept and track the active FMC route. The desired route is activated and modified through the FMC CDUs. In addition to en route guidance, the active routes can include terminal procedures such as SIDs, STARs and instrument approaches.

Engagement criteria must be met to use L NAV. There must be an active route in the FMC, capture criteria must be satisfied and the L NAV switch must be pressed.

L NAV capture criteria is divided into 2 categories. First, any aeroplane satisfies capture criteria when within 3 NM of the active route segment. Second, outside of 3 NM, the aeroplane must be on an intercept course of 90 degrees or less and intercept the active route segment before the active waypoint.

L NAV will automatically disconnect for several reasons. It will disconnect upon reaching the end of the active route or upon entering a route discontinuity. Additionally, it will disconnect upon either intercepting or missing the intercept of an approach path inbound track. Finally, either loss of capture criteria or selecting HDG SEL will disconnect L NAV.

## Examples of Outer Loop Inputs in Pitch

### *Altitude Hold (ALT HLD)*

The altitude hold mode gives pitch commands to hold the MCP selected altitude or the uncorrected barometric altitude at which the ALT HOLD switch was pressed. ALT HOLD engages in either of 2 conditions;

**ALT HOLD at the MCP selected altitude.** This is indicated by the annunciation of ALT HOLD and the ALT HOLD switch light extinguished.

**ALT HOLD not at the MCP selected altitude.** This is indicated by the annunciation of ALT HOLD and the ALT HOLD switch light illuminated.

ALT HOLD not at the MCP selected altitude occurs with either of the following:

- Pushing the ALT HOLD switch while not at the MCP selected altitude.
- Selecting a new MCP altitude while in ALT HOLD at the currently selected altitude.

ALT HOLD is inhibited after glide slope capture. When in ALT HOLD at the selected altitude, LVL CHG, V/S and V NAV climb and descend functions are inhibited until a new altitude is selected.

The altitude selected on the MCP is referenced to the Captain's barometric altimeter setting for the "A" autopilot and FDS, and to the First Officer's barometric setting for the "B" autopilot and FDS. After ALT HOLD engages, changes in the altimeter barometric settings do not change the selected altitude reference.

### *Altitude Acquire (ALT ACQ)*

The altitude acquire mode is the transition manoeuvre entered automatically from a V/S, LVL CHG, or V NAV climb or descent to a MCP selected altitude. The altitude acquire mode is also armed while climbing or descending in CWS with an autopilot engaged.

Altitude acquire engagement is annunciated ALT ACQ in pitch when levelling off in either V/S or LVL CHG. However, V NAV remains annunciated throughout the altitude acquire mode when levelling in V NAV.

ALT ACQ engagement is inhibited when the ALT HOLD switch is pressed or while the glide slope is captured.

### *IAS/MACH Hold (SPD)*

This will hold a selected IAS or Mach No. by comparing the selected value with the actual value from the ADC and pitching the aircraft up or down to decrease or increase speed.

### *Vertical Speed (V/S)*

The V/S mode gives pitch commands to hold the selected vertical speed and engages the auto-throttle in the SPEED mode to hold the selected airspeed. The V/S mode has both an armed and engaged state.



Pressing the V/S switch engages the V/S mode unless the ALT HOLD is engaged or after glide slope capture.

V/S engaged is annunciated, the Vertical Speed Display changes from blank to present vertical speed and desired vertical speeds can be selected with the vertical speed thumb-wheel.

The V/S mode becomes armed if, while in the ALT HOLD at the selected altitude, a new MCP altitude is selected which is more than 100 ft different than the previously selected altitude. V/S armed is annunciated and the V/S mode can be engaged by moving the vertical speed thumb-wheel.

The V/S mode automatically engages when the altitude acquire mode is engaged and a new altitude is selected which is more than 100 ft different than the previously selected altitude. The V/S mode annunciates engaged and existing vertical speed appears in the vertical speed display. The commanded V/S can be changed with the vertical speed thumb-wheel. Vertical speeds can be selected which command flight toward or away from the selected altitude.

### **Level Change Mode (LVL CHG)**

The LVL CHG mode co-ordinates pitch and thrust commands to make automatic climbs and descents to preselected altitude at selected airspeeds. A LVL CHG climb or descent is initiated by selecting a new altitude and engaging the LVL CHG mode.

During a LVL CHG climb, the annunciations are MCP SPD for pitch and N1 for the autothrottle (A/T). During a LVL CHG descent, the annunciations are MCP SPD for pitch and RETARD for the A/T while reducing the thrust toward idle. When at idle thrust, ARM is annunciated for the A/T.

If a speed mode was active prior to engaging LVL CHG, the previous speed is retained as the target speed for the LVL CHG mode. If the LVL CHG mode is engaged with no active speed mode, the IAS/Mach display and airspeed cursors synchronize to existing speed and present speed becomes the LVL CHG target speed. After LVL CHG mode engagement, the target speed can be changed with the MCP speed selector.

### **Vertical Navigation Mode (V NAV)**

With the V NAV mode engaged, the FMC commands AFDS pitch and A/T modes to fly the vertical profile selected on the FMC CDUs. The profile includes preselected climbs, cruise altitudes, speeds, descents and can also include altitude constraints at specified waypoints. The profile may end with an ILS approach to the destination airfield.

Pressing the V NAV switch selects the V NAV mode provided FMC performance initialization is complete. The mode selector switch illuminates, the MCP IAS/Mach display becomes blank and the airspeed cursors are positioned at the FMC commanded airspeed. The FMA displays are V NAV SPD or V NAV PTH for the AFDS pitch mode and FMC SPD, N1, RETARD or ARM for the A/T mode.

V NAV climbs and descents are constrained by the selected MCP altitude. V NAV commanded speeds can be changed with the FMC CDUs.

During V NAV path cruise flight, selecting a lower MCP altitude arms the FMC to automatically begin the descent upon arrival at the FMC calculated top of descent point.

During a V NAV path descent, V NAV remains engaged until:

Glide slope capture, or

Another pitch mode is selected, or

Flaps are extended beyond 15°, or

L NAV is disengaged without glide slope capture.

Proper MCP altitude selections ensure correct altitude alerting.

### Autopilot in Operation

The modes of operation of the autopilot during the various flight phases can be seen from the following table:

PHASE	ROLL	PITCH	AUTOTHROTTLE
Take-off	TOGA	TOGA	THR REF
Climb	LNAV, HDG or VOR	FLCH SPD, VNAV or V/S	THR REF, SPD or MACH
Cruise	LNAV, HDG or VOR	ALT HOLD, VNAV	SPD or MACH
Descent	LNAV, HDG or VOR	FLCH SPD, VNAV or V/S	THR REF, SPD or MACH
Approach	LNAV, HDG or LOC	ALT, G/S, FLARE	SPD
Land	ROLLOUT		
Go-around	TOGA	TOGA	THR REF

### Other Autopilot Features

#### Roll and Pitch Modes

Approach mode (APP) allows localizer and glide slope elements of the ILS system to be coupled to the roll and pitch channels of the autopilot to allow automatic control down to decision height, or to effect a fully automatic landing.

#### Control Wheel Steering (CWS)

A Control Wheel Steering mode (CWS) is provided in some automatic flight control systems (Boeing 737-400), its purpose being to enable the pilot to manoeuvre his aircraft in pitch and/or roll, through the automatic control system without disconnecting the autopilot. The signals for the A/P are produced by transducers in the control column.

Pressing a CWS engage switch, engages the A/P pitch and roll axes in the CWS mode and displays CWS P and CWS R on the FMAs.

With CWS engaged, the A/P manoeuvres the aeroplane in response to control pressures applied by either pilot. The control pressure is similar to that required for manual flight. When control pressure is released, the A/P holds existing attitude.

If aileron pressure is released with 6 degrees or less angle of bank, the A/P rolls the wings level and holds the existing heading. This heading hold feature with bank less than 6 degrees, is inhibited when any of the following conditions exist:

- Below 1500 ft RA with the landing gear down.
- After FD VOR capture with TAS 250 kt or less.
- After FD LOC capture in the APP mode.

Pitch CWS with a CMD Engage switch selected

The pitch axis engages in CWS while the roll axis is in CMD when:

A command pitch mode has not been selected or was de-selected.

A/P pitch has been manually overridden with control column force. The force required for override is greater than normal CWS control column force. This manual pitch override is inhibited in the APP mode with both A/Ps engaged.

CWS P is annunciated on the Flight Mode Annunciators (FMA) while this mode is engaged. Command pitch modes can then be selected.

When approaching a selected altitude in CWS P with a CMD engage switch selected, CWS P changes to ALT ACQ and when at the selected altitude, ALT HOLD engages.

If pitch is manually overridden while in ALT HOLD, the annunciator changes to CWS P. If control force is released within 250 ft of the selected altitude, CWS P changes to ALT ACQ and the A/P returns to the selected altitude and ALT HOLD engages. If the elevator force is held until more than 250 ft from the selected altitude, pitch remains in CWS P.

Roll CWS with a CMD engage switch selected

The roll axis engages in CWS while the pitch axis is in CMD when:

A command roll mode has not been selected or was de-selected.

A/P roll has been manually overridden with control wheel force. The force required for override is greater than the normal CWS control wheel force.

CWS R is annunciated on the FMA while this mode is engaged.

CWS R with a CMD engage switch illuminated can be used to capture a selected radio course while VOR/LOC or APP mode is armed. Upon intercepting the radial or localizer, the FD and A/P annunciation changes from CWS R to VOL/LOC engaged and the A/P tracks the selected course.

### ***Touch Control Steering (TCS)***

Touch Control Steering (TCS) also permits a pilot to manoeuvre his aircraft in pitch or roll, but unlike CWS the appropriate automatic control channels and servomotors are disengaged while the TCS button is held depressed while the pilot flies the aircraft to the desired attitude using

manual control. The autopilot is re-engaged by release of the TCS button and the autopilot continues to again hold the aircraft in the attitude in which it was left.

## Autopilot Limitations and Operational Restrictions

### *Autopilot Disengage Warnings*

The "A/P" light flashes red and a tone sounds when an autopilot has disengaged.

The warning can be reset by pressing either disengage light or either A/P disengage switch.

The A/P warning light stays at steady red if:

Stabilizer is out of trim below 800 ft RA on a dual channel approach.

Altitude Acquire mode is inhibited during an A/P go-around. (Stabilizer not trimmed for single A/P operation. See A/P Go-around).

Disengage light test switch is held in position 2 (red filament test).

Automatic ground system test fail.

The light will illuminate a steady amber when the disengage light test switch is held in position 1 (amber filament test).

The light will flash amber if A/P automatically reverts to CWS pitch or roll while in CMD. The light will reset when either light is pressed or another mode is engaged.

### *Maximum Pitch and Bank Angles*

During normal autopilot operation the maximum angles are:

**Pitch**     $\pm 10^\circ$

**Roll**      $\pm 30^\circ$

These limits are, however, not stipulated legally and will vary from aircraft to aircraft.

### *Gain Adaption*

Variations in flight parameters such as altitude, speed, aircraft load, configuration and rate of manoeuvre, will have an effect on the handling characteristics of an aircraft. It is therefore necessary to incorporate 'gearing' elements within flight control systems which will adapt the parameters to the aircraft so that their effect on handling characteristics is reduced. In automatic systems the response is altered by changing the 'gain' of the system to a given level of input signal. This can be likened to changing gear ratios in a mechanical system.

Gain adaptation is particularly important for maintaining handling characteristics with changes in IAS during the different phases of flight and is similar to the gain scheduling in the flight director system.

### *Approach/Land Mode*

During an autoland sequence the autopilot has to execute many important manoeuvres. These are described in the Autoland notes.

## Flight Management System

The autopilot can form part of the overall Flight Management System (FMS). This may also be designated the Automatic Flight Control System (AFCS) or the Flight Management and Guidance System (FMGS). It provides manual or automatic modes of control throughout the entire flight envelope from take-off to landing and roll-out. All the subsystems of the FMS are fully integrated and have levels of redundancy to achieve a high level of reliability. Redundancy is accomplished by providing two or more systems of each type so a failure of one system will not affect the operation of the complete system.

The AFCS and FMS will be checked completely during the preflight checks. During these checks all the automated systems will be engaged, tested and their various safety devices tested. The FMS will be checked for the correct information and any additional information will be entered.



# Chapter 27 Autoland

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## Introduction

The approach and landing manoeuvre is the most difficult one demanded of a pilot in that it entails control of the aircraft in all three axes simultaneously as well as control of airspeed through engine power changes. The pilot has to:

- Align the aircraft with the runway centreline
- Achieve a sink rate of about 2 feet per second before touchdown
- Reduce the airspeed from  $1.3V_s$  to about  $1.15V_s$  by progressive reduction of engine power
- Level the wings before actual landing yaw the aircraft to remove any drift angle (drift “kick off” or de-crabbing).

An automatic landing system that takes over from a pilot must be able provide guidance and control better than that required of the pilot.

Autopilots have for a long time now been able to fly most of the approach allowing the pilot to concentrate on navigating the approach correctly. The pilot would then take over at decision height and continue to land manually. Aircraft that are fitted with all the equipment required for a fully automatic landing may, due to lack of required ground equipment or simply for pilot experience requirements, only be allowed to carry out an auto-approach. Essentially all the procedures are carried out as for an autoland, but when decision height is reached the pilot will take over manually.

## Autoland System

### Objective

In order to achieve the objective of automatic landing, the operation of an automatic flight control system must be of such a nature that it will:

Not disturb the flight path as a result of an active malfunction

Have adequate authority for sufficiently accurate control along the required flight path

Warn of a passive failure

Allow the intended flight manoeuvre to be completed following an active or a passive failure.

### Requirements

To enable an aeroplane to complete an automatic landing the autoland system requires:

A minimum of **two independent autopilots** capable of following ILS signals

Two independent **radio altimeters** to give accurate height from the ground information

**Category 3 ILS** ground installation at the airport.

### Autoland Status

The number of autopilots required also depends upon the autoland status of the aircraft. These fall into two main categories:

### *Fail-passive (Fail-soft)*

This is defined as the ability of the system to withstand a failure without endangering passenger safety, and without producing excessive deviations in the flight path but removing its capability to complete an automatic landing.

The minimum number of autopilots required for a fail-passive capability is two.

### *Fail-operational (Fail-active)*

This status is defined as the ability of a system to withstand a failure without affecting the overall functioning of the system and without causing degradation of performance beyond the limits required for automatic landing.

The system requires a minimum of three autopilots. However, it is possible for an aircraft to have a fail-operational category with only two autopilots provided that there is suitable duplicate monitoring for each channel.

## The Automatic Landing Sequence

### *Profile*

During cruise and initial stages of approach to land, the control system operates as a single channel system, controlling the aircraft about its pitch and roll axes, and providing the appropriate flight director commands. The profile of an automatic approach, flare and landing sequence is shown in *Figure 27.1* and is based on a system that utilizes triple digital flight control computer channels, allowing for redundancy to operate in the fail-operational and fail-passive conditions already defined.

### *Status Annunciator*

Depending upon the number of channels that are armed and engaged, the system performs what are termed a '**LAND 2**' status or '**LAND 3**' status autoland. Thus '**LAND 2**' signifies there is dual redundancy of engaged flight control computers, sensors and servos (**fail-passive operation**) while '**LAND 3**' signifies triple redundancy of power sources, engaged flight control computers, sensors and servos (**fail-operational**). Each status is displayed on an autoland status annunciator.

### *Approach*

Since multi-channel operation is required for an automatic landing, at a certain stage of the approach, the remaining two channels are armed by pressing an '**APPR**' switch on the flight control panel. The operation of this switch also arms the localizer and glide slope modes. Both of the 'off-line' channels are continually supplied with the relevant outer loop control signals and operate on a comparative basis the whole time.

### *Radio Altimeter*

Altitude information essential for vertical guidance to touchdown is always provided by signals from a radio altimeter which becomes effective as soon as the aircraft's altitude is within the altimeter's operating range (typically 2500 feet).

### *AFS Radio Altimeter Loss*

Two independent radio altimeters provide RA to the respective FCC. The Captain's radio altimeter also provides RA information to the A/T.

With a radio altimeter inoperative, do not use the associated FCC or the A/T, if affected, for approach and landing, i.e. failure of a single radio altimeter causes the autoland system to fail passive.

## The Sequence

An example of an autoland sequence (for a Boeing aircraft) is described below.

### *Approach (APP) mode*

The approach mode arms the AFDS to capture and track the localizer and glide slope. It can be engaged for dual or single autopilot operation (autoland with dual only). Dual autopilot (A/P) approach operation is described first.

Approach mode allows both A/Ps to be engaged at the same time. Dual A/P operation provides fail-passive control through landing flare and touchdown or an automatic go-around. During fail-passive operation, the flight controls respond to the A/P commanding the least control movement.

One VHF Nav receiver must be tuned to an ILS frequency before the approach mode can be selected. For a dual A/P approach, the second VHF NAV receiver must be tuned to the ILS frequency and the corresponding A/P engaged prior to 800 ft RA.

### *Localizer and Glide Slope Armed*

After setting the localizer frequency and course, pressing the APP switch selects the APP mode. The APP switch illuminates and VOR LOC and G/S armed is annunciated. The APP mode permits selecting the engagement of the second A/P. This arms the second A/P for automatic engagement after LOC and G/S capture and when descent below 1500 ft RA occurs.

The localizer can be intercepted in the HDG SEL, CWS R or L Nav modes. Either the LOC or G/S can be captured first (although it is most common to capture LOC then G/S).

### *Localizer Capture*

The localizer capture point is variable and depends on the intercept angle and rate of closure, but does not occur at less than  $\frac{1}{2}$  a dot deviation. Upon LOC capture, VOR LOC annunciates captured, 1 CH is annunciated for the A/P status, the previous roll mode disengages and the aeroplane turns to track the LOC.

### *Glide Slope Capture*

The G/S can be captured from above or below (although from below is generally preferred). Capture occurs at  $\frac{2}{5}$  dot deviation. G/S annunciates captured, the previous pitch mode disengages, the APP switch light extinguishes if the localizer has also been captured, aeroplane pitch tracks the G/S and the annunciated N1 thrust limit for the A/T is GA.

After LOC and G/S are both captured the APP mode can be exited only by pressing the TOGA switch or by disengaging the A/P and turning off both FD switches or re-tuning a VHF Nav receiver.

### *After Localizer and Glide Slope Capture*

The A/Ps will disengage and the FD command bars will retract to indicate an invalid ILS signal.

**At 1500 feet Radio Altitude (RA)**

Shortly after capturing both LOC and G/S and descending below 1500 ft RA, the second A/P couples with the flight controls, FLARE mode armed is annunciated and the A/P go-around mode arms but is not annunciated. ROLL OUT mode if available will also now arm. The autoland status will also now be annunciated as either "LAND 2" (or "LAND 3" for fail-operational aircraft).

The pitch and roll axes cannot be manually overridden into CWS. Attempts to do so will result in A/P disengagement.

**800 ft RA**

The second A/P must have been engaged by 800 ft RA to execute a dual A/P approach. Otherwise, engagement of the second A/P is inhibited on descending through 800 ft RA.

**400 - 330 feet RA**

The stabilizer is automatically trimmed an additional amount nose-up, with the elevators neutralizing and holding the pitching up moment. If the A/Ps subsequently disengage, forward control column force may be required to hold the desired pitch attitude. This biasing aids the flare and in the event of a subsequent fail-passive the aircraft will tend to pitch nose-up to prevent a hard contact with the ground and aids the initiation of a go-around.

If FLARE is not armed by approximately 350 ft RA, both A/Ps automatically disengage.

**At 45 feet Gear Altitude (GA)/ 50 ft RA**

The A/P flare manoeuvre starts at approximately 50 ft RA and is completed at touchdown. FLARE engaged is annunciated and the FD command bars retract. Also:

The stabilizer trim is again automatically trimmed an additional amount nose-up.

The **FLARE** mode is automatically engaged (replacing G/S) to reduce the vertical speed from around 10 - 12 feet per second at 50 feet to 1 - 2 ft/sec at the point of touchdown.

It uses the rate of reduction of radio height as the controlling signal.

The A/T begins retarding thrust at approximately 27 ft RA so as to reach idle at touchdown.

The **gear altitude** calculation, which is pre-programmed into the computer, is based upon radio altitude, pitch attitude, and the known distance between the landing gear, the fuselage and the radio altimeter antenna.

**At about 5 feet GA:**

The flare mode is disengaged and there is transition to the touchdown

LOC disengages

Roll-out mode (if available) will engage.

**At about 1 foot GA:**

The pitch attitude of the aircraft is decreased to 2°

**At touchdown**, a command signal is supplied to the elevators to lower the aircraft's nose and so bring the nose landing gear wheels in contact with the runway and hold them there during the **roll-out**.

**When reverse thrust is applied** the autothrottle system is automatically disengaged. Irrespective of reverse thrust deployment, the A/T automatically disengages approximately 2 seconds after touchdown.

The automatic flight control system remains on until manually disengaged by the flight crew, which is when the autoland sequence is considered to be completed.

## Other Features of Autoland

### *Runway Alignment*

Although the yaw channel has not been mentioned any auto-flight system capable of an autoland must be capable of 'kicking off drift' prior to touchdown. This is known as **runway alignment** mode and will typically be armed at the same time as the flare mode and engaged at less than 100'. During the approach from 1500' the yaw channel will compute the difference between heading and track, and when align mode engages the rudder deflects to align the aircraft with the runway centre line before touchdown. This manoeuvre is known as **de-crabbing**, or drift 'kick-off'.

### *Roll-out*

Another function of Cat 3 autoland systems is roll-out which gives steering commands on the ground proportional to localizer deviation along the centre line. These commands can show left/right steering guidance through a rotating 'barbers pole' indicator known as a para-visual display (PVD). Alternatively automatic steering can be achieved by applying deviation signals to the rudder channel and nosewheel steering to keep the aircraft on the centre line throughout the ground roll.

### *A/P Go-around Mode*

The A/P go-around (GA) mode requires dual A/P operation and becomes armed when FLARE armed is annunciated. The A/P GA mode cannot be engaged before flare arm is annunciated or after the A/P senses touchdown.

Note: if the GA mode is selected after touchdown and prior to A/T disengagement, the A/Ps will disengage and the A/Ts may command GA thrust, with the procedure being flown manually.

Pressing either TOGA switch engages the GA mode and GA engaged is annunciated for the AFDS. The MCP IAS/Mach display becomes blank and the airspeed cursors are positioned at the AFDS commanded speed. Command airspeed is the flap manoeuvring speed.

### *A/P GA Pitch Control*

Upon GA engagement, the thrust levers advance toward the reduced GA N1. The A/P initially commands a 15 degree nose-up pitch attitude, and the airspeed cursors display manoeuvring speed for the flap setting. When a programmed rate of climb is established, the A/P controls pitch to hold airspeed based on the normal flap manoeuvring speed.

### *A/P GA Roll Control*

With the GA mode engaged, the A/Ps maintain the aeroplane ground track existing at GA engagement.

### *Leaving A/P GA mode*

Below 400 ft RA, the A/Ps must be disengaged to change either pitch or roll modes from GA. Above 400 ft RA, other pitch and roll modes can be selected.

If the roll mode is changed first, the selected mode engages in single A/P roll operation and is controlled by the A/P which was first engaged. Pitch control remains in the dual A/P GA mode.

The pitch mode cannot be changed from GA until sufficient nose-down trim has been input to allow single A/P operation. This nose-down trim is automatically added to reset the previous trim inputs that were applied during the auto-approach. If the pitch mode is the first to be changed from GA, the selected pitch mode engages in single A/P operation and is controlled by the first A/P that was engaged for the approach. The second A/P disengages and the roll mode changes to CWS R.

With pitch engaged in GA, ALT ACQ engages when approaching the selected altitude and ALT HOLD engages at the selected altitude if the stabilizer position is satisfactory for single A/P operation.

The transition from GA to ALT ACQ is normally successful if the selected altitude is at least 1000 ft above the GA engagement altitude. A higher selected altitude may be required if full GA thrust is used.

If stabilizer trim is not satisfactory for single A/P operation, ALT ACQ is inhibited and the A/P disengage lights illuminate steady red and pitch remains in GA. To extinguish the A/P disengage lights, a higher altitude can be selected or the A/Ps disengaged.

### *Approach (APP) Mode / Single A/P*

A single A/P ILS approach can be executed by engaging only one A/P after pressing the APP switch. Single A/P approach operation is the same as for dual, with the following exceptions:

A/P status of 1 CH is annunciated for the entire approach after localizer capture.

Full automatic flare and touchdown capability is not available. FLARE is not annunciated and stabilizer trim bias is not applied.

An A/P GA is not available.

The diagram on the next page shows an automatic landing. See if you can correctly fill in the pitch and roll armed and engaged modes in the annunciator boxes at each stage of the approach and landing. I've started you off at point A as Approach mode has been selected.

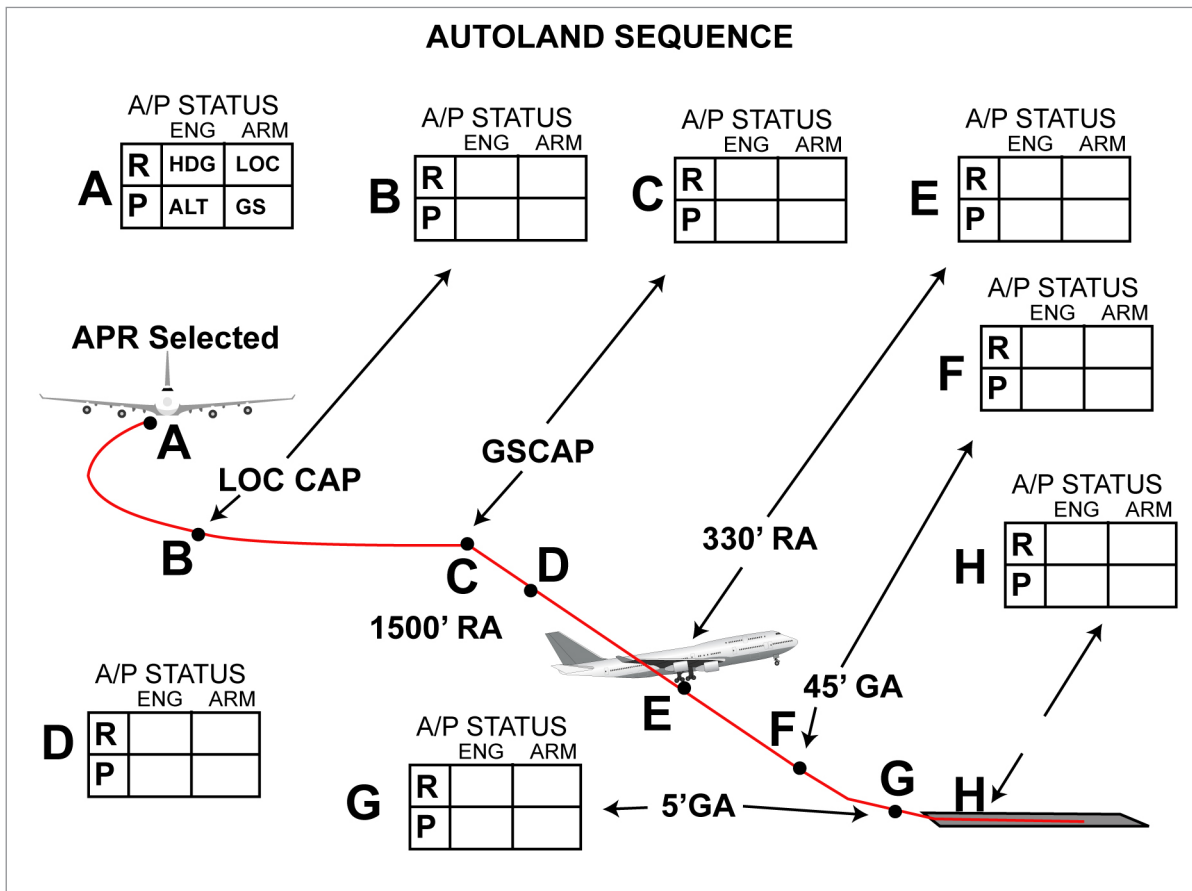


Figure 27.1 Autoland Sequence

## Appendix 3A

### ILS

An Instrument Landing System (ILS) is a short-range navigational aid which provides azimuth and vertical guidance during the approach to an airport runway. The system comprises ground-based transmitting elements and also receiving elements carried on board the aircraft.

The **ground-based** elements are:

- A localizer transmitter which sends runway azimuth approach information.
- A glide slope transmitter which provides vertical approach information.
- Marker beacons which transmit information about the distance to the runway threshold. Marker beacons have been replaced with Distance Measuring Equipment (DME) in most installations.

The **airborne** elements are:

- A localizer signal receiving antenna (usually the same antenna as the one used for the VOR).
- A glide slope signal receiving antenna.
- A dual ILS receiver installation.
- An indicator which shows whether the aircraft is on the correct approach path. Loc and GS deviation.
- A marker beacon antenna and receiver (or DME).
- Marker lights on the main instrument panel.

## Weather Minima

In low visibility operations, the weather limits for landing are given in the following terms:

**Runway Visual Range (RVR)**, which is an instrumentally derived value that represents the range at which high-intensity lights can be seen in the direction of landing **along the runway**. The measurements are transmitted to the air traffic controller who can inform the pilot of the very latest visibility conditions.

**Decision Height (DH)** This is the wheel height above the runway elevation at which a go-around must be initiated by the pilot, unless adequate visual reference has been established, and the position and approach path of the aircraft have been visually assessed as satisfactory to safely continue the approach or landing.

Minimum values of DH and RVR are known as 'weather minima' and are specified by the national licensing authorities for various types of aircraft and airports. When the traffic controller advises that the **RVR** is above the specified minimum, the pilot may descend to the specified decision height and if, by then, he has sighted a sufficiently large segment of the ground to enable him to be confident of his judgement, he may carry on and land. He must otherwise overshoot, and either enter the holding pattern pending another approach, or divert to an alternative airport. During the approach, the pilot's line of sight is down the glide path and not along the runway, and this gives rise to another factor, called '**slant visual range**', which a pilot must take into account in order to avoid misinterpretation of visual cues.

## The ICAO Categorization of Low Visibility Landing Capabilities

### *All Weather Operations*

The term 'all weather operations' is frequently used in connection with automatic landing systems and in describing low weather minima. This is a term which can, and sometimes is, taken to mean that there are no weather conditions that can prevent an aircraft from taking off and landing successfully. This is not the case, because no automatic system can, for example, perform the landing task in wind conditions in excess of those for which the aircraft has been certified, this being primarily governed by the controllability characteristics and strength factors of the aircraft. Similarly, no automatic system can land an aircraft on a runway which has a surface which is not fit for such an operation because of contamination by water, slush or ice.

### *Category of Operation*

The definitions of the main categories are illustrated in *Figure 27.2*, overleaf.

The three categories also serve as an indication of the stages through which automatic approach and automatic landing development progress, and thereby designate the capabilities of individual automatic flight control systems. In addition, they designate the standards of efficiency of the ground guidance equipment available at airports, namely ILS localizer and glide path, and approach, runway and taxiway lighting.

### *Category 1*

A precision instrument approach and landing with a decision height no lower than 60 m (200 ft), and with either a visibility not less than 800 m, or a runway visual range not less than 550 m



### Category 2

A precision instrument approach and landing with decision height lower than 200 ft but not lower than 100 ft, and a runway visual range not less than 300 m.

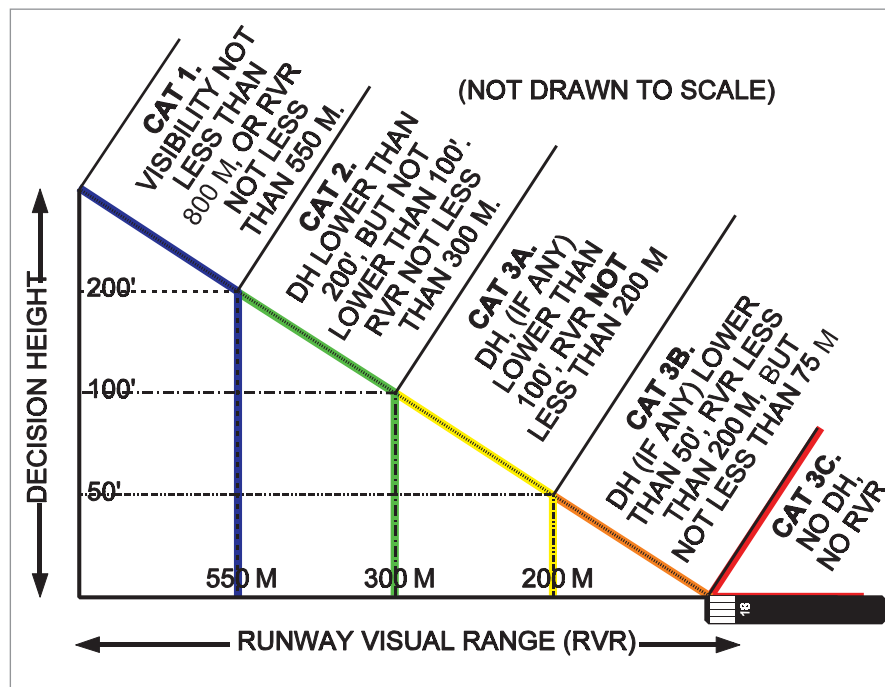


Figure 27.2 Categories of low visibility landing capabilities

### Category 3A

A precision instrument approach and landing with a decision height lower than 100 ft and a runway visual range not less than 200 m.

### Category 3B

A precision instrument approach and landing with a decision height, if any, lower than 50 ft and a runway visual range less than 200 m but not less than 75 m.

### Category 3C

To and along the surface of the runway and taxiways without external visual reference.

## Alert Height

The alert height is a specified radio height, based on the characteristics of the aircraft 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 aircraft 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** the alert height, it would be ignored and the **approach continued** to complete the autoland.



Chapter  
**28**  
Autothrottle

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## Introduction

An autothrottle system is a computer controlled, electromechanical system which can control the thrust of an aircraft's engines within specific design parameters. The throttle position of each engine is controlled to maintain a specific value of thrust in terms of:

- Fan Speed (N1)
- Engine Pressure Ratio (EPR) or
- Target Airspeed (set by SPD on mode control panel)

Thrust is the **force** generated by the engines. The throttles control the thrust and in some aircraft the preferred name for the throttles is thrust levers. It is worth noting that although there are thrust computation systems there is, as yet, no direct indicator of thrust value in use. Instead N1 and EPR are used to indicate a measure of engine thrust.

Using the above modes the autothrottle can control either engine thrust or aircraft speed from the beginning of the take-off roll until the system is disconnected after an automatic landing.

## Autothrottle System

### System Components

The autothrottle can also be called the Thrust Management System (**TMS**) that works in conjunction with the autopilot and the FMS.

*Figure 28.1* shows a basic autothrottle system and signal interfacing between various aircraft systems and sensors.

### Inputs

These would include:

- Mode selection and A/T Arm switch on the MCP
- TAS, Mach No. and TAT from the ADC
- Attitude and acceleration from the IRS
- N1 speed and/or EPR from engine sensors
- Angle of attack from AoA sensor
- Radio altitude from the radio altimeter
- Air/ground logic from the landing gear switch
- Reverse thrust requirement from the engine accessory unit

#### Plus

Thrust command from the FMS or thrust mode selection from the trust mode select panel

- A/T Disconnect switch on the throttles
- PLA (power lever angle) position from transducers
- Flap position

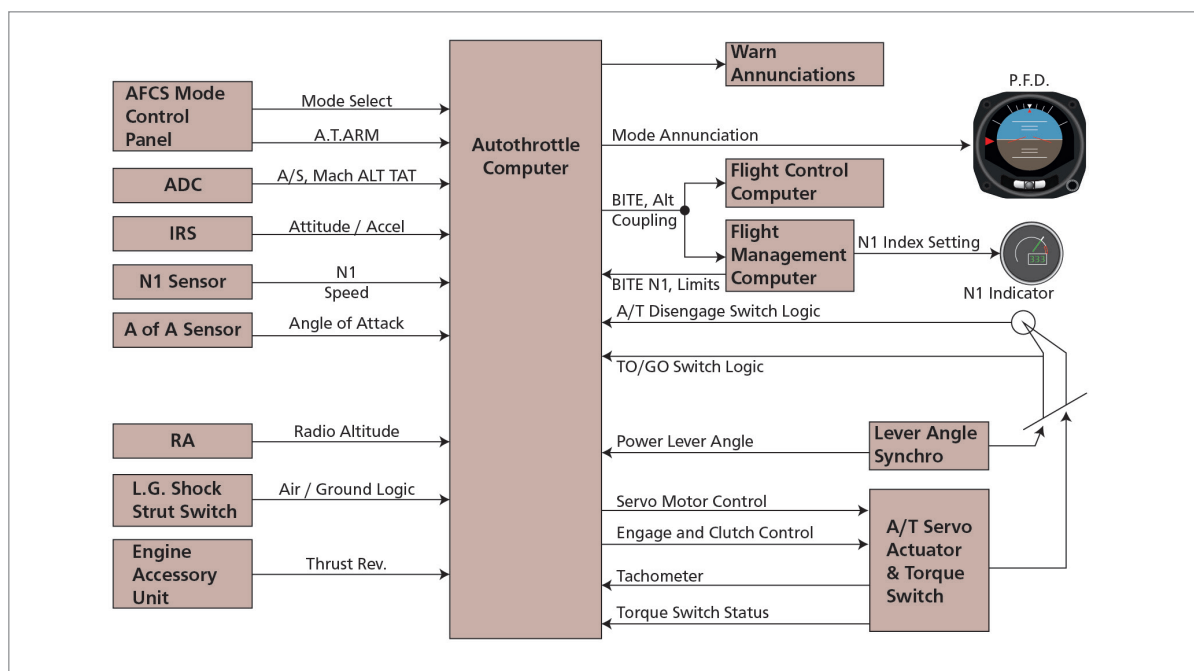


Figure 28.1

### Outputs

The main outputs from the system would be signals to:

- A/T servo actuator to move the throttles
- A/T disengage circuit
- BITE (built-in test equipment) circuits in the FCC and the FMC
- Mode annunciation to the EFIS symbol generator
- Thrust limits and to the EICAS / ECAM display
- Failure warnings annunciations (lamp and/or aural, electronic display)

### Feedback

The autothrottle system compares the actual values with reference values and passes control signals to the servomotors of the thrust levers. In order to control the speed at which the thrust levers are moved, there is a suitable feedback from the servo actuators to the TMC.

### Boeing 737-400 General

The A/T system provides automatic thrust control from the start of the take-off through climb, cruise, descent, approach and go-around or landing. In normal operation, the FMC provides the A/T system with N1 limit values.

The A/T moves the thrust levers with a separate servomotor on each thrust lever. Manually positioning the thrust levers does not cause A/T disengagement unless 10 degrees of thrust lever separation is exceeded during a dual channel approach after FLARE armed is annunciated. Following manual positioning, the A/T may reposition the thrust levers to comply with computed thrust requirements except while in the HOLD and ARM modes.

### **Power Management Control (PMC)**

The thrust control system consists of a hydromechanical Main Engine Control (MEC) unit and a PMC unit mounted on each engine. The MEC schedules fuel to provide the thrust called for by the Thrust Lever setting on the flight deck. This fuel flow is further refined electronically by the Power Management Control (PMC) without moving the thrust levers.

The PMC uses MEC power lever angle, N1 speed, inlet temperature and pressure to adjust, or trim, the MEC to obtain the desired N1 speed. The PMC adjusts fuel flow as a function of thrust lever angle.

The PMC provides a constant thrust climb feature once the thrust lever is set at the beginning of climb. Thus, when thrust is set for climb, the PMC automatically maintains that thrust throughout the climb profile with no further thrust lever adjustments. If the thrust lever is repositioned, the PMC maintains the setting corresponding to the new thrust lever angle.

The PMC includes failure detection and annunciation modules which detect PMC failures and provide a signal to the crew. For detectable failure conditions, the PMC schedules a slow N1 drift over approximately 30 seconds and then illuminates the PMC INOP light, the ENG system annunciator light and the MASTER CAUTION lights. For a PMC failure, the PMC can be selected OFF by a switch on the aft overhead panel. The engine speed is then controlled by the hydromechanical MEC only. The PMC INOP light is suppressed below starter cut-out engine speed.

### **A/T - PMC Operation**

The A/T system operates properly with the PMCs ON or OFF. In either case, the A/T computer controls the FMC N1 limits.

During A/T operation, it is recommended that both PMCs be ON or both OFF, as this produces minimum thrust lever separation. A/T take-offs may be performed with both PMCs OFF.

### **A/T Engagement and A/T Disengagement**

Moving the A/T Arm switch to ARM, arms the A/T for engagement in the N1, MCP SPD or FMC SPD mode. The A/T Arm switch is magnetically held at ARM and releases to OFF when the A/T becomes disengaged.

Any of the following conditions or actions disengages the A/T:

Moving the A/T Arm switch to OFF.

Pressing either A/T Disengage switch.

An A/T system fault is detected.

2 seconds have elapsed since landing touchdown.

Thrust levers become separated by more than 10 degrees during a dual channel approach after FLARE is annunciated.

A/T disengagement is followed by A/T Arm switch releasing to OFF and the A/T Disengage light flashing red.

The A/T Disengage lights can be extinguished by any of the following actions:

- Returning the A/T Arm switch to ARM
- Pressing either A/T Disengage light
- Pressing either A/T Disengage switch

The A/T Disengage lights do not illuminate when the A/T automatically disengage after landing.

### Take-off Mode

The take-off mode is engaged by pressing either TOGA switch with the aeroplane on the ground, the A/T armed and the desired take-off N1 thrust limit selected from an FMC CDU. The A/T annunciation changes from ARM to N1 and the thrust levers advance toward take-off thrust.

The A/T sets take-off thrust. THR HLD annunciates at 84 kt (64 kt for aeroplanes with earlier model A/T computers) to indicate that the A/T cannot change thrust level position, but thrust levers can be repositioned manually.

After lift-off, the A/T remains in the THR HLD until 400 ft RA is reached and approximately 18 seconds have elapsed since liftoff. A/T annunciation then changes from THR HLD to ARM. Reduction to climb thrust can now be made by pressing the N1 switch.

Until 2½ minutes after lift-off, automatic reduction to climb thrust is inhibited when in LVL CHG or V/S mode. If V NAV, ALT ACQ or ALT HOLD is engaged during this 2½ minute period, automatic thrust reduction occurs normally.

### N1 Mode

The A/T maintains thrust at the N1 limit selected from an FMC CDU. N1 is annunciated for the A/T and the N1 switch illuminates. Pressing the N1 switch changes the A/T mode from N1 to ARM.

If an engine fails while the A/T is in the N1 mode, the thrust lever of the failed engine will advance forward a few degrees and then return to or below the other thrust lever position.

### Speed Mode

The speed mode is available throughout the flight once the take-off phase is completed. Pressing the MCP Speed Select switch selects the speed mode if compatible with the engaged AFDS pitch mode. MCP SPD is annunciated for the A/T mode and the Speed Mode switch illuminates. The speed or Mach shown in the MCP IAS/MACH display is the target speed. The A/T will not set power above the displayed N1 limit; however, the A/T can exceed an N1 value that has been manually set by the N1 manual set knob. If an engine fails while the A/T is in a speed mode, both thrust levers advance together to maintain the target speed.

When on final approach in landing configuration, it is not recommended to set the A/T command speed to allow for wind or gust corrections. Through airspeed and acceleration sensing, the A/T corrects for normal wind gusts. Higher command speed settings result in excessive approach speeds. The recommended A/T approach speed setting is  $V_{REF} + 5$ .

Below 400 ft RA, A/T thrust level response rate and engine power levels are sufficient to place the engines in the rapid acceleration range.



### **FMC Speed Mode**

The FMC SPD mode is an A/T mode which is commanded by the FMC during V NAV operation. When engaged, the MCP IAS/Mach display is blank, the airspeed cursors are positioned at the FMC commanded airspeed and the A/T maintains this commanded speed. The A/T is limited to the N1 value shown on the thrust mode annunciators.

### **N1 Equalization**

The A/T attempts to equalize N1 through the dual servo individual thrust lever control. Equalization control is limited to 8 degrees of thrust lever separation.

### **ARM Mode**

The A/T annunciates ARM when the A/T Arm switch is at ARM and no A/T mode is engaged. The thrust levers can be manually positioned without interference from the A/T system while ARM is annunciated.

The A/T automatically transfers to ARM from the SPEED or N1 mode when the mode is deselected by pressing the respective mode selector switch while the switch light is illuminated.

### **Descent Retard Mode**

The A/T engages and annunciates RETARD during LVL CHG and V NAV descents. RETARD changes to ARM when the thrust levers reach the aft stop or when they are manually prevented from reaching the aft stop.

### **Landing Flare Retard Mode**

During landing, the RETARD mode engages, reduces thrust and annunciates RETARD 2½ seconds after FLARE mode engagement or at 27 ft RA, whichever occurs first. During a non-precision or visual approach with flaps extended to 15 or greater and the AFDS not in ALT ACQ or ALT HOLD, the A/T RETARD mode engages at 27 ft RA. The A/T automatically disengages approximately 2 seconds after landing touchdown.

### **Go-around Mode**

With the A/T Arm switch at ARM, the A/T go-around mode is armed when descending below 2000 ft RA, with or without the AFDS engaged. Once armed, the A/T go-around mode can be engaged until 2 seconds have elapsed after landing touchdown.

Pressing either TO/GA switch engages the A/T go-around mode. GA is annunciated for the A/T and the thrust levers advance to the reduced go-around thrust setting. This setting produces a 1000 to 2000 fpm rate of climb. After reaching reduced go-around thrust, pressing either TO/GA switch the second time signals the A/T to advance thrust to the full go-around N1 limit.

After reaching reduced or full go-around thrust, the A/T GA mode can be terminated by selecting another AFDS pitch mode or when ALT ACQ annunciates engaged.

During a single engine FD go-around, the A/T will increase thrust to the full N1 limit.

### **Autothrottle Disengage Switches**

Pressing an Autothrottle Disengage switch disengages the autothrottle (A/T). The A/T Disengage light flashes and the A/T ARM switch on the MCP trips off. Pressing the Autothrottle Disengage switch a second time extinguishes the A/T warning.

After an automatic A/T disengagement, pressing the Autothrottle Disengage switch will extinguish the A/T warning.

### Autothrottle Disengage Light

The A/T Disengage light will flash red if the A/T disengages for any reason.

The A/T Disengage light will illuminate steady red when the Disengage Light Test Switch is held in position 2 (red filament test position) and steady amber when the Disengage Light Test Switch is held in position 1 (amber filament test position).

The A/T Disengage light flashing amber indicates an A/T airspeed error if speed is not held within +10 or -5 knots of the commanded speed when all of the following conditions exist:

- In flight
- Flaps not up
- A/T engaged in MCP SPD, or FMC SPD mode

An automatic test of the A/T flashing amber function is performed if the A/T is engaged and the following conditions exist:

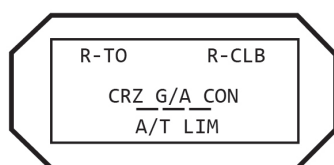
- MCP SPD or FMC SPD is the active A/T mode
- More than 150 seconds after lift-off
- Flaps extended

The A/T amber light flashes for 2 seconds, remains extinguished for 2 seconds and then flashes for 2 seconds again.

### Thrust Mode Annunciator Panel (TMA)

On the Boeing 737-400 the Thrust Mode Annunciator panel is located on the centre instrument panel, above the N1 rpm indicators. It displays the active N1 limit reference mode for auto throttle and manual thrust control. N1 limits are also displayed on the N1 rpm indicator cursors with the reference knobs pushed in.

N1 limits are normally calculated by the FMC. When FMC N1 limit calculations become invalid, or if either engine N1 is less than 18%, A/T LIM is annunciated. The autothrottle computer then calculates a single N1 limit for the affected engine(s).



Centre Instrument Panel

R - Reduced. Can appear with TO and CLB.

TO - Take off.

CLB - Climb.

CRZ - Cruise.

G/A - Go-around.

CON - Continuous.

--- FMC is not computing thrust limits.

A/T LIM - Autothrottle limit. Indicates that A/T computer is calculating a degraded N1 limit for the affected engine or engines.

### **Flexible Take-off**

In situations where take-off can be executed without the need for full engine power (such as light weight take-off from a long runway), then reduced power may be used, thereby reducing engine wear and increasing their life. This is called the Flexible Take-off mode (and in the Airbus aircraft there is a detent position for the thrust levers labelled FLEX TO). The most basic way to achieve the reduced thrust is to manually set a lower rpm setting. To achieve a reduced power setting with the autothrottle we can select a temperature on the control panel that is higher than the ambient airfield temperature. This causes the thrust computation system to calculate a lower limiting EPR or N1, thereby producing reduced power for the take-off.

### **Turbulence**

When operating in light to moderate turbulence the autothrottle can stay engaged unless performance is poor.

There will be increased thrust lever activity and airspeed excursions of 10 to 15 knots are to be expected. The autothrottle is not to be used in severe turbulence.

### **FADEC**

The system example comes from the Airbus series of aircraft.

#### **Function**

FADEC stands for full authority digital engine control. It provides complete engine management throughout all phases of flight and performs the following functions:

- Gas generation control (fuel flow, acceleration/deceleration, variable bleed valve and variable stator vane schedules, turbine clearance control, idle setting)
- Engine limit protection (over-speed N1 and N2)
- Power management (control of engine thrust rating, computation of thrust parameter limits, auto-thrust system demand, thrust lever position manual demand)
- Automatic engine starting sequence (control of start valve, fuel, ignition, monitoring N1, N2, FF, EGT)
- Manual engine starting sequence (passive monitoring of start valve, fuel, ignition, N1, N2, FF, EGT)

In performing its functions it takes into account such variables as power demanded, air bleed for air conditioning and de-icing, temperature, static pressure and engine accessory selection.

#### **Advantages**

The FADEC system reduces crew workload, provides engine limit protection, improves engine life and saves fuel and maintenance down time.

#### **Components**

FADEC consists of an electronic engine control (EEC) plus a fuel metering unit, sensors and peripheral units. There are suitable interface circuits between the EEC and its peripheral units. There are 2 FADEC channels per engine, one in control and one in standby for redundancy.

Each channel is powered by the aircraft's A/C supply before and during the initial start and then by an individual internal magnetic alternator above 12% engine rpm.

Thrust control is provided by a FADEC dedicated to each engine. Thrust selection is achieved by means of the thrust levers when in manual mode or the Flight Management and Guidance System (FMGS) when in automatic mode. Thrust rating limit is provided by the FADEC according to the thrust lever position both for manual and automatic thrust.

### **Fuel Control**

In addition to the high pressure fuel pumps and shut-off system there are again 2 main systems controlling engine performance. The Hydromechanical Unit (HMU) is modulated by the FADEC. It provides control of fuel flow to the combustion chamber, control of fuel hydraulic signals to actuators, and over-speed protection.

The Fuel Metering Valve (FMV) transforms FADEC orders through a torque motor / servo valve into fuel flow to the engine nozzle. The FMV resolver provides an electrical feedback signal proportional to the FMV position. The bypass valve regulates a constant pressure drop across the FMV to ensure that metered fuel flow is proportional to FMV position.

The FADEC computes fuel flow necessary to hold the target N1. To obtain this N1, the N2 is allowed to vary without exceeding N2 min and N2 max. The FADEC will also vary N2/N1 in order to maintain RPM under varying load conditions, maintain bleed air production, avoid engine stalls or flameout. With the Airbus FADEC it even modulates the cooling airflow around the engine in order to expand or contract the casing to control the compressor and turbine clearances at nominal settings.

### **Thrust Levers**

The thrust levers are only moved manually (unlike many other autothrottle systems). They move over a sector which is divided into 4 segments defined by 5 detents or stops. Thrust lever position is transmitted to the FADEC which computes and displays the thrust rating limit and the N1 TLA. Note that there is no reverse idle detent. When the idle stop is cleared by pulling up the reverse thrust levers, reverse idle is selected.

### **Thrust Rating Limit**

Thrust rating limit is computed according to the thrust lever position. If the thrust lever is set in a detent, the FADEC will select the rating limit corresponding to this detent. If the lever is set between 2 detents the FADEC will select the rating limit corresponding to the higher mode.

## **Thrust Control**

### **Manual Mode**

The engines are in the manual mode provided that the ATS function is:

- Not armed
- Armed and not active (thrust lever is not in the ATS operating range and/or no alpha floor condition)

In these conditions, each engine is controlled by the position of the corresponding thrust lever. Thrust modulation is performed by the pilot moving the thrust lever from IDLE to TO/GA position. Each position of the thrust lever within these limits correspond to an N1. When the thrust lever is positioned in a detent, the corresponding N1 is equal to the N1 rating limit computed by the associated FADEC.

When the thrust lever is set in the FLX-MCT detent:

### *On the ground*

When the engine is running, the flex TO thrust rating is selected provided a flex TO temperature greater than the current TAT has been selected through the MCDU. Otherwise the MCT thrust is selected.

### *After TO*

A change from FLX to MCT is achieved by setting the thrust lever to TO/GA or CL position and then back to MCT. After that FLX rating setting is not possible.

MAX TO power is always available by pushing the thrust lever fully forwards.

### *Automatic Mode*

In the auto-thrust mode (A/THR function active), the thrust is computed by the FMGC and is limited to the value corresponding to the thrust lever position (except if the alpha floor mode is activated).

### *Inputs*

- Air data parameters from the air data inertial reference system.
- Operational commands from the engine interface unit ( target N1 )
- Thrust lever angle (TLA)
- Engine Pressure sensors ( N1 and N2 )
- Temperature (EGT)
- Fuel flow.

### *Outputs*

- Data outputs necessary for the FMGS
- Thrust parameters and TLA to the FMGS
- Control signals to the fuel metering unit
- Command N1 to the ECAM display.



Chapter  
**29**  
Yaw Dampers

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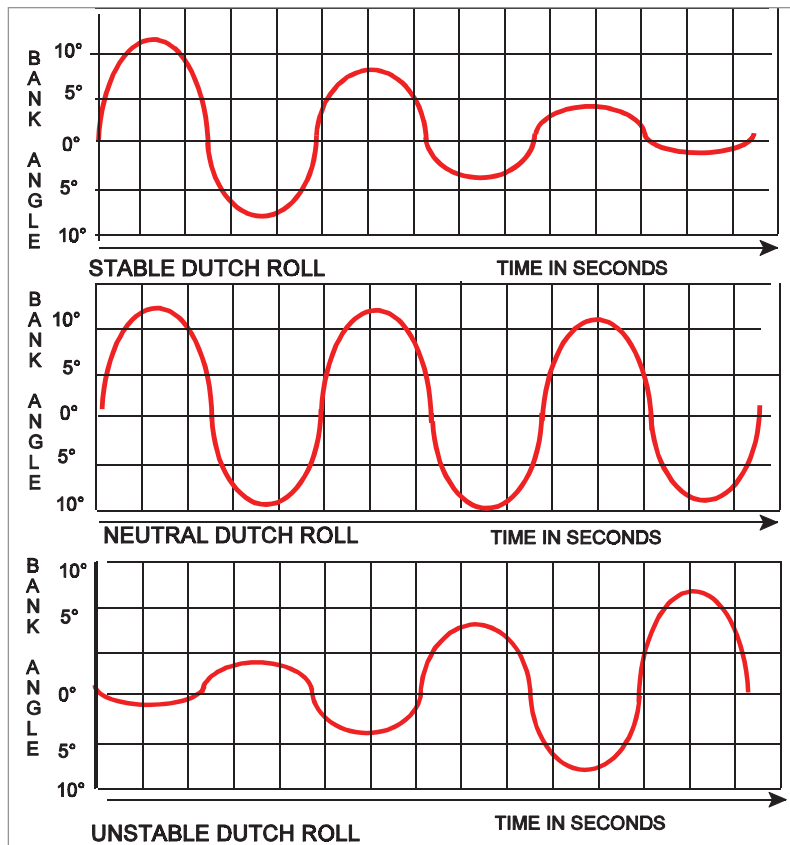


Figure 29.1 Dutch roll stability

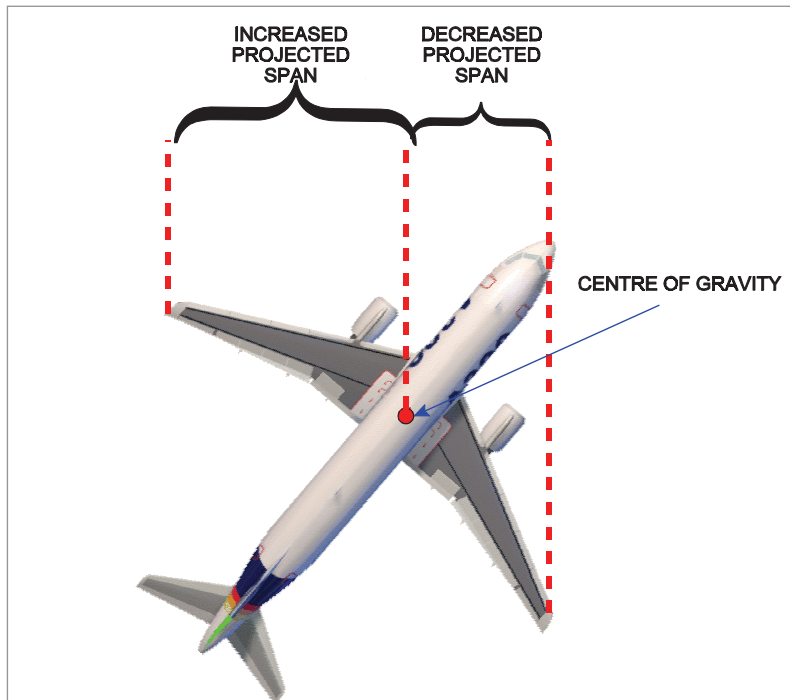


Figure 29.2 Change in effect aspect ratio with yaw



## Dutch Roll

Dutch roll is caused by the interplay between lateral stability around the longitudinal axis (roll) and longitudinal stability around the vertical axis (yaw) of an aircraft in flight. An aircraft with an excess of lateral stability will by default have poor directional stability and therefore will be susceptible to Dutch roll.

More simply, any disturbance of an aircraft in yaw directly causes a secondary disturbance in roll and vice versa. Stability is the aircraft's natural tendency to resist any disturbance and return to the same conditions that existed before the disturbance occurred. If lateral stability is greater than directional stability then Dutch roll will be the result.

Consider an aircraft disturbed by a gust causing it to yaw. As the aircraft yaws, one wing will travel slightly faster through the surrounding air and the other wing will travel slightly slower. The fast wing will produce slightly more lift than before and the slower wing will produce slightly less. This obviously will produce a roll.

As lift increases then lift induced drag will increase so the faster, higher wing will produce more drag and the low wing will produce less. This causes a yawing moment in opposition to the initial disturbance and the whole process is reversed.

Swept wing further exacerbate the tendency for Dutch roll because the forward going wing undergoes a reduction in effective wing sweep, further increasing the lift it produces, while the retreating wing experiences an increase in effective wing sweep, again reducing its lift.

All airline aircraft will be statically stable, in that they will naturally try to return to the undisturbed condition. The amount they are dynamically stable depends on the amount of damping force available. Aerodynamic damping is dependent on the change in the relative airflow, which affects the angle of attack and the TAS. As an aircraft climbs its true airspeed must increase to maintain the lift pressure. This increase in TAS means that the relative angle of attack for the aerodynamic surfaces is reduced for the same given disturbance, and so the corrective force supplied by that surface is reduced, so damping effectiveness is reduced at high altitude.

## Function of a Yaw Damper

To increase the damping forces at altitude could mean an increase in the overall size of the stabilizing surfaces but this would also increase drag. Another option is to produce an aircraft that is dynamically stable at lower and middle altitudes and have a system to automatically counter Dutch roll. This system detects the yaw then applies rapid, small and effective rudder deflections stopping the Dutch roll before it starts. This system we call a **Yaw Damper**.

The yaw damper will either be the third axis of an autopilot system or will be an addition to the third axis of autopilot control.

The yaw damper will be on for the duration of the flight and will provide turn co-ordination, runway alignment and assist during asymmetric thrust.

Large, modern, airliners generally have poor Dutch roll characteristics and so tend to have 2 and even 3 yaw damper systems. This is because the only way to control an aeroplane with poor Dutch roll tendencies, and a failed yaw damper system, is to operate at a much lower altitude where the TAS is reduced.

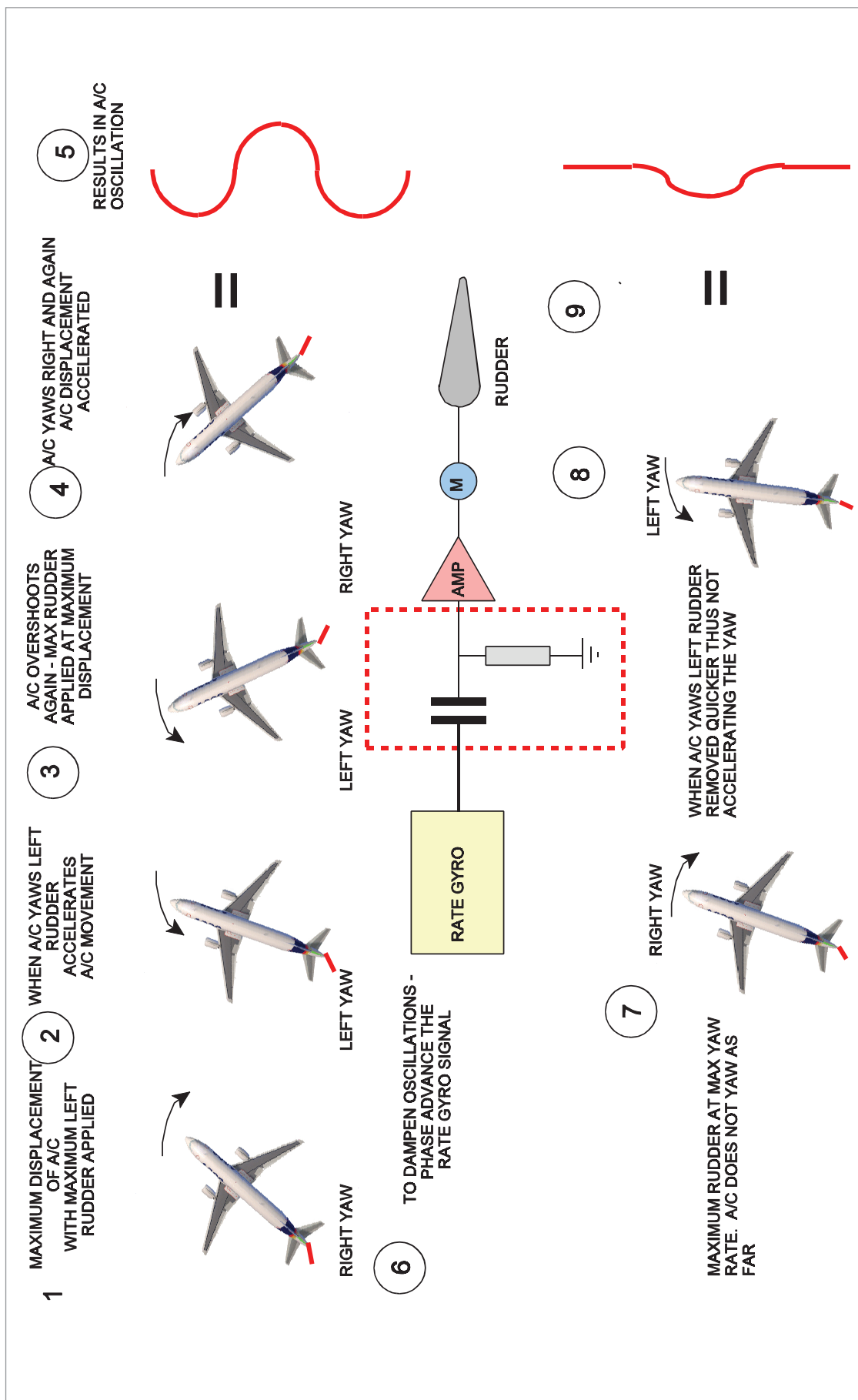


Figure 29.3 Principle of phase advance

## The Yaw Damper

### Typical Yaw Damping Signal Processing

#### Sensing

A single axis rate gyro, or in modern aeroplanes an input from the IRS, is used to sense the aeroplane's motion about the normal or vertical axis. The objective being to sense the yawing motion as quickly as possible and produce a correcting demand signal to the servo/actuator which then feeds signals to the rudder control system to apply rudder in opposition to the yaw.

#### Phase Advance

Phase advance is a means of applying the damping application as soon as possible. The reason for this is that damping must be applied when the **rate of disturbance** is at its greatest, not when the disturbance has moved to the point where the natural stability of the fin has arrested the disturbance. If a rudder application is applied at the same instant the fin starts to return the aircraft, both the combined forces will over correct and cause the aircraft to overshoot. *Figure 29.3* tries to explain when the ideal damping is required.

The phase advance accelerates the signal so that the rudder input occurs at the point of maximum yaw and cancels the signal quickly so as not to exaggerate the yawing effect.

### Dutch Roll Filter

The system as described, however, would also interpret a normally commanded turn as a yaw and act in opposition to produce an unco-ordinated turn (in fact the rudder would be exacerbating the co-ordination problem by introducing the full amount of its rudder authority in the opposite direction to that required). So the yaw damper system needs to be able to distinguish commanded turn inputs from yaw due to a disturbance or apparent Dutch roll.

When an aircraft is turned, the aircraft rolls about the longitudinal axis in the direction of the turn and also YAWS about the vertical axis. A yaw damper is provided to dampen Dutch roll, not to prevent the aircraft following a turn command. Therefore only the Dutch roll frequency must be acted upon by the yaw damper.

The Dutch roll frequency is based on the natural yawing frequency of the aircraft. The frequency is relatively LOW and will differ slightly with aircraft type.

Yaw dampers must be designed to allow the Dutch roll frequency to control the rudder but block other frequencies. *Figure 29.5* illustrates a typical Dutch roll filter and circuit action.

Initially, while the rate of turn is building up to the constant rate, the Dutch roll filter output also builds, then falls off to nothing when the rate of turn becomes constant. The reverse, with opposite polarity as the filter capacitor discharges, occurs as the aircraft levels out on completion of the turn. Therefore, whilst the turn is constant the filter output is zero. This results in no rudder demand.

*Figure 29.4* shows an aircraft yawing at the Dutch roll frequency. Since the rate of turn is constantly changing, the output from the rate gyro is constantly changing. The DC graph at the bottom of *Figure 29.4* is the Dutch roll filter output.

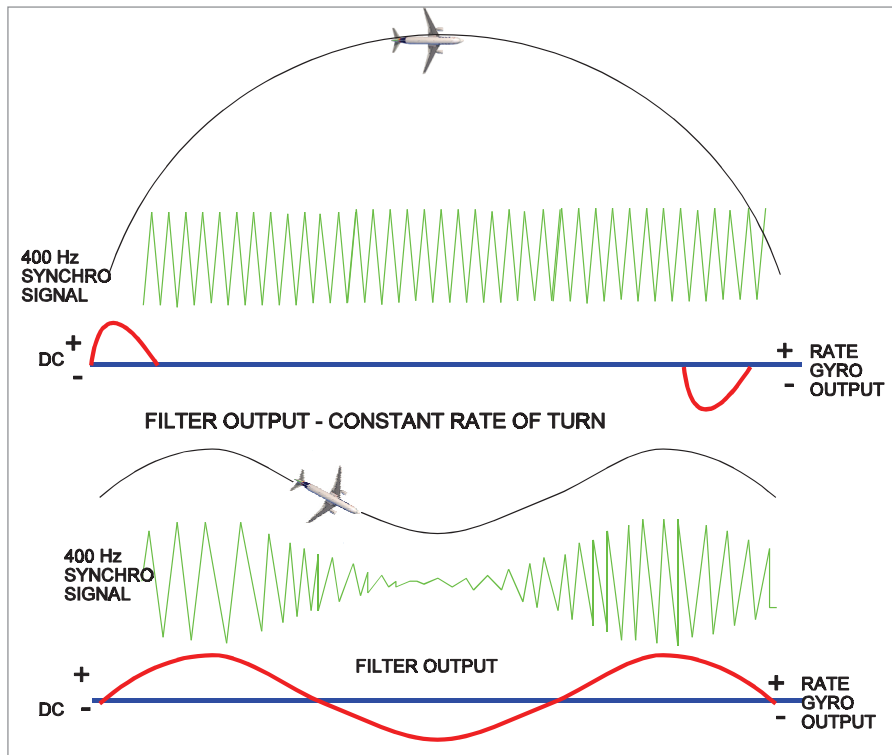


Figure 29.4 Filter output - Dutch Roll

The DC polarities are the greatest when the rate of turn is the greatest and reverse when the direction of turn (rate of gyro signal) reverses.

Figure 29.5 is a super simplified yaw damper, illustrating mainly Dutch roll filter. The Dutch roll filter is a NARROW BAND PASS FILTER designed to pass only signals which change the frequency of the Dutch roll. The rate gyro produces outputs for all turns, but only those related to Dutch roll will appear at the input to the servo amplifier driving the rudder servomotor.

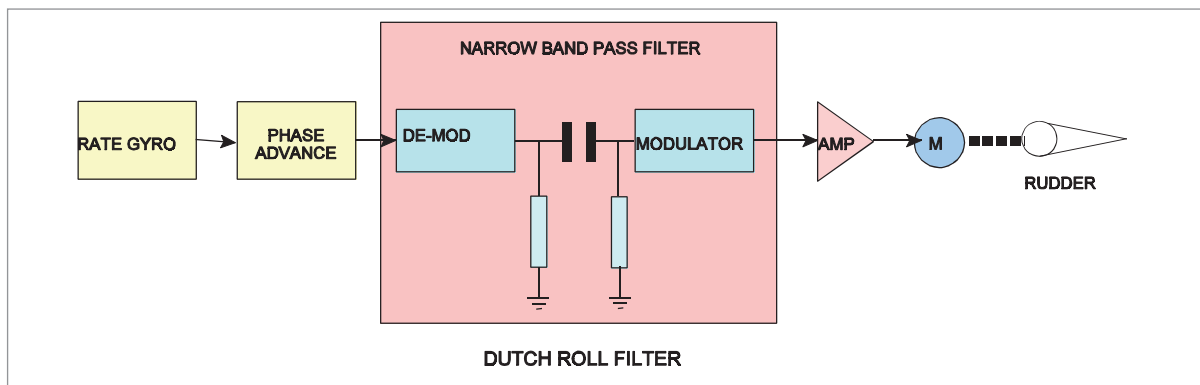


Figure 29.5 Basic Yaw Damper Showing Dutch Roll Filter

### Rudder Control Computing Authority

As removal of oscillations does not require a large rudder movement and to reduce the hazard posed by a yaw damper runaway, yaw damper authority is normally only about 3-6 degrees left and right of centre. If 2 yaw damper systems are operating on a single span rudder their authority is accumulative, i.e. singly each system could move the rudder by 3 degrees, together they can move it by 6 degrees.

On aircraft with a split rudder surface, if one yaw damper system fails then the aircraft has only  $\frac{1}{2}$  the yaw damper protection that it originally had. This is allowed for in the design and operation of the aircraft. As even this small input may cause overstress in certain flying conditions, some aircraft have an input to the yaw damper computer from the CADC to schedule the gain of the yaw damper inputs for the ambient flying conditions. At high speed therefore the yaw damper authority may be even further reduced to avoid an overstress condition.

## Operating Modes

### *Synchronization*

This is to prevent yaw axis engagement transients by cancelling servomotor outputs by an inverting integrator. RL1, see [Figure 29.6](#), is energized prior to yaw damper engagement. Any amplifier output is fed back through the integrator, the inversion through the integrator cancels any transients present. On engagement of the yaw damper, RL1 is de-energized.

### *Engaged Mode*

Providing the interlock logic is good, the engage solenoid engages allowing the yaw damper elements of the Power Control Unit (PCU) to pressurize. The rate gyro signal is phase advanced and applied to the demodulator. The demodulator converts the signal to a DC signal where the output polarity will represent the AC signal input phase. The yaw damper frequency is passed by the bandpass filter which blocks all other frequencies. Consequently, the yaw damper does not oppose normal turn manoeuvres and does not respond to aircraft vibration and bending. The modulator restores the AC signal maintaining the appropriate phase. The servo amplifies the signal and applies it to the transfer valve which in turn drives the yaw damper actuator which drives the main actuator. The maximum deflection of the rudder is  $3^\circ - 4^\circ$  in either direction.

The Linear Variable Differential Transformer (LVDT) position feedback is applied to SP2 to cancel the processed rate gyro signal when the corresponding change in rudder position is appropriate for the rate of yaw change. Position feedback is also applied through the energized relays of RL2 which is fed back to SP2. The purpose of this is to ensure that the rudder will always return back to the neutral position. The rudder can be affected by crosswinds. If the crosswind is strong, the position feedback voltage may not be large enough to drive the rudder to the central position. The position feedback voltage now causes INT 2 to ramp up, increasing the voltage at SP2. This increase in position feedback starts to drive the rudder back to the neutral position. As the rudder returns the position feedback voltage decreases allowing the integrator to run down.

### *Yaw Damper Testing*

Actuation of the yaw damper test switch to either the right or the left applies a voltage to the yaw damper rate gyro torquing coil which torques the rate gyro and simulates aircraft movement. The rudder position indicator responds to this action of an output from the position transducer. If the switch is moved to the left the indication will first move to the left and then back to the centre; on release of the switch the indication will move to the right and back to the centre. The reverse will happen if the switch is first moved to the right.

### *Yaw Damper Indications*

Yaw damper controls and indications are fairly simple. On a typical aircraft (fitted with 2 yaw dampers), the main indications will be a panel or part thereof that contains an on/off switch, a test switch or button and a failure light for each of the yaw dampers. The failure light indicates many faults in the system: loss of hydraulic or electrical power, logic failure in the yaw damper computation system, loss of input from the rate gyro. The switch can then be used to isolate

the inoperative yaw damper to avoid spurious rudder inputs.

Also involved with the monitoring of the yaw dampers is a small rudder trim indicator, one for each yaw damper system fitted. It indicates the demands on the rudder by moving left and right of centre. It is mainly used during the testing of the yaw dampers during preflight checks. If the yaw damper test switch is operated for one of the yaw dampers, a test signal operates a small torquing coil on the yaw damper rate gyro. This moves the gyro and fools the yaw damper system into thinking that a yaw condition exists. A pass at test is indicated by the position indicator moving in the direction tested and back to centre. Moving the switch to the left simulates a yaw in one direction and the position indicator should move to the left also. If the switch is operated to the right then the rudder should move to the right.

Some aircraft do not have test switches and rely on the yawing movement of the aircraft during taxi to test the yaw damper system. So in a left turn the yaw damper signal will move left and the yaw damper will then move the rudder left and the signal will return to the neutral position.

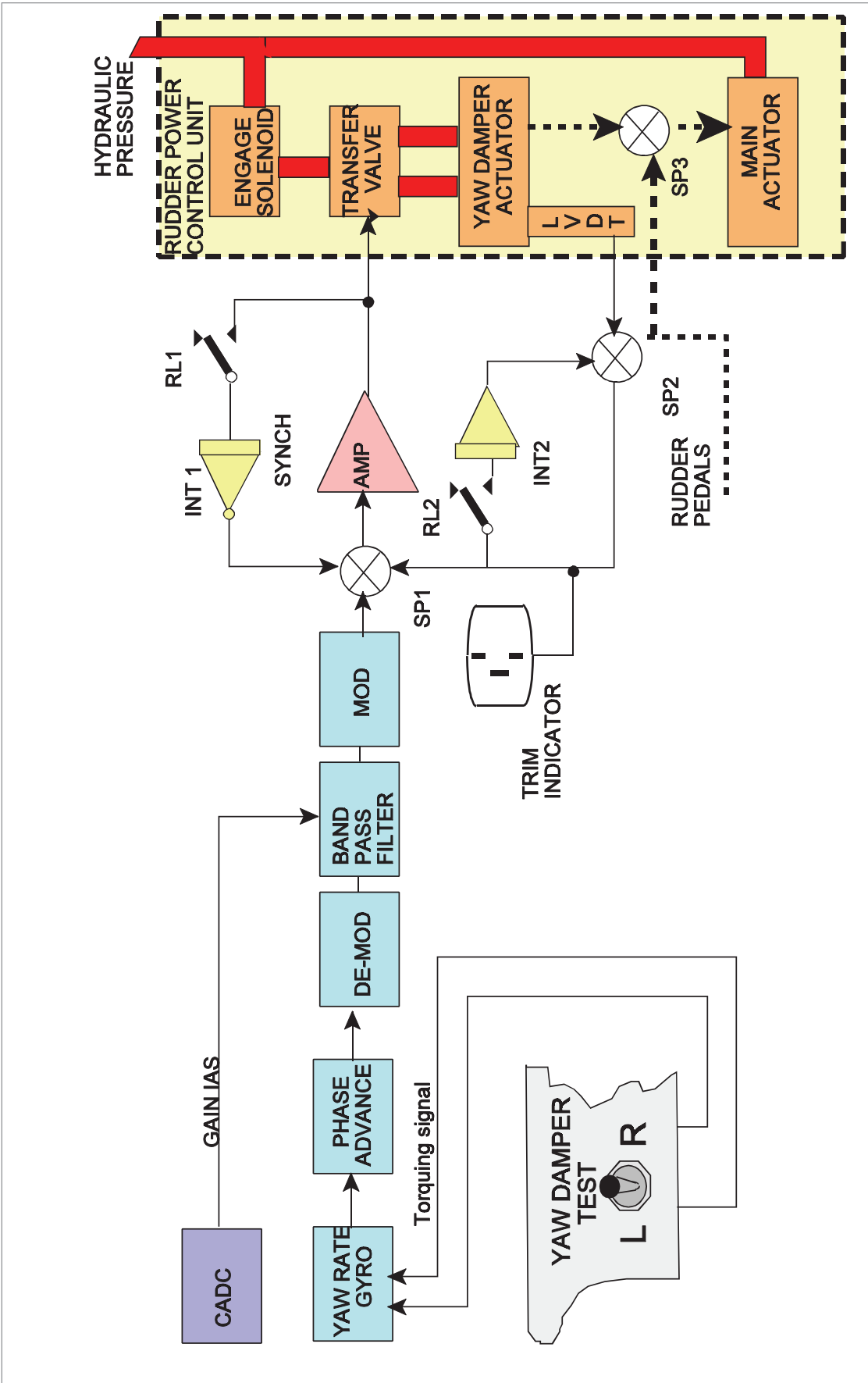


Figure 29.6 Series yaw damper schematic diagram





# Chapter **30** Control Laws

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## Introduction

The autopilot must be able to manoeuvre the aircraft logically and safely in a similar manner to the way a human pilot should. This means ensuring the autopilot does not break aircraft limitations in terms of speed, load factor, pitch and bank limits etc. However, the autopilot should be able to use a satisfactory amount of the performance of the aircraft otherwise the manoeuvres would take too long to execute.

For example consider a light aircraft on a VFR navigational exercise. Overhead the next waypoint the pilot requires an 80° left turn to proceed to the next point. The bank angle used will be determined by that turn. The pilot will not generally choose under normal circumstances to turn with 5° of bank as that would mean the turn taking an inordinate amount of time. Conversely the pilot would not normally choose a 90° banked turn as that would be excessively hard, possibly breaking load factor limitations as well as probably causing a loss of height. A turn using 30° - 45° angle of bank would be the most sensible. If the turn required was about 10° instead of 80° then banking to 30° would generally be considered to be too harsh.

The autopilot needs to be able to apply similar logic to all its actions. Whenever the autopilot is required to make a correction either for stability or when referring to a particular flight path, the control response will always be in proportion to deviation or corrective action required but only up to a limit that prevents the autopilot attempting a manoeuvre that would cause an excursion from the safe flight envelope.

## Boeing 737-400 Autopilot Limiting and Reversion Modes

### *Command Speed Limiting and Reversion Modes*

To prevent the AFS from causing a flight envelope excursion there is a system of command speed limiting and reversion modes. AFS command limiting and reversion is independent of the stall warning and airspeed/Mach warning systems.

### *Command Speed Limiting*

The AFS provides speed, pitch and thrust commands to avoid exceeding the following limit speeds:

- $V_{MO} / M_{MO}$ .
- Wing flap limiting speeds.
- Landing gear speeds.
- Minimum speeds

The commanded speed can be equal to, but will not exceed, a limit speed.

Speeds greater than  $V_{MO} / M_{MO}$  cannot be selected from the MCP. Speeds can be selected which exceed flap and gear limiting speeds or that are less than the minimum flight speed.

Minimum speed is based on an angle of attack and is approximately  $1.3V_s$  for the current flap configuration. It is sensed by the angle of attack vanes, one on either side of the forward fuselage.

If a speed greater than a placard speed or less than the minimum speed is selected, the AFS allows acceleration or deceleration to slightly short of the limit, then commands the limit speed. The over-speed or under-speed limiting symbol appears in the MCP IAS/Mach display when the commanded speed cannot be reached.

Either pitch or thrust, whichever is engaged in a speed mode, attempts to hold the limit speed. The commanded limit speed and MCP speed condition symbol remain until another speed is selected which does not exceed the limit. A speed 15 kt greater than the minimum speed must be selected to remove the under-speed symbol.

### Reversion Modes

During some flight situations, speed control by the AFDS or A/T alone could be insufficient to prevent exceeding a limit speed. If this occurs, AFDS and A/T mode automatically revert to a more effective combination. The reversion modes are:

Placard Limit reversion.

Minimum airspeed reversion.

Mode reversion occurs slightly before reaching the limit speed. Both the AFDS and the A/T have reversion modes which activate according to the condition causing the reversion.

### Placard Limit Reversion

When one of the placard limits (gear, flap or  $V_{MO} / M_{MO}$ ) is reached, the over-speed limiting symbol appears in the MCP IAS/Mach display and the following occurs:

- If not in AFDS or A/T speed control and the A/T is armed, the A/T reverts to SPEED mode and controls speed to the placard limit.
- If in AFDS or A/T speed control, no reversion is necessary. The AFDS or A/T, whichever is controlling speed, holds speed slightly below the placard limit.
- If the A/T is not available, no reversion response to gear or flap placard speeds is available. The AFDS reverts to speed control for  $V_{MO} / M_{MO}$  speed limiting.

### Minimum Speed Reversion

The AFDS and A/T do not control speed to a speed which is less than the minimum speed for the current flap configuration. This speed is approximately  $1.3V_S$ . Minimum speed, FMC speed or selected speed, whichever is higher, becomes the AFS commanded speed. If actual speed becomes equal to or slightly less than the minimum speed, the under-speed limiting symbol appears in the MCP IAS/Mach display and if operating in the V/S mode, the AFDS reverts to LVL CHG.

The AFS commands a speed 5 kt greater than the minimum speed. Selecting a speed 15 kt greater than the minimum speed reactivates normal MCP speed selection control. The AFDS commands nose-down pitch to increase airspeed if the thrust levers are not advanced. When actual speed becomes 15 kt greater than minimum speed, the under-speed limiting symbol disappears.

The A/P disengages and the FD command bars retract when in LVL CHG climb with a command speed equal to minimum speed and a minimum rate of climb cannot be maintained without decelerating.

No minimum speed reversion is available when the A/T is OFF and the AFDS is in ALT HOLD, ALT ACQ or after G/S capture.

## Flight Envelope Protection

### Introduction

Flight envelope protection is taken to the extreme by fly-by-wire aircraft with the aim of ensuring that the aircraft remains **within** the normal flight envelope in all phases of flight. The system prevents the envelope being violated during **extreme situations**, such as:

- Windshear
- Very high turbulence
- Mid-air collision avoidance
- GPWS or TCAS activation
- Mismanagement by the crew.

The purpose of the flight envelope protection is to:

- Give **full authority** to the pilot in order to consistently achieve the **best possible aircraft performance** in those extreme conditions.
- **Reduce the risks** of overcontrolling / overstressing the aircraft.
- Provide the pilot with an easy, instinctive and immediate procedure to achieve the best possible performance when required.

### The Protection

The system provides protection in all phases of flight to prevent the aircraft exceeding the limits for the following parameters:

- Angle of attack
- Speed
- Pitch attitude
- Bank angle
- Load factor

### High Angle of Attack

The protection enables the pilot to execute a rapid pull-up manoeuvre in an emergency situation (as in a mid-air collision avoidance situation) at **maximum angle of attack**, max, without overcontrolling the aircraft. The technique requires simply that the pilot "snatch stick fully back". If the aircraft exceeds the normal flight envelope for any reason, the pilot is immediately made aware of the situation by the pitch auto-trim stop and the aft pressure required on the stick to keep the flight path. The high angle of attack protection is an aerodynamic protection but thrust is required to maintain the flight path and the auto-thrust function would automatically provide TOGA thrust when the aircraft reaches a certain value (called floor ) before it gets to max. The input to the circuit is the angle of attack and the output is applied to the elevators and the auto-thrust.

## High Speed

High speed protection circuits prevent the aircraft from reaching  $V_d / M_d$  by adding a positive nose-up g demand to the pilot demand on the stick; this **demand is proportional to the amount of speed overshoot beyond  $V_{MO} / M_{MO}$** . This enables a pilot to enter a steep dive rapidly by pushing the stick forward, safe in the knowledge that the high speed protection will prevent the aircraft from exceeding the design speed limits.

The inputs to the unit would be airspeed/Mach No. from the air data computer and the output is applied to the elevators.

## Pitch Attitude

The pitch attitude protection, which is only available in fly-by-wire aeroplanes, enhances the high angle of attack protection and the high speed protection. The circuit reduces the pitch demand of the stick when the aircraft reaches the pre-defined maximum pitch attitude values which are:

**30° nose-up and 15° nose-down.**

The input is the pitch angle from the attitude gyros and the output is applied to the elevators.

## Bank Angle

On a commercial aircraft the bank angle does not normally exceed 30°. However, in certain circumstances higher bank angles might be required. Bank angle protection, which is only available in fly-by-wire aeroplanes, allows the pilot to achieve any roll manoeuvre efficiently and prevents the aircraft entering into an uncontrollable state. For example, the limits of bank angle for an Airbus aircraft are:

- 67° in the normal flight envelope
- 45° when high AoA is triggered
- 40° when high speed protection is triggered.

After a roll manoeuvre if the pilot releases the stick, the aircraft would return to a bank angle of 33°. The bank angle limit is achieved by **reducing the roll rate demand progressively** as the bank angle increases.

## Load Factor

A commercial aircraft is designed to withstand a maximum load factor, beyond which structural damage is likely to occur. In conventional aircraft where no protection is provided the pilot has to assess the instantaneous g load and could overstress the aircraft in an urgent situation.

Load factor protection, available in fly-by-wire aeroplanes, is provided by sensing the g load on the aircraft with accelerometers. The g load limiter protects the aircraft against overstress by maintaining it within its structural limitations while allowing the pilot to react immediately to an evasive manoeuvre. The load factor protection is linked to the high angle of attack protection.

## Autopilot Gain Adaption / Gain Scheduling

In the same way as the Flight Director System uses gain scheduling to reduce demands when in close proximity with the ground in order to ensure that the FD system does not demand a manoeuvre that would endanger the aircraft, the autopilot has a comparable system. This ensures that for example, during an autoland the autopilot's pitch and roll authority is significantly reduced as the aircraft nears the ground. An example of this may be an aircraft that has an autopilot which when used in the manual mode may have bank angle limited to  $45^\circ$ . During VOR or Localizer tracking this may be reduced to  $30^\circ$  as that is deemed all that is necessary. However, during the final phase of an automatic approach or an autoland this may be reduced to  $15^\circ$ .

Gain adaption may also be used to alter the autopilot's limits to allow for differing aircraft performance at different altitudes and speeds. Although artificial feel is provided to give pilots awareness of control forces, autopilots could easily ignore artificial feel inputs and overstress the aircraft. So an input from the ADC to the autopilot may be used to reduce the autopilot's authority in proportion to  $Q$ .





Chapter

# 31

## AFCS Revision Questions

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## Question Paper 1

1. **A single axis autopilot system:**
  - a. provides stabilization about the normal axis
  - b. provides control about the pitch axis
  - c. is unsuitable for use in powered aircraft
  - d. provides control about the roll axis
  
2. **A single axis autopilot may also be called:**
  - a. altitude hold
  - b. wing leveller
  - c. pitch control loop
  - d. auto-stabilization loop
  
3. **An autopilot:**
  - a. is a system which will maintain a preselected altitude
  - b. is a system which will maintain a preselected airspeed
  - c. is an auto-stabilization system
  - d. is an outer loop control system
  
4. **The fundamental components of an autopilot control loop are:**
  - a. rate gyro, servomotor, error signal generator
  - b. rate gyro, servomotor, torque limiter
  - c. torque limiter, error signal generator, servomotor
  - d. servomotor, rate gyro, torque limiter, error signal generator
  
5. **A device in a closed loop control system in which a small power input controls a much larger power output in a strictly proportionate manner is:**
  - a. an amplifier
  - b. a servomechanism
  - c. a powered flying control unit
  - d. a rate gyro
  
6. **An automatic flight control system:**
  - a. is another name for an autopilot system
  - b. applies flight data to the autopilot system
  - c. is automatically disengaged by a gpws alert
  - d. can only be used in EFIS equipped aircraft
  
7. **An aircraft has yaw damping included in its auto-stabilization system. An essential requirement of such a system is:**
  - a. a three axis autopilot system
  - b. parallel connected servomotors
  - c. automatic maintenance of C of G position
  - d. INS inputs to the CADDC

- 31  
AFCS Revision Questions
8. **Automatic flight systems may be capable of controlling the aircraft flight in:**
- azimuth, elevation and velocity
  - azimuth and velocity only
  - azimuth only
  - azimuth and elevation only
9. **An automatic flight control system is fitted with control wheel steering (CWS).**
- The autopilot must be disengaged before the pilot can input manoeuvring commands
  - Manoeuvring commands may be input by applying normal forces to the control yoke without first disengaging the autopilot
  - Manoeuvring commands may be input using pitch and turn controls on the automatic flight system control panel, without first disengaging the autopilot
  - The CWS is only there for steering on the ground
10. **During an approach to an autoland at 1500 feet:**
- off line channels are manually engaged, flare mode is armed
  - localizer is controlling the roll channel, off line channels are automatically engaged and flare mode is armed
  - localizer is controlling the roll channel, stabilizer is trimmed nose up and roll out is armed
  - provided both localizer and glide slope signals are valid LAND 3 will illuminate
11. **What type of autoland system would be required for the landing to continue following a single failure below alert height?**
- Fail-soft
  - Fail-passive
  - Fail-operational or fail-active
  - Land 2 system
12. **Inputs to the rudder channels initially originate from:**
- servomotors
  - compass gyro and gyro for ah
  - compass gyro and turn and slip gyro
  - AH gyro and turn and slip gyro
13. **An automatic flight system which can safely continue with an automatic landing after a system failure is a:**
- fail-redundant system
  - fail-passive system
  - three axis system
  - fail-operational system
14. **Altitude Select and Altitude Hold are examples of:**
- inner loop functions in pitch
  - manometric functions from the ADC
  - interlocking functions
  - outer loop functions in roll

15. During an autoland the caption LAND 2 is illuminated. The system is:
- fail-active or fail-operational
  - fail-passive
  - approaching decision height
  - requiring a crew input
16. For an autoland system to meet FAIL-PASSIVE criteria it must:
- have suitable system redundancy
  - withstand a system failure without excessive deviations from flight path
  - can continue with an autoland below alert height
  - can continue with an autoland above alert height
17. During an autoland at 50 ft AGL (45' GA) the pitch control of the autopilot is ..... and the roll control is .....
- glide slope      localizer
  - glide slope      roll out
  - flare              roll out
  - flare              localizer
18. During an autoland approach:
- flare is engaged at 1500' AGL
  - localizer roll control is disengaged just prior to touchdown
  - flare is disengaged prior to touchdown at 5' GA
  - glide slope is the engaged pitch mode until 5' GA
19. In an autoland at 1000' AGL with two autopilots engaged:
- the armed roll mode would be LOCALIZER
  - the engaged roll mode would be GLIDE SLOPE
  - the engaged pitch mode would be FLARE
  - the engaged roll mode would be LOCALIZER
20. An automatic flight control system in which the application of normal forces on the control column allows the pilot to input demands to the autopilot is a:
- control wheel steering
  - touch control steering
  - series connected system
  - parallel connected system
21. If a fault develops in a triplex autopilot system during an approach, the system will revert to:
- fail-passive and the landing may continue
  - fail control wheel mode
  - fail-operational
  - a manual disconnect

22. **Central Air Data Computers (CADCs) transmit data concerning:-**
- airspeed, altitude and decision height
  - airspeed, altitude and Mach number
  - airspeed, attitude and Mach number
  - airspeed and altitude only
23. **Inner loop stability is obtained by:-**
- inputs from the Air Data Computer
  - manometric locks
  - rate gyro displacement
  - raw data feed to the data control bus bar
24. **The autothrottle is used to control some factors during the three primary control modes; they are:**
- EPR, Mach and Speed
  - EPR, wheel and speed
  - EPR, Mach and altitude
  - EPR, wheel and altitude
25. **The mode that enables the pilot to manoeuvre his aircraft in pitch and roll by use of the automatic control system is called the:**
- control wheel steering (CWS) mode that allows the pilot to control the aircraft, and when the wheel is released, the aircraft holds the newly established attitude
  - touch control steering that will permit the pilot to control the aircraft via the air data computer
  - control wheel steering mode which will disengage the servomotors
  - touch control steering mode which will prevent the flaps retracting
26. **Touch control steering:**
- prevents aerodynamic feedback
  - will only operate while the flaps are down
  - allows the pilot to control the aircraft with the servomotors disengaged
  - engages the servomotors during manual operation in pitch and roll
27. **A system which can still function without degradation of performance after a failure has:**
- fail-passive ability
  - fail-soft ability
  - fail-operational ability
  - fail-symbol ability
28. **During a CAT 2 ILS automatic approach, the source for altitude information is the:**
- basic altitude capsule stack
  - radar altimeter which becomes effective below about 2500 feet
  - radio altimeter which becomes effective below about 2500 feet
  - mode comparator sensor

29. **Heading hold mode relates to control in:**
- the height lock via the CADC
  - the pitch channel via the inner loop
  - the roll channel via the outer loop control source
  - the manometer mode of the CADC
30. **The system which allows the pilot to control the aircraft with the servomotors engaged is called:**
- touch control steering
  - control wheel steering
  - the electronic inner/outer axis loop
  - the outer loop control
31. **The type of automatic landing system which would necessitate a manual landing after a system failure during an automatic approach is:**
- fail-passive
  - fail-safe
  - fail-active
  - fail-operational
32. **After a failure of one of the necessary redundant systems below alert height you would:**
- continue the descent but revert to a higher DH
  - carry out a missed approach
  - disengage autoland and take over manually
  - continue descent and land automatically
33. **When localizer and glide slope are captured at 1500 feet during an automatic landing sequence, two other functions will be activated at the same time; they are:**
- touchdown mode and roll out mode
  - flare mode arm and touchdown mode
  - flare mode engage and roll out mode
  - flare mode arm and off line channels engaged
34. **A fundamental requirement of a closed loop servomechanism is:**
- a stable reference device
  - an interlock control
  - a tacho-generator
  - feedback
35. **ALT HOLD is an example of:**
- inner loop control in the roll axis
  - outer loop input to the pitch channel
  - outer loop control about the longitudinal axis
  - inner loop control in the pitch axis

36. **A rate gyro:**
- a. has three degrees of freedom, two gimbals and a transducer
  - b. senses rate of turn and positions an indicator on the EHSI
  - c. supplies rate and displacement information to the computer
  - d. controls the outer loop inputs
37. **To prevent servomotor runaway from producing excessive demands to the control surface:**
- a. a gyro damper is fitted
  - b. a torque limiter is fitted
  - c. a gyro limiter is fitted
  - d. a torque converter is fitted
38. **Auto-trim is functional:**
- a. in the pitch and roll channel with the autopilot engaged
  - b. in the pitch channel only with the autopilot engaged
  - c. in the pitch channel only with the autopilot disengaged
  - d. in the pitch and roll channel with the autopilot disengaged
39. **L NAV is an..... input to the .....channel using data from the.....**
- a. outer loop, pitch, FMC
  - b. inner loop, pitch, ADC
  - c. outer loop, roll, FMC
  - d. inner loop, roll, ADC
40. **In an aircraft which requires a Mach trim system it will apply inputs to the horizontal stabilizer:**
- a. all the time
  - b. at high Mach numbers with the autopilot engaged
  - c. at Mach one with the autopilot engaged or not
  - d. at high subsonic speeds with the autopilot engaged or not



## Question Paper 2

1. **With the autopilot engaged in the Alt mode the Captain alters the barometric setting. The aircraft**
  - a. maintains its altitude
  - b. changes its altitude in accordance with the change in pressure setting
  - c. switches barometric input over to the 1st Pilot setting
  - d. trips out of altitude hold
  
2. **Control wheel steering enables a pilot to:**
  - a. taxi the aircraft on the ground
  - b. manoeuvre the aircraft in the air while the autopilot is engaged
  - c. alter the flight path while the autopilot is engaged by applying a breakout force
  - d. manoeuvre the aircraft with the autopilot disengaged
  
3. **Autopilot synchronization in an aircraft:**
  - a. requires that the interlocks are made before the autopilot will engage
  - b. ensures that, when the autopilot is engaged, the take-over is effected smoothly and without snatching on the control system
  - c. requires that the aircraft is trimmed out before the autopilot can be engaged
  - d. needs at least two alternators running in parallel
  
4. **The rules for the use of auto-trim are that it:**
  - a. can be engaged without the autopilot
  - b. usually operates on all three axes
  - c. is not needed if the autopilot is engaged
  - d. operates only in conjunction with the autopilot
  
5. **The JAR OPS requirements for single pilot operation under IFR state that the aircraft must be fitted with:**
  - a. a single axis autopilot
  - b. a two axis autopilot
  - c. a three axis autopilot
  - d. a two axis autopilot with autothrottle
  
6. **JAR 25 operational requirements for the installation of automatic pilot state that the system must have:**
  - A. automatic synchronization
  - B. quick release controls on both control wheels.
  - a. Only statement A is correct
  - b. Only statement B is correct
  - c. Both statements are correct
  - d. Neither statement is correct

7. Consider the following statements regarding flight envelope protection:
- A** High speed protection prevents the airspeed from exceeding  $V_{MO}/M_{MO}$   
**B** High angle of attack protection comes in when the aircraft reaches the stalling AoA
- a. Only statement A is correct  
b. Only statement B is correct  
c. Both statements are correct  
d. Neither statement is correct
8. The control laws for an autopilot are known as:
- a. normal law and emergency law  
b. alternate law and direct law  
c. normal, alternate and emergency laws  
d. normal, alternate and direct laws
9. An autoland system that, in the event of an autopilot failure, continues to function without degradation of performance beyond the limits required automatic, would be one with the status:
- a. fail-passive  
b. fail-safe  
c. fail-operational  
d. duplex
10. The autoland sequence is considered to be complete when:
- a. reverse thrust is engaged  
b. the autopilot is manually disengaged by the pilot  
c. the aircraft touches down  
d. the aircraft reaches the end of the runway
11. The autothrottle will come on automatically even with the A/T switch OFF when:
- a. in a FBW aircraft the AoA reaches a critical value called floor  
b. the AoA reaches the stalling angle  
c. TOGA button is pressed  
d. reverse thrust is selected in flight
12. An aircraft on autopilot is engaged in the VOR mode and loses the VOR signals as it flies through the VOR cone of silence. The autopilot:
- a. automatically switches to Heading mode  
b. decouples from the VOR and disconnects  
c. tunes to the next VOR on the route  
d. decouples from the VOR and flies the last heading for a fixed period

13. For an aircraft with a non-synchronized autopilot system, 'snatching' of the controls by the autopilot when engaging or disengaging can be prevented by:
- the pilot ensuring that the aircraft is trimmed out before selecting or disengaging the autopilot
  - being in a straight and level position
  - disengaging the auto-trim
  - switching on the yaw dampers
14. With the autopilot in CWS the pilot manoeuvres the aircraft and releases control. The aircraft will maintain:
- heading and altitude
  - heading, speed and attitude
  - altitude and attitude
  - attitude at the time of release
15. Autopilot corrections affecting pitch are carried out by:
- auto-trim only
  - auto-trim and elevators
  - elevators only
  - autothrottle
16. For a commercial aircraft operating with a single pilot in IFR the minimum requirement is that the autopilot should have control in:
- three axes
  - Heading mode
  - Altitude Hold and Heading mode
  - Altitude Hold, Heading mode and Speed

## Answers - Paper 1

1	2	3	4	5	6	7	8	9	10	11	12
d	b	c	a	b	a	a	a	b	b	c	c
13	14	15	16	17	18	19	20	21	22	23	24
d	b	b	b	d	c	d	a	a	b	c	a
25	26	27	28	29	30	31	32	33	34	35	36
a	c	c	c	c	b	a	d	d	d	b	c
37	38	39	40								
b	b	c	d								

## Answers - Paper 2

1	2	3	4	5	6	7	8	9	10	11	12
a	b	b	d	b	b	d	d	c	b	a	d
13	14	15	16								
a	d	b	c								

Chapter

# 32

## Flight Warning Systems

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The Flight Warning System (FWS) . . . . .	.435
FWS Components . . . . .	.436



## Introduction

The purpose of the Flight Warning System (FWS) is to produce cautions and warnings for the crew to increase their situation awareness and to give them suitable indications of the action necessary to avoid impending danger.

The proliferation of various warning systems in today's aircraft poses a severe problem in that the crew could be confused by the multiplicity of warnings. It is therefore necessary to install an integrated flight warning system that will prioritize the warnings. By producing warnings relevant to a particular stage of flight and inhibiting other warnings the system enables the crew to respond to the warning posing the most immediate threat to safety.

## Levels of Alerts

The alerting and warning system produces the following levels of alerts:

- **Warnings or Level A alerts.** These require immediate crew action. Warnings must attract the pilot's attention in sufficient time for appropriate action to be taken.
- **Cautions or Level B alerts.** These require immediate crew alertness and subsequent crew action.
- **Advisories or Level C alerts.** These require crew alertness.

## Warnings in General

The alerting and warning messages are presented to the crew in visual, aural and sensory (tactile) forms.

### Visual

The level of alert is indicated by colours as follows:

- Warnings are presented in **red**
- Cautions are shown in **amber** or **yellow**
- Advisories are any colour except **red** or **green**. (On EICAS panels they are also **amber**)

These visual indications can be presented in two different forms:

- **Electronic Screens.** Alerts and warnings appear in coloured text or symbols on various electronic screens (flight, navigation, engine and aircraft system displays).
- **Lights or Flags.** Red lights or reflective flags signify warnings and require remedial action if flight is to continue. An amber light or flag is used to indicate that a system or equipment is approaching a limit of normal function and that corrective action is necessary to prevent further deterioration and consequent failure.

Additionally, master warning and caution lights are normally provided and are located near the centre of scan in front of each pilot.

### *Aural*

An audible warning is mandatory if the pilot is required to assume control. This can be in a variety of forms depending upon the type of aircraft. The alert can be in the form of sounds or synthetic voice messages or a combination of both. For multiple alerts, the warnings are prioritized: Stall, Windshear, GPWS, ACAS.

### **Warnings**

Boeing aircraft produce the following aural warnings:

- A bell accompanies fire messages
- A siren accompanies warnings on cabin altitude, configuration and overspeed
- A wailer accompanies autopilot disconnect
- Synthetic voice messages for ground proximity, windshear, airborne collision avoidance

Airbus aircraft produce:

- continuous repetitive chimes (red warnings)
- cavalry charge (autopilot disconnect)
- cricket sound (stall warning)
- synthetic voice (GPWS, TCAS warnings)

### **Cautions**

Beepers with various tones or chimes or musical chords are used to caution the crew to potential threats to safety.

### *Sensory*

A vibratory mode on the controls is used to indicate stall approach and demands immediate action to avert loss of control. In some aircraft a stick-pusher provides guidance to prevent a further deterioration of the situation that demanded the vibratory warning.

To rationalize warnings systems, a Master Warning Indicator light is often provided near the centre of scan. In older systems the crew member would then refer to a Master Centralized Warnings Panel (CWP) where warnings were assembled in a rational order and annotated. In the modern Electronic Flight Instrumentation Systems (EFIS) most of the alerts and warnings appear on appropriate electronic screens together with associated aural messages and master warning lights.



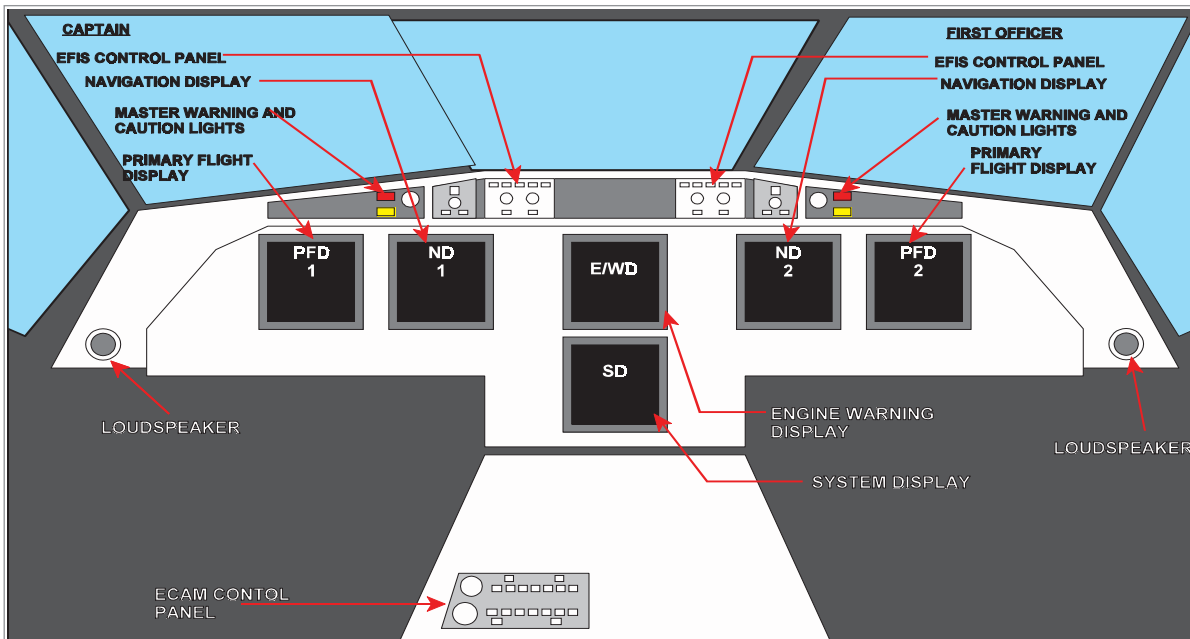


Figure 32.1 Shows the cockpit displays and warnings of an Airbus A320

## The Flight Warning System (FWS)

### General

The Flight Warning System generates alerts and warnings for the following situations:

- Engine and airframe systems malfunctions
- Aerodynamic limits exceeded
- Presence of external hazards

### Engine and airframe systems malfunctions

These are dealt with in detail in the Engines and the Systems sections of the course.

### Aerodynamic limits

If aerodynamic limits are exceeded the FWS provides the following alerts to the crew:

- Altitude Alerting
- Overspeed Warning
- Stall Warning

These are dealt with in the next chapter.

### External hazard warning

The external hazards that constitute a threat to aircraft safety are proximity to terrain and to other aircraft. These hazards can be avoided by the use of:

The Ground Proximity Warning System and

The Airborne Collision Avoidance System. These are dealt with in Chapters 34 and 35.

### FWS Components

The FWS system comprises:

#### Inputs

There are inputs from various sources including hundreds of engine and airframe sensors, air data sensors, GPWS and ACAS systems.

#### A processing unit

This is made up of one or two flight warning computers.

#### Outputs

The outputs are classified either as alerts or as warnings and are generated according to the nature of the malfunction or threat to safety. Alerts can be visual (amber lights or text on VDUs) or aural (chimes or tones). Warnings are given in the form of red lights or by red text on electronic screens (steady or flashing) as well as aural signals (siren, bell, hooter). Additionally there are red and amber lights on the glare shield in front of the pilots to act as attention getters. A block diagram of a Boeing 767 warning and alert system is shown at [Figure 32.2](#).

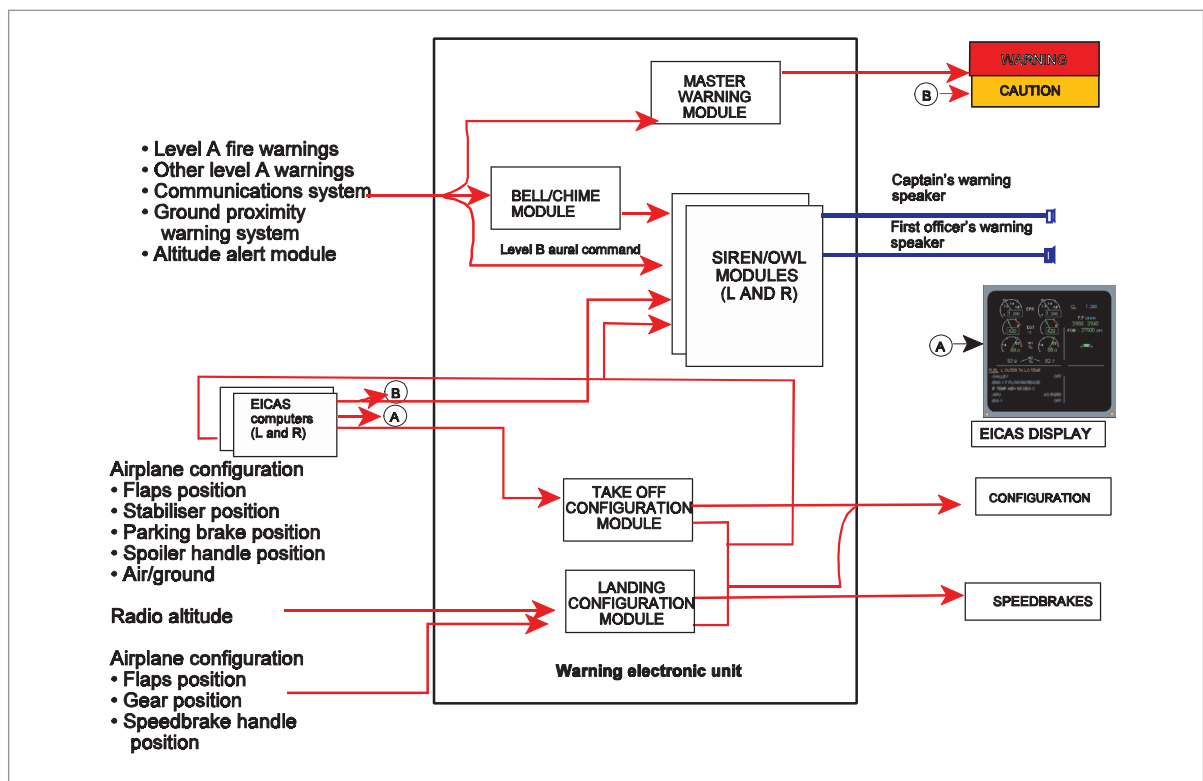


Figure 32.2 Warning and alerting system (Boeing 767)

Chapter

**33**

**Aerodynamic Warnings**

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## Introduction

The Flight Warning System alerts the crew if there are deviations from certain aerodynamic parameters like altitude, airspeed and angle of attack. The system therefore provides the following alerts and warnings:

- Altitude Alerting System
- Overspeed Warning
- Stall Warning

## Altitude Alerting System

### Function

The function of the altitude alerting system is to warn the pilots that the aircraft is approaching or deviating from the altitude selected on the autopilot control panel. It does this in certain height bands above and below the selected altitude.

### Operation

The height bands within which altitude alerting operates are typically 300 feet to 900 feet for Boeing aircraft and 250 feet to 750 feet for Airbus aircraft. *Figure 33.1* shows the operation of altitude alerting on a Boeing 747-400.

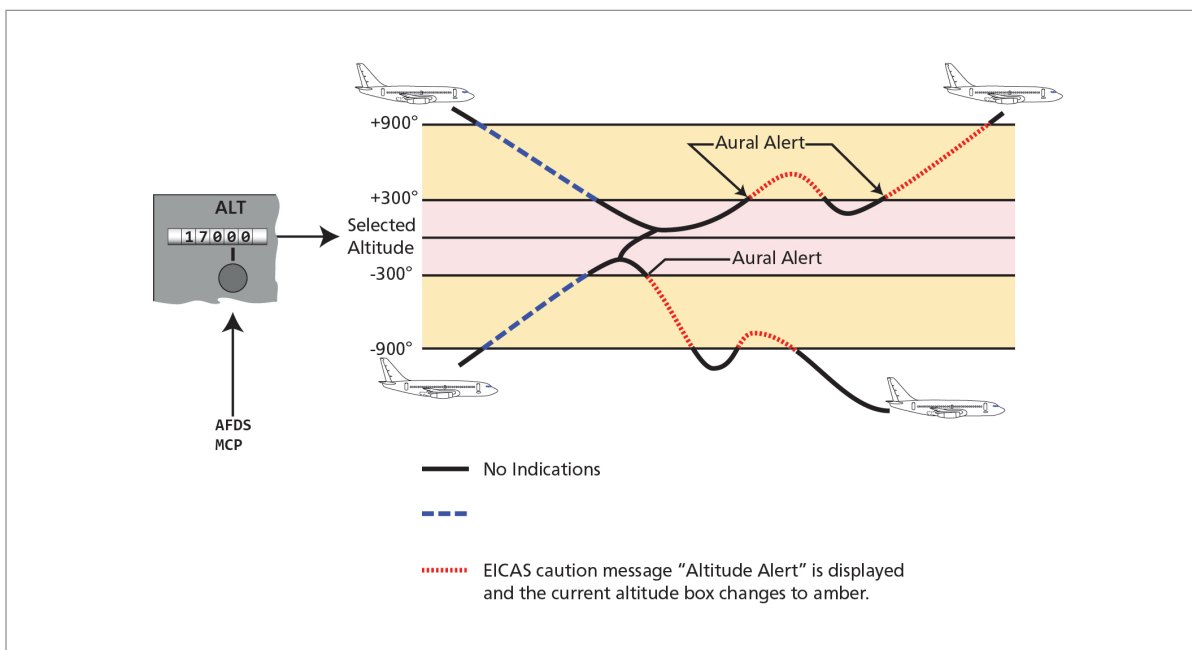


Figure 33.1 Altitude alert (Boeing 747-400)

### Approaching a Selected Altitude

At 900 feet prior to the selected altitude a white box will be displayed around the current altitude display on the PFD, together with a momentary audible alert. At 300 feet prior to the selected altitude the white box disappears.

### Deviation from Selected Altitude

At 300 feet from the selected altitude:

- Master caution lights illuminate
- Continuous caution beeper sounds
- EICAS caution message ALTITUDE ALERT is displayed
- Current altitude box changes to amber

The audible alert and master caution light can both be cancelled by pressing the light. The warning ceases on return to within 300 feet of the selected altitude, or when the aircraft exceeds 900 feet from the selected altitude.

### Block Diagram

Figure 33.2 shows a block diagram of a Boeing 767 altitude alerting system. When the aircraft approaches the selected altitude the advisory light on each electric altimeter illuminates. If the aircraft deviates by more than 300 feet from the selected altitude the system generates a level B warning (i.e. a caution) consisting of a level B message on the EICAS display, an alert tone from the speakers and illumination of the master caution (amber) light and the ALT ALERT light.

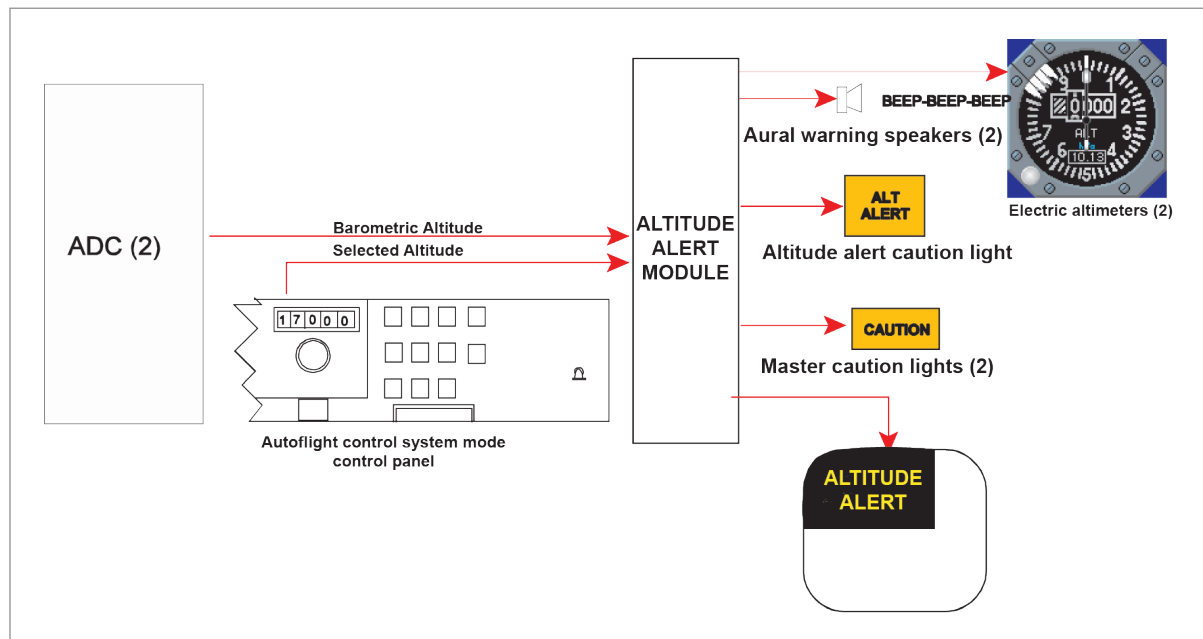


Figure 33.2 Altitude alerting system (Boeing 767)

### System Inhibition

Altitude alerting is inhibited in flight whenever glide slope is captured or when landing flaps are selected with the gear down.

### Regulatory Requirement

It is a requirement for EASA registered Commercial Air Transport aircraft to be equipped with an altitude alerting system if it is:

- a turboprop aircraft weighing more than 5700 kg or has more than 9 passenger seats or
- a turbojet aircraft

The altitude alerting system must be capable of:

- alerting the crew on approaching the preselected altitude
- alerting the crew by at least an aural signal when deviating above or below a preselected altitude

## Overspeed Warning

### Function

The purpose of the overspeed warning system is to alert the flight crew if the airspeed exceeds the  $V_{MO} / M_{MO}$  limits calculated by the air data computer (ADC).

### Operation

Whenever an overspeed situation occurs in an aircraft with electronic instrumentation the system:

- sounds the siren or horn
- illuminates the red master WARNING lights
- displays the message OVERSPEED on the EICAS upper display in red

The warning continues while the overspeed situation exists and cannot be cancelled by depressing the red master WARNING light switch.

The system obtains its input from the air data computers (ADCs) via the flight warning system. It can be tested on the ground before flight by pressing a test switch which would then sound the siren or horn as appropriate for that aircraft.

In case of system failure the pilot would receive no warning if  $V_{MO}$  or  $M_{MO}$  is exceeded.

### Displays

The maximum allowable speed is shown on the airspeed indicator by means of a barber's pole on a conventional instrument and on the airspeed tape on the primary flight display or EADI of an EFIS display. These are shown in [Figure 33.3](#) and [Figure 33.4](#). The barber's pole indicates the  $V_{MO}$  up until the  $M_{MO}$  becomes limiting. The barber's pole will then move anti-clockwise to indicate the maximum allowable speed. As altitude increases when climbing at a constant indicated airspeed the  $M_{MO}$  when expressed as an indicated airspeed will decrease.

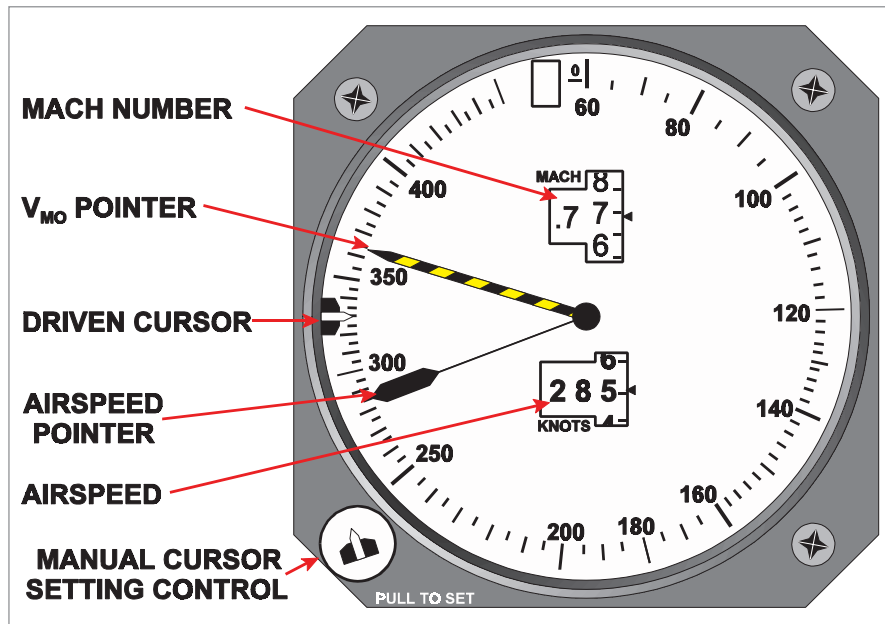


Figure 33.3 Conventional ASI with V<sub>MO</sub> pointer

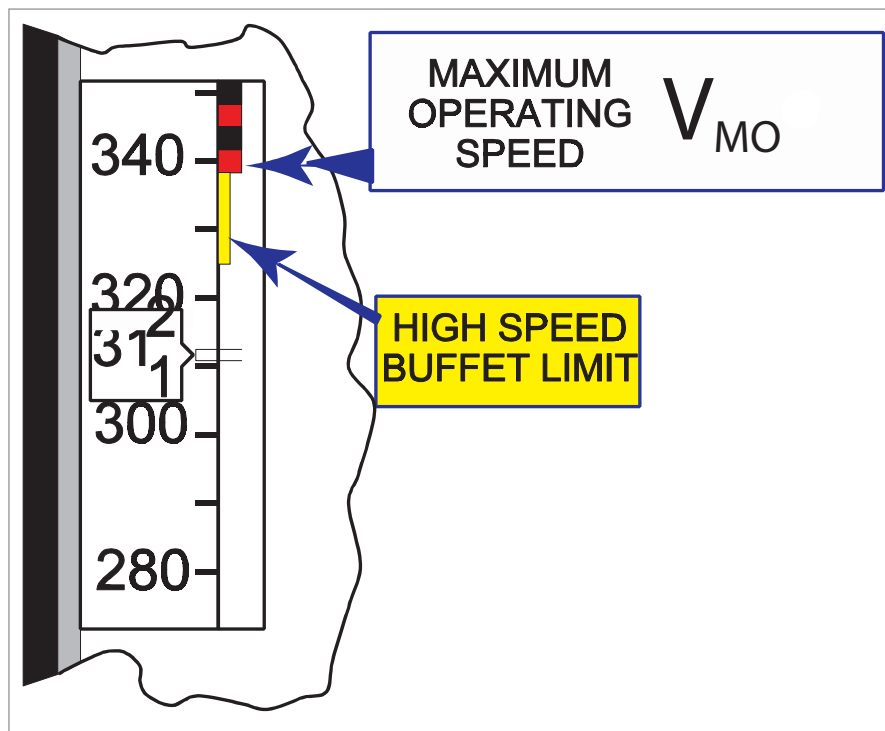


Figure 33.4 Overspeed warning on PFD



## Stall Warning System

### Function

The purpose of the stall warning system is to warn the pilot of an impending stall. It does so when the aircraft approaches the stalling angle of attack for the current speed and configuration of the aircraft.

### Stall Warning Systems

The simplest form of system, and one which is adopted in several types of small aircraft, consists of a hinged-vane-type sensor mounted in the leading edge of a wing so that the vane protrudes into the airstream. In normal level flight conditions, the airstream maintains the vane in line with the relative air flow. If the aircraft's attitude changes such that the angle of attack (AoA) increases, then, by definition, the airflow will meet the leading edge at an increasing angle, and so cause the vane to be deflected. When it reaches the angle at which the warning unit has been preset, the vane activates a switch to complete a circuit to an aural warning unit in the cockpit.

In larger types of aircraft, stall warning and prevention systems are designed to perform a more active function, such as 'stick-shaker' or 'stick-push or nudger' type.

### The Warning

The regulatory margin between the stall and the stall warning is 5 knots or 5% of the CAS whichever is the greater.

The warning provided can be in the form of tactile, aural or visual or a combination of these signals. Most aircraft have warning provided by stick-shakers which vibrate the control column as well as produce a rattling noise. In fly-by-wire systems the warning consists of a cricket (insect) sound, a synthetic voice STALL message and the red master WARNING light illumination.

The stall warning must continue until the angle of attack is reduced to approximately that at which the stall warning is initiated.

### Operation

The stall warning module processes the signals from the various inputs to produce appropriate stall warning output signals. The system has the following inputs:

- angle of attack
- flap and slat positions
- landing gear weight-on position
- airspeed

The angle of attack sensors are usually located on either side of the front fuselage. Sensing relays denote the positions of the flaps and slats. Since the pitch attitude of the aircraft is also changed by the extension of flaps or slats the angle of attack signal has to be modified when these are extended. During take-off when the nosewheel lifts off, microswitches operate to make the stall warning system active. The airspeed is usually derived from the ADC.

The output signals from the system can be applied to:

- a stick-shaker motor
- an angle of attack indicator aural warning
- synthetic voice warning
- red master WARNING light

### Stall Protection

A stall protection system may be fitted to large commercial aircraft, to prevent them from entering the stall. In fly-by-wire systems, the flight computer will not allow the aircraft to approach the stall. In other aircraft an output from the AFCS advances the throttles to full power if there is a deceleration to below  $1.2V_s$ . On aircraft which have a T-tail a stick-pusher may be fitted, to push the control column forward in the event that the aircraft slows to 2 kt above the stall speed. This will prevent the aircraft entering a deep stall, from which there is little or no chance of recovery.

### Components

A block diagram of the component parts of a stall warning system and an angle of attack sensor are shown at [Figure 33.5](#).

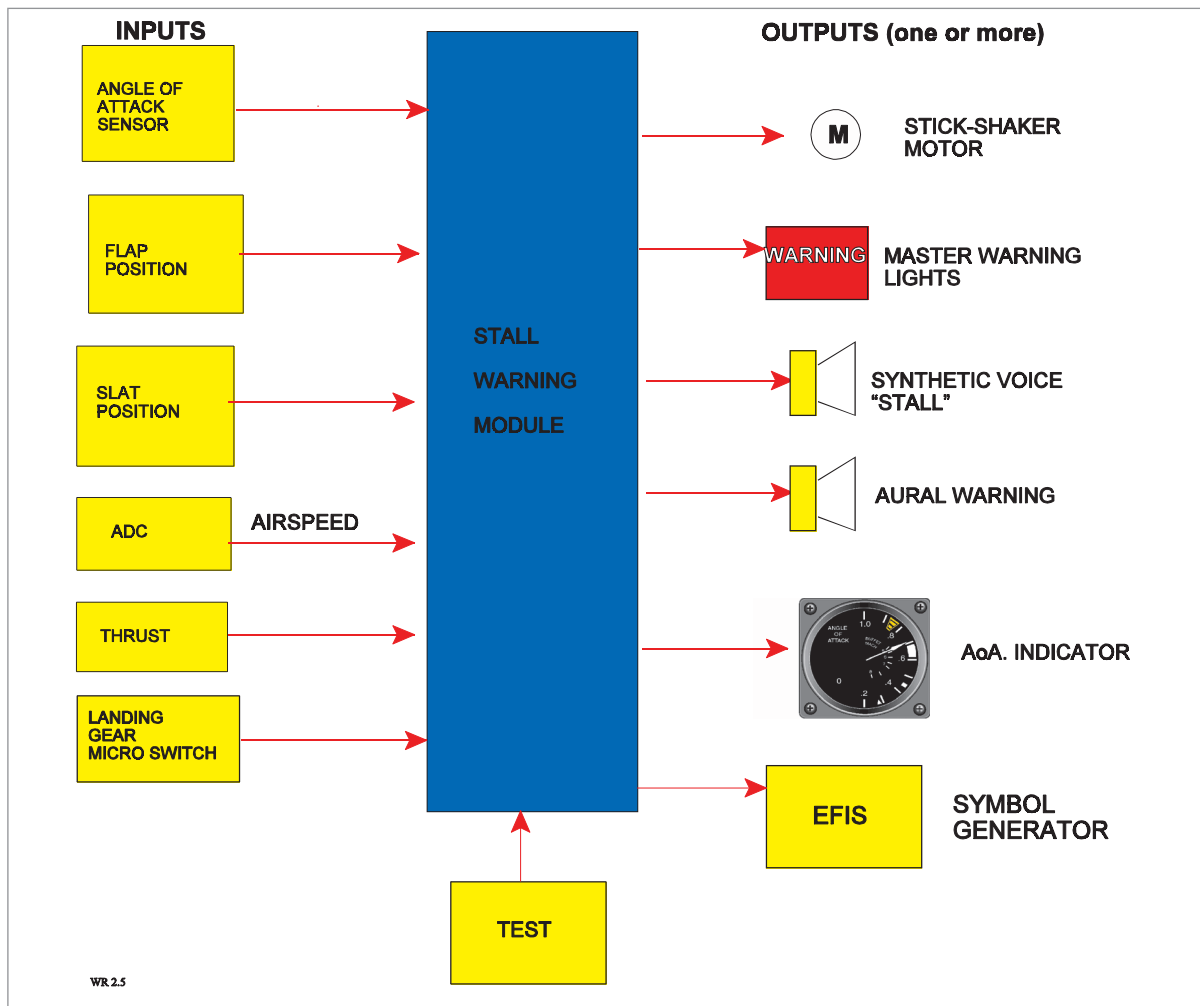


Figure 33.5 components of a stall warning system

### Angle of Attack Sensing

The angle of attack (AoA), or alpha ( $\alpha$ ) angle, also known as the aerodynamic incidence, is the angle between the chord line of the wing of an aircraft and the direction of the relative airflow, and is a major factor in determining the magnitude of lift generated by a wing. Lift increases as the AoA increases up to some critical value at which it begins to decrease due to separation of the slow-moving air (the boundary layer) from the upper surface of the wing, which, in turn, results in separation and turbulence of the main airflow. The wing, therefore, assumes a stalled

condition, and since it occurs at a particular angle rather than a particular speed, the critical AoA is also referred to as the stalling angle. The angle relates to the design of aerofoil section adopted for the wings of any one particular type of aircraft, and so, of course, its value varies accordingly; typically it is between  $12^\circ$  and  $18^\circ$  for straight wings but may be as high as  $30^\circ$  or  $40^\circ$  for swept or delta wings.

### Angle of Attack Probes

The two types in current use are the conical slotted probe and the vane detector; the conical slotted probe is shown in *Figure 33.6* and the vane type in *Figure 33.7*. The vane detector is a counter-balanced aerodynamic vane which positions the rotor of a synchro. Both types are protected against ice formation by a heater.

The conical probe extends through the aircraft skin perpendicular to the flow of air. The probe is attached to a paddle inside the transmitter housing. The probe and paddle are free to rotate. Two sets of slots in the probe allow pressure variations, caused by changes in airstream direction, to be transmitted through separate air passages to opposite sides of a paddle chamber. When the pressure acting on one side of a paddle is greater than the pressure on the other side, the paddle and probe rotate until the pressures are equal. The probe thus positions itself to determine the angle of attack of the aircraft. The probes or vane detectors are mounted on both sides of the fuselage, usually forward of the wing line, to compensate for side-slip/yaw. Angle of attack probes send information to the stall warning system, the ADC, and, if fitted, Flight Envelope Protection Systems and Angle of Attack Indicators.

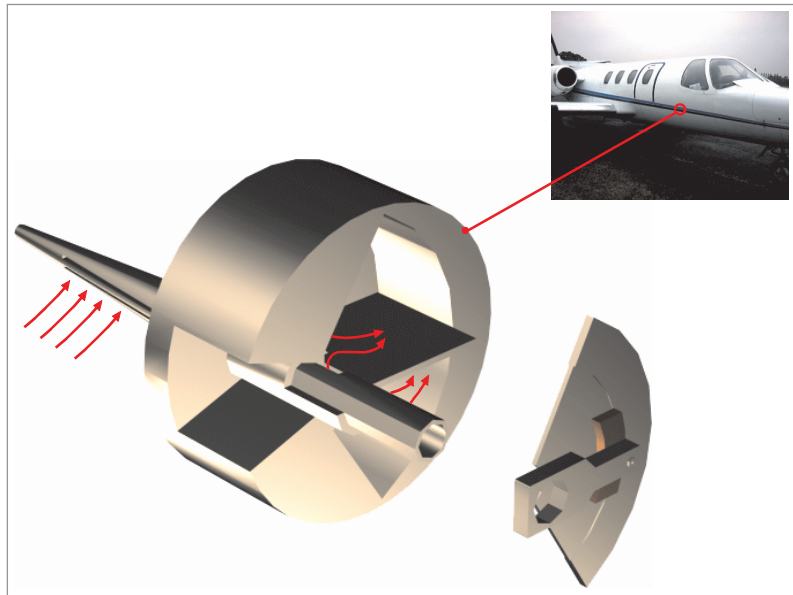


Figure 33.6

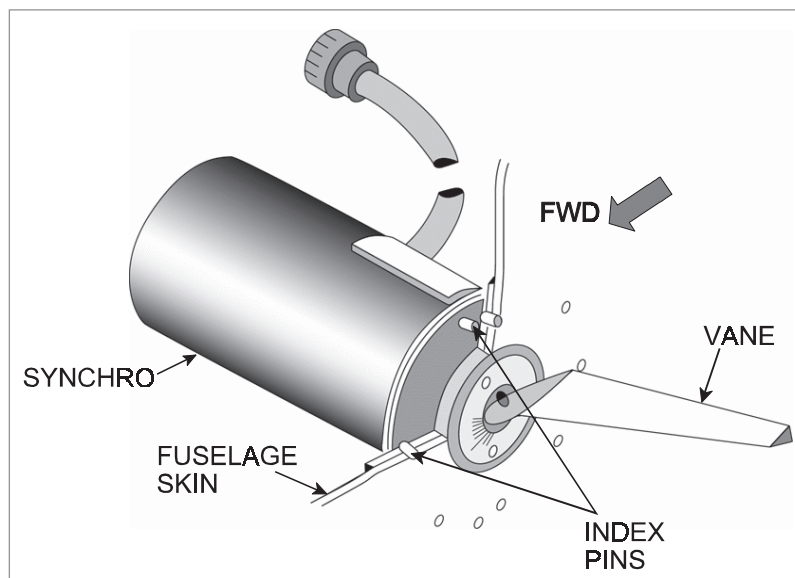
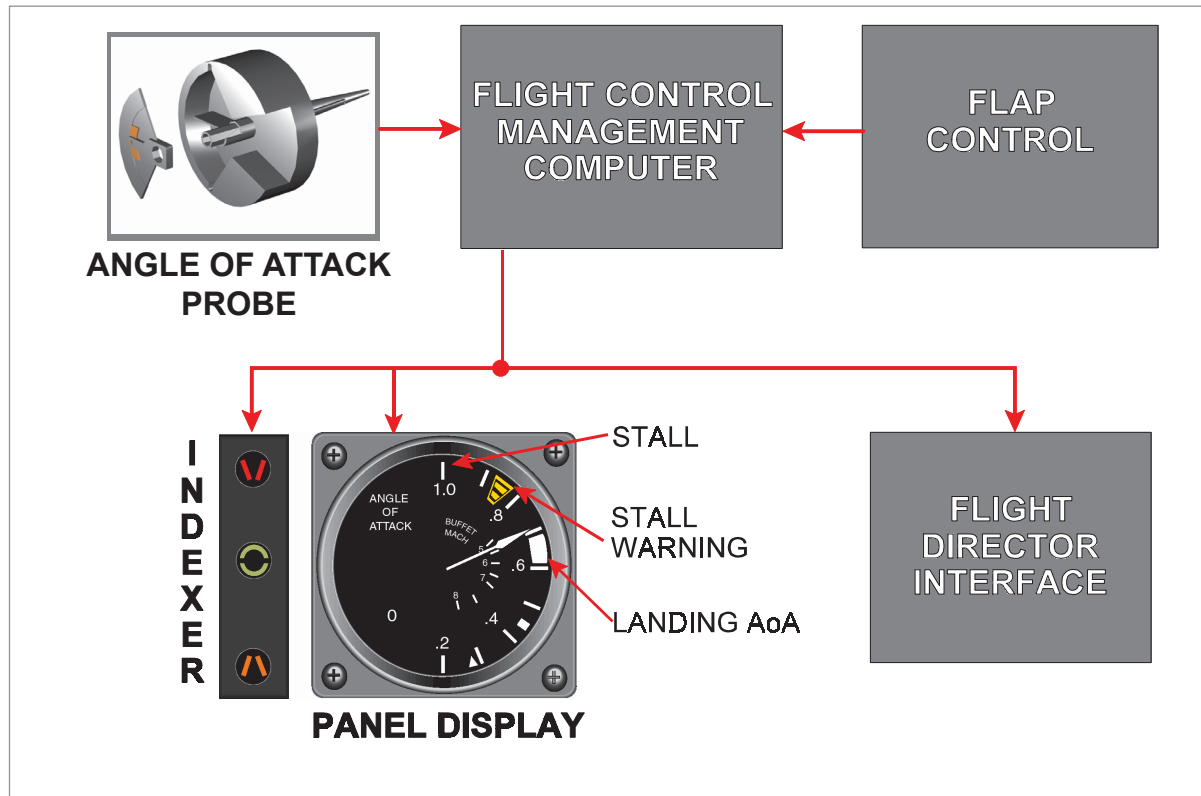


Figure 33.7 Angle of attack sensor

### Angle of Attack Indicators

These may be fitted in addition to the stall warning system. A simple schematic lay-out of the installation is shown in *Figure 33.8*.



*Figure 33.8*

### Configuration Warning

Some aircraft incorporate configuration warning systems for take-off and landing. Landing configuration would give an audible warning if certain throttles are retarded without the landing gear locked down. There may also be warnings for flaps. (This would not be required on aircraft fitted with GPWS, discussed in the next chapter).

The Take-off Configuration Warning (TOCW), as the name suggests, lets the pilot know, by means of an audible warning, that the aircraft is not in the correct configuration for take-off. Different aircraft may have different parameters for activation, but typically the TOCW would sound if throttles are advanced with:

- Flaps not in take-off position
- Slats not in take-off position
- Stabilizer trim outside take-off range
- Spoilers/speedbrakes deployed
- Remotely operated flight control locks not disengaged
- External doors/hatches not locked closed
- Parking brake applied

# Chapter 34

## Ground Proximity Warning System

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## Introduction

The aim of the system is to give **visual and audible warning** signals to a pilot when the aircraft's **proximity to the terrain poses a potential threat to its safety**. Although not a foolproof means of preventing a collision with the earth's surface, **GPWS enhances flight safety** and can prevent those accidents which could result from crew errors or distraction, malfunction or misinterpretation of navigational equipment, or inappropriate ATC instructions.

*Figure 34.1* shows the three elements of a GPWS: inputs, outputs and a central processing unit. The central processing unit will also indicate a computer failure and any failures of the six input signals. The system operates between **50' and 2450' actual height above the surface** and automatically selects the correct mode of operation.

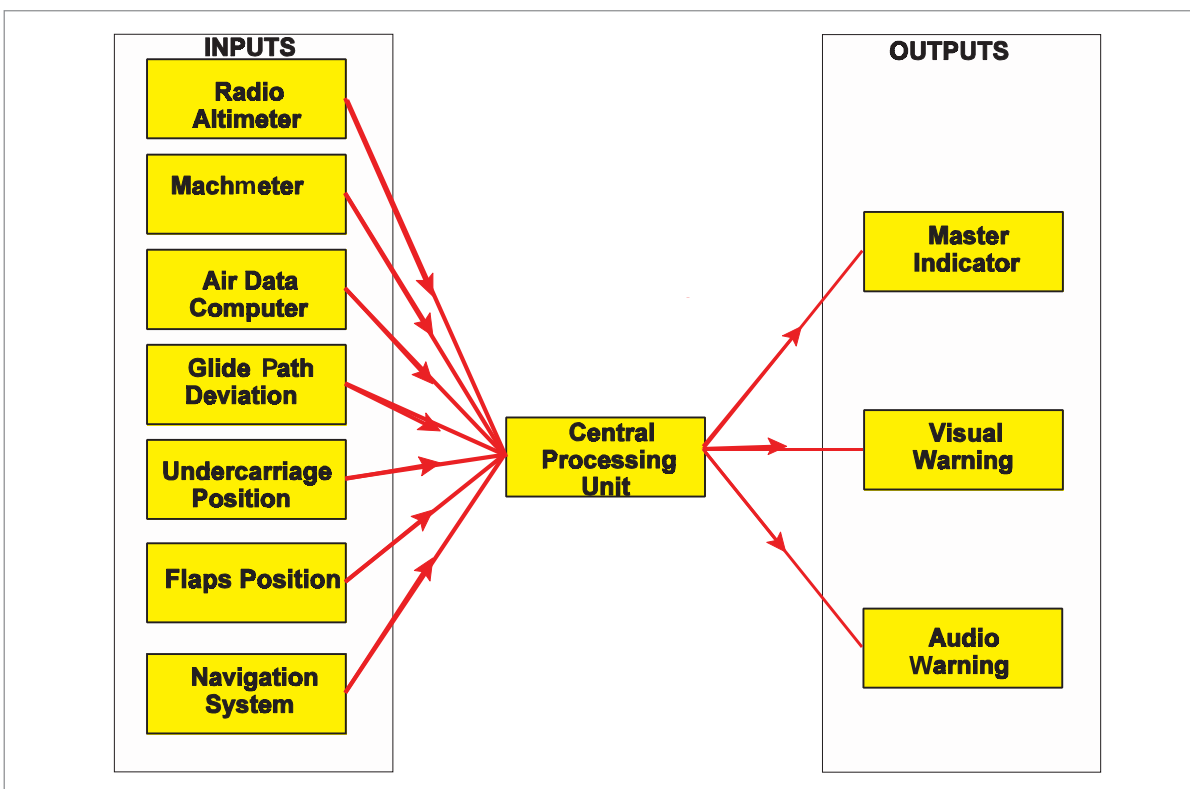


Figure 34.1

## Definitions

**ALERT:** A caution generated by the EGPWS equipment.

**WARNING:** A command generated by the EGPWS equipment.

## Types of Warnings/Alerts

### Genuine

The equipment provides a warning in accordance with its technical specification.

### Nuisance

The equipment provides an alert in accordance with its technical specification, but the pilot is flying an accepted safe procedure.

### False

A fault or failure in the system causes the equipment to provide a warning that is not in accordance with its technical specification.

### Boeing 737 Mark II EGPWS

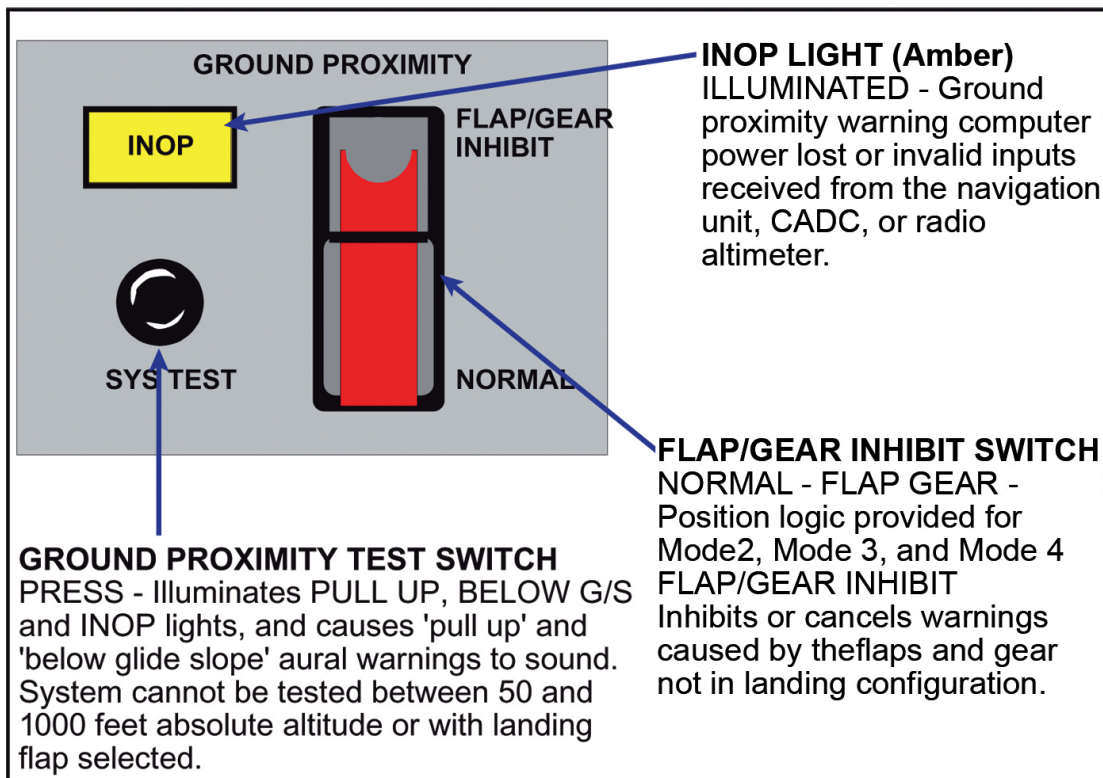
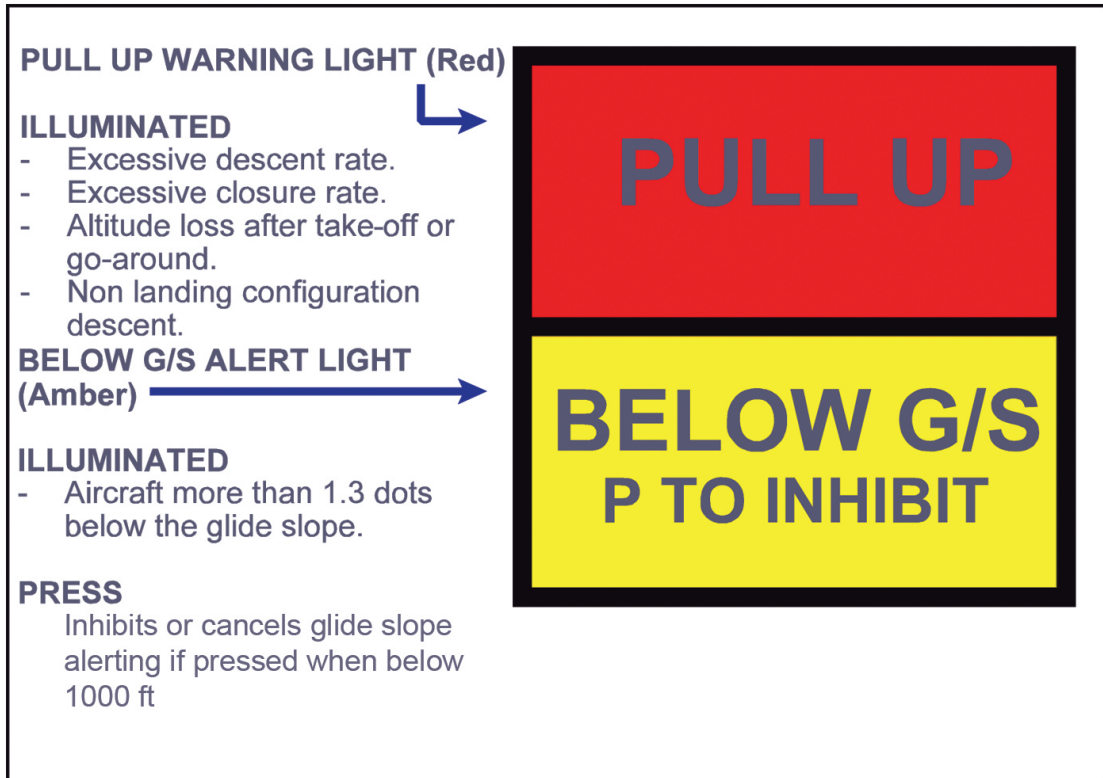


Figure 34.2 First officer's instrument panel



## A Table of the GPWS Operating Modes

GPWS MODE		ADVANCED EQUIPMENT Alert                      Warning	
1. Excessive descent rate		'Sink Rate'	'Whoop Whoop Pull Up'
2. Excessive terrain closure rate		'Terrain Terrain'	'Whoop Whoop Pull Up'
3. Altitude loss after take-off or go-around		'Don't Sink'	-
4. Unsafe terrain clearance while not in the landing configuration	4A. Proximity to terrain - Gear not locked down	'Too Low gear'	'Too Low Terrain'
	4B. Proximity to terrain - Flaps not in a landing position	'Too Low Flaps'	'Too Low Terrain' (see note below)
5. Descent below glide slope		'Glide slope'	-
6. Descent below 'minimums'	6A.	'Minimums'	-
	6B.	'Bank angle'	-
7. Windshear warning		-	'Windshear'

**Note:** Although some manufacturers of GPWS equipment may show in their literature 'Too Low Terrain' to be an alert, the view of the CAA is that the response to this should be as for a warning.

## MODE 1 - Excessive Barometric Descent Rate

Mode 1 has two boundaries and is independent of aircraft configuration. Penetration of the **first boundary** generates an aural alert of **"SINK RATE"** repeated each 1.5 seconds.

Penetrating the **second boundary** causes the repeated warning of **"WHOOOP, WHOOP PULL UP"**, until the rate of descent has been corrected.

### MODE 1

AURAL ALERT - SINK RATE, SINK RATE

AURAL WARNING - 'WHOOOP WHOOP PULL UP'

VISUAL - PULL UP

**MODE 1**

AURAL ALERT - SINK RATE, SINK RATE  
 AURAL WARNING - 'WHOOOP WHOOP PULL UP'

VISUAL - **PULL UP**

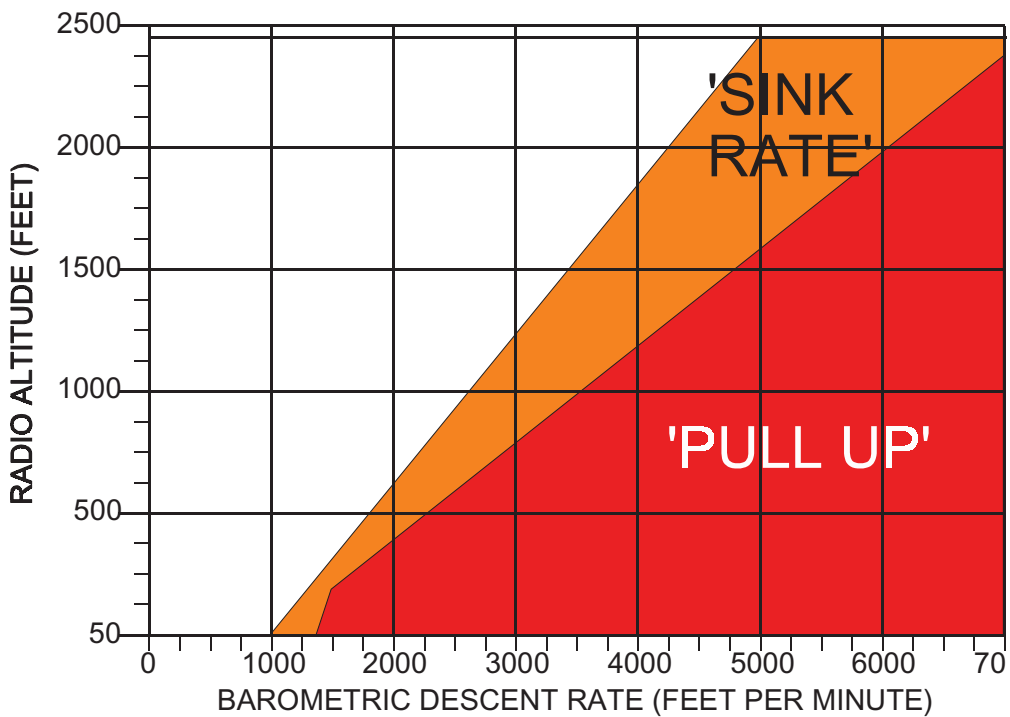
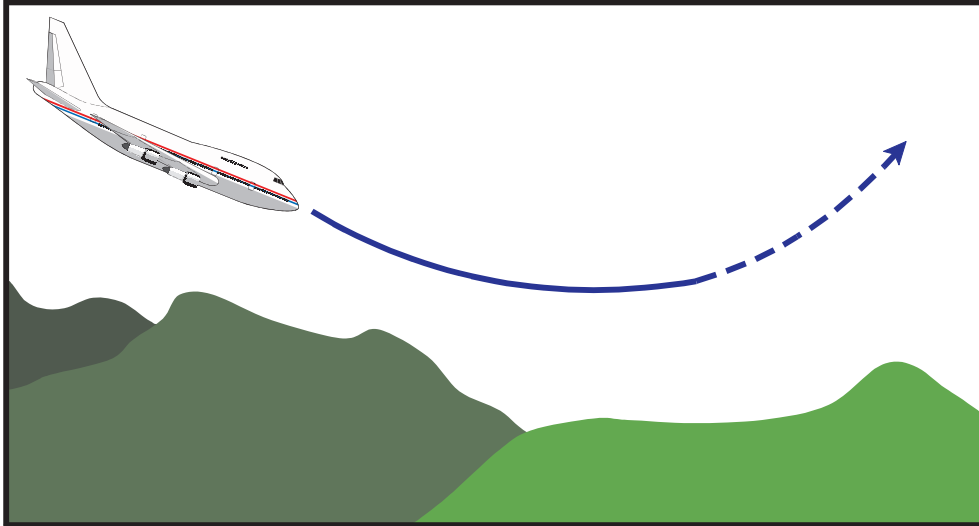


Figure 34.3 MODE 1

## MODE 2 - Excessive Terrain Closure Rate

Mode 2 monitors **Mach number, radio altitude rate of change, barometric altitude and aircraft configuration.**

Mode 2 has two boundaries. Penetrating the **first boundary** causes an **aural alert** of **"TERRAIN, TERRAIN"**, followed by the repeated aural warning **"WHOOOP, WHOOOP PULL UP"**. After leaving the PULL UP area, the repeating TERRAIN message will again be heard while in the terrain portion of the envelope. If both boundaries are penetrated while in the landing configuration, only the repeating TERRAIN aural alert will occur. The terrain message is repeated each 1.5 seconds.

**As Mach number increases from 0.35 to 0.45 with gear up, the highest radio altitude at which Mode 2 alert warning will occur is increased to 2450 feet.** This higher portion of the envelope is inhibited with the flap override switch in the FLAP OVRD position.

### MODE 2

AURAL ALERT - 'TERRAIN, TERRAIN'

AURAL WARNING - 'WHOOOP WHOOP PULL UP'

VISUAL - PULL UP

MODE 2

AURAL ALERT - 'TERRAIN, TERRAIN'  
 AURAL WARNING - 'WHOOOP WHOOP PULL UP'

VISUAL -

**PULL UP**

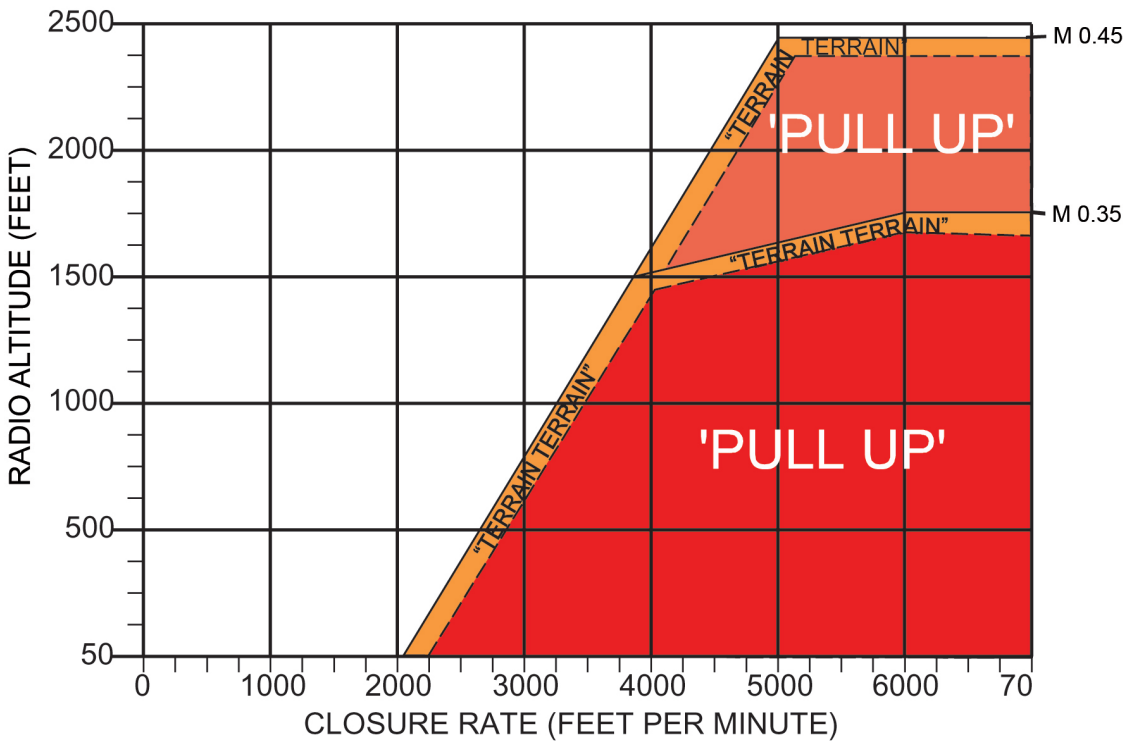
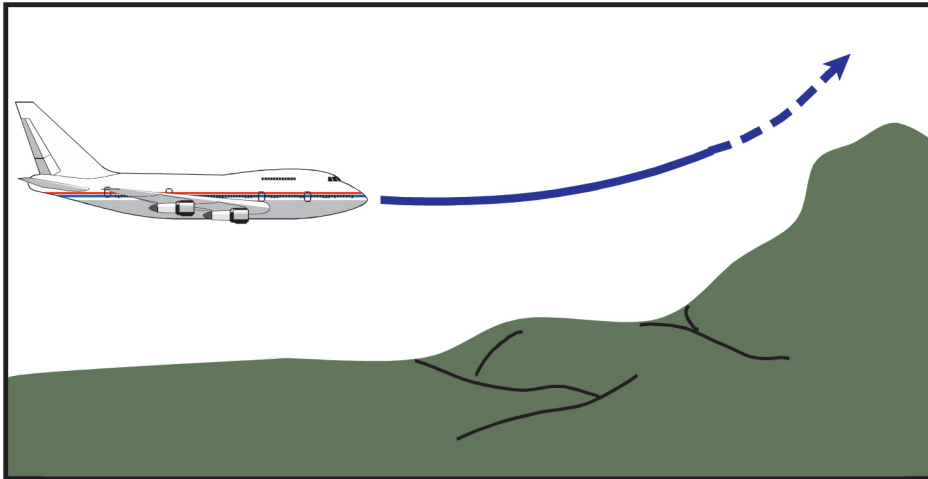


Figure 34.4 MODE 2

## MODE 3 - Altitude Loss after Take-off or Go-around

Mode 3 provides an alert if a descent is made during initial climb or go-around. The aural alert is a voice message of **"DON'T SINK"**, repeated each 1.5 seconds until the flight condition is corrected.

Mode 3 is effective between **50 and 700 feet** radio altitude and generates the alert when the accumulated barometric loss equals approximately **10 percent** of the existing barometric altitude.

Mode 3 does not **arm during the descent until below 200 feet** radio altitude.

### MODE 3

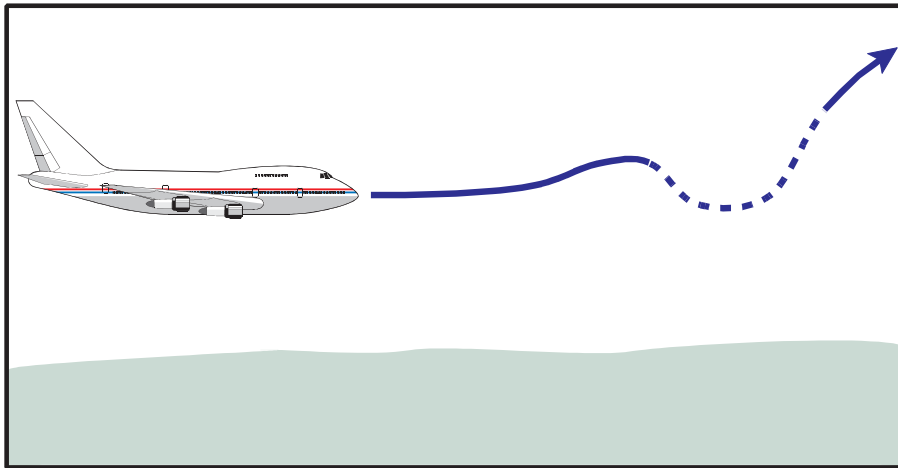
AURAL ALERT - "DON'T SINK"

VISUAL - PULL UP

MODE 3

AURAL ALERT - "DON'T SINK"

VISUAL -



Note: Mode 3 arms when the aeroplane descends below 200 ft in the landing configuration

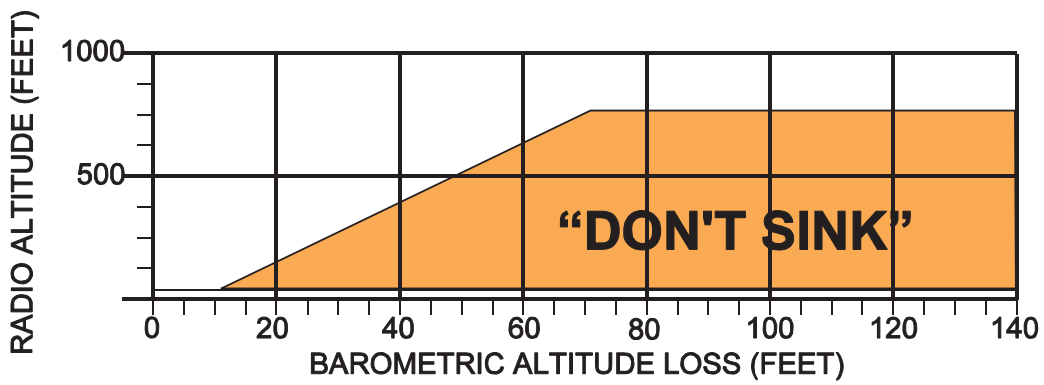


Figure 34.5 MODE 3

## MODE 4A - Unsafe Terrain Clearance with Landing Gear Not down

The terrain clearance mode, with gear retracted, is armed after take-off upon climbing through 700 feet radio altitude.

When this envelope is penetrated at less than Mach 0.35, the aural alert "TOO LOW GEAR" is sounded. When the envelope is penetrated at more than Mach 0.35, the aural alert "TOO LOW TERRAIN" is sounded and the upper boundary of the envelope is increased to 1000 feet radio altitude. The applicable voice message is repeated each 1.5 seconds until the flight condition has been corrected.

### MODE 4A

AURAL ALERT - "TOO LOW GEAR" or "TOO LOW TERRAIN"

VISUAL - PULL UP



MODE 4A

AURAL ALERT - "TOO LOW GEAR"  
"TOO LOW TERRAIN"

VISUAL -

**PULL UP**

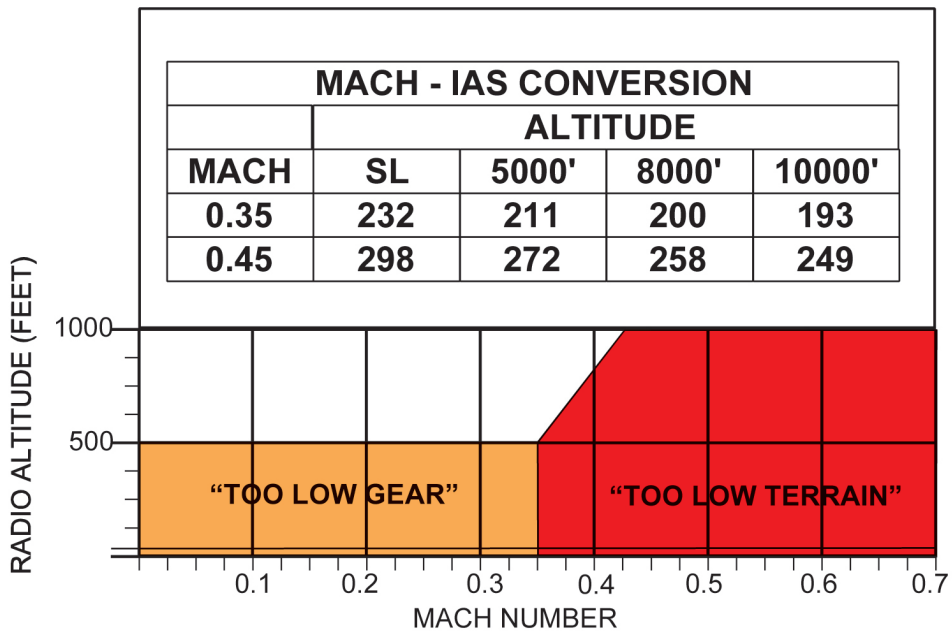
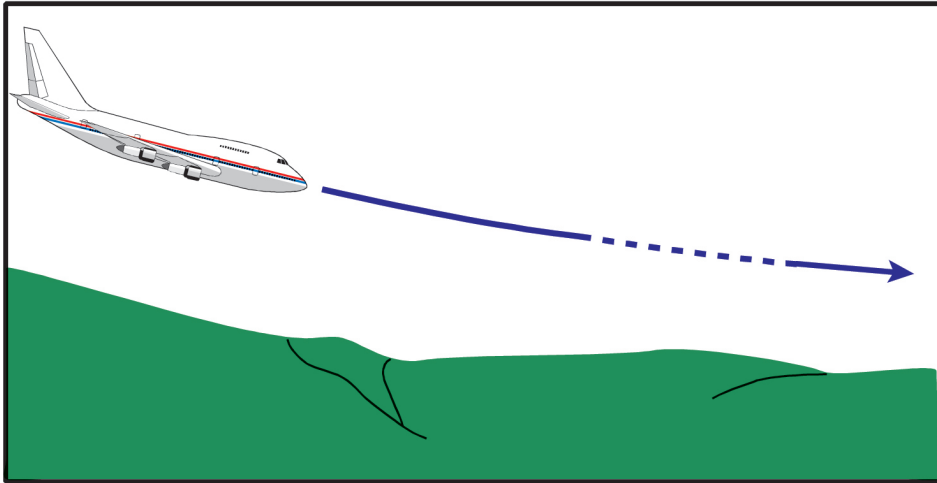


Figure 34.6 MODE 4A

## MODE 4B - Unsafe Terrain Clearance with Flaps Not in Landing Configuration

This mode provides an alert **when the gear is down** and the **flaps are not in the landing position**. If the envelope is penetrated at **less than Mach 0.28** with the flaps not in the landing position, the aural alert of **"TOO LOW FLAPS"** is sounded.

When the envelope is penetrated at **more than Mach 0.28**, the aural alert of **"TOO LOW TERRAIN"** is sounded and the upper boundary of the envelope is increased to **1000 feet radio altitude**.

The applicable voice message is repeated each 1.5 seconds until the flight condition has been corrected. **The "TOO LOW GEAR" alert takes priority over the "TOO LOW FLAPS"**. The too low flaps alert and associated too low terrain alert are **inhibited** with the flap inhibit switch in the **FLAP OVRD** position.

### MODE 4B

AURAL ALERT - "TOO LOW FLAPS" or "TOO LOW TERRAIN"

VISUAL - PULL UP

MODE 4B

AURAL ALERT - "TOO LOW FLAPS"  
"TOO LOW TERRAIN"

VISUAL -

**PULL UP**

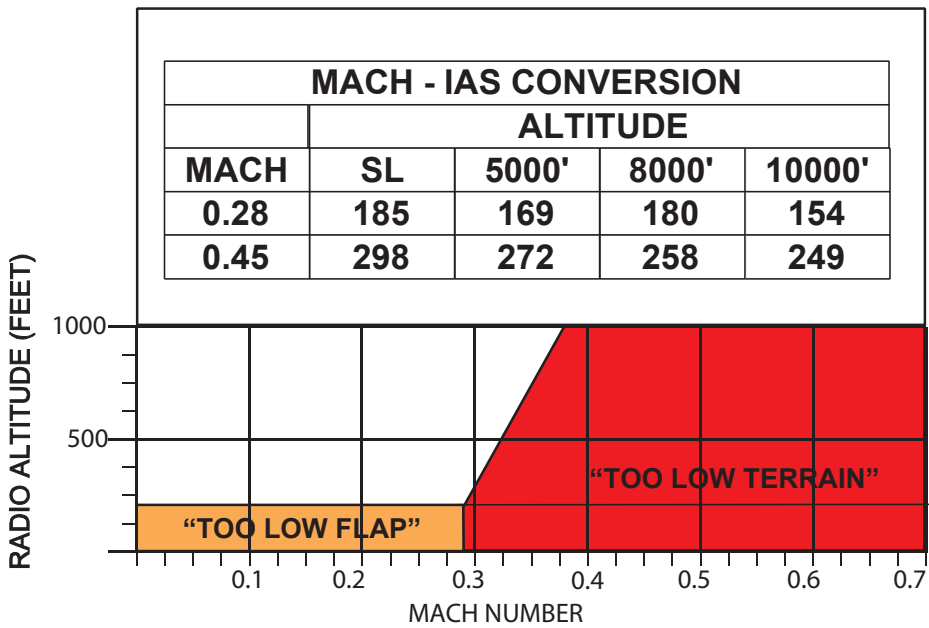
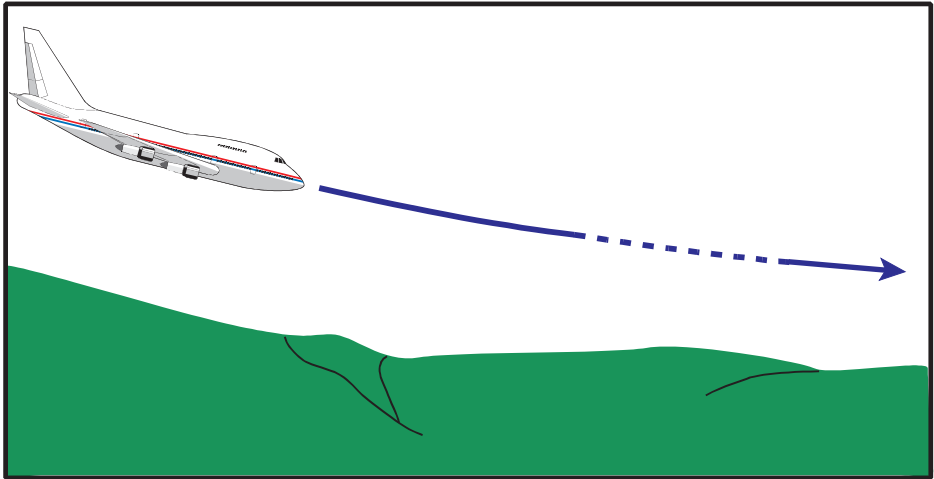


Figure 34.7 MODE 4B

## MODE 5 - Below Glide Slope Deviation Alert

This mode alerts the flight crew of a descent of more than 1.3 dots below an ILS glide slope. The envelope has two areas of alerting, **soft** and **hard**. In the soft area the alert is a reduced volume voice message of "GLIDE SLOPE". In the hard area, (approximately 2 dots below the glide slope) a full volume "GLIDE SLOPE GLIDE SLOPE" occurs. In both areas, the voice message repetition rate is increased as the glide slope deviation increases and the radio altitude decreases. The mode is armed when a valid signal is being received by the glide slope receiver, gear is down and the radio altitude is 1000 feet or less.

The mode may be cancelled or inhibited by pressing either pilot's **BELOW G/S** light while below 1000 feet radio altitude. The mode will re-arm when climbing above 1000 feet radio altitude. **Mode 1 to 4 aural alerts and warnings have priority over mode 5 aural alerts**, however both **PULL UP** and **BELOW G/S** lights could be illuminated at the same time.

### MODE 5

AURAL ALERT - "GLIDE SLOPE"

VISUAL - BELOW G/S P TO INHIBIT

**MODE 5**

AURAL ALERT - "GLIDE SLOPE"

VISUAL -

BELOW G/S  
P TO INHIBIT

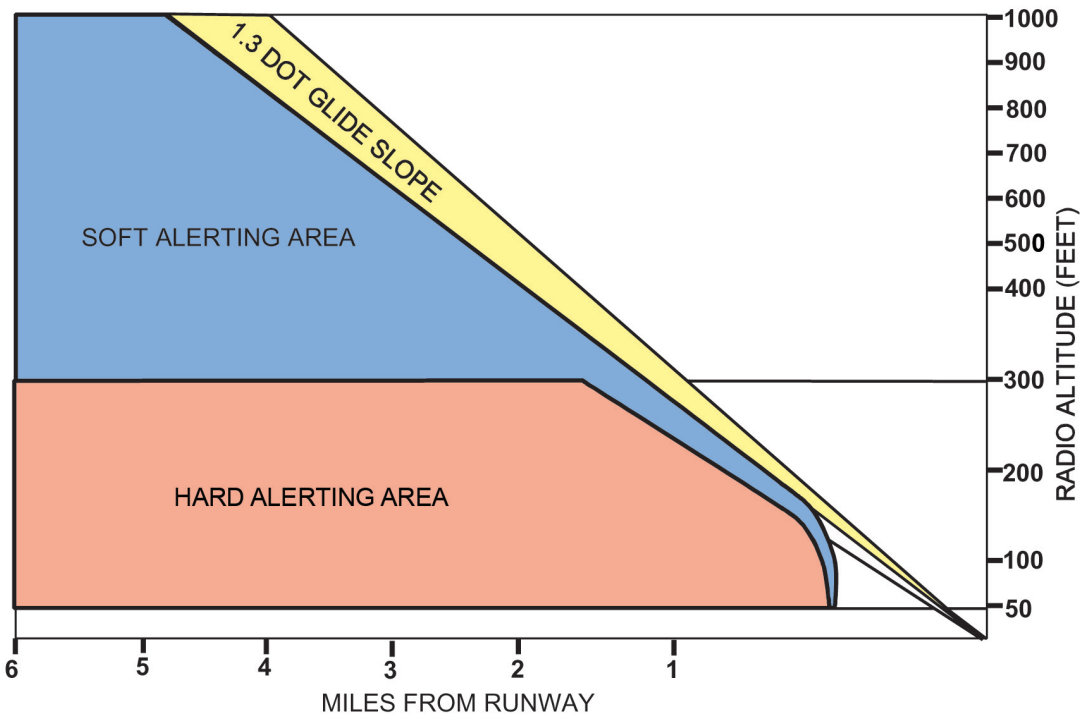
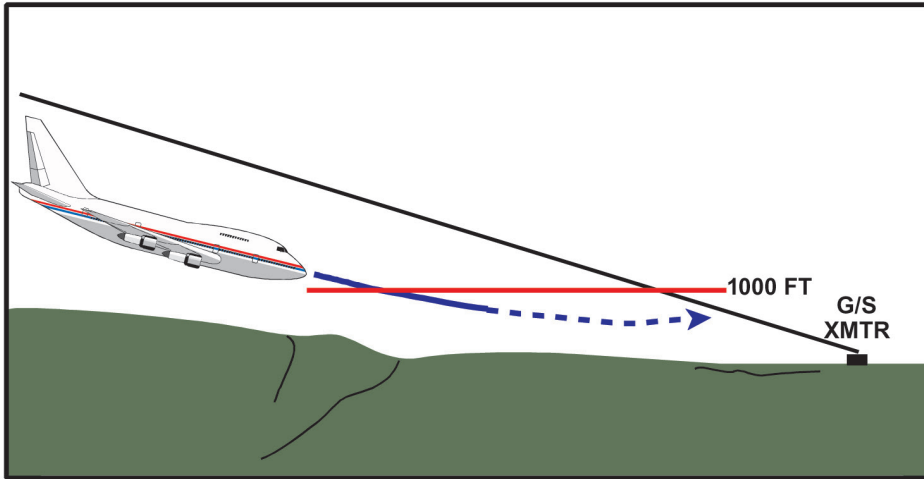


Figure 34.8 MODE 5

### MODE 6A - Below Selected Minimum Radio Altitude

Mode 6A provides an **aural alert** if a descent is made below the minimum decision altitude cursor in the radio altimeter. This mode operates **between 50 and 1000 feet** of radio altitude.

This alert is aural only and consists of **"MINIMUMS, MINIMUMS"** sounded once.

The mode is re-armed when the radio altitude becomes greater than that selected with the altitude cursor.

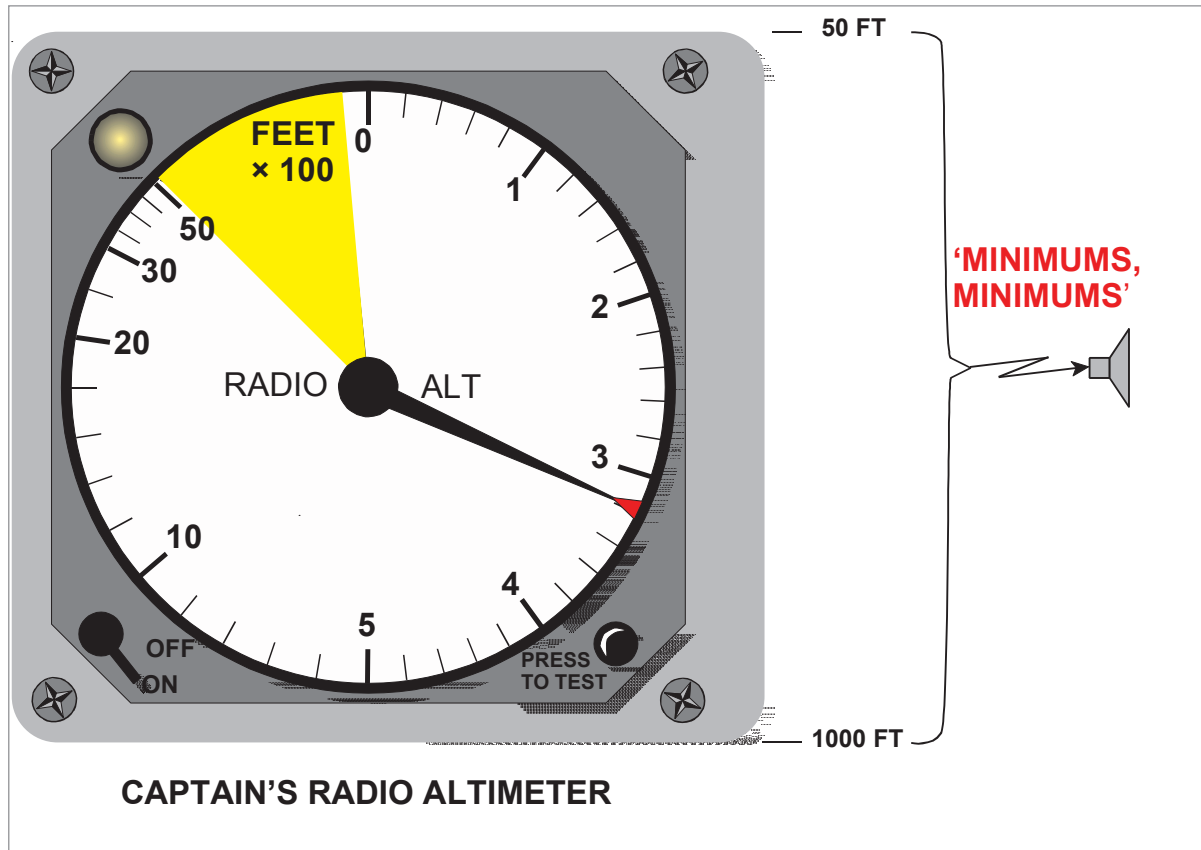


Figure 34.9 MODE 6

## MODE 6B Altitude Call-outs and Bank Angle Alert

Call-outs of selected altitudes and minimums is available. The call-outs used are a customer option but for example may consist of calls at 200 ft and 100 ft to decision height, or absolute height call-outs from the radio altimeter with respect to the ground.

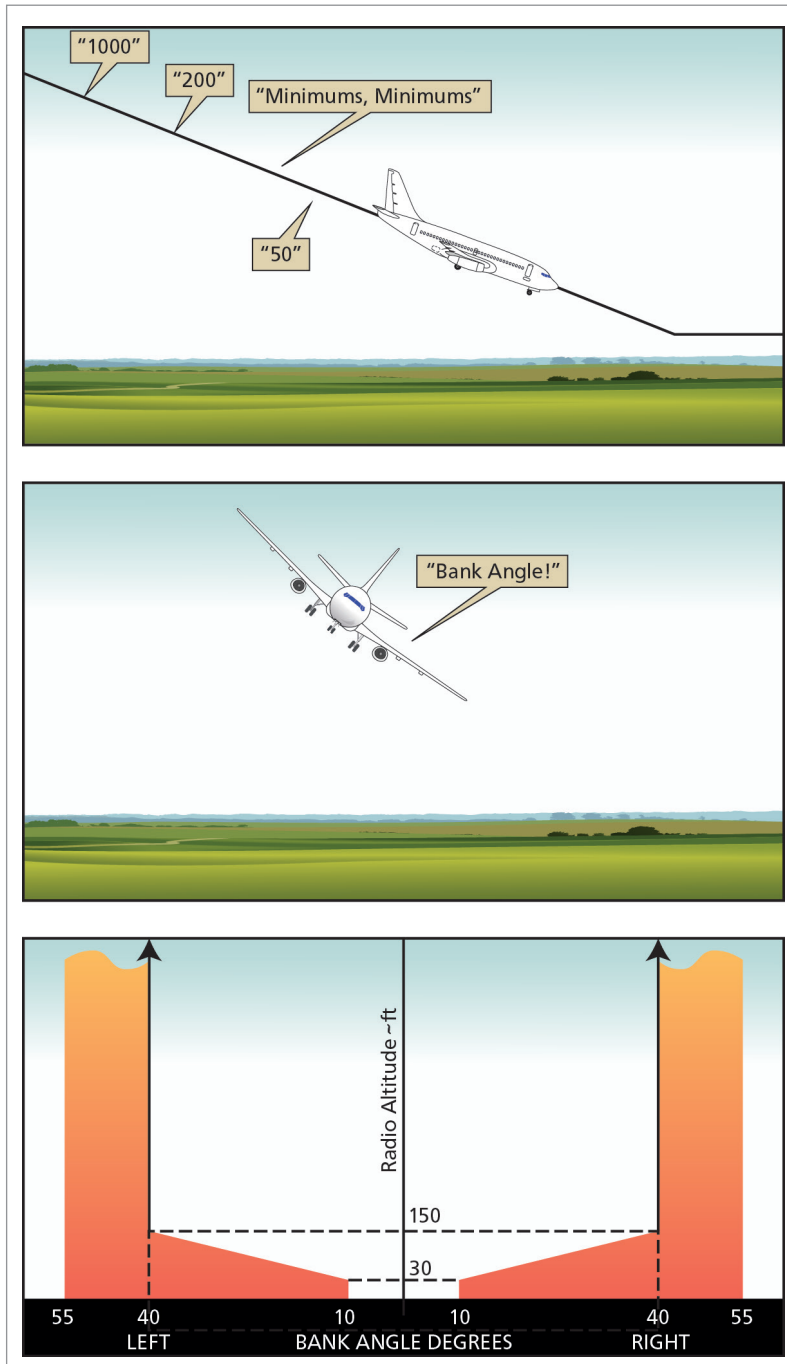


Figure 34.10 MODE 6B

"BANK ANGLE" can be used to alert crews of excessive roll angles. The bank angles are not an aircraft manoeuvre limit, but rather a limitation decided upon by the airline. However, the limit reduces with proximity to the ground due to the reduced wing tip clearance to prevent wing tip or engine damage during take-off and landing.

### MODE 7 Windshear Alerting

Visual and aural windshear warnings are given when several parameters such as ground speed, airspeed, barometric height and rate of descent and radio altitude, indicate the initial conditions of entering an area of windshear. Again as with the terrain threat display there is no scanning beam looking ahead to avoid the condition entirely. Rather the benefit from the system is derived from the fact that it allows the pilot to initiate the windshear go-around procedure earlier, giving the aircraft a greater probability of avoiding an accident. Mode 7 warnings take priority over all other modes.

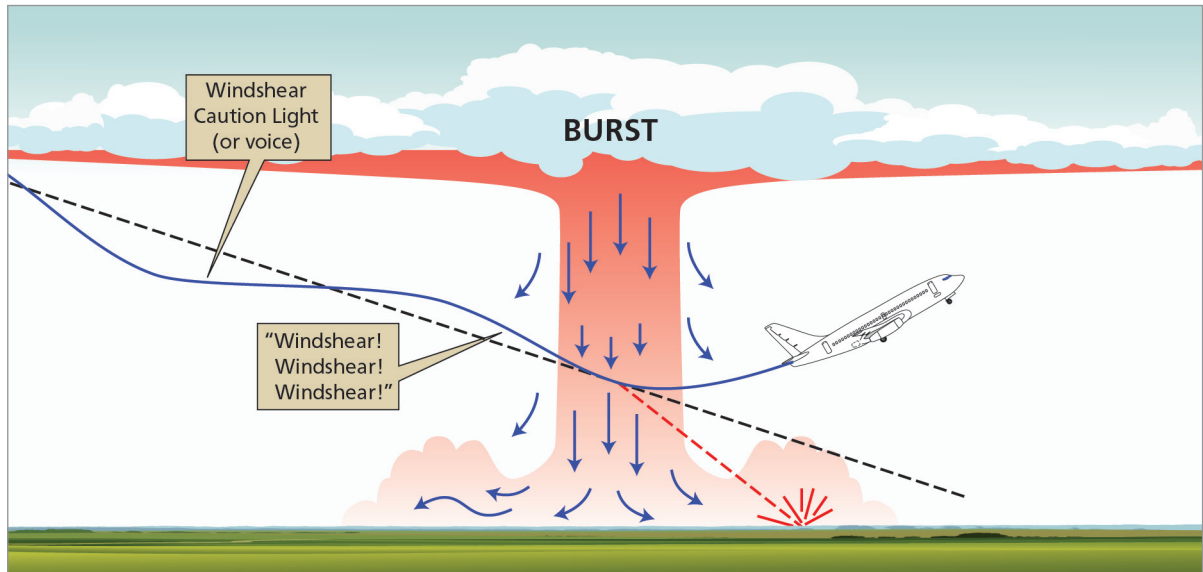


Figure 34.11 MODE 7 Windshear Alerting



## Action to Be Taken on Receiving a Warning

The response to all alerts or warnings should be **positive and immediate**; establishing the cause of GPWS activation should take second place.

There is a risk that repeated experience of unwanted alerts/warnings may reduce confidence in the system. Hence, **flight crews should report ALL alerts/warnings to the operator** thereby ensuring that appropriate analysis and remedial action can be taken. There is a GPWS operation reporting form for this purpose.

The immediate response to a **warning** must be to **level the wings and to initiate a maximum gradient climb** which should be maintained until the aircraft attains the **minimum safe altitude** for that part of the route being operated. Modification is permissible only in exceptional circumstances such as the necessity to follow a curved path for azimuth terrain avoidance.

When established in the climb every effort shall be made to determine the cause of the warning and to verify the aircraft's position. **The only circumstances when a climb to this altitude may be discontinued are when:**

- The cause of the warning has been positively identified and the warning has ceased
- The aircraft is operated by day in clear visual conditions
- 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

## Enhanced Ground Proximity Warning System (EGPWS)

### Terrain Awareness System (TAWS)

GPWS does not "look ahead" and any Mode 2 warning when flight is towards high ground will be dependent upon the steepness of the terrain. Hence, a sheer cliff ahead will not generate a mode 2 warning and any subsequent warning due to rising ground beyond the cliff will be delayed until the aircraft is over that ground. However, this limitation has been overcome by **Enhanced GPWS** with the **Terrain Threat Display**. This uses essentially an electronic map of the world (giving ground elevation) and information from the aircraft's navigational system (be that INS/GPS or any combination). Given the location of the aircraft, its course and height (either from the ADC or derived from GPS) a display can be created showing the locations of terrain that could threaten the safety of the aircraft.

Using this system EGPWS can warn of approaching high terrain even when that terrain is not in close enough proximity to initiate a mode 2 warning. This terrain threat display and warning will be initiated in sufficient time to comfortably avoid any threat of flight into terrain (approx 40-60 seconds for an alert, 20-30 seconds for a warning). The terrain is shown in shades of green, yellow and red and the display indicates terrain not only below the aircraft but also ahead of its flight path. At a certain time before predicted impact the warning will issue a **"Caution Terrain"** or **"Terrain Ahead"** message. If the system database has obstacle data the alert may be **"Obstacle Ahead"**. The threat terrain will turn solid yellow. If the situation is allowed to deteriorate so as to close further with the high ground, the second message **"Whoop Whoop Pull Up"** will sound and the most threatening terrain will turn solid red. This will happen at sufficient spacing to avoid impact with the terrain but this time using more positive control movements.

The accuracy of this display is, however, linked to the accuracy of the navigational equipment. A poor nav fix or a malfunctioning nav system will result in dangerously inaccurate display. Some pilots have been found to be using the threat display to “thread” their way through high terrain. This is, of course, a gross misuse of the system and is strongly advised against.

The terrain display can be selected by the pilot, or may be automatically activated whenever the terrain becomes a threat. The threat display may be incorporated with the weather radar display, the navigational display or it may have its own Plan Position Indicator (PPI).

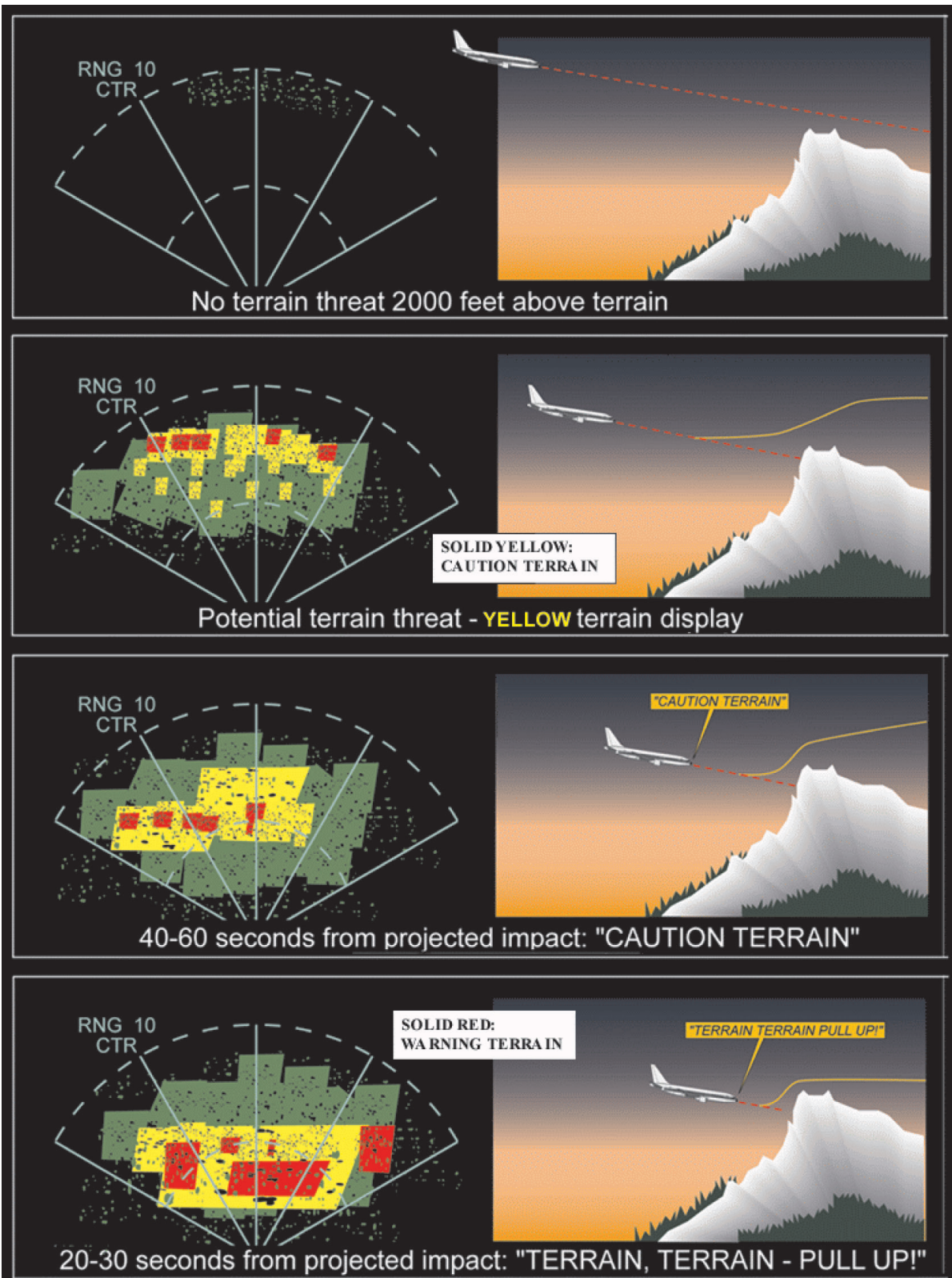


Figure 34.12 Terrain display

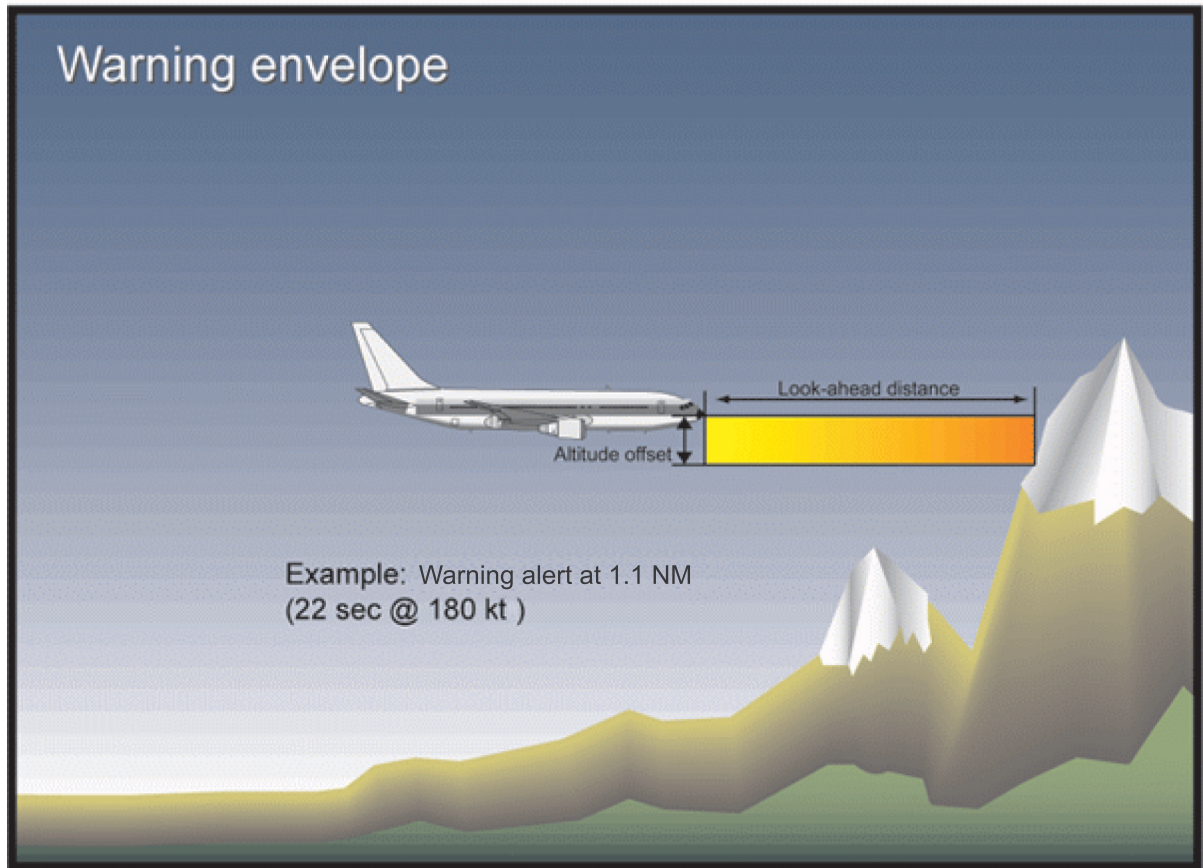
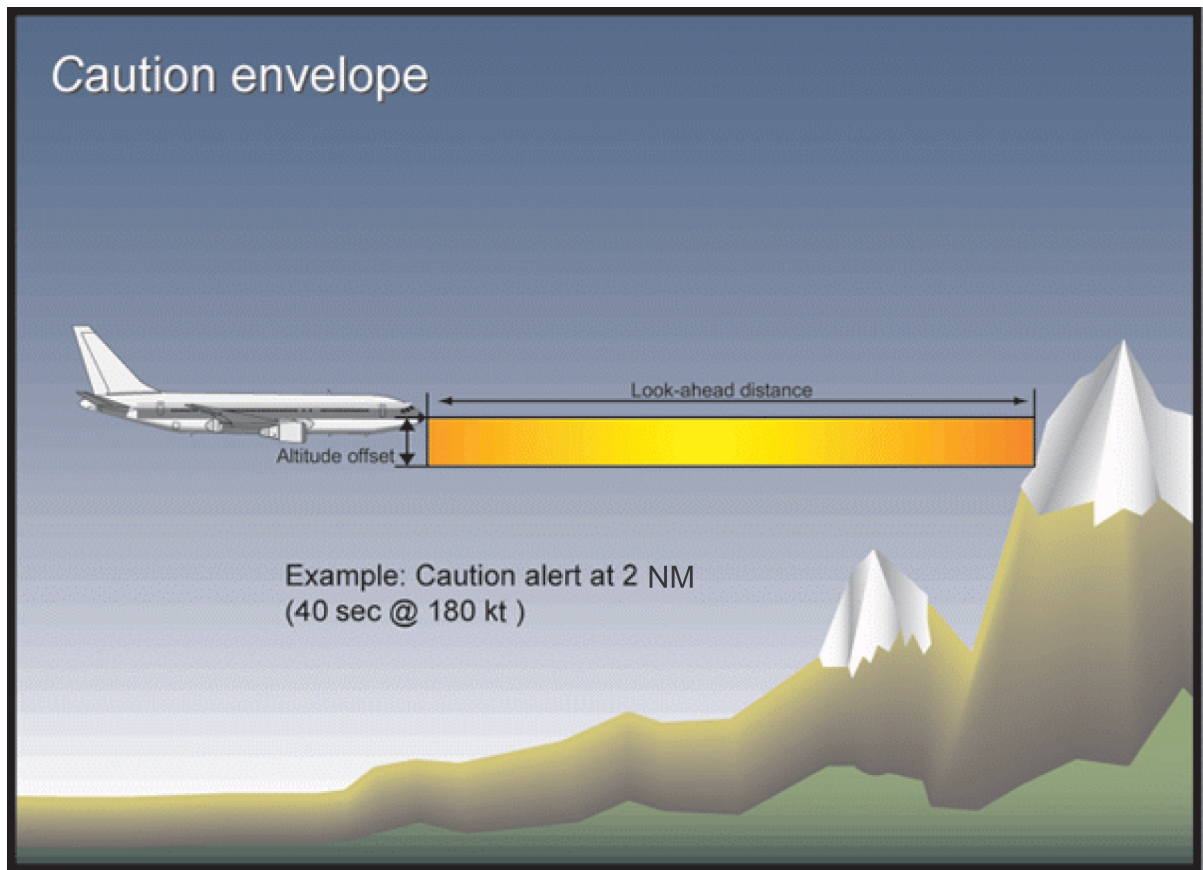


Figure 34.13 "Look ahead" warning system

## Terrain Clearance Floor

This alerts the crew to possible premature descent for non-precision approaches regardless of aircraft configuration. It uses the present aircraft position with respect to the runway. Presently the EGPWS database is separate from the FMS, and therefore not subject to regular updates. This is likely to change in the future, especially when the Runway Awareness and Advisory System (RAAS) is in general use. This system is designed to enhance situational awareness on the ground, and reduce the likely-hood of runway incursions.

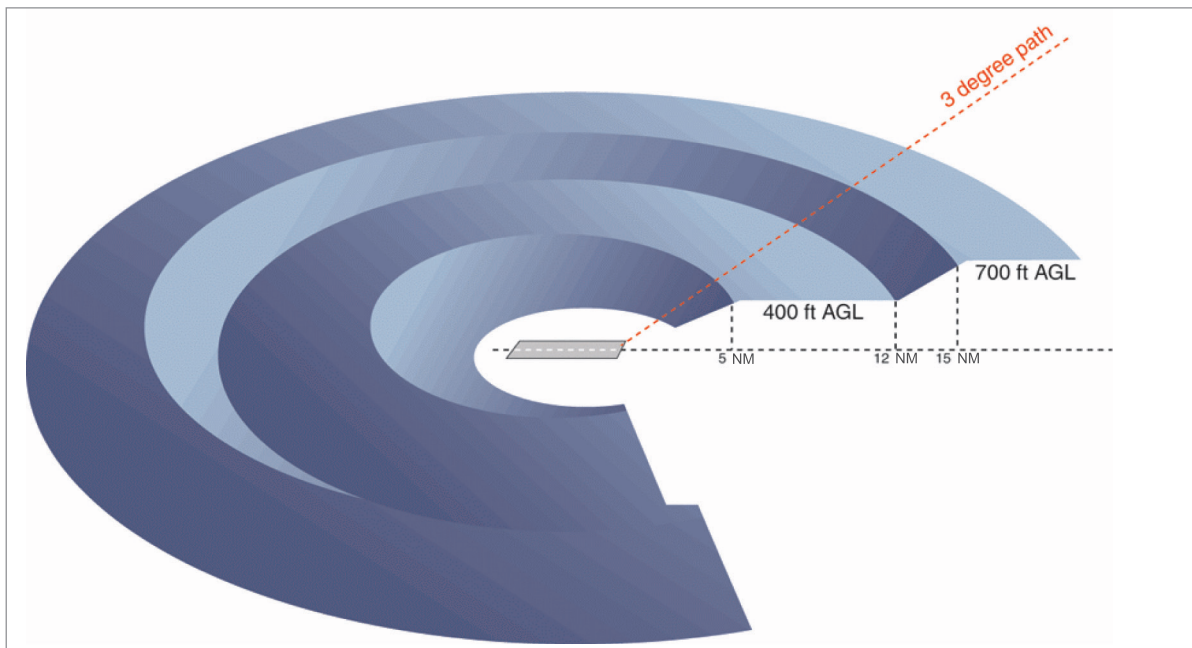


Figure 34.14 Terrain clearance floor

## Integrity Testing

The GPWS is provided with built-in test equipment (BITE) which allows all its functions and visual/audible warnings to be tested prior to flight; the preflight BITE is inhibited in flight. The test is initiated by the pilot pressing the test switch.

During flight the system is continuously monitored to confirm its serviceability and any in-flight failure is automatically indicated on the flight deck. A short confidence check is possible while airborne, but this is not a full BITE check.

## Inhibition of EGPWS Modes

The EGPWS must not be de-activated (by pulling the circuit breaker) except for approved procedures. Instructions on inhibition must include a statement that no person may de-activate the EGPWS except in accordance with the procedures stated in the Operations Manual.

- Inhibition of the glide slope mode may be desirable when a glide slope signal is present but the aircraft is deliberately being flown without reference to it, e.g. the pilot may have discontinued the ILS, to land on a different runway, or is performing a localizer only approach.
- Inhibition may also be required when the gear or flap position inputs are known to be non-standard.

## Questions

1. **The GPWS the alert/warning information is provided by a radio altimeter with:**
  - a. a downward transmitting beam whose dimensions are in the order of 60 and 30 in the fore/aft and the athwartship axes
  - b. a downward transmitting beam whose dimensions are in the order of 30 and 60 in the fore/aft and the athwartship axes
  - c. a forward transmitting beam
  - d. a downwards transmitting radio beam
  
2. **The GPWS would provide visual and audible warning to a pilot if the aircraft descended:**
  - a. to below 500' radio altitude with flaps not in the landing position and speed below Mach .28
  - b. to below 500' radio altitude with flaps not in the landing position and speed below Mach .35
  - c. to below 200' barometric altitude with flap not in the landing position and speed below Mach .28
  - d. to below 200' radio altitude with flap not in the landing position and speed below Mach .28
  
3. **The Ground Proximity Warning mode 5 provides a visual and audible warning to the pilot if the aircraft:**
  - a. descends below 500 ft radio altitude with gear retracted
  - b. is below 1000 ft radio altitude and more than 1.3 dots below the ILS glide path
  - c. descends below 200 ft radio altitude with flaps retracted
  - d. sinks more than approximately 10% of accumulated altitude
  
4. **The GPWS uses inputs from:**
  - a. the radio altimeter, the ILS receiver, the Air Data Computers and the landing gear position indicators
  - b. the radio altimeter, the Air Data Computers, the landing gear position indicators and the flap position indicators
  - c. the radio altimeter, the Air Data Computers, the ILS receiver, the landing gear position indicators and the flap position indicators
  - d. the radio altimeter and the ILS receiver
  
5. **The Ground Proximity Warning mode 4A provides a visual and audible warning 'Too Low Gear' to the pilot if the aircraft descends below \_\_\_\_\_ with landing gear retracted.**
  - a. 200 feet radio altitude with speed below M 0.28
  - b. 200 feet barometric altitude with speed below M 0.28
  - c. 500 feet radio altitude with speed below M 0.35
  - d. 500 feet barometric altitude with speed below M 0.35

6. An aircraft 'goes around' after descending to a radio alt of 190 feet. As power is applied a power unit is lost and some height is lost. The GPWS would provide an alert when the aircraft had lost about:
- 10 feet
  - 20 feet
  - 50 feet
  - 100 feet
7. GPWS mode one gives warning of:
- excessive descent rate
  - height loss after take-off/missed approach
  - unsafe terrain clearance when not in the landing configuration
  - excessive terrain closure rate
8. GPWS, mode three gives warning of:
- excessive descent rate
  - height loss after take-off/missed approach
  - unsafe terrain clearance when not in the landing configuration
  - excessive terrain closure rate
9. GPWS, mode two operates between:
- 50 ft and 2450 ft AGL
  - 50 ft and 1800 ft AGL
  - 50 ft and 700 ft AGL
  - 50 ft and 500 ft AGL
10. With reference to GPWS:
- In all six modes the audible alerts and warnings are accompanied by the red flashing PULL-UP light
  - Mode 4A activates when the aircraft descends below 500 ft radio altitude at a speed less than Mach .35 with the landing gear retracted
  - Mode 4A activates when the aircraft descends below 500 ft barometric altitude at a speed less than Mach .28 with the landing gear retracted
  - Mode 6 re-arms when the aircraft leaves the hard alerting area
11. GPWS mode 3 will operate if altitude loss occurs before you have acquired:
- 700 ft barometric altitude gain
  - 500 ft terrain clearance
  - 200 ft barometric altitude gain
  - 700 ft terrain clearance
12. At what radio altitude is Mode 4 armed?
- 500 ft
  - 700 ft
  - 200 ft
  - 790 ft

13. **Mode 4 gives warning of:**
- a. excessive descent rate
  - b. height loss after take-off/missed approach
  - c. unsafe terrain clearance when not in the landing configuration
  - d. excessive terrain closure rate





## Answers

1	2	3	4	5	6	7	8	9	10	11	12
b	d	b	c	c	b	a	b	a	b	d	b

13
c

Chapter  
**35**

**Airborne Collision and Avoidance System**

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## Introduction

Today's higher traffic densities and greater speed differences have generated a need for an Airborne Collision Avoidance System. Although ICAO named it ACAS, the principle manufacturer in the US referred to its system as Traffic alert and Collision Avoidance System (TCAS), and this name is now a widely accepted alternative. The system is designed to provide an additional margin of safety and keep commercial aircraft **clear of conflict**, independently of Air Traffic Control. An aircraft must carry a transponder and have the facility to interrogate other aircraft transponders. Presently there are four systems in use, I, II, III and IV, each with increasing levels of protection. All commercial air transport turbine powered aircraft registered in Europe greater than 5700 kg MTOM or with more than 19 passenger seats must have at least ACAS II (version 7).

## TCAS I

**TCAS I** is a first generation collision avoidance system and simply warns the crew of other traffic in the vicinity of their aircraft. It will detect and display range and approximate relative bearing. If the TCAS display aircraft and the intruder are carrying Mode C, relative altitude will also be displayed. It encourages flight crew to look for the conflicting traffic by generating visual and aural warnings - **TRAFFIC ADVISORIES (TAs)**:

"Traffic, Traffic".

It does not give any resolution advisory information, i.e. a course of action to follow.

## TCAS II

**TCAS II** detects intruders in the TCAS aircraft's vicinity, assesses the collision risk and presents warnings to the crew in the form of **TAs** and **Resolution Advisories (RAs)** e.g.

"Climb" "Increase Climb" "Descend" "Increase Descent" "Monitor Vertical Speed" "Decrease Climb" "Decrease Descent"

Thus, RAs offer manoeuvring advice **in the vertical plane** to resolve conflict. If your aircraft and the intruder both have Mode S data link transponders the system will co-ordinate the RAs to provide complementary vertical avoidance instructions. The rest of this chapter deals with TCAS II only and discusses both visual and audible TAs and RAs in detail.

## Principle

TCAS II operates on the secondary radar principle using the normal SSR frequencies of 1030 MHz and 1090 MHz, but in an air to air role. Using this principle the TCAS system creates two protective three dimensional bubbles around the TCAS equipped aircraft (*Figure 35.1*).

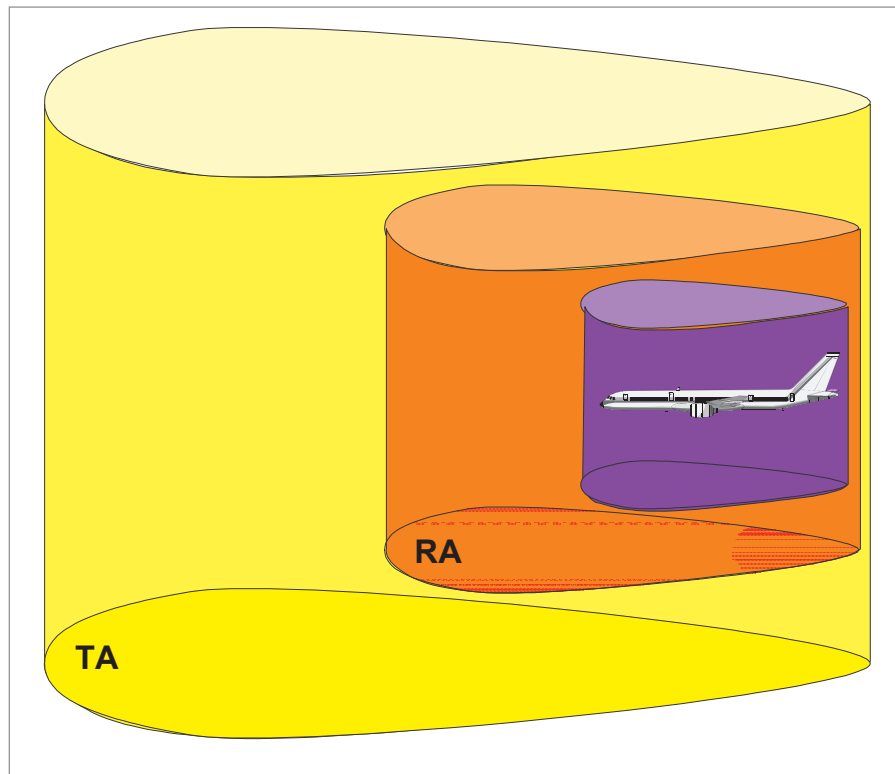


Figure 35.1

## Aircraft Equipment

For aircraft to be visible to a TCAS equipped aircraft they must have a minimum of a **Mode "A" Transponder**. If the transponder is switched off, or is unserviceable, the intruding aircraft are invisible to the TCAS equipment and a collision risk exists. **Mode A transponders transmit no height information and therefore the information available to the TCAS equipment is two dimensional only** and therefore can only give TAs.

**Mode "C" Transponder** equipped intruders broadcast height information to the TCAS equipment and the system becomes **three dimensional** and can now give both TAs and RAs. **Mode "S" Transponder** TCAS equipped intruders as well as broadcasting height information allow a discrete data link to be established between them. This data link will allow avoidance manoeuvres to be mutually resolved.

## Operation

The **range** of an intruder is determined by measuring the time lapse between transmission of an interrogation, and receiving the response (Radar Principle). An interrogation signal is sent out approximately once a second; however in densely populated airspace, where many aircraft are being monitored, a system of reduced surveillance is employed, where the interrogation period is extended to every 5 seconds, with priority given to the closest. Normal detection range is approximately 30 NM, reducing to 5 NM when operating reduced surveillance. The **bearing** of an intruder is determined by a directional antenna (*Figure 35.2.*). Because of the wavelengths involved and the necessarily small size of the antennas, **bearing resolution is the least accurate parameter**. TCAS II never offers collision avoidance commands in the horizontal plane; only in the form of climb or descend.

The **relative height** of an intruder is found by comparing its Mode “C” height with the TCAS equipped aircraft’s height.

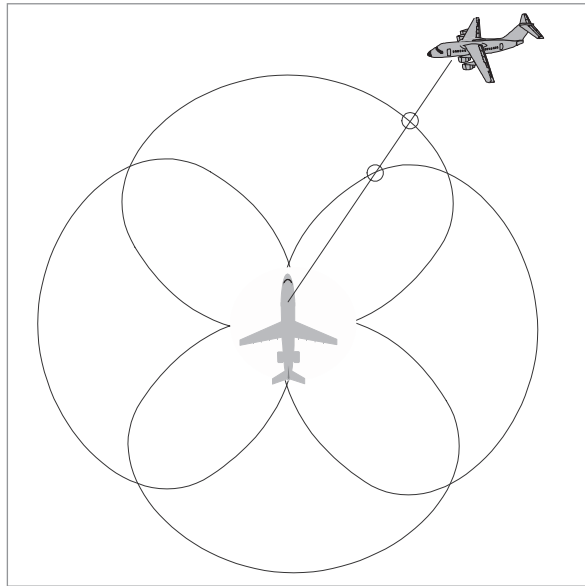


Figure 35.2 TCAS Bearing determination

## System Interconnections

Figure 35.3 shows a TCAS installation in a commuter/feeder airliner. The heart of the system is the TCAS receiver-transmitter-computer unit controlled by a combined ATC/SSR/TCAS control panel. The TCAS displays in this installation are a dedicated TCAS Plan Position Indicator (PPI), and the red and green sectors on the vertical speed tape of the Primary Flight Display (PFD). Electronic Attitude Director Indicator (EADI). On other aircraft the symbols may be displayed on an electronic VSI, or the Electronic Horizontal Situation Indicator (EHSI/ND). A synthetic voice issues TCAS commands.

The TCAS upper and lower antennas are directional while the Mode “S” antennas are omnidirectional.

The TCAS also has feeds from the radio altimeter to modify the RAs received when in close proximity to the ground. There are no instructions given at all when the aircraft is below 400 ft AGL, no descent RAs are given below 1000 ft AGL and no increase rate of descent commands below 1450 ft AGL. The system will also take aircraft configuration/performance into consideration when deciding an avoiding action. When the aircraft has gear and/or flap deployed its climb performance will be poor so TCAS will avoid giving climbing demands for an RA. A feed from the ADC tells the ACAS the aircraft’s height, so that, at high altitudes, it will not give a climbing RA if the aircraft is close to its performance ceiling. A further feed from the IRS gives inertial vertical acceleration.

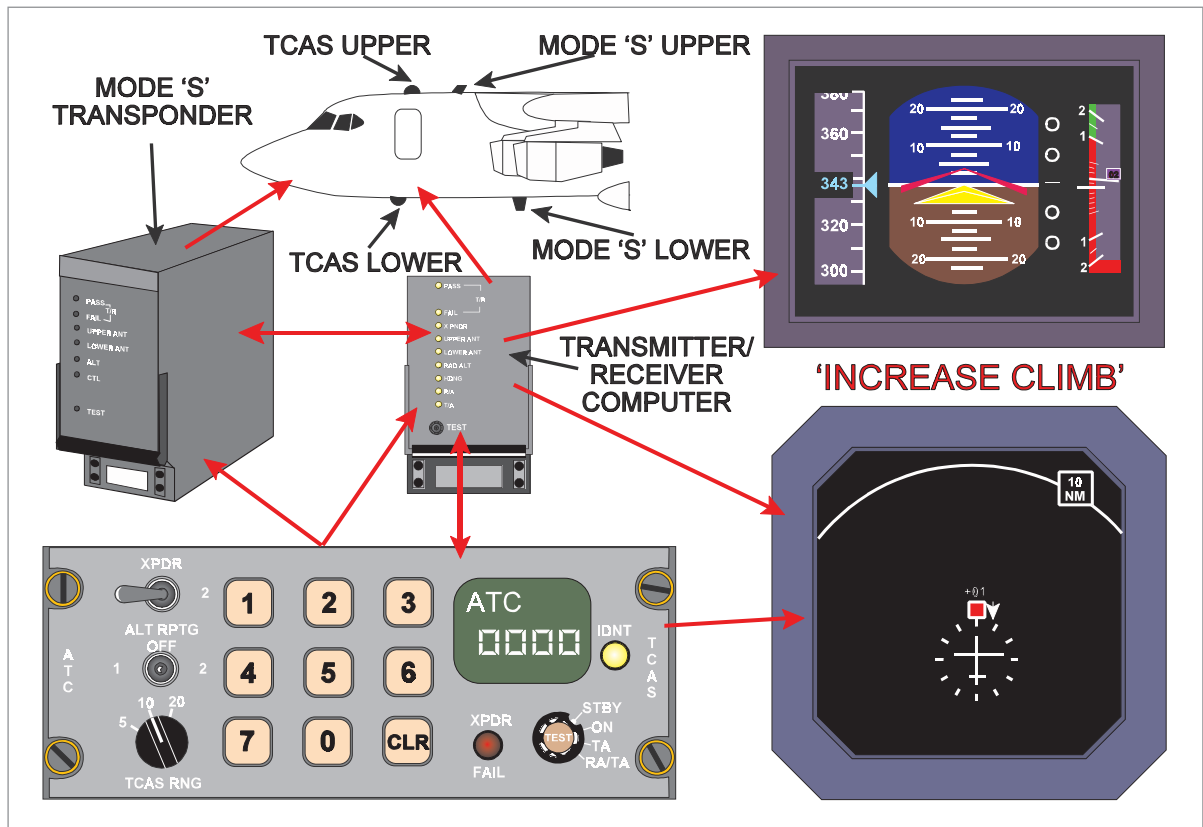


Figure 35.3 TCAS aircraft installation

### Synthetic Voice Prioritization

Modern aircraft use a synthetic voice to give warning advice to the crew. The voice is used for various systems including windshear detection, ground proximity warnings, including height call-outs, and TCAS. The synthetic voice is prioritized as follows:

**Stall Identification/Stall Prevention.** (Stick-shake/Stick-push). The synthetic voice is inhibited during stick shake/stick push operation.

**Windshear.** The detection of performance decreasing windshear takes first priority with the synthetic voice, inhibiting both GPWS and TCAS warnings.

**Ground Proximity Warning System (GPWS).** Detection of approach to terrain takes priority over TCAS announcements.



## Traffic Advisories (TAs) Resolution Advisories (RAs)

The TCAS can monitor a large number of aircraft simultaneously (up to 30 can be displayed) and can produce resolution advice against multiple conflicts. It bases advice on the target aircraft's Closest Point of Approach, (CPA) which is denoted by the Greek letter  $T$  (tau) in calculations. (The calculations are not required knowledge and are therefore not included in this book). Depending upon the setting of the TCAS function switch on the control panel, the equipment level of intruder aircraft and the phase of flight of the TCAS aircraft, TCAS will generate the following.

- a) **Traffic Advisories (TAs)** exist when an intruder penetrates the outer bubble caution area and is between 45 and 35 seconds from the collision area. TAs appear as **solid amber circles** on the TCAS display and are accompanied by the synthetic voice saying "Traffic, Traffic". This is a potential collision threat.
- b) **Resolution Advisories (RAs)** exist when an intruder penetrates the inner bubble warning area and is between 35 and 15 seconds from the collision area. RAs appear as **solid red rectangles** on the TCAS display accompanied by various synthetic voice warnings. RAs indicate a serious collision threat. (See [Figure 35.4](#)).

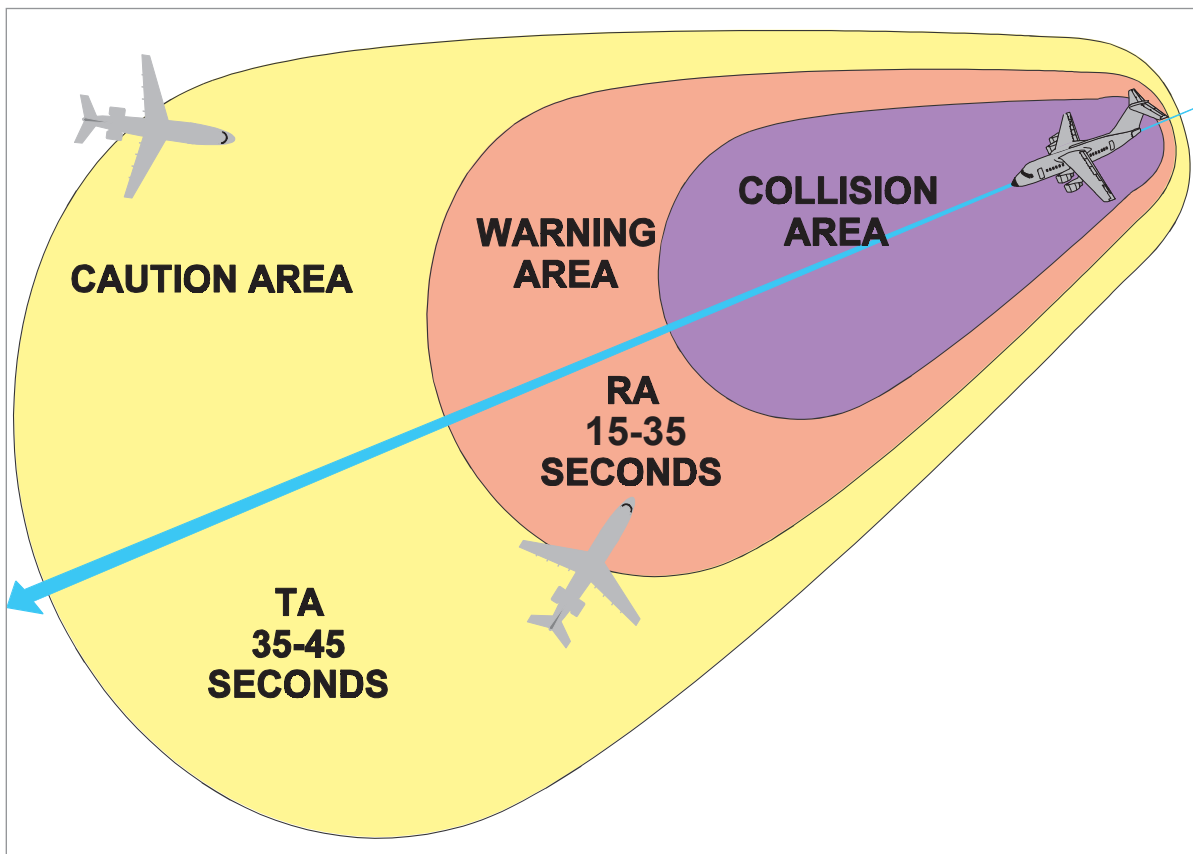


Figure 35.4

## Resolution Advisories

It should be remembered that it is not the job of the ACAS system to maintain standard Air Traffic separation minima. If RAs are responded to smoothly and immediately, then the intended vertical separation varies from 300 to 600 ft (700 above FL410) depending on the Sensitivity Level (SL) which decreases with altitude.

Resolution Advisories come in two forms:

**Preventative Advisories** are situations where no collision risk exists unless a change of level is initiated by either aircraft. The synthetic voice advisory is **"Monitor Vertical Speed"**.

**Corrective Advisories** are situations where a collision risk exists and a manoeuvre is necessary to avert it. The synthetic voice produces the appropriate command.

Figure 35.5 shows examples of preventative and corrective RAs displayed on the vertical speed tape of the primary flight display.

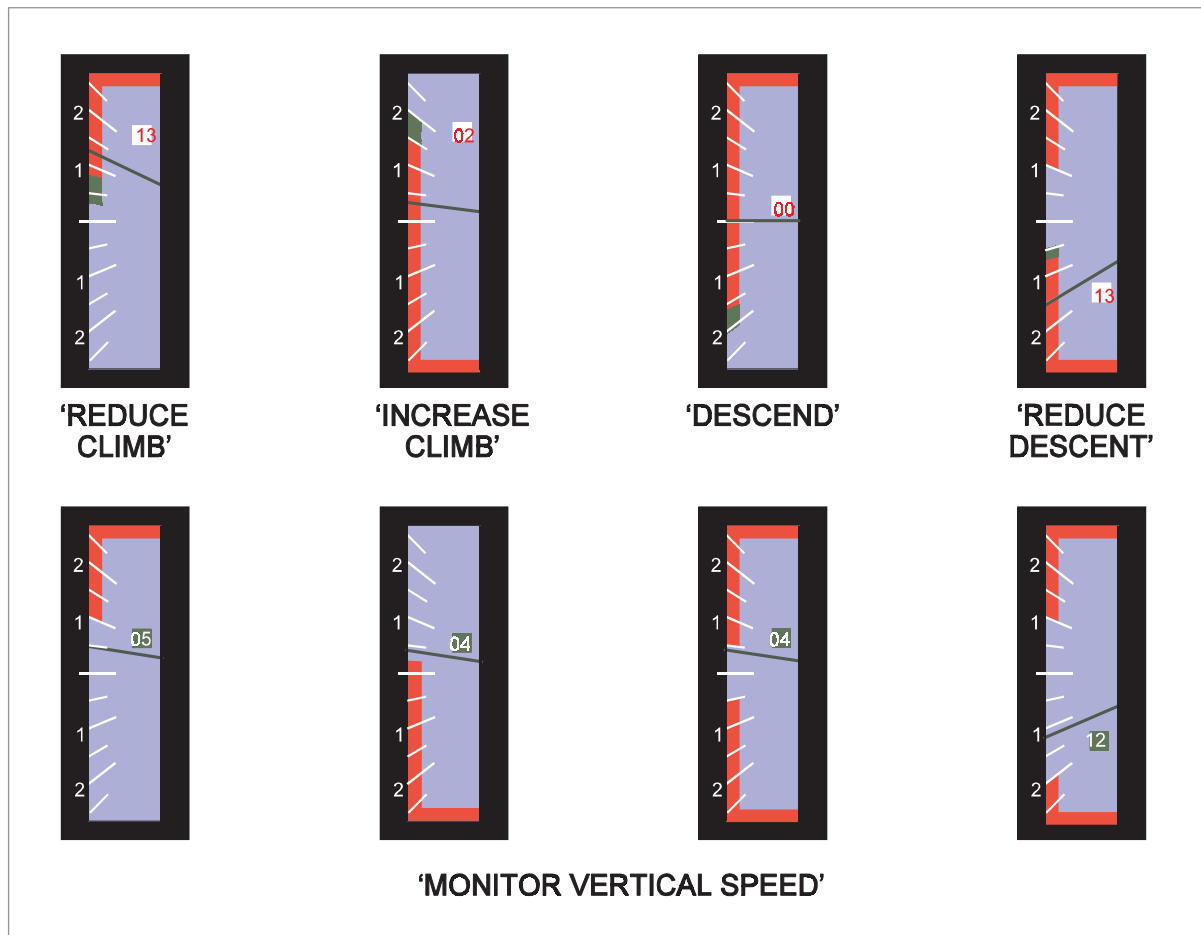
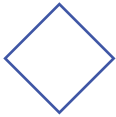


Figure 35.5 Corrective and preventative resolution advisories

## Proximate Traffic/Other Traffic



**Proximate Traffic** appears as a solid cyan diamond and represents transponder equipped aircraft within 6 NM and within +/- 1200 feet relative height. TCAS does not consider this traffic a threat and displays it to improve crew situational awareness.



**Other Traffic** appears as hollow cyan diamonds which represent transponder equipped aircraft within range of the display and within +/- 2700 feet relative height (+8700/-2700 in the climb, +2700/-8700 in the descent. This being a function of the "Above/Below" switch). Again it is displayed to improve situational awareness.

The predicted flight paths of Proximate and Other Traffic do not penetrate the Collision Area of the TCAS aircraft.

## Data Tag

If the intruding aircraft is transmitting mode C the traffic symbols will also have an associated altitude TAG which shows relative altitude in hundreds of feet. To indicate whether an intruder is climbing, flying level or descending:

A **+ sign** indicates the intruder is above, **- sign** below.

A **trend arrow** ↑ or ↓ appears alongside the symbol when the intruder's vertical rate is **500 feet per minute or greater**.

No altitude number or trend arrow will appear beside an intruder that is non-altitude reporting. If TCAS II direction finding techniques fail to locate the azimuth of another aircraft a **no bearing** message appears on the screen.

## Off Scale Traffic Advisory

When TCAS is tracking an intruder that is outside the selected display range, but has entered the caution or warning areas, one-half of the appropriate symbol will appear at the appropriate bearing at the edge of the display area. The symbol will appear in its proper colour and have its data tag displayed providing there is room. (See [Figure 35.15](#)).

### TCAS Displays

TCAS range and bearing information can appear on a variety of displays:

#### Dedicated Plan Position Indicator

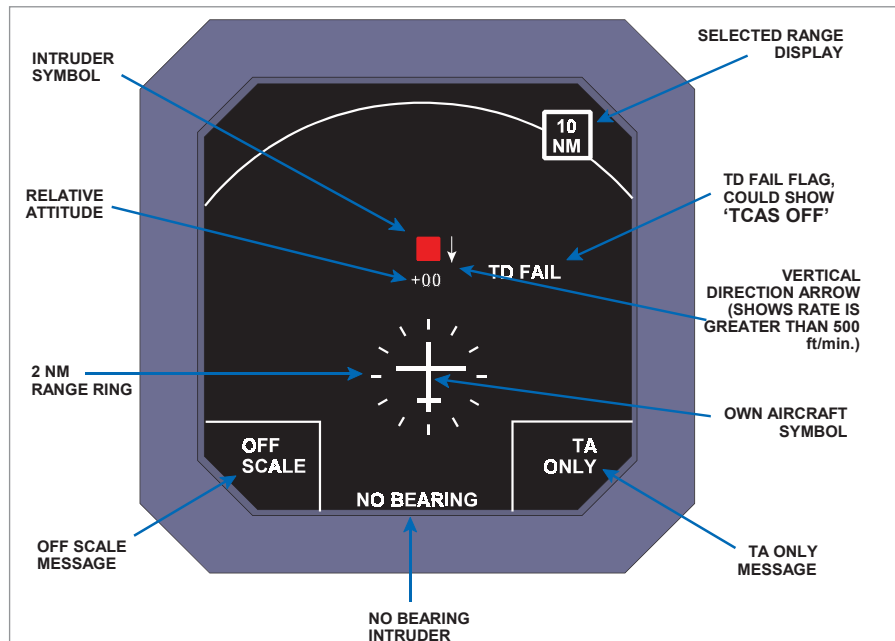


Figure 35.6 TCAS PPI

#### Electronic Vertical Speed Indicator

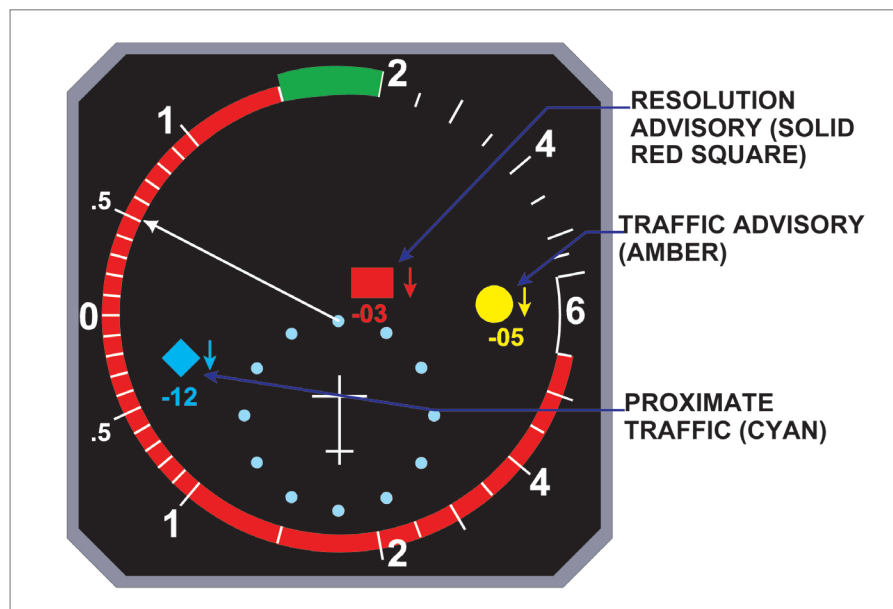


Figure 35.7 Electronic VSI

Superimposed on Navigation Display of EFIS Equipped Aircraft



Figure 35.8 Navigation display in MAP mode showing TCAS

Combined TCAS and SSR Control Panel

The control panel is produced in various forms but all perform the same functions. The TCAS controls are as follows: See Figure 35.9.



Figure 35.9 ATC Transponder/TCAS control panel

### Function Switch

- Standby - warm-up power is applied to the system, but it is not operational.
- On - the transponder only is operational (your aircraft will be visible to ACAS equipped aircraft).
- TA - the transponder and TCAS are now operational but only Traffic Advisories are generated. "TA ONLY" will be indicated on the TCAS display. TA would only be selected in accordance with specific procedures. For example, after an engine failure, when the aircraft performance is reduced or if parallel runway ops are in force.
- RA/TA - the transponder and TCAS are operational and both Resolution Advisories and Traffic Advisories are generated.
- TEST - pressing the centre TEST button on the function switch initiates a full Built-in Test Equipment (BITE) of the system. After completion of a successful test the synthetic voice will respond with "TCAS SYSTEM TEST OK". If the system test is unsuccessful the voice response is "TCAS SYSTEM TEST FAIL".

### TCAS RNG (Range)

This will select the range of the TCAS display either 5, 10, or 20 NM. It does not alter the range at which aircraft are detected or when warnings are given.

### TCAS Traffic Advisories on Electronic VSI

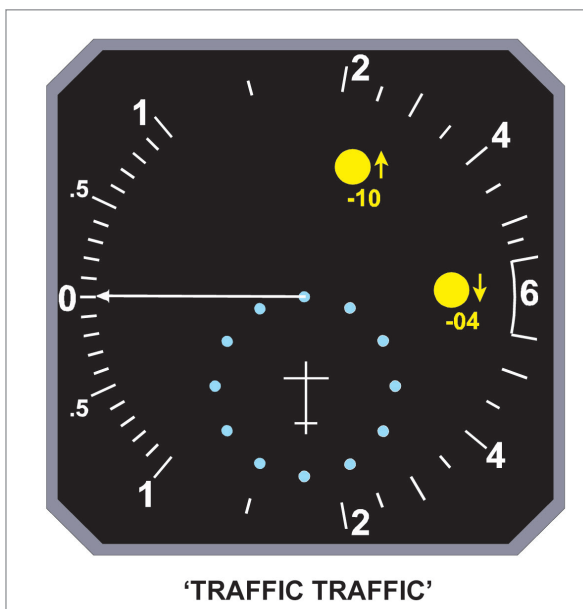


Figure 35.10 Traffic advisory

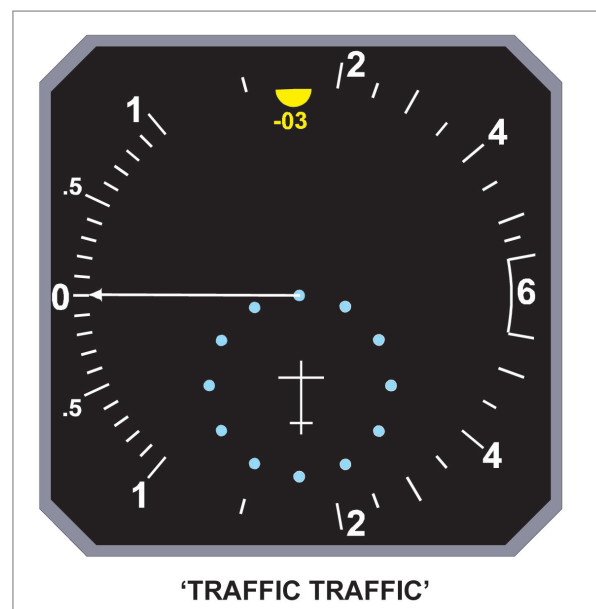


Figure 35.11 Off scale traffic advisory

TCAS Preventative Resolution Advisories on Electronic VSI

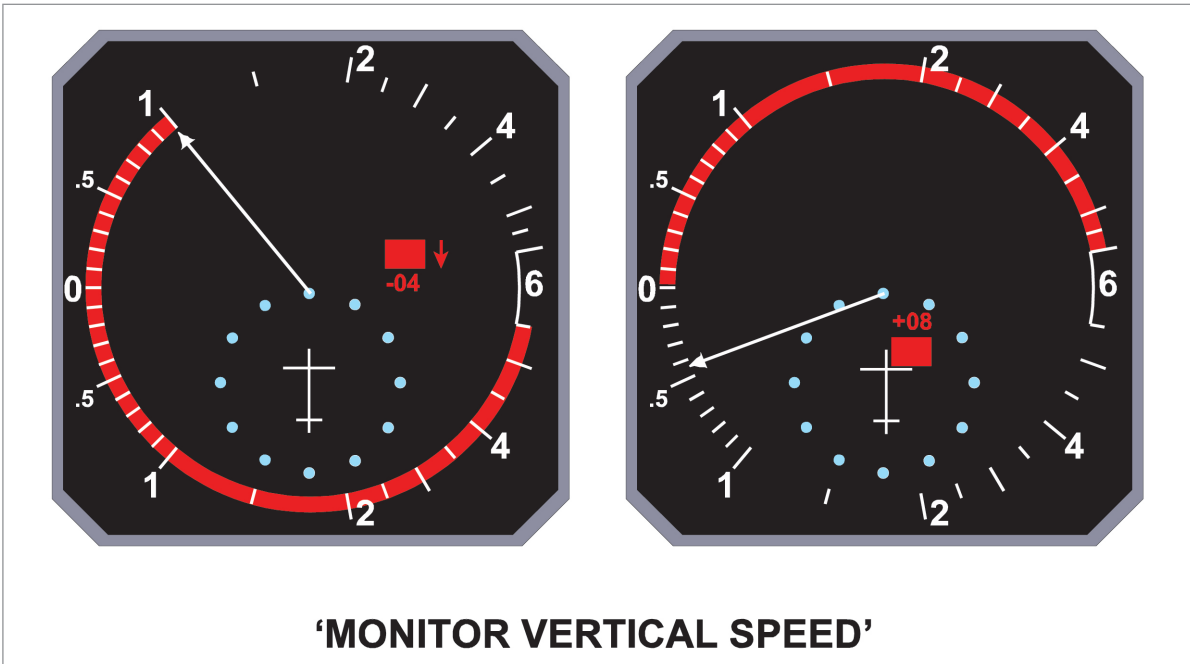


Figure 35.12 Preventative resolution advisories

TCAS Corrective Resolution Advisory on Electronic VSI



Figure 35.13 Corrective RA

### TCAS Test Format on Electronic VSI

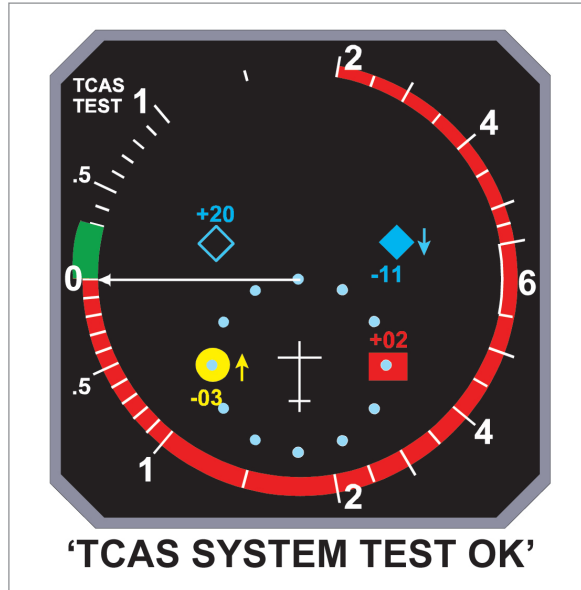


Figure 35.14 Test display

### No Bearing Advisories

If TCAS is unable to track the bearing of an intruder, possibly due to antenna screening, the RA or TA will appear lower centre of the display appropriately colour coded. Up to two lines of information can be displayed.

“TA 2.2- 04” means the intruder is creating a TA 2.2 NM away 400 below and the up arrow indicates the intruder is climbing at 500 fpm or greater.

It is important to realize that TCAS’s ability to compute a Traffic or Resolution Advisory is not degraded by lack of bearing information.



Figure 35.15 No Bearing RA and TA



## Action to Be Taken on Receiving TAs and RAs

- **Action on Receiving a TA.** TAs alert flight crews to the possibility that an RA may follow, which could require a flight path change. Flight crews should assimilate the information provided by the TA and commence a visual search of that part of the sky. They should also prepare to respond to an RA if the situation worsens. If the potential threat cannot be seen and continues to give cause for concern, flight crews should seek advice from ATC. Do not manoeuvre on the basis of a TA alone.
- **Action on Receiving an RA.** Pilots are to initiate the required manoeuvre immediately, adjusting flight path, aircraft power and trim accordingly. Crew members not involved in executing this manoeuvre should confirm that the sky ahead is clear of other aircraft and continue the visual search for the established threat. They are to inform ATC as soon as possible of any deviation from an ATC clearance.
- **Disregarding RAs.** 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. For this reason:
  - **RAs may be disregarded only when pilots visually identify the potentially conflicting traffic and decide no deviation from the current flight path is needed.**
  - **If pilots receive simultaneously an instruction to manoeuvre from ATC and an RA, and both conflict, the advice given by TCAS should be followed.**



## Questions

1. **On receipt of a TCAS RA your action is to:**
  - a. initiate the required manoeuvre immediately
  - b. make a note of the details
  - c. request a flight clearance deviation from ATC
  - d. do nothing until a TA is received
2. **Which of the following statements concerning TCAS is correct:**
  - a. TCAS II provides avoidance instructions in the vertical and horizontal planes
  - b. TCAS II cannot provide information on non-SSR equipped intruders
  - c. TCAS II requires Mode S to be fitted to other aircraft
  - d. TCAS II provides advice on which way to turn
3. **With reference to Traffic Collision Avoidance Systems. The difference between TCAS I and II is that:**
  - a. TCAS II can provide Traffic Advisories and Resolution Advisories whilst TCAS I can only provide Traffic Advisories
  - b. TCAS II can only be fitted to large aircraft which carry more than 30 passengers whilst TCAS I can be fitted to any aircraft
  - c. TCAS I can be fitted to aircraft which carry transponders with Mode A only whilst TCAS II can only be fitted to aircraft whose transponders include either Mode C or Mode S
  - d. TCAS II can only be fitted to aircraft which are equipped with EFIS
4. **The aural messages provided by TCAS II are:**
  - a. Threat, Climb; Threat, Descend
  - b. Climb left; Climb right; Descend left; Descend right
  - c. Climb; Descend; Increase Climb; Increase Descent
  - d. Turn Left, Turn Right, Increase Turn, Decrease Turn
5. **With reference to Traffic Collision Avoidance Systems:**
  - a. RAs may be disregarded only when the pilot visually identifies the potentially conflicting traffic and decides that no deviation is necessary and has the clearance confirmed by ATC
  - b. RAs may be disregarded only when the pilot visually identifies the potentially conflicting traffic and decides that no deviation is necessary and has advised ATC of the other aircraft's proximity
  - c. RAs must never be disregarded
  - d. RAs may be disregarded only when the pilot visually identifies the potentially conflicting traffic and decides that no deviation is necessary

## Answers

1	2	3	4	5
a	b	a	c	d

Chapter  
**36**

Flight Data Recorder

---

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## Introduction

Commercial aircraft have a flight recorder which records various aircraft parameters during the entire duration of the flight. The main function of the flight data recorder (FDR) is to preserve the aircraft data in order to determine the cause of any aircraft accident. It is also used to gather information for trend analysis and trouble shooting. In smaller aircraft the FDR may be combined with a cockpit voice recorder.

## FDR Designs

The FDR records the last 10 or 25 hours of aircraft data on a digital storage device housed in a shock resistant box that is painted red and located at the rear of the aircraft, normally under the fin. On the front of the unit is an underwater locating device (ULD).

## FDR Components

The FDR consists of the following components (see *Figure 36.1*):

- a recording system
- a control unit on the overhead panel
- a control unit on the pedestal
- Data Interface and Acquisition Unit (DIAU)

Information from the DIAU is sent to the Aeroplane Condition Monitoring System (ACMS), together with inputs from aircraft systems, such as air conditioning, auto-flight, flight controls, fuel, landing gear, navigation systems, pneumatic systems, APU and engines. This information is used for system monitoring and trend analysis, and can be downloaded via the aircraft printer, or via ACARS or the ATSU.

The control unit on the overhead panel also controls the cockpit voice recorder (CVR). A spring-loaded switch labelled GND CTL can be selected ON or AUTO as follows:

- ON            The CVR and the DFDR are energized and the ON light is lit
- AUTO        The CVR and the DFDR are energized:
  - on the ground with one engine running
  - in flight (with engine running or stopped).

The control on the pedestal consists simply of a push-button labelled 'EVENT' which sets an event mark on the DFDR recording. This acts as a kind of bookmark to enable the "event" to be found rapidly on the recording at a subsequent analysis.

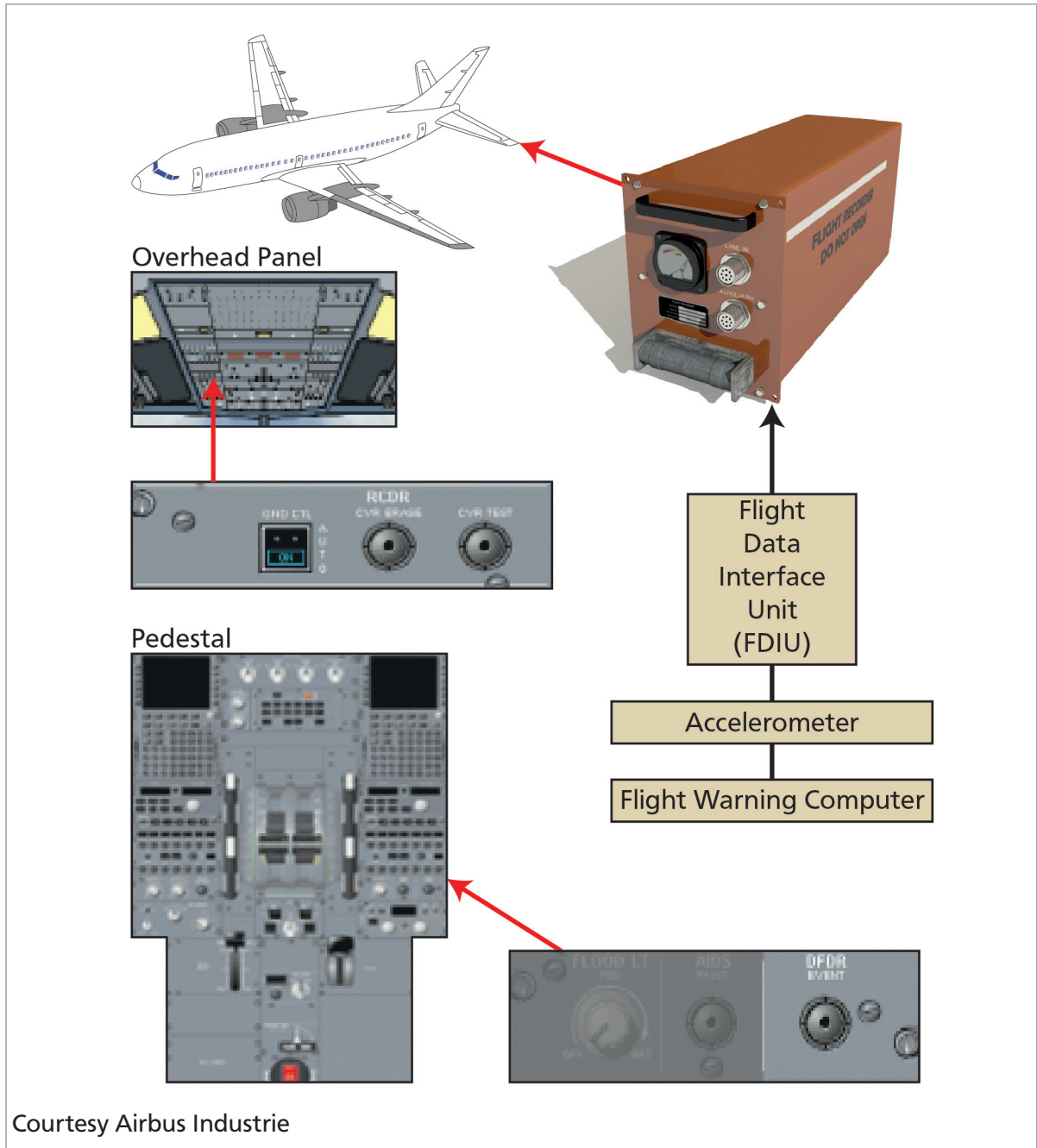


Figure 36.1 Digital flight data recorder

When on the ground the FDR is automatically stopped 5 minutes after the final engine shut-down.

A block diagram of a digital flight data recorder system for a Boeing 767 aircraft is shown in Figure 36.2.



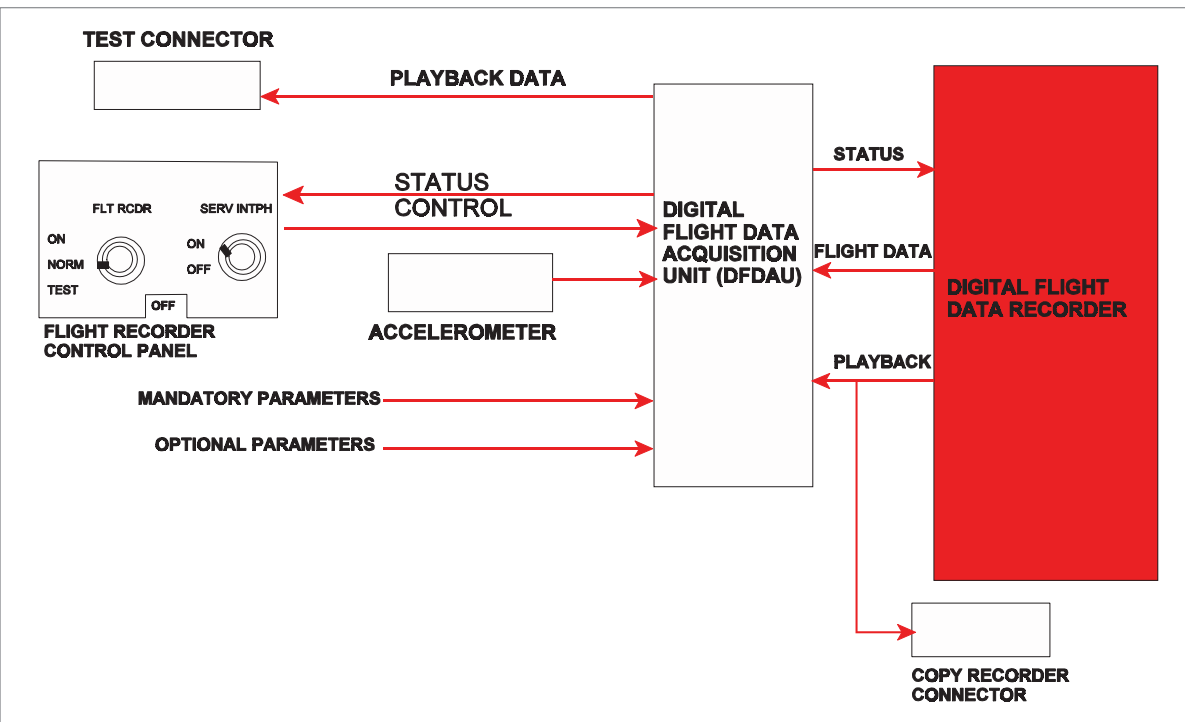


Figure 36.2 Digital flight data recorder system (boeing)

### Aircraft Integrated Data Systems (AIDS)

AIDS processes data for various aircraft systems to ease maintenance tasks. This is done via a data management unit (DMU) that collects and processes data to compile reports for storage and for printing. Some of this information is sent to the FDR via the flight data interface unit (FDIU) for recording mandatory parameters of the flight. The rest of the information is recorded on a separate flight maintenance recorder from which data can be printed out for the purpose of maintenance.

Data stored on the DMU can be printed out both in flight as well as on the ground for the purpose of maintenance.

It is also possible to transmit relevant data from AIDS to ground at certain intervals so that the aircraft performance can be monitored from the ground. This is done via ACARS (Airborne Communications and Reporting System) on a VHF data link.

## Parameters Recorded

The mandatory aircraft parameters recorded on the FDR depend upon the age and size of the aircraft and are specified in OPs Regulations.

The **main parameters** are:

- time or relative time count
- attitude (pitch and roll)
- airspeed
- pressure altitude
- heading
- normal acceleration
- propulsive thrust/power on each engine
- cockpit thrust/power lever position
- flaps/slats configuration or cockpit selection
- ground spoilers and/or speed brake selection

**Additional parameters** include the following:

- positions of primary flight controls and trim
- radio altitude and navigation information displayed to the flight crew
- cockpit warnings
- landing gear position

## European Regulations

### *Carriage of FDR*

FDR equipment is required to be carried on EU registered commercial air transport aircraft. The rules regarding carriage depend on a number of factors, i.e. number of passenger seats, type of propulsion, Maximum Take-off Mass (MTOM) and date of first registration, as follows;

- All aeroplanes over 5700 kg MTOM, or turbine powered less than 5700 kg but with more than 9 passenger seats, registered since 1 April 1998 (Referred to as case 1 in OPs Regulations)
- All aeroplanes >5700 kg MTOM registered since 1 June 1990 (case 2)
- All turbine powered aeroplanes >5700 kg MTOM registered before 1 June 1990 (case 3)

For the aircraft >5700 kg MTOM the minimum recording time is 25 hours, and all the main parameters listed previously must be recorded. For aircraft less than 5700 kg MTOM, (i.e. turbine powered, with more than 9 passenger seats registered after 1 April 1998) the recording time may be no less than 10 hours.

Aircraft >27 000 kg MTOM must also record the additional parameters.

### Other Requirements

The other regulatory requirements are:

- The FDR 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. This is achieved by starting when the first engine is started, and automatically switching off 5 minutes after the last engine is shut down.
- The FDR must have a device to assist in locating that recorder in water.
- Aeroplanes of 5700 kg or less may have the FDR combined with the cockpit voice recorder. Aircraft above 5700 kg must have 2 recorders, either separate FDR/CVR, or 2 combined recorders.
- An aeroplane may be dispatched with an inoperative FDR provided that:
  - it is not reasonably practicable to repair or replace the FDR before flight
  - the aeroplane does not exceed 8 further consecutive flights
  - not more than 72 hours have elapsed since the unserviceability
  - any cockpit voice recorder required to be carried is operative (unless it is combined with the FDR)



Chapter

**37**

Cockpit Voice Recorder

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## Introduction

The principal function of a Cockpit Voice Recorder (CVR) system is to preserve, in the event of an air accident, vital information that is recoverable for use by the Accident Investigation Branch (AIB). The CVR automatically records the last 30 minutes (2 hours on some aircraft) of communications and conversations on the flight deck. It is operational whenever 115 volts AC power is applied to the aircraft, any engine is running, or the aircraft is airborne. The system comprises a tape recorder, a control unit, a monitor display and an area microphone. The units and their locations are shown in *Figure 37.1*.

## The Recording

The recorder is required, in accordance with OPs Regulations, to record from before the aircraft is capable of moving under its own power, until it is no longer able to do so. It should, however, also record pre-start procedures and post engine shut-down procedures. Parameters recorded are:

- voice communications transmitted from or received on the flight deck
- the aural environment of the flight deck
- voice communication of flight crew members using the aeroplanes interphone system
- voice or audio signals introduced into a headset or speaker
- voice communication of flight crew members using the PA system

## The Voice Recorder

The tape recorder is located inside a crash-proof metal box that is painted red or orange and normally placed at the rear of the aircraft, often adjacent to the flight data recorder. The high impact case should be able to withstand shock, high temperature and fire.

On the front of the unit is fitted an underwater locating device (ULD), that will emit a continuous series of ultrasonic pulses to help locate a submerged CVR. The unit is automatically activated by water and the battery will last several days.

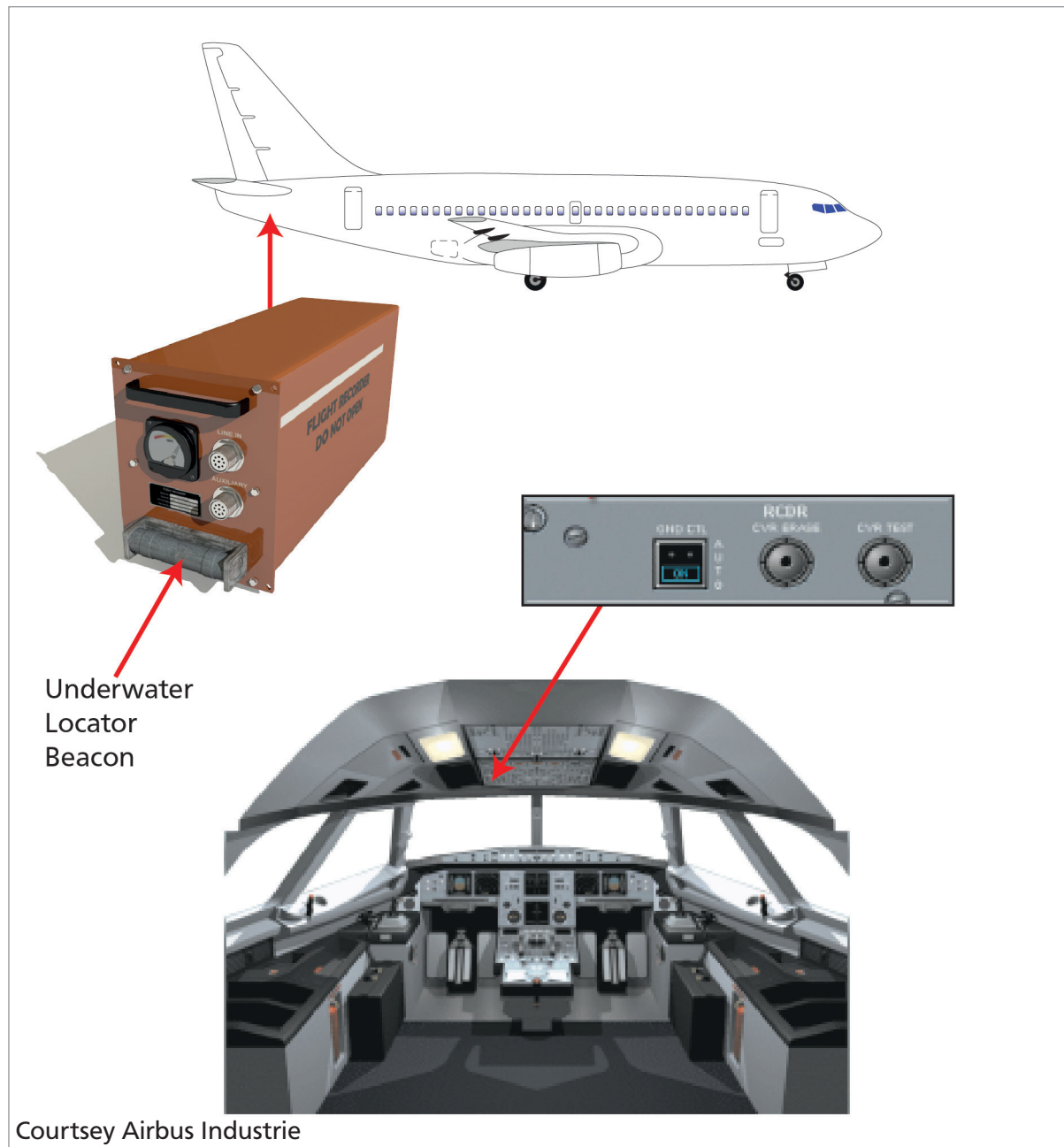


Figure 37.1 Cockpit Voice Recorder

## The Control Unit

This is mounted on the flight deck, usually in the roof panel. It contains monitoring and testing circuitry and also the area microphone to pick up general flight deck conversations and sounds. It has the following controls:

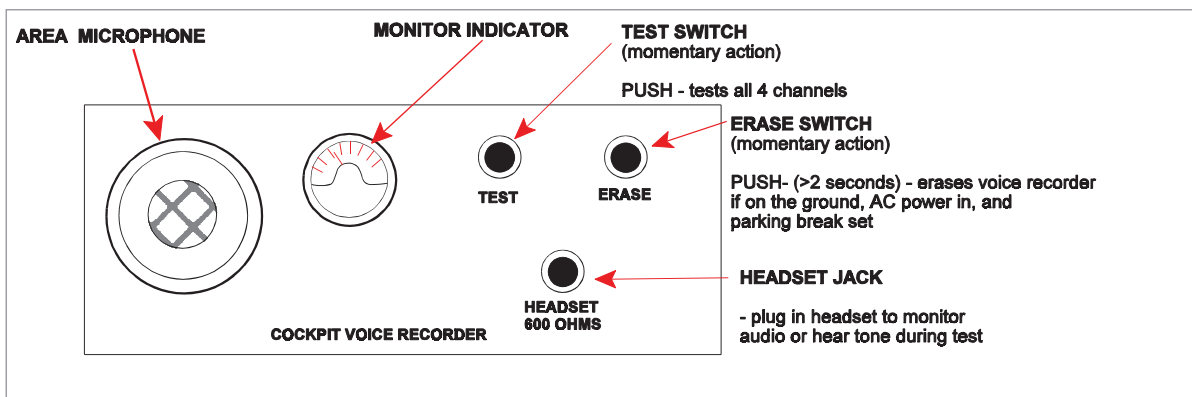
**AUTO / ON** When the switch is in the AUTO position the CVR will start to record when the first engine is started and will stop 5 minutes after the last engine is shut down. Selection of the ON position starts the CVR recording immediately and latches the switch in the ON position until first engine start, when it will click back to AUTO.



**CVR TEST** Pressing the TEST button activates an extensive set of functional tests which determine the integrity of the system using the BITE (built-in test equipment) facility. A successful self-test results in a visual 'good' indication (a status deflection needle or a status LED).

**ERASE** Erasure of the tapes is only possible with the aircraft on the ground, all engines stopped and the parking brake set. Suitable safety interlocks are installed to prevent inadvertent or airborne tape erasure. Additionally the erase button must be held depressed for at least 2 seconds before the circuit activates.

Some CVR control units will incorporate the area microphone as shown in *Figure 37.2*.



*Figure 37.2 CVR control unit*

## European Regulations

The rules for commercial air transport aircraft registered in the EU regarding CVRs are similar to, but not identical to, the requirements for FDRs as follows:

- All aeroplanes registered after 1 April 1998 which are >5700 kg MTOM or turbine powered, less than 5700 kg but with more than 9 passenger seats (Case 1)
- Turbine powered aeroplanes registered between 1 January 1990 and 31 March 1998 less than 5700 kg MTOM but with more than 9 passenger seats (Case 2)
- All aeroplanes with MTOM > 5700 kg (Case 3)

Minimum recording time is 30 minutes, except for aircraft >5700 kg MTOM registered after 1 April 1998, where the minimum recording time is 2 hours.

Rules for despatch with an unserviceable CVR are the same as for an unserviceable FDR. Aircraft less than 5700 kg MTOM may carry a combined CVR/FDR, but if the recorders fitted to aircraft >5700 kg MTOM are combined, they must be fitted with two.

## Questions

1. An altitude alerting system must at least be capable of alerting the crew on:
  1. Approaching selected altitude
  2. Abnormal gear/flap combination
  3. Excessive vertical speed
  4. Excessive terrain closure
  5. Excessive deviation from selected altitude
  6. Failure to set SPS or RPS as required
  - a. 1 & 3
  - b. 2 & 5
  - c. 4 & 6
  - d. 1 & 5
2. According to the regulations, when must the FDR on a 12 seat turbo prop a/c begin recording?
  - a. Switch on until switch off
  - b. From before the aircraft is capable of moving under its own power to after the a/c is no longer capable of moving under its own power
  - c. From lift-off until the weight on wheels switch is made on landing
  - d. At commencement of the taxi to turning off the runway
3. What is the GPWS mode 3 audible alert?
  - a. "don't sink, don't sink" followed by "whoop, whoop, pull up" if the sink rate exceeds a certain value
  - b. "don't sink, don't sink" followed immediately by "whoop, whoop, pull up"
  - c. "don't sink, don't sink" continuously
  - d. "Terrain, don't sink" continuously
4. What are the inputs to a modern jet transport aeroplane's stall warning system?
  1. AoA
  2. Engine rpm
  3. Configuration
  4. Pitch and bank information
  5. Control surface position
  6. Airspeed vector
  - a. 1 & 3
  - b. 1, 2, 3, 4, 5 & 6
  - c. 2, 4 & 6
  - d. 1, 3, 5 & 6

5. GPWS may indicate:

1. Excessive sink rate after T/O
  2. Excessive descent rate
  3. Excessive terrain closure
  4. Ground proximity, not in the landing configuration
  5. Upward deviation from glide slope
  6. Proximity to en route terrain
- 
- a. 1, 2, 3, 4
  - b. All of the above
  - c. 2, 3, 4, 5
  - d. 3, 4, 5, 6

## Answers

1	2	3	4	5
d	b	c	a	a



# POWERPLANT & SYSTEMS MONITORING INSTRUMENTATION

ATPL GROUND TRAINING SERIES

10

10



Chapter  
**38**

**Engine Instrumentation**

---

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## An Introduction to the Engine Instruments

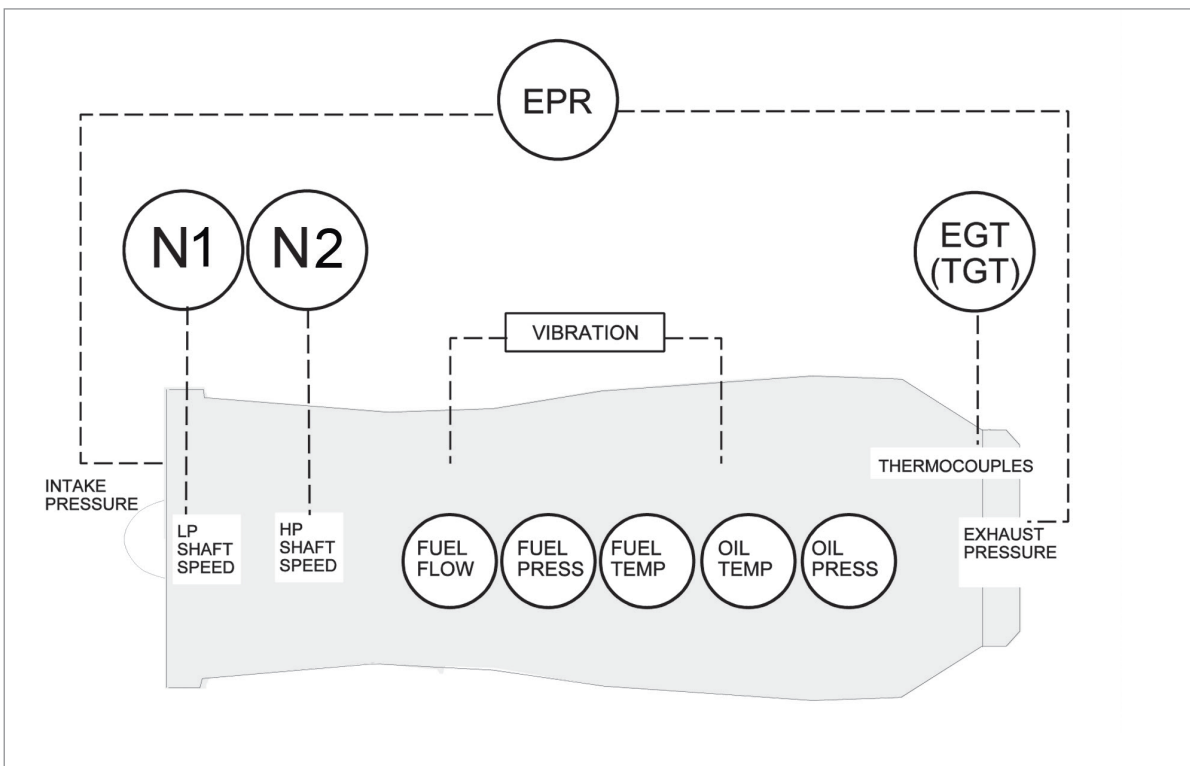
Just as it would be impossible to fly a large modern aircraft safely without the flight instruments, so would it also be impossible to fly it safely without the engine and aircraft systems instruments.

The engine instruments are divided basically into two categories, **Performance Indicators** or **Engine Condition Indicators**.

**Performance Indicators** are thrust indicating instruments such as the Engine Pressure Ratio (EPR) gauge or the Fan Speed (N1) gauge.

**Engine Condition Indicators** include the Exhaust Gas Temperature (EGT) gauge, Compressor Speed, Oil Pressure and Oil Temperature gauges. We will be discussing these and others in the following text.

*Figure 38.1* shows some of the parameters previously mentioned and the position of the sensors that are required to measure them.



*Figure 38.1* Some of the parameters required to be displayed

## Types of Display

There are two types of cockpit display, the analogue display (clockwork cockpit) or the electronic display (glass cockpit). In the former, there are a multitude of gauges displaying information, in the latter the display is shown on cathode ray tubes (CRTs) or liquid crystal display panels (LCDs) with light emitting diodes (LEDs) for digital displays. A small number of conventional gauges are retained in case of failure of the electronic displays.

Both types of display convey essentially the same information to the pilot, but the flexibility of the glass cockpit system means that it is now taking over as the preferred means of showing both flight and engine instrumentation.

## Thrust and Power Measuring Instruments

Thrust measuring instruments are of two basic types:

- the type that measures the jet pipe pressure, the **P7 gauge**.
- the type that measures the ratio of two parameters, the jet pipe pressure and the engine air intake pressure, the **EPR gauge**. (*Figure 38.2*)
- propeller-driven aircraft measure and indicate **Torque**. This is an indication of engine power. The propeller converts power into thrust.

On some large turbo-fan engines the integrated turbine discharge pressure and fan outlet pressure is compared to the compressor inlet pressure to produce what is called 'integrated' EPR.

Pitot tubes, suitably positioned, sense the pressures which are required to work the system; the tubes can either be connected directly to the indicator in the cockpit or to a pressure transmitter which is electrically connected to the indicator.

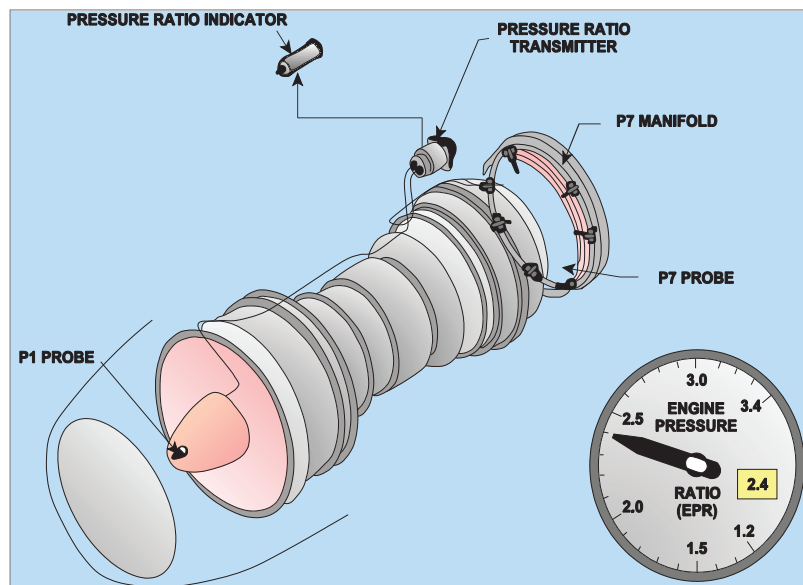


Figure 38.2 The engine pressure ratio (EPR) indicating system

The P7 system gauge can be marked in inches of mercury (inHg), pounds per square inch (psi), or a percentage of the engine's maximum thrust.

Although EPR can be indicated by either mechanical or electronic means, it is more normal to find the electronic system in use. This system uses two transducers which sense the relevant air pressures and vibrate at frequencies proportional to these pressures. A computer works out the electrical signal appropriate to the pressures and that signal is sent to the EPR gauge in the cockpit and to the engine management system.

The engine intake pressure will vary with changing ambient pressure and also with changing airspeed. An increase in forward airspeed during take-off will cause a drop in the reading on the EPR gauge. This is only an 'apparent drop' because it is only the ratio of two pressures, engine intake pressure and the jet pipe pressure, which is changing.

This apparent change is caused by a relative increase of the engine intake pressure with forward airspeed during take-off. The jet pipe pressure, being unaffected at low airspeeds, causes the ratio between the two pressures to fall.

This will be seen as a reduction in EPR on the gauge, which might cause the inexperienced engine operator to open the throttles further in an attempt to restore the loss, having eyes for only the one parameter, while the other parameters, (N1, N2, N3 & EGT) are in danger of exceeding their limits without his knowledge.

In an attempt to prevent this happening, most operators require that EPR is set before the aircraft has reached approximately 60 knots, no increase in engine power being allowed unless in emergency after this speed.

After take-off, as the airspeed increases beyond  $V_2$ , the increase in engine intake pressure is passed through the engine to the jet pipe, so changing the ratio back to that set on take-off.

## Engine Torque

Turboprops and turboshaft engines produce torque rather than thrust. The systems that produce indications of thrust for turbojet engines and turbofan engines are vastly different to those which produce indications of torque for turboprop and turboshaft engines. The torquemeter measures, and its indicator displays, the power being produced by the engine. Torque, by definition, is a force applied at a distance to a turning point. If applied to the **PLANE** formula given in Piston Engines Chap 1, the turning force is the product of the mean effective pressure **P** acting on the area of the piston **A** at distance **L** (the stroke is twice the throw of the crank) Therefore **P**, **L** & **A** can be replaced by the word **Torque**. **N** represents the number of cylinders and will remain constant, the only other variable is **E** the number of effective power strokes or rpm of the engine. **Power** can therefore also be expressed as **Torque** × **rpm**.

There are two main methods employed in measuring the torque of the engine. One uses oil pressure and the second is an electronic device. The units of measurement vary from system to system. The indicator gauges may be calibrated to read psi, inch or foot pounds, newton metres, brake or shaft horsepower, or percentage of maximum. Torque is measured between the engine and the reduction gearbox.

The oil torquemeter system makes use of a phenomenon that axial thrust (movement) is generated when **helically cut gears** are used to transfer power from one shaft to another. [Figure 38.3](#) shows how this end thrust can be utilized to provide an indication of the torque output of a turboprop or turboshaft engine.

As the gears in the propeller reduction gearbox rotate to drive the propeller, the amount of torque that they are transmitting attempts to move them axially. this axial force is proportional to the torque that is producing it.

The gears cannot be allowed to move axially because this would cause the teeth to no longer mesh with each other and the drive would fail, so the axial force has to be counteracted to maintain the gears in alignment. The force comes from passing engine oil through a filter and

then to a torquemeter pump which enables its pressure to be boosted to (in some cases) as much as 800 psi. This high pressure is allowed into cylinders which form the bearings within which the helical gear shafts rotate.

A small bleed hole in the wall of the cylinder will be covered by the gear shaft if it moves into the cylinder under increasing axial load, and this will cause the oil pressure within the cylinder to build up until it can move the gear shaft back to its original position. Conversely, if the load on the helical gear shaft decreases, the existing oil pressure will force its shaft slightly out of the cylinder. This uncovers the bleed hole allowing the balancing oil pressure to be reduced and so the gear shaft moves back into correct position within the cylinder.

If the oil pressure balancing the axial force is measured, it can be compared with reference figures which take into account the ambient pressure and temperature, and the performance of the engine, its power output, can be judged

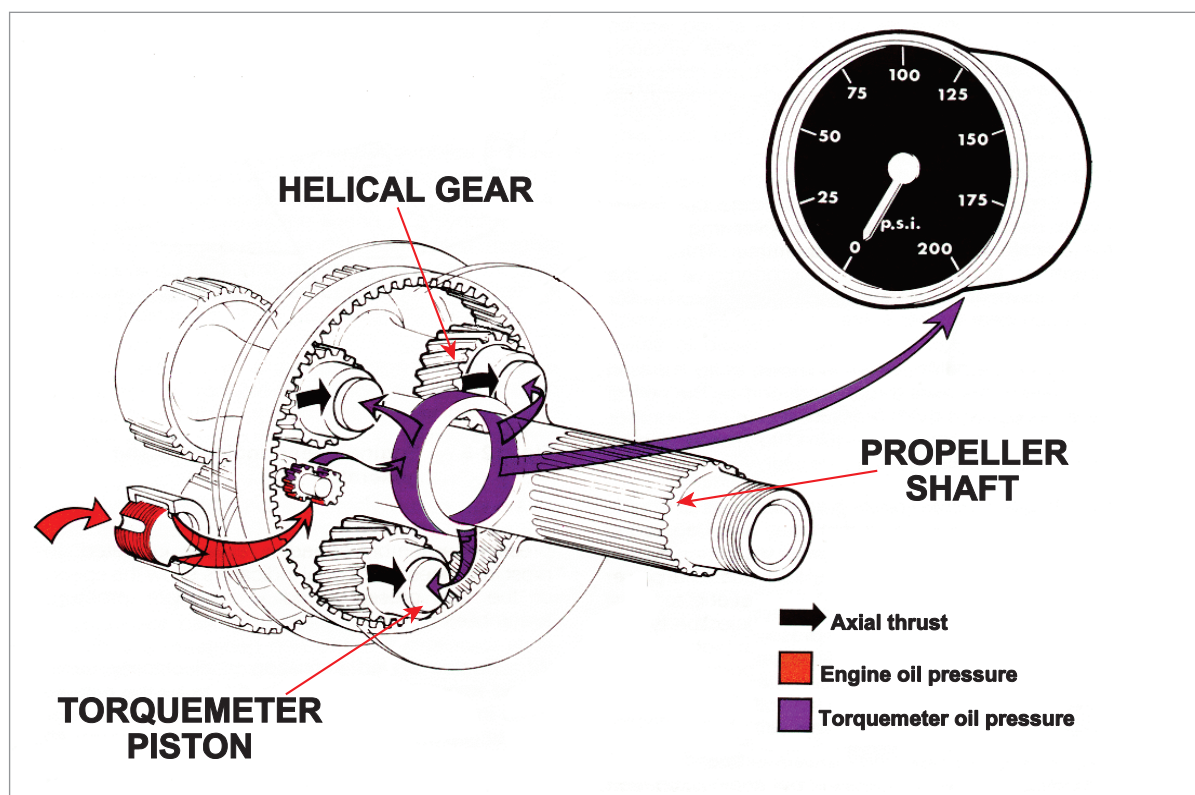


Figure 38.3 The torquemeter system

The electronic system comprises two concentric shafts. One, the **Torque Shaft**, is connected to both the engine and the **propeller's reduction gear box**. The second shaft, the **Reference Shaft**, is connected only to the engine. An exciter wheel (toothed gear) is formed at the forward end of each shaft. The exciter wheels rotate past an electromagnetic pick-up and produce an AC voltage. The exciter wheels are aligned at assembly, but as power is increased the torque shaft twists; this displaces the phase relationship of the voltages produced. The displacement is proportional to the change in power and is used to drive an indicator. This system is simple, and lighter than other systems, and it has proven to be very reliable in service. (Figure 38.4)

The torque indicator may indicate negative (windmilling propeller) as well as positive torque. The torque limits are colour coded and shown on the gauge. A red coloured band or marker indicates maximum limits. On a FADEC system these limits may be adjusted and set by the crew, and the indication can be presented in a digital read-out.

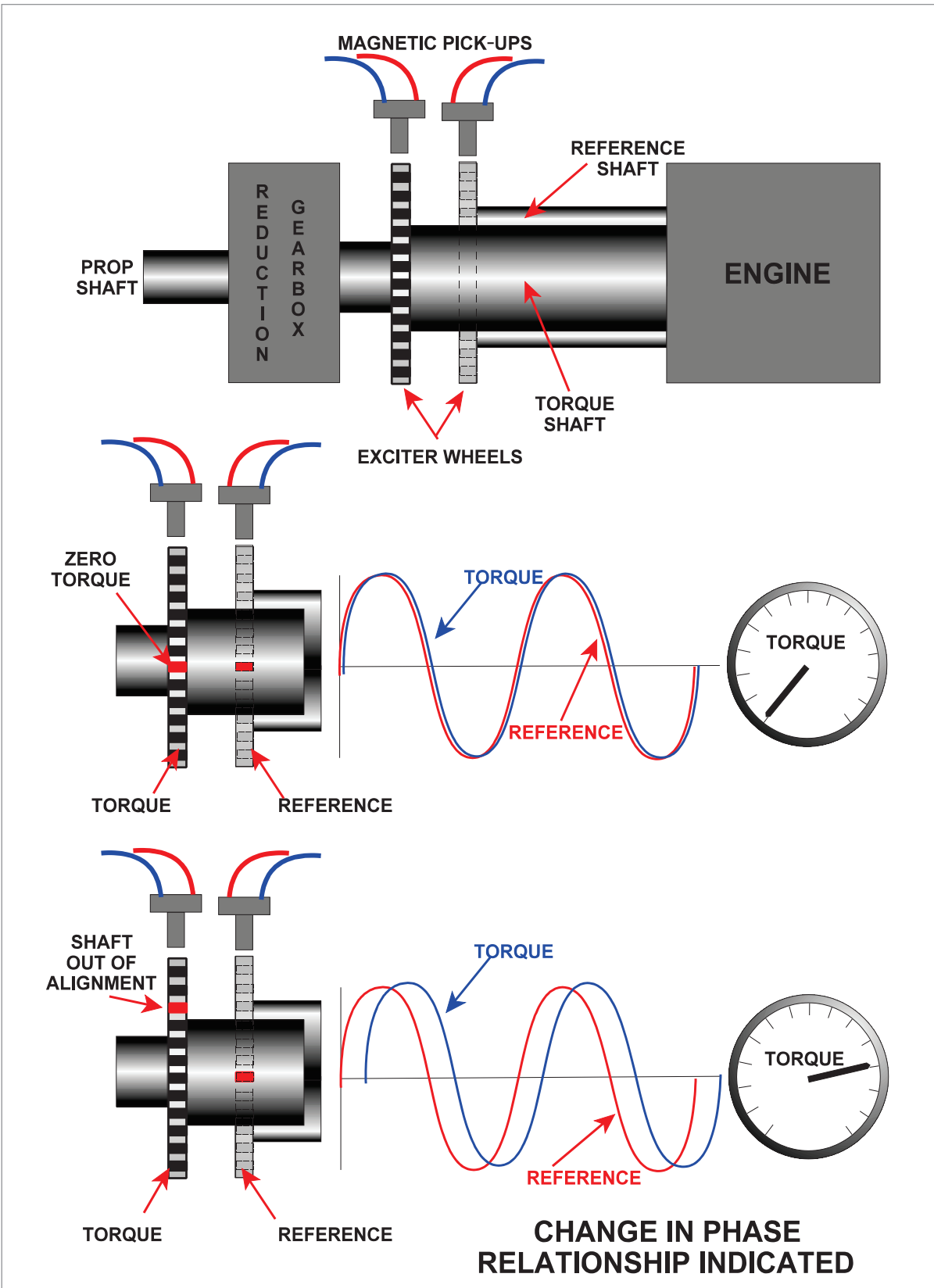


Figure 38.4 Electronic torque meter

## Engine rpm

The measurement of engine speed is of vital importance since, together with other parameters, accurate control and monitoring of the engine can be achieved. On piston engines it is crankshaft speed that is measured, whilst on gas turbine engines it is the speed of the compressor. The rpm indicator is called a **Tachometer (Tacho)**. There are three basic methods of measuring engine rotational speeds:

- **Mechanical (Magnetic) Tachometer.**
- **Electrical Generator System. (Tacho Genny)**
- **Inductive Probe System.**

There are no firm guidelines as to the application of each of the tachometer systems, although engine and aircraft design will dictate which system can be best utilized.

The **Mechanical Tachometer (Figure 38.5)** is now only found on older piston aircraft. It consists of a **Flexible Drive Shaft** that is connected to the flight deck **Tacho-indicator**. The input drive causes a magnet in the indicator to rotate. The magnet rotates inside a copper or aluminium drag-cup; this induces **Eddy Currents** in the drag-cup which oppose the magnetic field of the magnet. A torque is established which turns the drag-cup in the same direction as the permanent magnet. A shaft extends from the drag-cup and is connected to a pointer. The turning motion of the pointer is against the tension of a **Hairspring** which controls the drag-cup position and hence the position of the pointer. The flexible drive is driven at reduced speed, but true speed will be shown on the indicator. The indicator incorporates compensation devices for change in temperature.

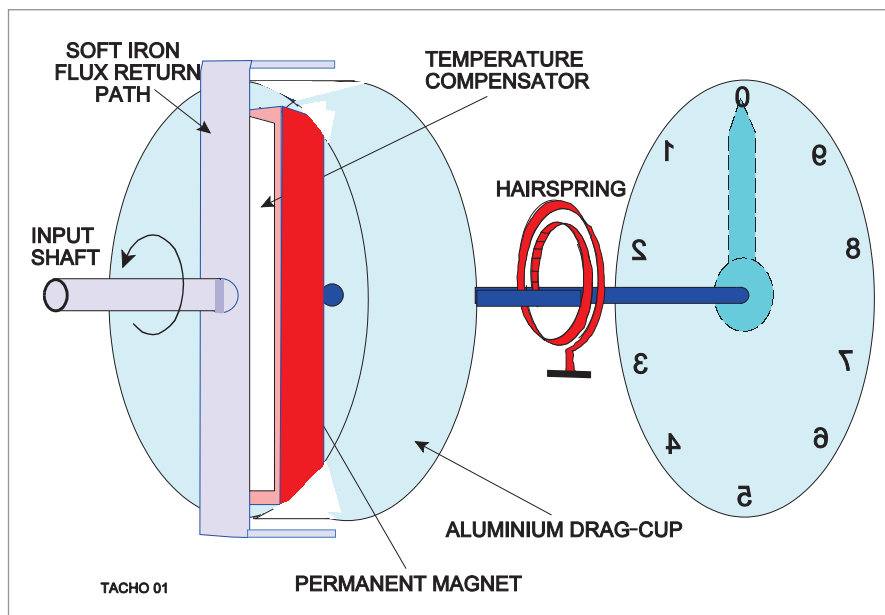


Figure 38.5 Mechanical tacho

The **Electrical Generator System** (*Figure 38.6*) is possibly the oldest form of engine speed measurement still in use on large aircraft. It utilizes a small three phase (tacho) generator, driven by the engine. The output of this generator is then taken to an indicator which consists of an asynchronous motor turning a drag-cup assembly which moves a pointer over a scale as in the mechanical system.

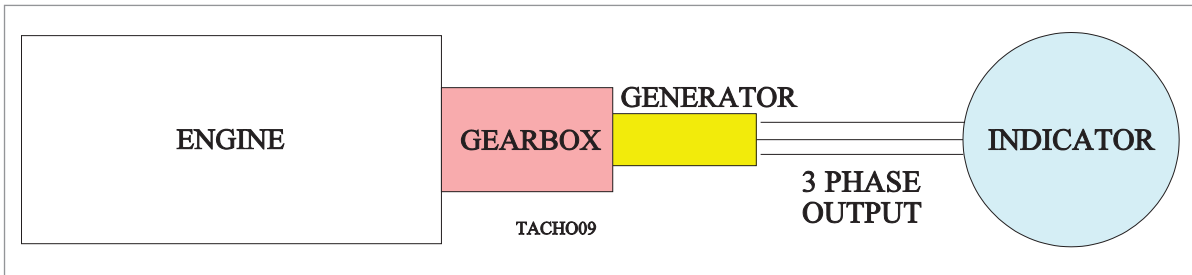


Figure 38.6 Tacho-generator

The indicator (*Figure 38.7*) can either show the actual revolutions per minute (not too common), or the speed as a percentage of maximum engine speed.

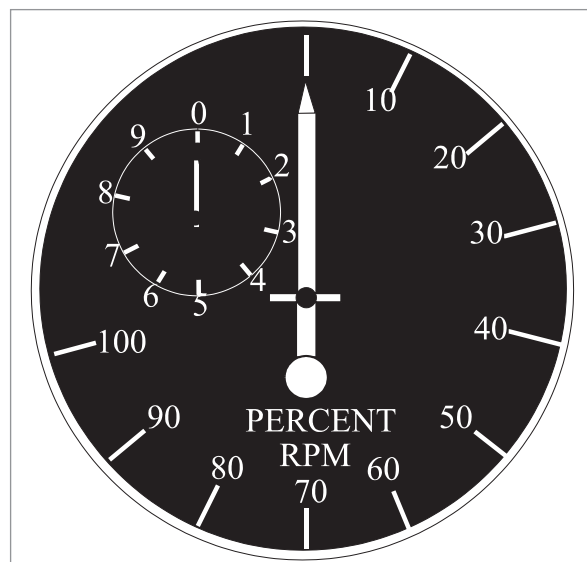


Figure 38.7 Percentage tacho-indicator

On twin or triple spool engines the speed of rotation of the high, intermediate and low pressure compressors can be displayed. These would be termed N3, N2 and N1 (N being the SI symbol for rotational speed). N1 and EPR are the parameters used to measure thrust in turbojets.

An overspeed pointer (trailing or limit pointer) can also be fitted concentrically with the main pointer, and is initially positioned at the appropriate max rpm graduation. If the main pointer exceeds this position, the limit pointer is carried with it. When speed is reduced the limit pointer will remain at the maximum speed reached. It can be reset by applying a separate 28 V DC supply to a solenoid in the indicator.

Although there would always be provision on the HP compressor spool for driving a tacho-generator through the high speed gear box, facilities may not always be available for driving tacho-generators from the intermediate and low pressure compressor shafts. If this is the case, a **Speed Probe**, shown in *Figure 38.8* and *Figure 38.9*, can be used to very good effect.

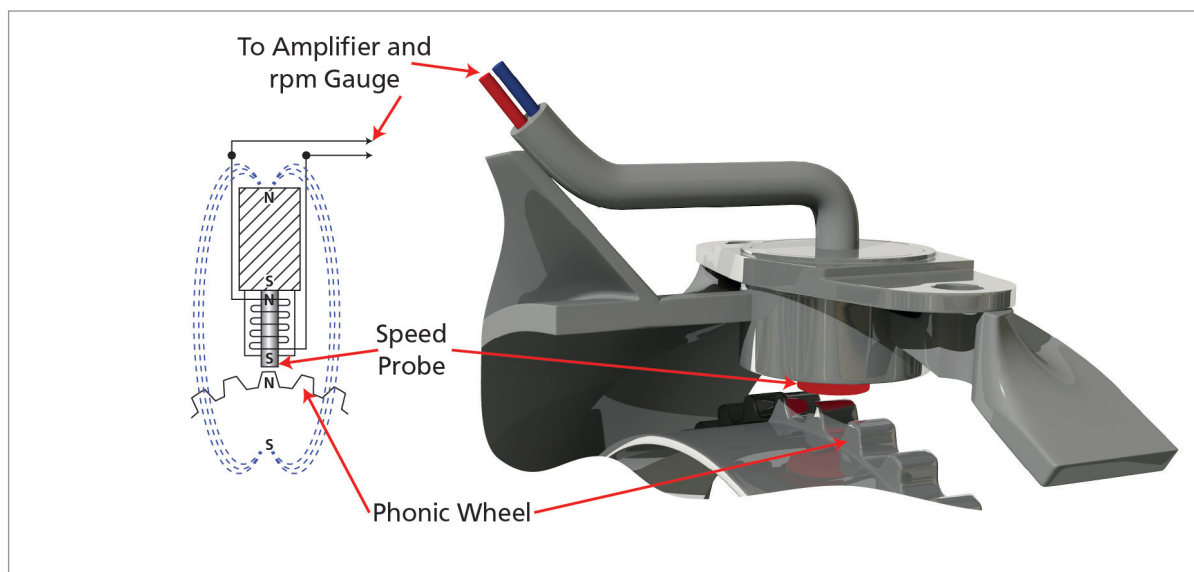


Figure 38.8 Measuring engine speed with a phonic wheel and a speed probe

The speed probe is positioned on the compressor casing in line with either a phonic wheel (Figure 38.8), or the actual fan blades (Figure 38.9). As the spool rotates, the magnetic flux in the probe or sensor head is altered. This changes the current flowing in the coil fitted inside the probe and the frequency with which it changes is directly related to the spool speed. This frequency is fed to an indicator in the cockpit to show the spool rotational speed.

In addition to providing an indication of spool speed, the tacho-generator or speed probe can both be used to provide a signal which will illuminate a warning lamp on the engine start control panel. This can tell the pilot not only that the engine is turning, but also whether the engine is turning in the correct direction. This is particularly important during engine start as it is used to inform the pilot when to open the HP Fuel Cock. This lamp is only illuminated during the start cycle.

An advantage of this system is the reduction in moving parts required in the engine, and that a number of separate electrical outputs additional to those required for speed indications can be provided, e.g. automatic power control and flight data acquisition systems.

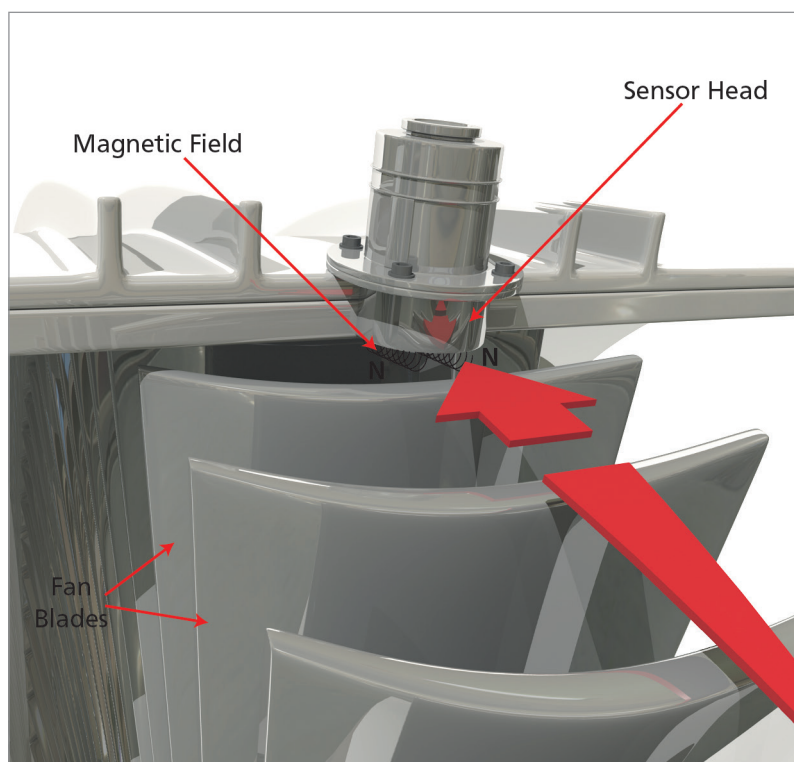


Figure 38.9 The use of a speed probe to measure fan speed



The tachometers mentioned in the text above receive their speed signals directly from speed sensors or via servo-operated systems. These indicators require a power source from the aircraft airborne power supply. In the event of power supply or signal failure, the indicator is returned to an **Off Scale** position, and a **Power Off Flag** may be displayed.

As previously stated, presentation of speed is now usually displayed as a percentage figure. It is only piston engine aircraft where the actual speed of rotation is displayed. Gas turbine engines have dial displays which show percentage speed, with 100% corresponding to the optimum turbine speed. Two scales are displayed. A main scale is calibrated 0 to 100% in 10% increments. A second pointer or digital counter displays speed in 1% increments. As well as digital read out, vertical ribbon displays are used.

In line with other instruments, coloured arcs or indicator lines are used to show ranges and limits of engine speed. Green represents normal operating range, with amber denoting caution. Red shows maximum or minimum speed, and ranges that are restricted because of excessive vibration. On a piston engine the reference rpm should also be placarded.

On multi-engine aircraft, to reduce structural vibration and noise the speed of all engines must be synchronized. It is impractical to have the pilot adjust the throttle of each engine manually to synchronize the speed, and individual indicators may vary in accuracy. In order to facilitate manual adjustment of speed an additional instrument known as a **Synchroscope** (Figure 38.10) is used. The instrument was designed at the outset for operation from the AC generated by the tachometer system. The instrument provides qualitative indication of the difference in speeds between two or more engines. One engine is selected as a master, the others are slaves to it. The instrument shows clearly whether a slave engine is running faster or slower than the master. An example of the dial presentation for synchrosopes for a twin and four-engine aircraft, and combined tacho and synchroscope are shown.

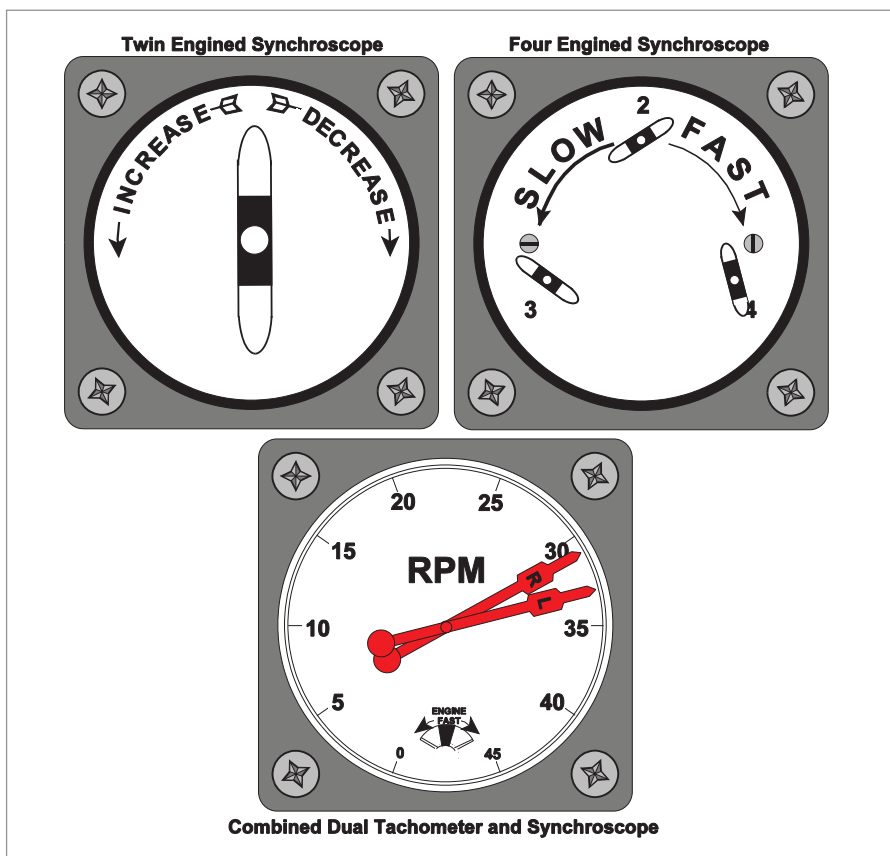


Figure 38.10 Synchrosopes

## Temperature Sensing Equipment

Piston and gas turbine aero engines are heat engines. The power they produce is directly proportional to the heat released during combustion of fuel. Engine components and systems are designed to withstand certain temperatures. If their limits are exceeded, they may fail. To allow safe operation the engine temperatures must be monitored. The effect of ambient temperatures as well as combustion must be considered. The following temperatures are monitored on piston and gas turbine engines: air inlet, piston cylinder heads, piston exhaust gas, gas turbine compressor outlets, turbines oil and fuel systems and internal air system.

The temperatures monitored may range from  $-56^{\circ}\text{C}$  to  $+1200^{\circ}\text{C}$ . Different sensors are used depending on the temperature range to be monitored. They fall broadly into two categories; high temperatures and low temperatures.

There are four major types of measuring devices. They are:

### *Expansion Type*

This relies on the principle that most solids, liquids and gases expand and contract with temperature changes, e.g. the mercury thermometer and bimetallic strip.

### *Vapour Pressure Type*

Liquids when subjected to a rise in temperature change their state from liquid to vapour. Therefore by measuring the pressure of the vapour an indication of temperature can be gained.

### *Electrical Type*

A change in temperature of an electrical conductor can cause a change in resistance of the conductor. Thus measuring the change in resistance can indicate the temperature of the conductor. This sensor is called the **Resistance Type** (temperature bulb). In addition, two dissimilar metals when joined together at their ends (a junction) can produce an electrical potential called a thermo EMF (Seebeck Effect). This is dependent on the temperature difference between the junctions. This is known as a '**Thermo-electric Type**' or '**Thermo-couple**'. This system is explained in detail later.

### *Radiation Type*

The radiation emitted by any body at any wavelength is dependent upon the temperature of that body. This is termed its 'emissivity'. If the radiation is measured and the emissivity is known, the temperature of the body can be determined. Such a measuring technique is known as **Pyrometry**.

Generally the expansion and vapour pressure sensors are used to indicate lower temperatures. They are direct reading, e.g. thermometer. The electrical and radiation sensors are used to measure higher temperatures and can be direct reading to a moving coil Indicator, e.g. piston engine exhaust gas temperature. However, most systems today use remote sensors that feed to servo-operated indicators after the signal from the sensor has been amplified. A **Ratiometer-type** indicating system can be used to obtain the greater accuracy required when indicating the temperature of critical component e.g. turbines.

The temperature of the gas passing through the turbine in a gas turbine engine is the most important parameter of those displayed on the engine instruments. Operation of the engine beyond the limits of turbine temperature, even for only a moment, is liable to cause excessive turbine blade creep which can be catastrophic if the rotating blades touch the casing of the engine.

The gas temperature must be monitored closely and automatic temperature limiting equipment is fitted to most gas turbine engines operating today. To enable this monitoring to be achieved temperature probes are inserted in the gas stream.

Temperature probes are formed from the junction of two dissimilar metals; when heated the junction generates a small voltage which is proportional to the actual temperature which produced it. The voltage can be measured on a milli-voltmeter and displayed in the cockpit as the temperature at the rear of the engine. A **Galvanometer** is a very sensitive instrument used to indicate these low voltages. The galvanometer uses a basic **Wheatstone Bridge Balancing** circuit that alters the magnetic field in a coil; this change produces a torque to drive an indicator.

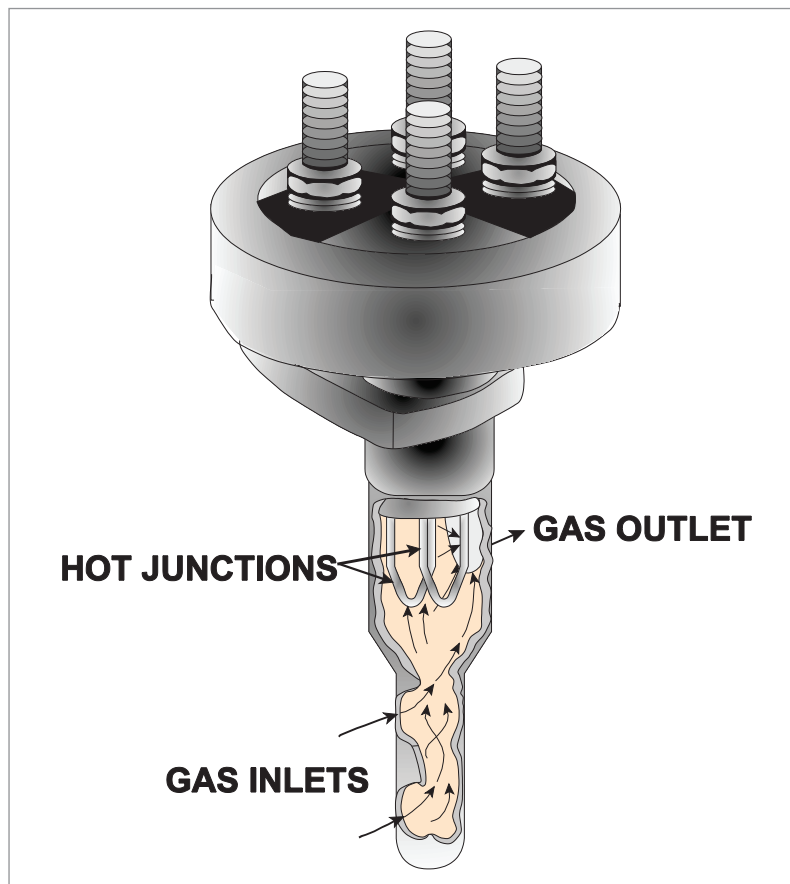


Figure 38.11 The gas flow over the probes and their electrical connections

Figure 38.11 shows how the probes, the hot junctions, are connected and also how the gas flows over them. The output from the probes is sent to the cockpit engine instrument, which is the cold junction, where the EMF is measured on a very sensitive milli-voltmeter to display the engine gas temperature.

Just one probe would obviously not supply enough information to accurately tell the pilot what was going on in the whole turbine, it could only inform him about the small part of the turbine that it was monitoring. It is therefore necessary to place a number of probes, electrically connected in parallel, all around the periphery of the engine or the exhaust system. This means that the gas stream is sampled in many more places and that the output is the average of all of the probes. This has an added advantage that if one probe is damaged, the effect on the reading on the gauge is minimal (a slight drop may occur). The actual position of the probes depends upon two things: the anticipated maximum temperature of the gas and the ability of the probe material to withstand that temperature.

The industry standard for the material used in the temperature probes in gas turbine engines is chromel (nickel chromium) and alumel (nickel aluminium). These two materials may not have the highest milli-voltage output of the materials available, but their ability to withstand very high temperatures coupled with a reasonable volts per degree ratio makes them ideal for the job.

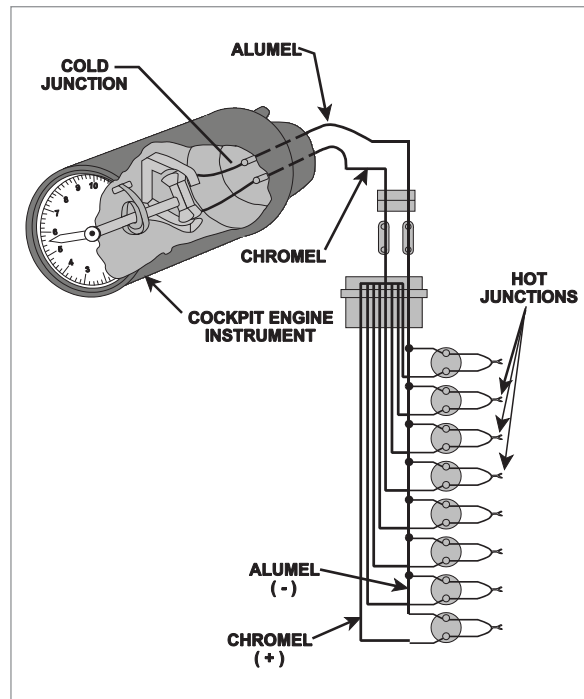


Figure 38.12 Thermocouple indicating system

**Note:** The system requires no power supply to indicate temperature however, if the signal is to be used to supply a temperature limiting system, the voltage will need to be amplified. This will be supplied by the aircraft's electrical system.

In the case of a system that supplies both a temperature limiting system (top temperature control) and a temperature indicating system, the probes will contain two hot junctions, one to feed the limiter and one to feed the indicator. As illustrated in [Figure 38.11](#).

The positioning of the probes within the engine depends on the temperature of the gas and the ability of the metal they are made of to withstand it. On engines where the temperature of the gas within the turbine is too high for the metal of the probes to withstand, they may be positioned after the turbine and the gauge calibrated to read 'exhaust gas temperature' (EGT). On other engines, it may be found convenient to combine the temperature probes with the pitot probes which measure exhaust gas pressure (P7), in this case the gauges will read 'jet pipe temperature' (JPT).

Obviously it would be ideal if the temperature could be sampled either before the turbine, called either 'turbine inlet temperature' (TIT) or 'turbine entry temperature' (TET), or inside the turbine, called 'turbine gas temperature' (TGT). In every case the position of the probes is dependent upon their ability to withstand the temperatures they encounter.

Actual blade temperature can be measured by the radiation method, with the use of an optical or pyrometer.

Air temperature is one of the basic parameters used to establish data vital to the performance monitoring of aircraft engines, e.g. thrust settings, fuel/air ratios settings etc. The temperature ideally required is that sensed at static conditions at various flight levels. This is called **Static Air Temperature (SAT)**. However, this is not possible for all types of aircraft or, in many instances, for one type of aircraft, for the measurements can be affected by the adiabatic compression with increased speed. Below Mach 0.2 the temperature is very close to SAT, but at higher Mach numbers an increase in skin friction will raise the air temperature. This increase is commonly referred to as '**Ram Rise**', and the temperature indicated called **Ram Air Temperature (RAT)** i.e. SAT plus the ram rise. The ram rise can be calculated mathematically as a function of Mach No. and for each type of aircraft tables or graphs can be included in flight manuals, or computed by air data computers to correct the indicators to SAT. The proportion of ram rise dependent on the ability of the sensor to sense or recover the temperature rise. The sensitivity in this case being expressed as a percentage and termed **Recovery Factor**. If for example, a sensor has a recovery factor of 0.80, it will measure SAT plus 80% of the ram rise.

For use at high Mach numbers, **Total Air Temperature (TAT)** is measured. The air is brought to rest (or nearly so) without addition or removal of heat. The temperature probes used have a high recovery factor (approximately 100%). TAT is equal to SAT + Ram Rise.

Temperature indicators use coloured arcs to show their operating range: green for normal, amber for caution and red for upper or lower limits.

## Pressure Gauges

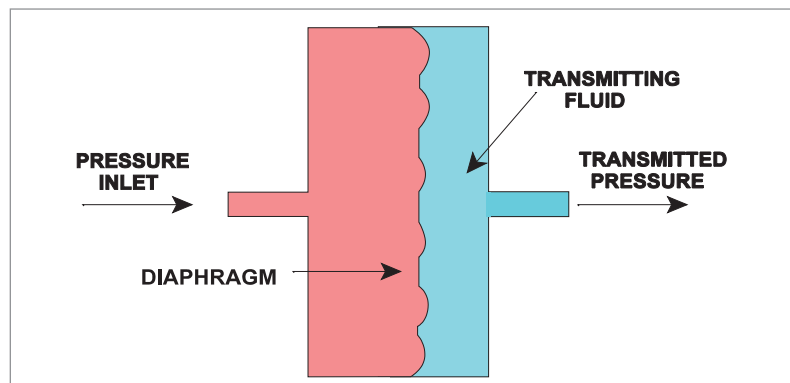
In many of the systems associated with the operation of the aircraft and its engines, liquids and gases are used, the pressures of which must be measured and indicated. The gauges and indicating systems fall into two categories: **Direct Reading** and **Remote Indicating**. Remote indicating is where a separate sensing element is connected to a pressure source at some remote point.

Pressure is defined as force per unit area. It is normally indicated either as pounds per square inch (psi) or inches of mercury (inHg); however, the measures of bar (1 bar = 14.5 psi) and pascal (1 bar = 100 000 pa) are sometimes referred to in aircraft manuals.

In connection with pressure measurement we are concerned with the following terms: **Absolute Pressure** and **Gauge Pressure**. Most pressure gauges measure the difference between absolute pressure and the atmospheric pressure. This is gauge pressure.

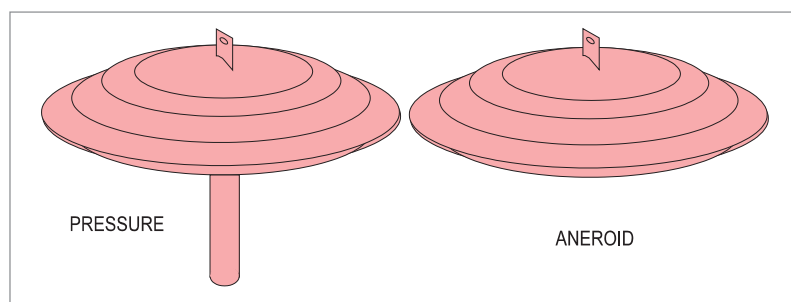
To actually measure pressure in a system, **Elastic Pressure Sensing Elements** are used in which forces can be produced by applied pressures and converted to mechanical movement. The movement can then operate a direct reading gauge or electrical transmitter. The sensing elements commonly used are **Diaphragms, Capsules, Bellows and Bourdon tubes**.

Diaphragms (*Figure 38.13*) consist of corrugated circular metal discs which are secured at their edge, and when pressure is applied they are deflected. Diaphragms are used to measure low pressures.



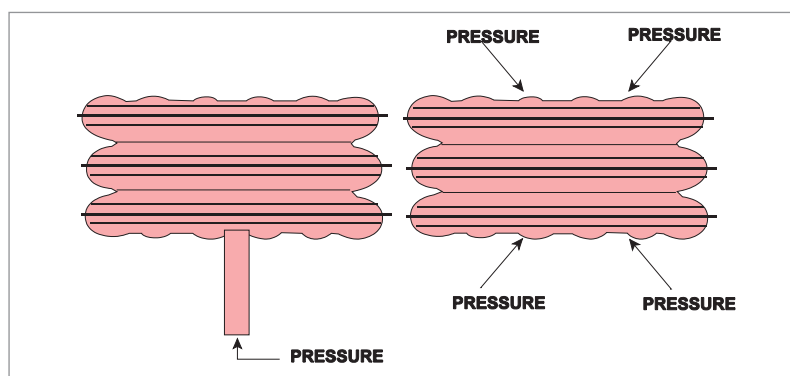
*Figure 38.13 A diaphragm pressure sensor*

Capsules (*Figure 38.14*) are made up of two diaphragms placed together and joined at their edges to form a chamber which may be sealed, called an **Aneroid**, or open to a pressure source and called a **Pressure capsule**. Like diaphragms they are used to measure low pressure, but they are more sensitive to small pressure changes.



*Figure 38.14 Pressure and aneroid capsules*

The bellows (*Figure 38.15*) type element can be considered as an extension of the corrugated diaphragm principle. It may be used for high, low or differential pressure measurement. It is typically used to measure pressures like the aircraft's LP booster pump output.



*Figure 38.15 Bellows sensors*

The Manifold Absolute Pressure Gauge or **MAP** (*Figure 38.16*) of a piston engine measures both pressure and differential pressure. Note this gauge measures **Absolute Pressure** and

indicates inches of mercury (inHg). When the engine is running this gauge can indicate less than atmospheric pressure. Earlier versions of this gauge were calibrated to read **Boost** in **psi** and called **Boost Gauges**. Under standard conditions the boost gauge will read 'zero' and the MAP gauge will read 30 inHg. This indication is called **Static Boost**.

The Bourdon tube (*Figure 38.17*) is about the oldest of the pressure-sensing elements. The element is essentially a length of metal tube with an elliptical cross-section shaped into a letter C. One end of the tube is sealed and called the free end. The other end is connected to the pressure source and fixed. When pressure is applied the tube tries to straighten; this movement is magnified to drive an indicator pointer. The Bourdon tube can be manufactured to indicate high or low pressures but is normally associated with higher pressures such as engine oil pressure.

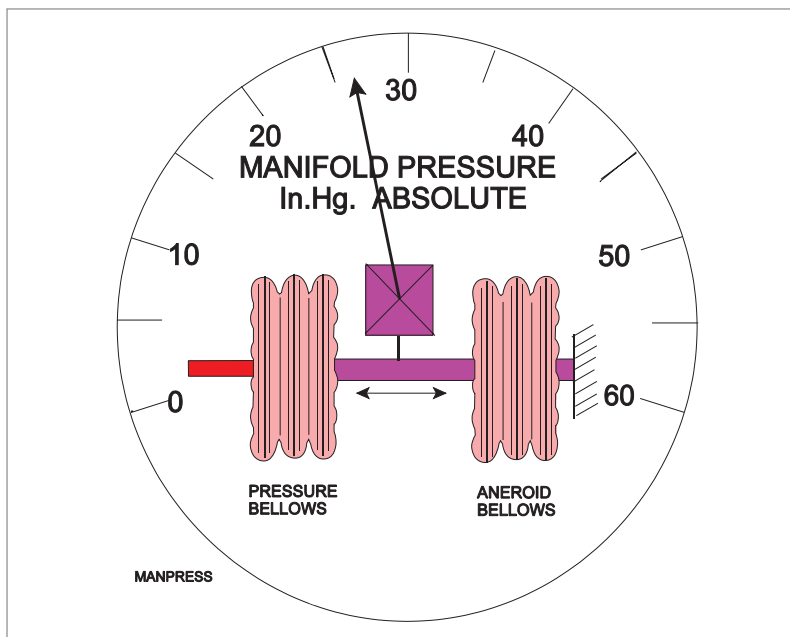


Figure 38.16 Manifold absolute pressure gauge

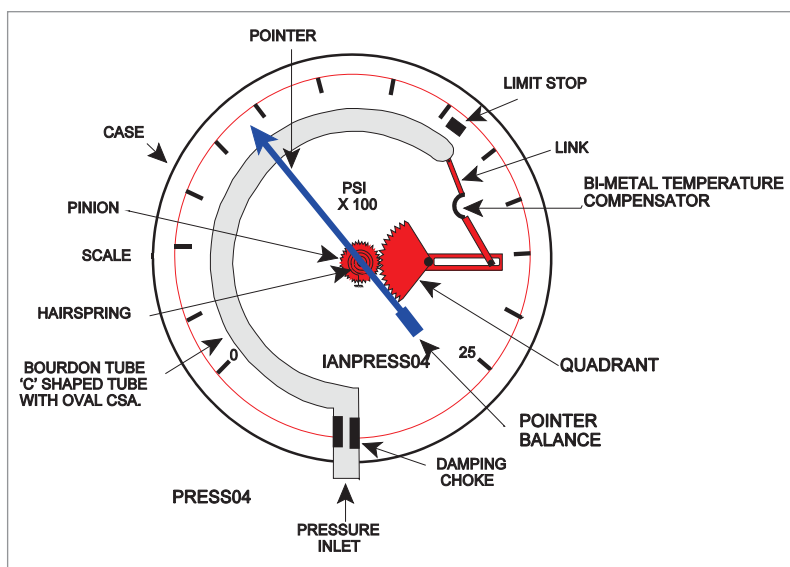


Figure 38.17 The Bourdon tube

It would be impractical to run an oil feed pipe from the outer engine of a Boeing 747 to a flight deck pressure indicator of the Bourdon tube type. To overcome this problem remote-indicating systems are used. They consist of two main components: a transmitter unit located at the pressure source and an indicator mounted on the appropriate panel. They have distinct advantages over direct reading gauges; for example, the pressure of hazardous fluids can be measured at their source and not brought into the cockpit. Also weight can be saved by reducing the length of pipelines. The transmitters feed varying current to an indicator and can be AC or DC in operation. These systems are covered later.

As well as indicating actual pressures, warnings can be displayed to the pilot by pressure operated switches. These switches can operate for low, high or incorrect differential pressures. A differential switch or gauge is subjected to pressure on both sides of its sensor.

Coloured arcs of green, amber or red are used to indicate the range and limits of the system.

### Engine Vibration

Vibration monitoring equipment (VME) is fitted to almost all commercial jet engine aircraft. Although gas turbine engines have an extremely low vibration level, any change in that level is usually indicative of damage which may lead to failure.

Warnings will be given in the cockpit if the vibration levels are exceeded and some systems have a continuous read-out of vibration levels.

The latest engines have the facility whereby the vibration level of each rotating assembly is monitored so that the source of the vibration can be pinpointed.

The principle upon which VME works requires either an input from a **Piezoelectric** crystal mounted strategically on the engine, or an input from a coil which will be affected by the movement of a **Magnet** mounted loosely within it. In either case, the frequency of the incoming vibrations will be filtered so that only those frequencies that are indicative of damage occurring will affect the output.

These systems utilize the principle that the magnet and piezo crystal, which are suspended within a fixed coil carrying 115 volts at 400 hertz, will move in sympathy with any vibration suffered by the engine. This will affect the current flowing through the coil into the amplifier and filter. The filter will erase any output which is normal to the engine and allow through to the amplifier any frequency that is considered to be harmful to the engine.

The result of this amplification is sent to the instrument via the rectifier and warning circuit. The needle will show the appropriate deflection for the amount of vibration being suffered by the engine at that time. If the level of vibration exceeds a predetermined amount, a warning light on the instrument illuminates. Vibration is measured and displayed in '**Relative Amplitude**' (Rel Ampl).



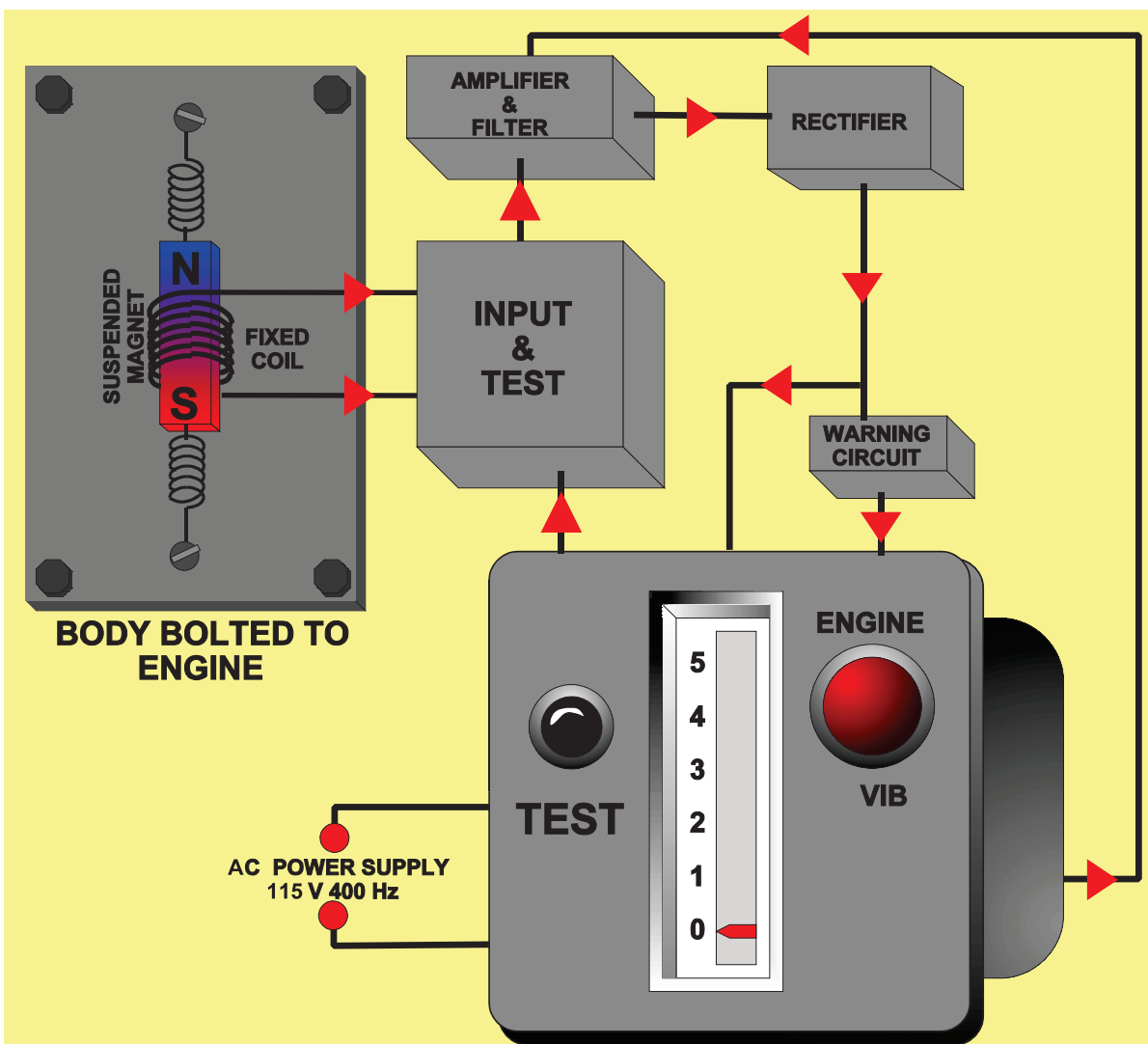


Figure 38.18 The circuitry of a vibration monitoring system

## Fuel Gauge

The measurement of the quantity of fuel in the tanks of an aircraft fuel system is an essential requirement, and in conjunction with measurements of the rate at which the fuel flows to the engine or engines it permits an aircraft to be flown at maximum efficiency.

There are two principal methods of indicating the quantity of fuel carried. Either the **Volume** (e.g. gallons) or **Mass** (kg or lb) are measured. The former is now only used on light aircraft as the **Mass** of the fuel is of more interest to the pilot. This assists the pilot in calculating the aircraft's 'all-up-weight' and also gives a better indication of the energy that can be released by the fuel. One pound of fuel has the same number of energy molecules regardless of temperature and volume.

The simplest form of volume indication is a float system. Early aircraft had a float which sat on the level of fuel. Attached to the float was a piece of wire that protruded out the top of the fuel tank. As the fuel level reduced so the wire disappeared from view. There have been many variations of this system. The most common of these is where the float moves to reposition a wiper on a variable resistor which alters the current to an indicator moving a pointer over a scale calibrated in volume. This is a DC powered system. (Figure 38.19)

The disadvantage of this system is that fuel tanks are rarely a symmetrical shape, therefore the float level is not a true measure of quantity. The gauge is set to be accurate at the low and empty positions. The system is also subject to errors whenever the aircraft manoeuvres and the attitude changes.

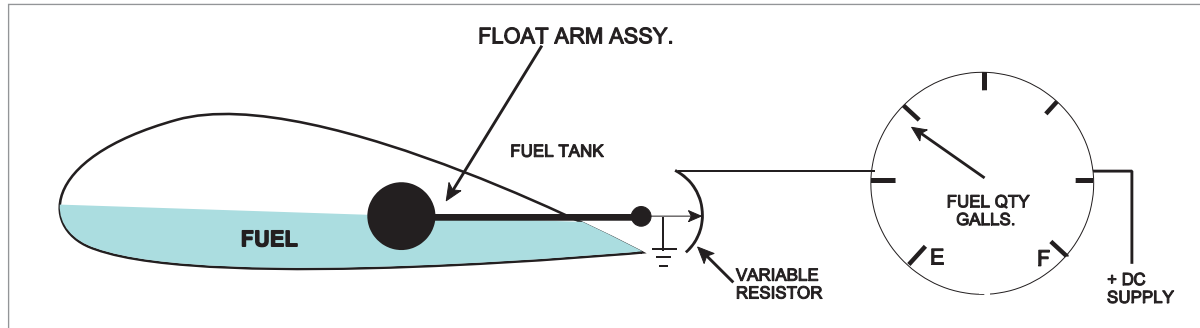


Figure 38.19 DC float type system

### Capacitance Type Fuel Gauge System

In its basic form a capacitance system (Figure 38.20) consists of a variable AC capacitor located in the fuel tank (Tank Unit Figure 38.21), an amplifier and an indicator. This system will indicate volume without the errors of the float system. If a correction voltage due to change in volume or temperature change is fed to the circuit, Mass of fuel will be indicated. A tank unit consists of two concentric aluminium alloy tubes which are held apart by pairs of insulating pins. The electrical connections are insulated and the unit is insulated from the tank. Co-axial connectors are used throughout.

Incorporated in the system are **Reference units**, which improve indication errors that would occur if the permittivity of the fuel changes from its normal value. The reference unit is located on the lower end of a tank unit and is always totally submerged in the unusable fuel level in the tank.

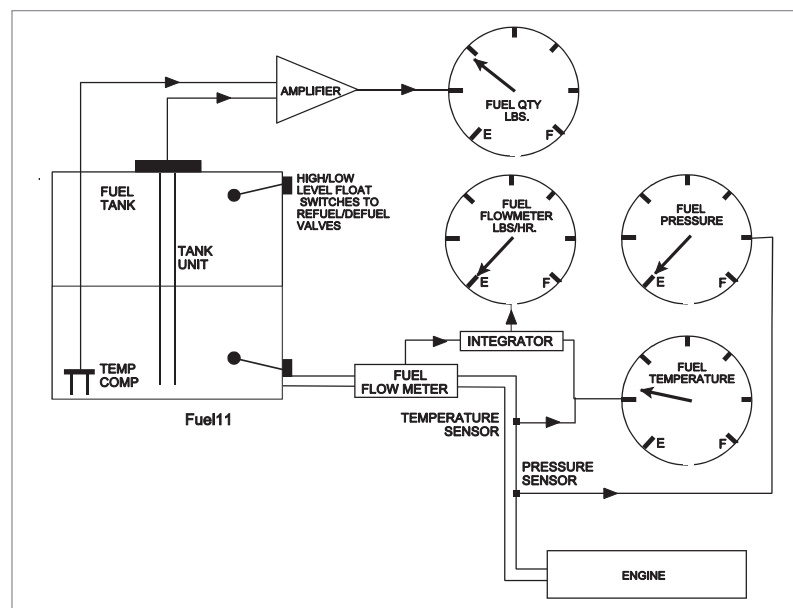


Figure 38.20

The principle of operation of the capacitance system is based on the use of fuel and air as the dielectric between parallel-plate capacitors of fixed area and a fixed distance between the plates. The only variable then being the ratio of fuel and air, which is determined by the quantity of fuel in the tank. Capacitance is measured in farads, the standard unit being the picofarad ( $10^{-12}$  farads). The capacitance depends on the following.

$$\text{Capacitance} = \text{Relative Permittivity} \times \frac{\text{Area of plates}}{\text{Distance between plates}} \quad \text{or} \quad C = E_r \times \frac{A}{D}$$

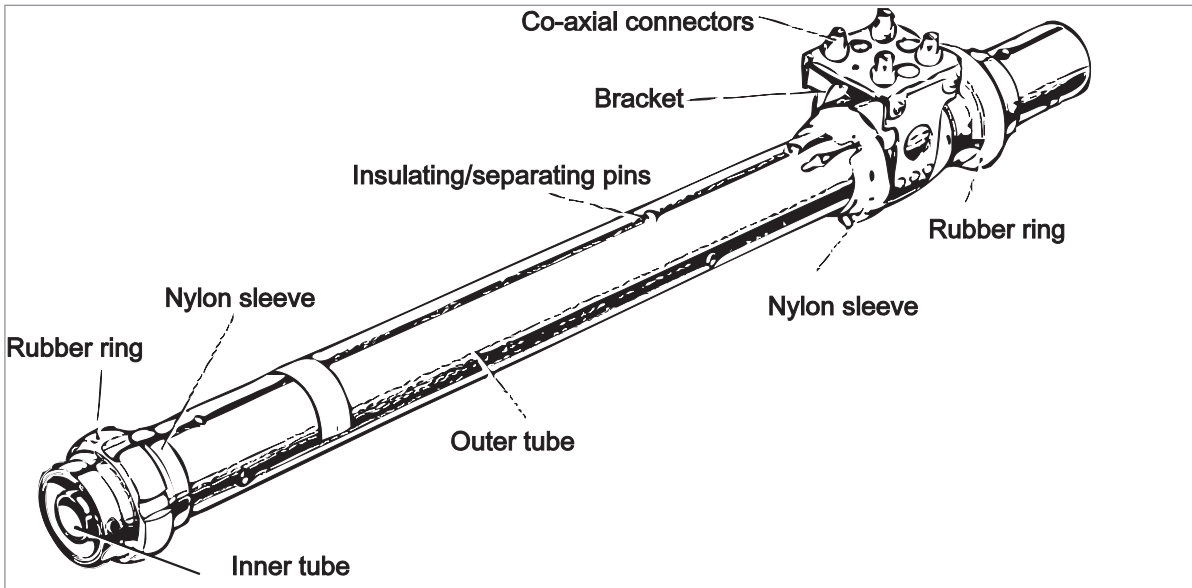


Figure 38.21 Capacitance tank unit

The **Relative Permittivity (Er)** is a number given as a ratio of the capacitance of a capacitor having a certain material as a dielectric to the capacitance of the same capacitor with a vacuum (or air) as its dielectric.

In an aircraft fuel system, the area of the plates and their distance apart remain constant, but the capacitance of the tank units will vary depending upon the level of fuel within the tanks.

The value of capacitance of a tank can be considered as two components. Ca (air) and Cf (fuel) and at any instance Tank Capacitance (Ct) = Ca + Cf. (Figure 38.22).

Typical Dielectric Values	
Material	Relative Permittivity
Impure water	0
Vacuum	1.0
Air	1.0006
Gasolene	1.95
Kerosene	2.10
Distilled Water	81.00

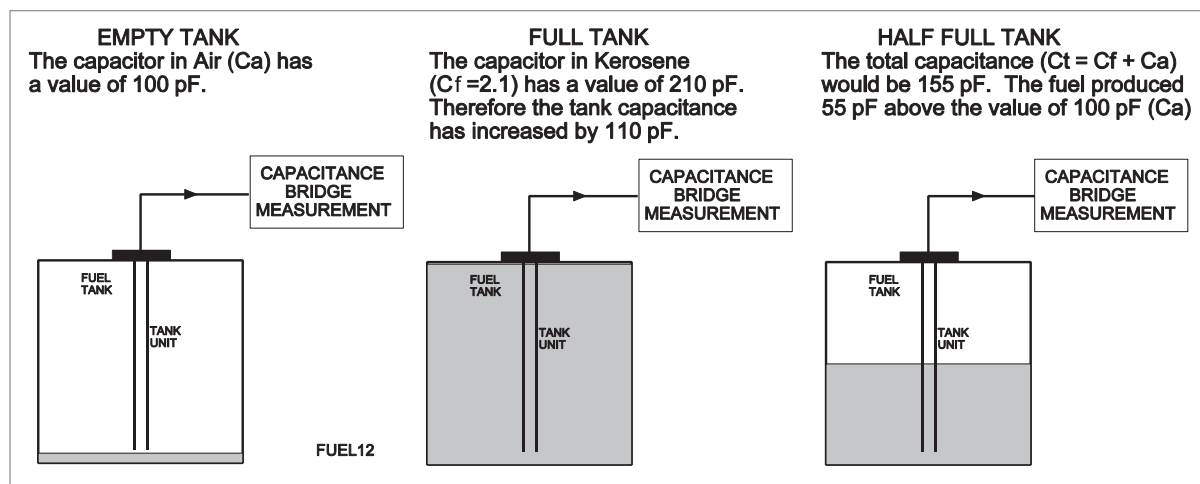


Figure 38.22

The pointer in the fuel quantity system measuring fuel by volume is directly related to:

- The change in fuel level.
- the ER- 1 value.

In an AC capacitance circuit the current is equal to the voltage over the capacitive reactance.

$$I \text{ (Current)} = \frac{V \text{ (Voltage)}}{X_c \text{ (Capacitive Reactance)}}$$

The capacitive reactance  $X_c$  is equal to  $1/(2\pi f c)$ . Since the voltage, frequency and  $2\pi$  are constants, as the fuel level and capacitance change current in the circuit changes.

Changes in temperature of the fuel will affect its **density, volume and dielectric value (Er)**. A decrease in temperature would cause a decrease in volume, increase in density and increase in Er. The circuit is compensated for changes in temperature and can now indicate **mass** of fuel which is of more value to the pilot.

For the calibration of gauges in terms of mass, an assumption is made that there is a constant relationship **between Er value and density ( $\rho$ )** for a given sample of fuel at a given temperature. Temperature of course is not a constant and a **Compensating Capacitor** circuit is incorporated in the system. This is fitted to the reference unit. The system will now sense changes in **Specific Gravity (SG)** of the fuel and so indicate mass.

The indicating system can incorporate an additional indicator known as the '**Fuel Totalizer**' which will indicate the sum of all the tank gauges. In the event of failure, the system will fail safe and drive the indicator slowly to the zero position. A test circuit is incorporated that when selected will simulate the emptying of the tank. When the switch is released the pointer should return to its original position.

If water is present in the tanks it will cause errors with the indicating system. The capacitors in the sensing units are effectively shorted, and the indicator is driven beyond the full scale.

If the unusable fuel supply for any tank exceeds one gallon, or 5% of the tank capacity, whichever is greater, a **red arc** must be marked on its indicator extending from the calibrated zero reading to the lowest reading obtainable in flight.

As well as the quantity of fuel measured, the rate of fuel consumed and the instantaneous rate of fuel flow can be shown. The **Fuel Flowmeter** can display volume flow or mass flow. Flow is proportional to the square root of pressure drop across an orifice. A simple flowmeter can be an adaption of a pressure gauge. This is used on many light piston engine injection systems. Most modern engines use an electrical sensor, which utilizes the change in torque or speed of a turbine (impeller). Typical construction consists of a light alloy casting with guide vanes and an electrical 'pick-off' coil. Inside the casting there is a helical vane impeller which has a magnet embedded in it. When the impeller rotates due to fuel flow, the pick-off coils will have a sinusoidal signal induced in it, at a frequency proportional to the speed of the rotor, which is proportional to the rate of volume flow. To measure mass flow the signal is corrected for temperature. (Figure 38.23)

The total consumption is obtained by integrating the rate of fuel consumption over time, this time is one hour. Units used for volume flow are gallons/hour or litres/hour, and for mass flow pounds/hour or kilograms/hour. A flowmeter that displays fuel consumed as well as fuel flow is broadly defined as an **Integrated** flowmeter. The flowmeter is located in the high pressure fuel line to the fuel spray nozzles (burners).

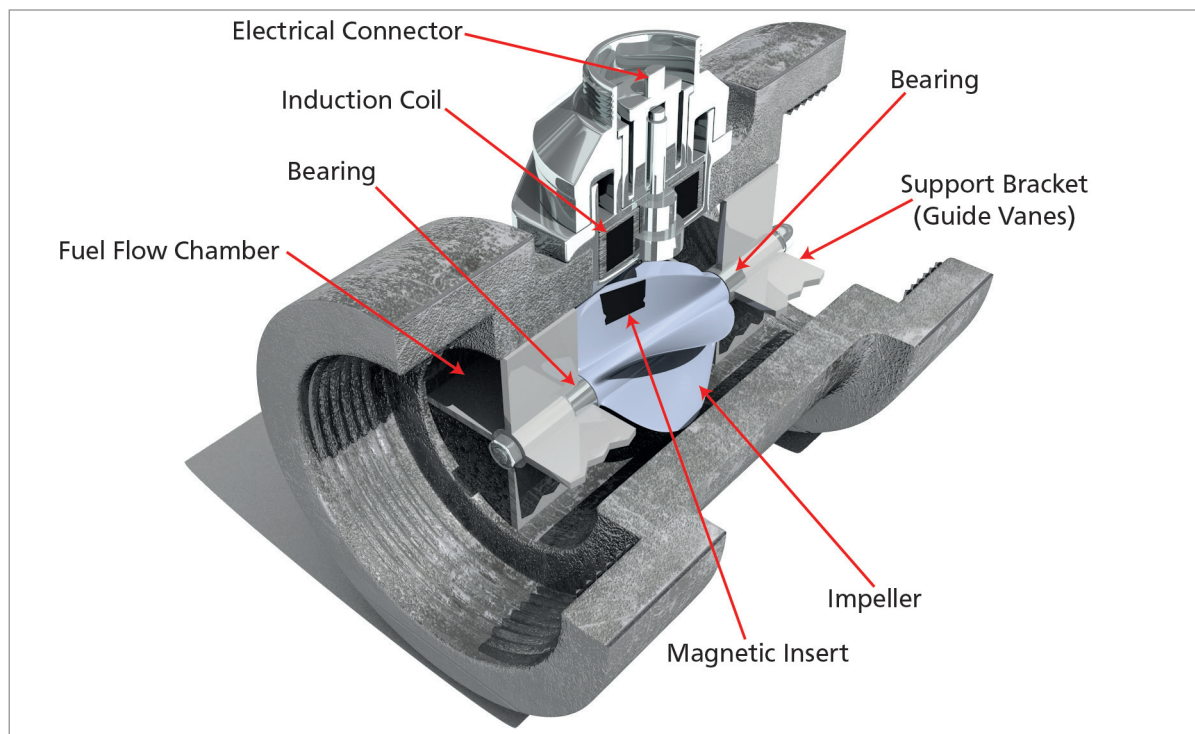


Figure 38.23 A typical fuel flowmeter

## Remote (Signal) Transmission System

To control an aircraft system may require the movement of a valve, flap or lever on the engine. The pilot may need to know the position of the control. Early systems had **Mechanical Feedback** to a position indicator in the flight deck. Most of the aircraft flying today employ remote indicating systems that can be either DC or AC operated. Whichever system is used each data transmission system employs a **Transmitter** located at the source to be measured and a **Receiver**, which acts on the information received.

### Flight Hour Meter

Some aircraft have a time monitor to record usage of the engines and systems in flight. This can be automatically switched via the “weight-on-wheels” switch or, more commonly, by an airspeed switch.

# Chapter 39

## Electronic Instrumentation

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## Electronic Instruments for Engine and Airframe Systems Control

The display of the parameters associated with engine performance and airframe systems control by means of Cathode Ray Tube (CRT) or LCD type display has, like those of flight instrument systems, become a standard feature of many types of aircraft.

The display units form part of two principal systems designated as Engine Indicating and Crew Alerting System (**EICAS**), and Electronic Centralized Aircraft Monitoring (**ECAM**) system. These systems were first introduced in Boeing 757 and 767 aircraft and the Airbus A310 respectively.

At the time of their introduction there were differing views on the approach to such operating factors as flight deck layouts and crews' controlling functions, the extent to which normal, alerting and warning information should be displayed, and in particular, whether engine operating data was required to be displayed for the whole of a flight, or only at various phases.

In respect of **EICAS**, engine operating data is displayed on screens, thereby eliminating the need for traditional instruments.

This data, as well as that relevant to other systems, is not necessarily always on display but in the event of malfunctions occurring at any time, the flight crew's attention is drawn to them by an automatic display of messages in the appropriate colours.

The **ECAM** system, on the other hand, displays systems' operation in checklist and schematic form, and as this was a concept based on the view that engine data needed to be displayed during the whole of a flight, traditional instruments were retained in the Airbus A310.

It is of interest to note, however, that in subsequent types produced by this manufacturer, e.g. A320, the ECAM system is developed to include the display of engine data in one of its display units.

Smaller aircraft with electronic flight decks, where space is limited, may have more compact displays, rather than the 6 screen layout common in most Airbus and Boeing aeroplanes. These may include EFIS for flight instruments, and a separate Engine and Warning Display (EWD) or electronic flight Instruments combined with engine indications on a combined Multi-function Display Unit (MFDU).

### EICAS

The basic **EICAS** system comprises two display units, a control panel, and two computers supplied with analog and digital signals from engine and system sensors as shown in the schematic functional diagram of [Figure 39.1](#).

The computers are designated 'Left' and 'Right', and only one is in control at a time; the other is on 'standby'. In the event of failure it may be switched in either manually or automatically. Operating in conjunction with the EICAS system are discrete caution and warning lights, standby engine indicators and a remotely-located panel for selecting maintenance data displays.

The system provides the flight crew with information on primary engine parameters (full-time), with secondary engine parameters and warning, caution and advisory alert messages displayed as required.

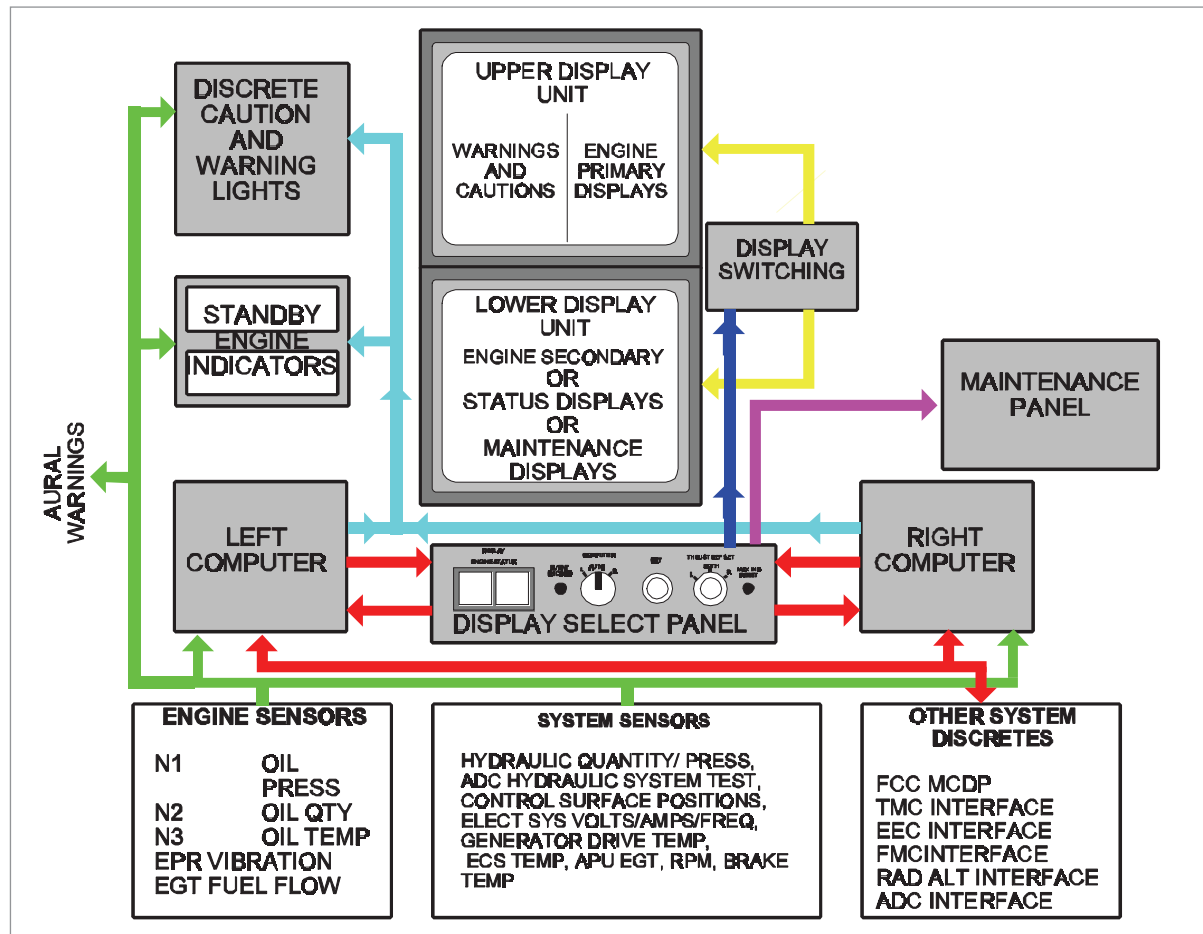


Figure 39.1 The EICAS functional diagram

## Display Units

The display units provide a wide variety of information relevant to engine operation, and operation of other automated systems, and they utilize colour shadow mask CRTs and associated card modules whose functions are identical to those of the EFIS units. The units are mounted one above the other as shown in [Figure 39.2](#).

The **upper unit** displays the primary engine parameters, N1 speed, EGT, and warning and caution messages.

In some cases this unit can also display EPR, depending on the type of engines installed and on the methods used to process data by the thrust management control system.

The **lower unit** displays secondary engine parameters, i.e. N2 speed, fuel flow, oil quantity, pressure and temperature, and engine vibration. In addition, the status of non-engine systems, e.g. flight control surface positions, hydraulic system, APU, etc., can also be displayed together with aircraft configuration and maintenance data.

The rows of 'V's shown on the upper display unit only appear when secondary information is being displayed on the lower unit.

Seven colours are produced by the CRTs and they are used as follows:

White	All scales, normal operating range of pointers, digital read-outs.
Red	Warning messages, maximum operating limit marks on scales, and digital read-outs.
Green	Thrust mode read-out and selected EPR / N1 speed marks or target cursors.
Blue	Testing of system only.
Yellow	Caution and advisory messages, caution limit marks on scales, digital read-outs.
Magenta	During in-flight engine starting, and for cross-bleed messages.
Cyan	Names of all parameters being measured (e.g. N1 oil pressure, TAT etc. and status marks or cues).

The displays are selected according to an appropriate display selection mode.

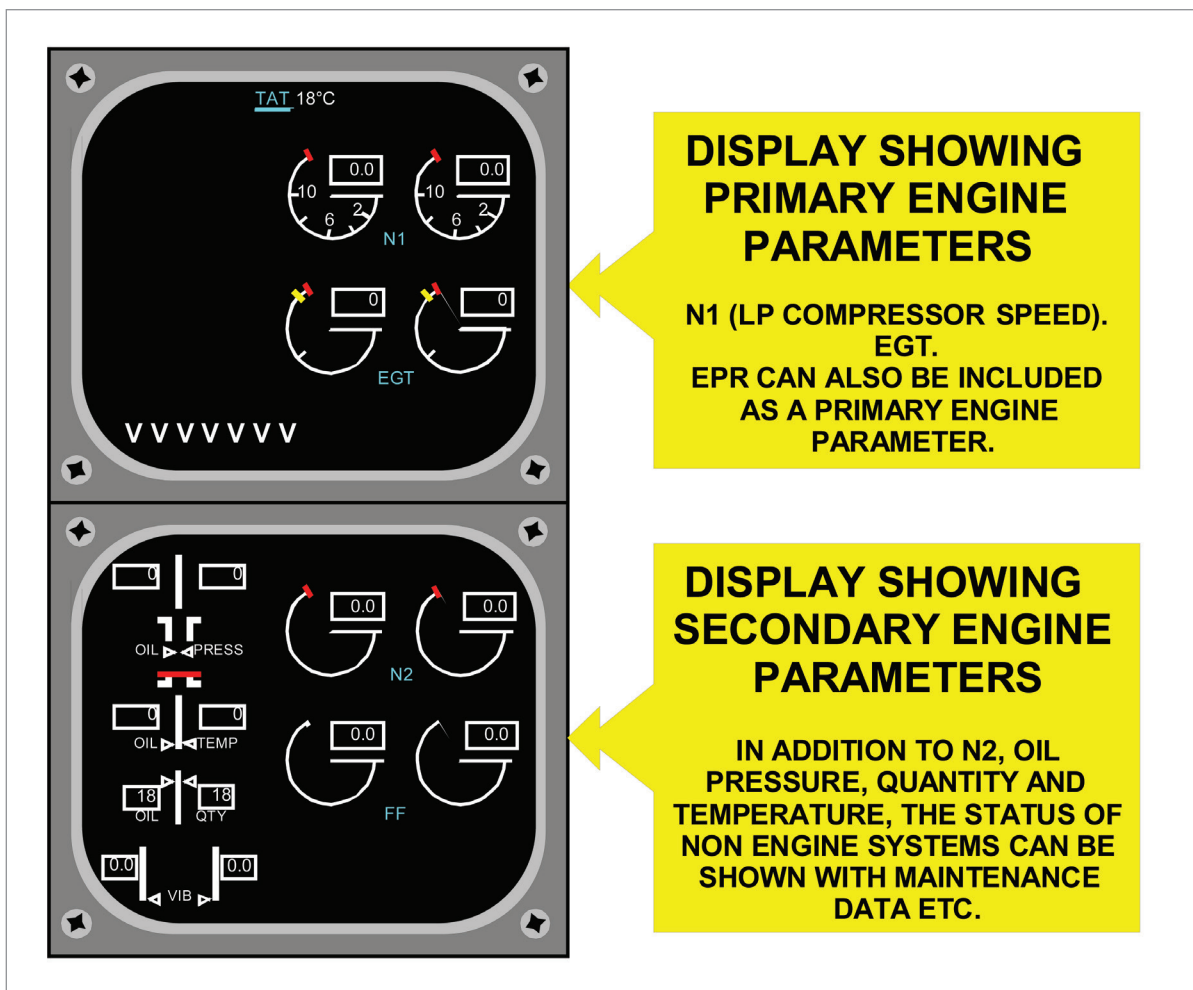


Figure 39.2 EICAS engine data displays

## Display Modes

EICAS is designed to categorize displays and alerts according to function and usage, and for this purpose there are three modes of displaying information: (i) **operational**, (ii) **status** and (iii) **maintenance**.

Modes (i) and (ii) are selected by the flight crew on the display select panel, while mode (iii) is selected on the maintenance panel which is for the use of ground engineering staff only.

## Operational Mode

The **operational mode** displays the engine operating information and any alerts required to be actioned by the crew in flight. Normally only the upper display unit presents information, the lower one remains blank and can be selected to display secondary information as and when required.

## Status Mode

When selected, the **status mode** displays data to determine the dispatch readiness of an aircraft, and is closely associated with details contained in an aircraft's Minimum Equipment List.

The display shows positions of the flight control surfaces in the form of pointers registered against vertical scales, selected sub-system parameters, and equipment status messages on the lower display unit. Selection is normally done on the ground either as part of preflight checks of dispatch items, or prior to shut-down of electrical power to aid the flight crew in making entries in the aircraft's Technical Log.

## Maintenance Mode

This mode provides maintenance engineers with information in five different display formats to aid them in trouble-shooting and verification testing of the major sub-systems. The displays, which are presented on the lower display unit, are not available in flight.

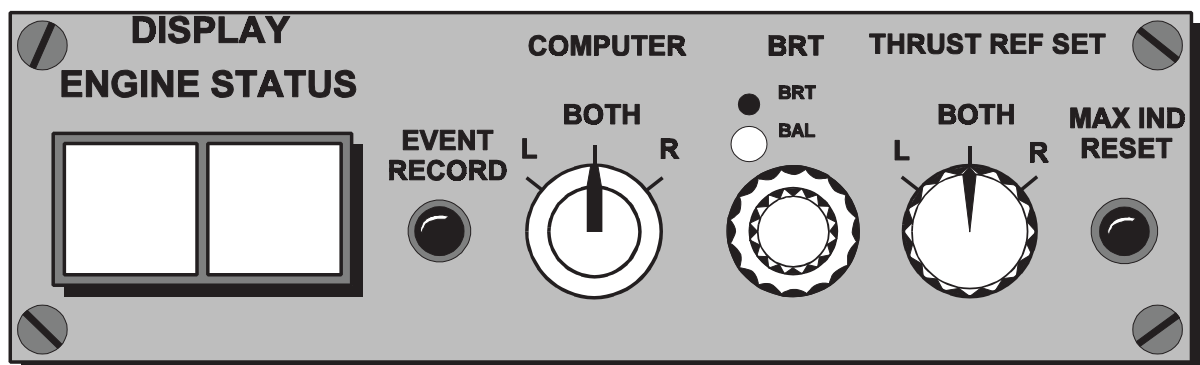


Figure 39.3 EICAS display select panel

## Display Select Panel

This panel, as indicated in [Figure 39.3](#), permits control of EICAS functions and displays and can be used both in flight and on the ground. It is normally located on the centre pedestal of an aircraft's flight deck, and its controls are as follows:

- **Engine Display Switch.** This is of the momentary-push type for removing or presenting the display of secondary information on the lower display unit.
- **Status Display Switch.** Also of the momentary-push type, this is used to display the status mode information referred to earlier, on the lower display unit. The display is known as a 'status page', an example of which is shown in [Figure 39.4](#).
- **Event Record Switch.** This is of the momentary-push type and is used in the air or on the ground, to activate the recording of fault data relevant to the environment control system, electrical power, hydraulic system, performance and APU. Normally if any malfunction occurs in a system, it is recorded automatically (called an 'auto event') and stored in a non-volatile memory of the EICAS computer. The push-switch also enables the flight crew to record a suspected malfunction for storage, and this is called a 'manual event'. The relevant data can only be retrieved from memory and displayed when the aircraft is on the ground and by operating switches on the maintenance control panel.

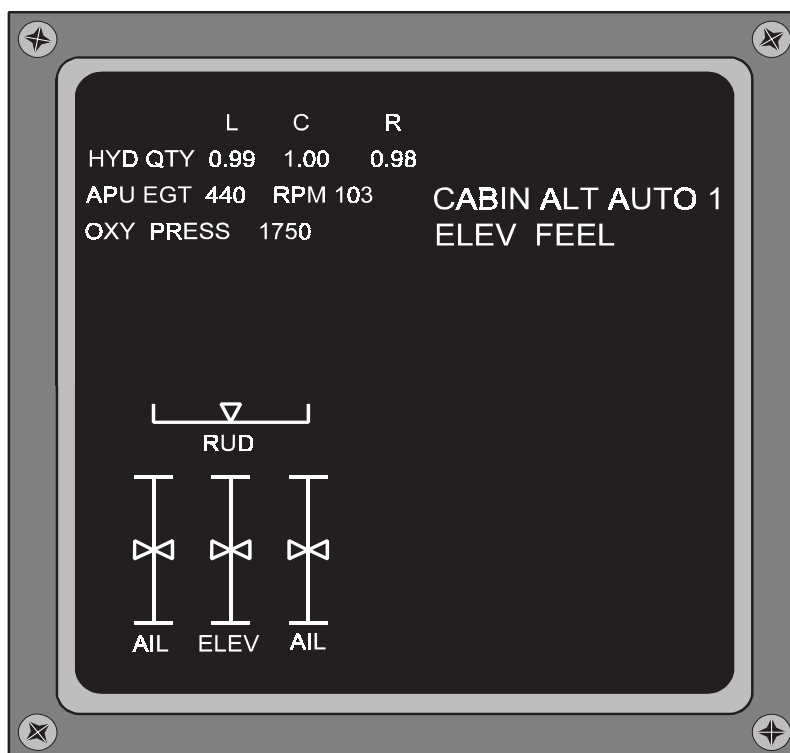


Figure 39.4 Status Mode Display

- **Computer Select Switch.** In the 'AUTO' position it selects the left, or primary, computer and automatically switches to the other computer in the event of failure. The other positions are for the manual selection of left or right computers.
- **Displays Brightness Control.** The inner knob controls the intensity of the displays, and the outer knob controls brightness balance between displays.
- **Thrust Reference Set Switch.** Pulling and rotating the inner knob positions the reference cursor on the thrust indicator display (either EPR or N1) for the engine(s) selected by the outer knob.
- **Maximum Indicator Reset Switch.** If any one of the measured parameters e.g. oil pressure, EGT, should exceed normal operating limits, this will be automatically alerted on the display units. The purpose of the reset switch is to clear the alerts from the display when the excess limits no longer exist.

## Alert Messages

The system continuously monitors a large number of inputs (typically over 400) from engine and airframe systems' sensors and will detect any malfunctioning of systems.

If this should occur, then appropriate messages are generated and displayed on the upper display unit in a sequence corresponding to the level of urgency of action to be taken.

Up to 11 messages can be displayed, and at the following levels:

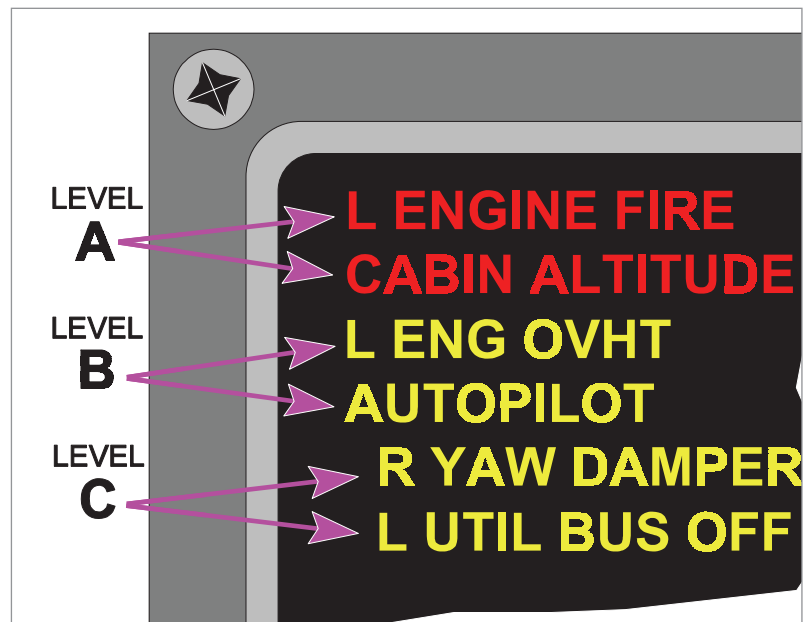


Figure 39.5 Alert message levels

**Level A** - Warning requiring immediate corrective action. They are displayed in red. Master warning lights are also illuminated, and aural warnings (e.g. fire bell) from a central warning system are given.

**Level B** - Cautions require immediate crew awareness and future crew action. They are displayed in amber, and also by message caution lights. An aural tone is also repeated twice.

**Level C** - Advisories requiring crew awareness. Also displayed in amber. No caution lights or aural tones are associated with this level.

The messages appear on the top line at the left of the display screen as shown in [Figure 39.5](#).

In order to differentiate between a caution and an advisory, the latter is always indented one space to the right.

The master warning and caution lights are located adjacent to the display units together with a 'CANCEL' switch and a 'RECALL' switch. Pushing the 'CANCEL' switch removes only the caution and advisory messages from the display; the warning messages cannot be cancelled. The 'RECALL' switch is used to bring back the caution and advisory messages into the display. At the same time, the word '**RECALL**' appears at the bottom of the display.

A message is automatically removed from the display when the associated condition no longer exists. In this case, messages which appear below the deleted one each move up a line.

When a new fault occurs, its associated message is inserted on the appropriate line of the display. This may cause older messages to move down one line. For example, a new caution message would cause all existing caution and advisory messages to move down one line.

If there are more messages than can be displayed at one time, the whole list forms what is termed a 'page', and the lowest message is removed and a page number appears in white on the lower right side of the list.

If there is an additional page of messages it can be displayed by pushing the 'CANCEL' switch. Warning messages are carried over from the previous page.

## Display Unit Failure

If the lower display unit should fail when secondary information is being displayed on it, an amber alert message appears at the top left of the upper display unit, and the information is transferred to it as shown in [Figure 39.6](#).

The format of this display is referred to as 'compact', and it may be removed by pressing the 'ENGINE' switch on the display select panel. Failure of a display unit causes the function of the panel 'STATUS' switch to be inhibited so that the status page format cannot be displayed.

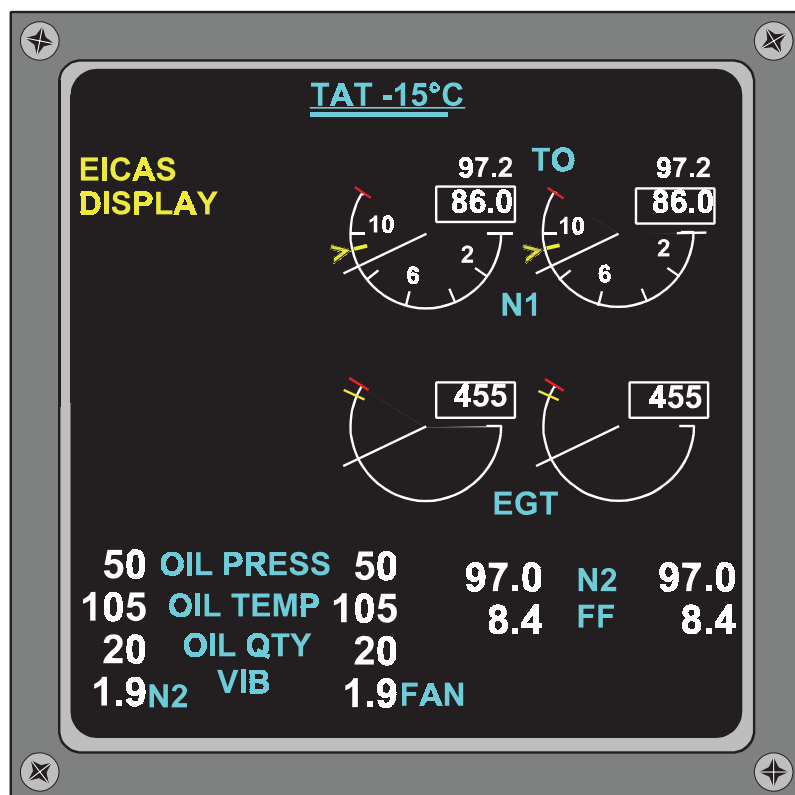


Figure 39.6 The 'compact format' display

### Standby Engine Indicator

This indicator provides primary engine information in the event that a total loss of EICAS displays occurs.

As shown in *Figure 39.7*, the information relates to N1 and N2 speeds and EGT and the displays are of the LCD type. Operating limit values are also displayed.

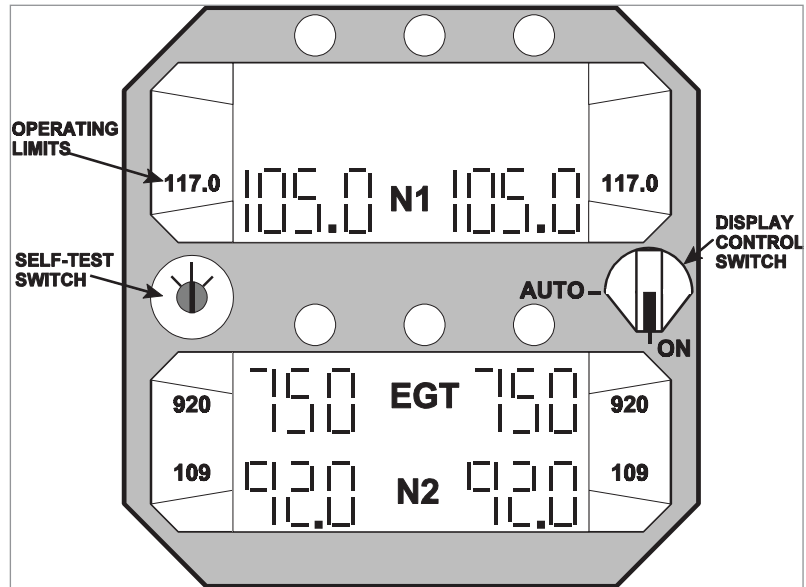


Figure 39.7 The standby engine indicator

### Electronic Centralized Aircraft Monitoring (ECAM)

The units comprising this system, and as originally developed for the Airbus A310, are shown in the functional diagram of *Figure 39.8*.

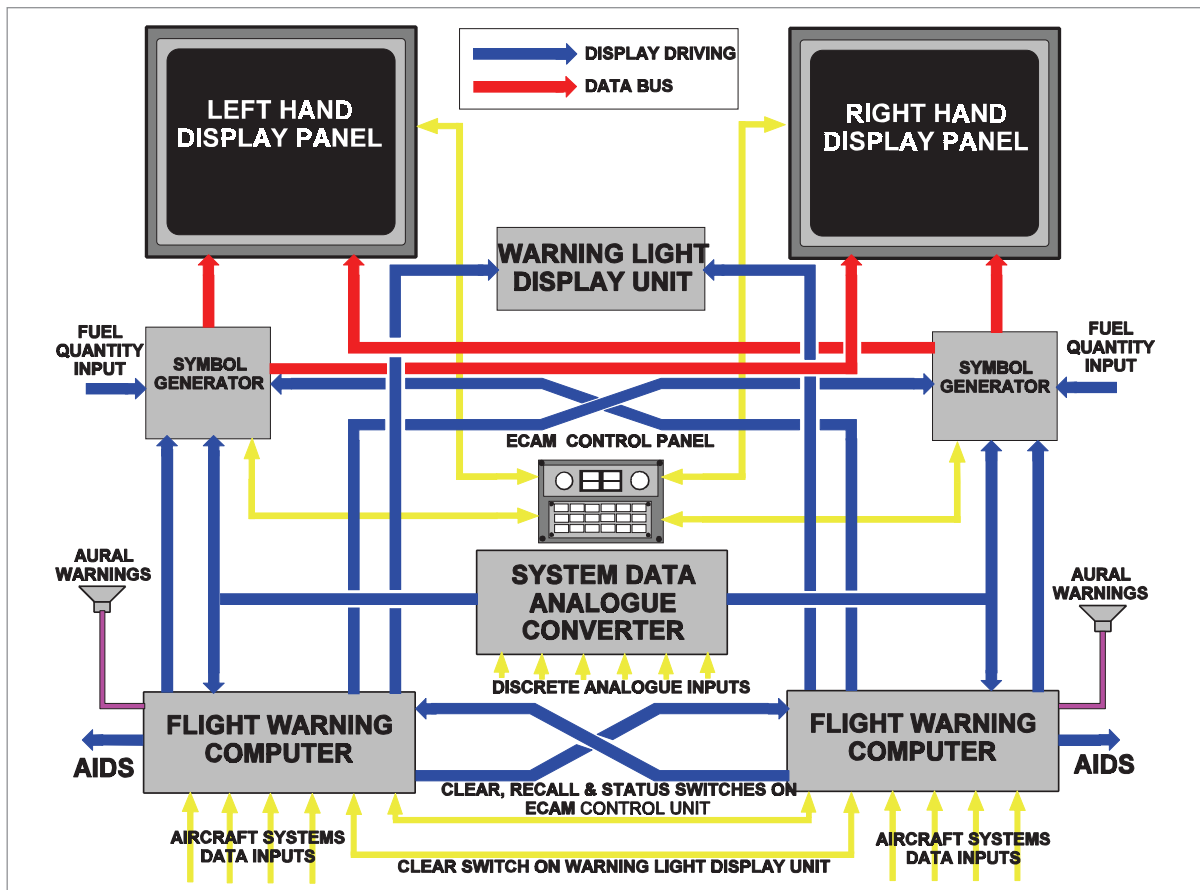


Figure 39.8 ECAM system functional diagram



As far as the processing and display of information are concerned, the ECAM system differs significantly from EICAS in that data relates essentially to the primary systems of the aircraft, and is displayed in check-list and pictorial or synoptic format.

Engine operating data is displayed by conventional types of instruments as noted in the introduction to this chapter. Other differences relate to display locations and selection of system operating modes.

## Display Units

These units may be mounted side-by-side; the left-hand unit is dedicated to information on the status of systems, warnings and corrective action in a sequenced check-list format, while the right-hand unit is dedicated to associated information in pictorial or synoptic format.

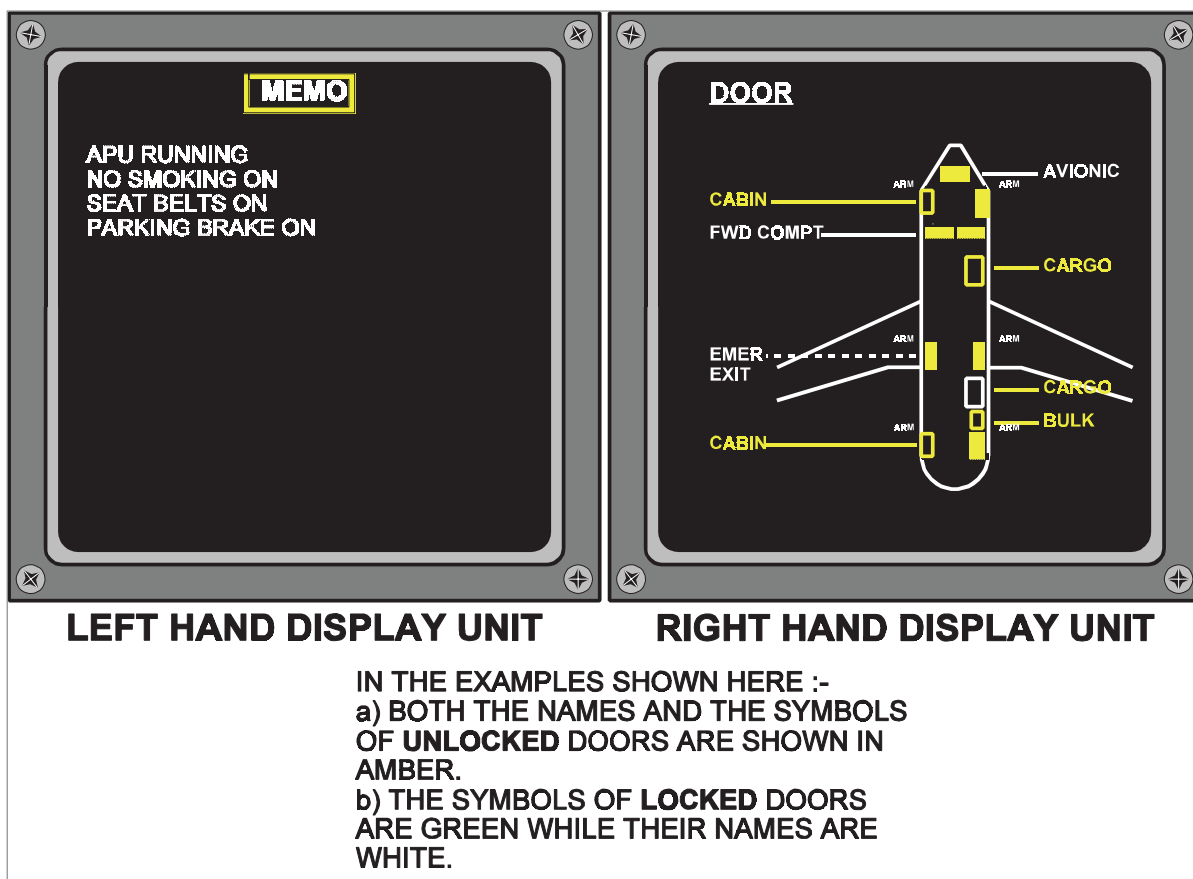


Figure 39.9 Preflight phase-related mode display

## Display Modes

There are four display modes, three of which are automatically selected and referred to as:

- Flight Phase-related
- Advisory (mode and status)
- Failure-related modes

The fourth mode is manual

- Aircraft System Display

## The Flight Phase-related Mode

In normal operation the automatic 'flight phase-related mode' is used, and in this case the displays are appropriate to the current phase of aircraft operation, i.e. preflight, take-off, climb, cruise, descent, approach, and after landing.

An example of a preflight phase is shown in *Figure 39.9*, the left-hand display unit displays an advisory memo mode, and the right-hand unit displays a diagram of the aircraft's fuselage, doors, and arming of the escape slides deployment system.

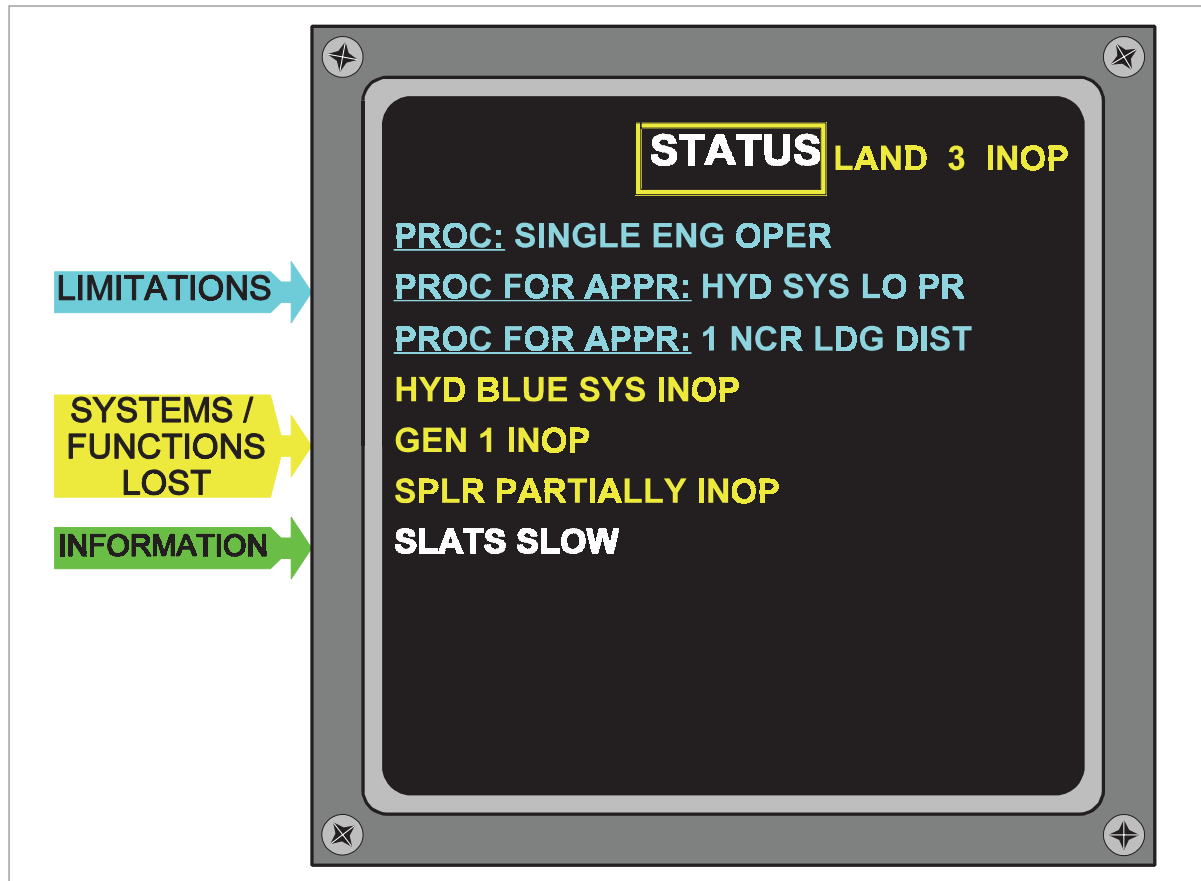


Figure 39.10 Example of the status display

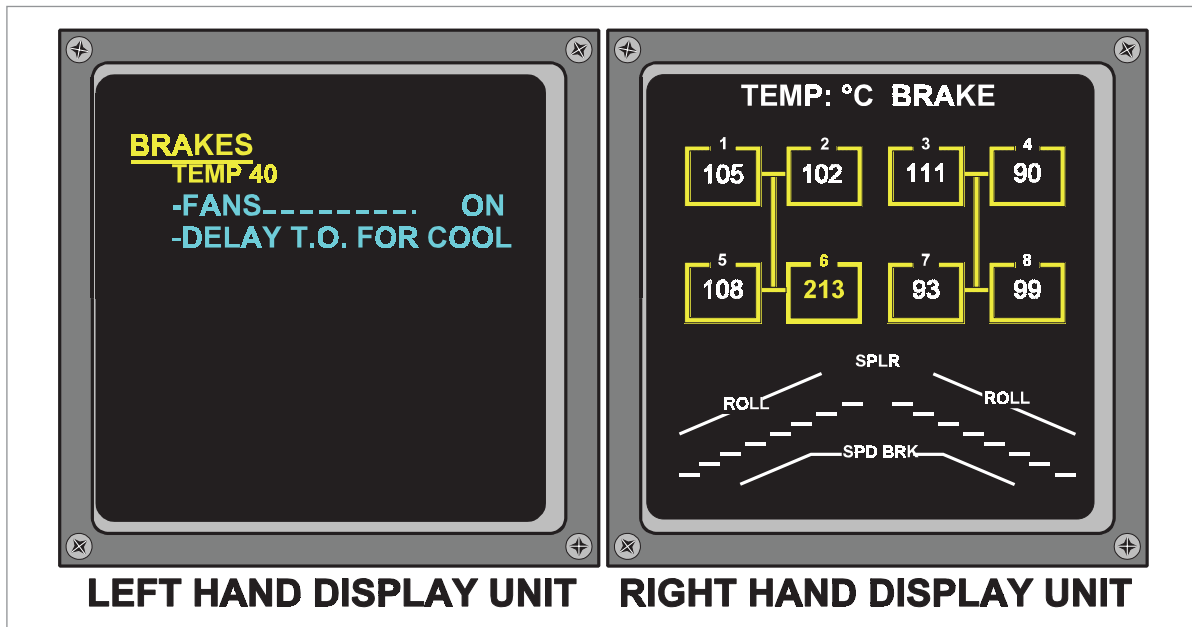
## Advisory (Mode and Status)

Status messages, which are also displayed on the left-hand display unit, provide the flight crew with an operational summary of the aircraft's condition, possible downgrading of autoland capability, and as far as possible, indications of the aircraft status following all failures except those that do not affect the flight. The contents of an example display are shown in *Figure 39.10*.

## Failure-related Mode

The failure-related mode takes precedence over the other two automatic modes and the manual mode.

An example of a display associated with this mode is shown in [Figure 39.11](#).



*Figure 39.11 The failure-related mode display*

In this case, while taxiing out for take-off, the temperature of the brake unit on the rear right wheel of the left main landing gear bogie has become excessive.

A diagram of the wheel brake system is immediately displayed on the right-hand display unit, and simultaneously the left-hand unit displays corrective action to be taken by the flight crew. In addition, an aural warning is sounded, and a light (placarded 'L/G WHEEL') on a central warning light display panel is illuminated.

As the corrective action is carried out, the instructions on the left-hand display are replaced by a message in white confirming the result of the action. The diagram on the right-hand display unit is appropriately 'redrawn'. In the example above, the 'failure related mode' displaces warning relates to a single system, and by convention such warnings are signified by underlining the system title displayed.

In cases where a failure can affect other sub-systems, the title of the sub-system is shown 'boxed', as for instance in the display shown in [Figure 39.12](#).

Warnings and the associated lights are cleared by means of 'CLEAR' push-button switches on either the ECAM control panel or a warning light display panel.

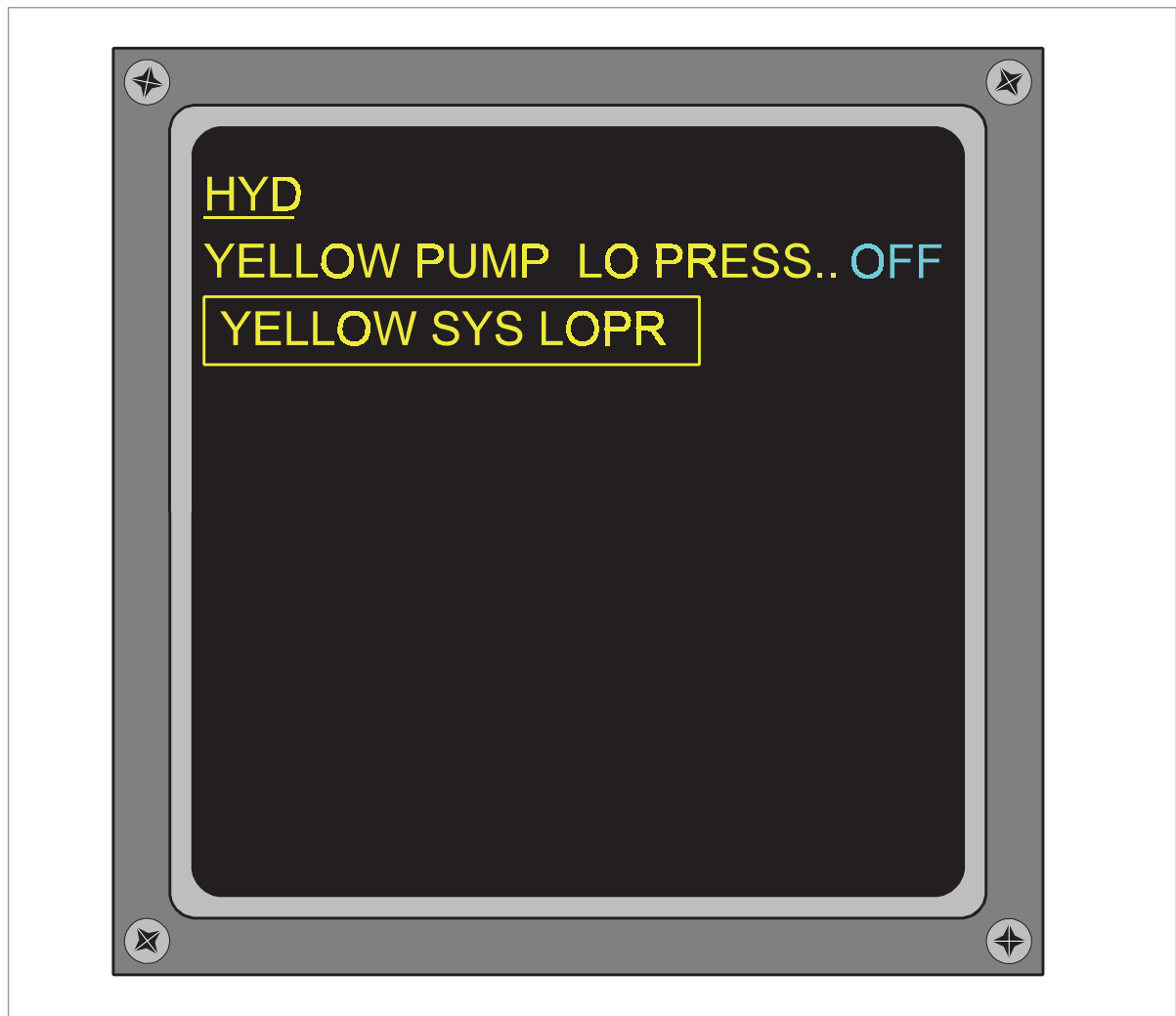


Figure 39.12 A display showing how a failure affects a sub-system

## The Fourth Mode (Manual), the Aircraft System Display Mode

This mode permits the selection of diagrams related to any one of 12 of the aircraft's systems for routine checking, and also the selection of status messages provided no warnings have been triggered for display. The selections are made by means of illuminated push-button switches on the system control panel.

### The 'ECAM' Control Panel

The layout of the 'ECAM' control panel is shown in *Figure 39.13*, and all switches, with the exception of those for display control, are of the push-button, illuminated caption type.

**SGU Selector Switches.** These control the respective symbol generator units, and the lights are off in normal operation of the system. The 'FAULT' caption is illuminated amber if a failure is detected by an SGU's internal self-test circuit. Releasing a switch isolates the corresponding SGU, and causes the 'FAULT' caption to extinguish, and the 'OFF' caption to illuminate white.

**Synoptic Display Switches.** These permit individual selection of synoptic diagrams corresponding to each of 12 systems and illuminate white when pressed. A display is automatically cancelled whenever a warning or advisory occurs.

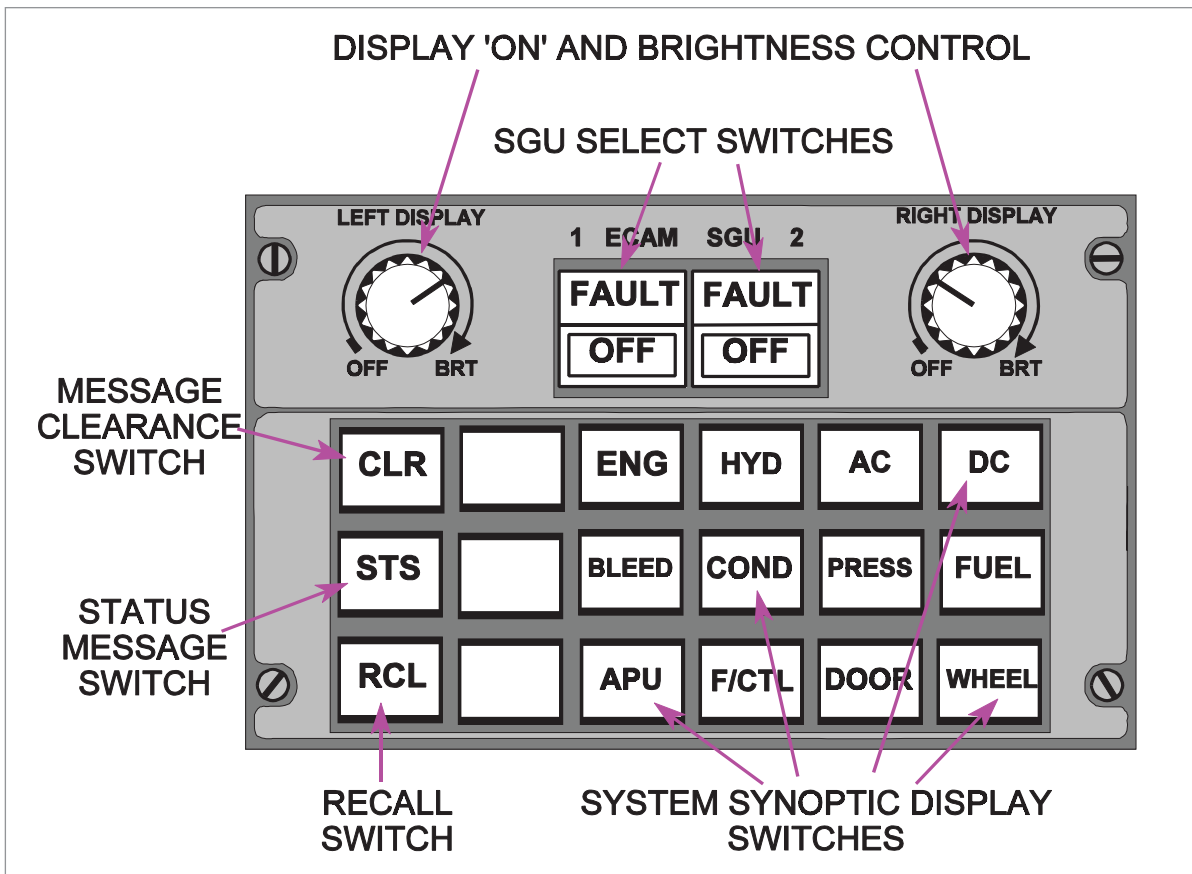


Figure 39.13 The ECAM control panel

**CLR Switch.** The light in the switch is illuminated white whenever a warning or status message is displayed on the left-hand display unit. The switch is pressed to clear messages.

**STS Switch.** The Status Switch permits manual selection of an aircraft status message if no warning is displayed; illuminated white. Pressing the switch also causes the 'CLR' switch to illuminate. A status message is suppressed if a warning occurs or if the 'CLR' switch is pressed.

**RCL Switch.** The Recall Switch enables previously cleared warning messages to be recalled provided the failure conditions which initiated them still exist. Pressing the switch also causes the CLR switch light to illuminate. If a failure no longer exists the message 'NO WARNING PRESENT' is displayed on the left-hand display unit.

## Questions

1. **With an Engine Indicating and Crew Alerting System:**
  - a. the secondary display will show continuously the engine primary instruments
  - b. the primary display unit will continuously show the engine primary instruments such as N1, N2, N3 and maybe oil pressure
  - c. the primary engine display will continuously show the engine primary instruments such as N1, EGT and maybe EPR
  - d. the primary engine instruments are N1, EGT and EPR and are on the primary and secondary display units
  
2. **The electronic engine display system with three automatic modes is:**
  - a. the Electronic Centralized Aircraft Monitor, with the fourth mode manual
  - b. the Electronic Centralized Aircraft Monitor, with the fourth mode flight phase related or manual
  - c. the Engine Indicating and Crew Alerting System, with the fourth mode manual
  - d. the Engine Indicating and Crew Alerting System, with the fourth mode a manual cross-over from the Electronic Centralized Aircraft Monitor System
  
3. **The display modes for the Engine Indicating and Crew Alerting System are:**
  - a. operational, status and maintenance of which status and maintenance are automatic
  - b. flight phase related, advisory and failure related
  - c. operational, status and maintenance
  - d. operational, flight phase related and status
  
4. **With an Engine Indicating and Crew Alerting System lower display unit failure:**
  - a. a compact message will only appear on the upper display unit
  - b. a compact message will only appear on the central display unit
  - c. a compact message will appear both on the upper display unit and the captain's Electronic Flight Instrument System
  - d. a compact message will appear on the upper display unit when the status button is pressed on the control panel
  
5. **With an Electronic Centralized Aircraft Monitoring type of system:**
  - a. the display units have two control panels and with any system failure the control will be from the port control box only
  - b. the left display unit shows warning and corrective action in a check list format
  - c. the two display units are only fitted side by side
  - d. the left display unit shows the synoptic format and the right or lower unit shows the corrective format
  
6. **The Engine Indicating and Crew Alerting System alert messages are shown on the upper display unit in three forms:**
  - a. Level 'C' are warnings that require immediate corrective action
  - b. Level 'A' are cautions that require immediate crew awareness and possible action
  - c. Level 'B' are advisories requiring crew awareness
  - d. and these messages appear on the top left of the upper display unit

7. **The electronic engine display system will have:**
- one primary and one secondary display unit for an EICAS and a change over selector to change to the ECAM mode if necessary
  - two display units for ECAM and three display units for EICAS
  - either EICAS or ECAM but not both
  - an interconnect to the EFIS symbol generators in an emergency
8. **The Electronic Centralized Aircraft Monitor (ECAM) type of system shows a:**
- checklist format on the left display panel and schematic form always automatically on the right display unit
  - checklist format on the left display unit and the right, or lower display unit, a diagram or synoptic format
  - synoptic format on the left display unit and a warning and corrective action display on the right or lower display unit
  - continuous primary engine display on the primary display unit
9. **The electronic display system that has three automatic modes plus one manual is the:**
- Electronic Management and Control Section
  - Electronic Indication and Fail Safe system
  - Electronic Indication and Crew Alert system
  - Electronic Centralized Aircraft Monitor
10. **A boxed message shown as an electronic engine display system fault is one that:**
- affects other sub-systems and is used in the Engine Indicating and Crew Alerting System
  - does not affect any other system
  - does not affect any other system and is used in the Engine Indicating and Aircraft Monitor system
  - affects other sub-systems and is used in the Electronic Centralized Aircraft Monitor type of system
11. **An engine fire indication on an electronic engine display is shown:**
- on the primary display panel in red
  - on the secondary display panel in amber
  - on both the Electronic Flight Instrument System and Engine Indicating and Crew Alerting System secondary panels
  - only on the Flight Management Computer primary panel
12. **An engine electronic system which in normal conditions of flight shows only the primary engine instruments is:**
- an EICAS system with EPR, EGT and  $N_2$  shown on the primary instruments
  - an ECAM system with the primary engine instruments displayed on the lower screen
  - an EICAS system with the primary engine instruments displayed on the primary screen, the secondary screen being blank
  - an ECAM system with the primary engine instruments displayed on the primary screen, the left screen being blank

## Answers

1	2	3	4	5	6	7	8	9	10	11	12
c	a	c	a	b	d	c	b	d	d	a	c





# REVISION QUESTIONS

## ATPL GROUND TRAINING SERIES

10

10



# Chapter 40 Revision Questions

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## Flight Instruments Questions

1. **A 2 axis gyro measuring vertical changes will have:**
  - a one degree of freedom, vertical axis
  - b two degrees of freedom, vertical axis
  - c one degree of freedom, horizontal axis
  - d two degrees of freedom, horizontal axis
  
2. **The properties of a gyro are:**
  1. mass
  2. rigidity
  3. inertia
  4. precession
  5. rotational speed
  - a. 1, 2 & 3
  - b. 2 & 4
  - c. 2 & 3
  - d. 1 & 3
  
3. **An aircraft fitted with a DRMC upon landing in a northerly direction will indicate:**
  - a no change
  - b oscillation about north
  - c a turn towards east
  - d a turn towards west
  
4. **Which of the following will affect a direct reading compass?**
  1. ferrous metals
  2. non-ferrous metals
  3. electrical equipment
  - a. 1 only
  - b. 1 & 3
  - c. 1 & 2
  - d. all 3
  
5. **A vibrator may be fitted to an altimeter to overcome:**
  - a. friction
  - b. hysteresis
  - c. lag
  - d. pressure error
  
6. **An aircraft is flying at constant indicated altitude, over a warm airmass. The altimeter reading will be:**
  - a. correct
  - b. greater than the real altitude
  - c. less than the real altitude
  - d. oscillating around the correct altitude

7. **The Machmeter consists of:**
- an airspeed indicator with Mach scale
  - an airspeed indicator with an altimeter capsule
  - an altimeter corrected for density
  - a VSI and altimeter combined
8. **CAS is IAS corrected for:**
- position and instrument error
  - instrument, pressure and density error
  - relative density only
  - compressibility
9. **A DGI has:**
- one degree of freedom & a horizontal spin axis
  - two degrees of freedom & a vertical spin axis
  - two degrees of freedom & a horizontal spin axis
  - one degree of freedom & a vertical spin axis
10. **An aircraft is flying at an indicated altitude of 16 000 ft. The outside air temperature is  $-30^{\circ}\text{C}$ . What is the true altitude of the aircraft?**
- 16 200 ft
  - 15 200 ft
  - 18 600 ft
  - 13 500 ft
11. **The main cause of error in a DRMC is:**
- parallax in the rose
  - turning
  - magnetic deviation
  - latitude
12. **QNH is:**
- the airfield barometric pressure
  - the setting that will give zero indication on the airfield
  - the equivalent sea level pressure at the airfield
  - the setting that will indicate airfield height
13. **What is the Schuler period?**
- 21 minutes
  - 84 minutes
  - 1 oscillation in azimuth
  - 63 minutes

14. The vertical reference of a data generation unit is:
- horizontal axis with 1 degree of freedom
  - vertical axis with 1 degree of freedom
  - horizontal axis with 2 degree of freedom
  - vertical axis with 2 degree of freedom
15. The torque motor of a gyro stabilized magnetic compass:
- precesses the directional gyro
  - takes its input from the flux valve
  - moves the heading pointer
  - moves the Selsyn stator
16. A factor giving an error on a direct indicating compass would be:
- crosswinds – particularly on east/west headings
  - parallax due to oscillations of the compass rose
  - acceleration on east/west headings
  - turning through east/west headings
17. A rate integrating gyro is used in:
- inertial attitude unit
  - autopilot system
  - stabilizer servomechanism system
  - inertial navigation unit
  - rate of turn indicator
- 1, 2, 3, 4, & 5
  - 1 & 4
  - 2, 3, & 5
  - 2, 3, & 4
18. The errors of a DGI are:
- earth rate
  - transport wander
  - banking when pitched up
  - annual movement of poles
  - mechanical problems
- 2, 3, & 5
  - 3, 4, & 5
  - 1, 2, 3, & 5
  - all 5
19. An Air Data Computer (ADC) obtains altitude from:
- outside air temperature
  - barometric data from static source
  - time elapsed for signal to travel to and return from the earth
  - difference between absolute and dynamic pressure

20. If the needle and the ball of a Turn & Slip indicator both show right, what does it indicate?
- Turn to left & too much bank
  - Turn to right & too much bank
  - Turn to left & too little bank
  - Turn to right & too little bank
21. What formula gives the total temperature ( $T_T$ ) from the static temperature ( $T_S$ )?
- $T_T = T_S (1 + 0.2 M^2)$
  - $T_T = T_S (1 + 0.2 K_r M^2)$
  - $T_T = T_S / (1 + 0.2 K_r M^2)$
  - $T_T = T_S (1 - 0.2 M^2)$
22. The Inertial Strap down Unit of an IRS is programmed with co-ordinates during alignment in order to:
- establish the trihedron with reference to the earth
  - establish true or magnetic heading
  - check the function of the laser gyros
  - compensate for aircraft movement
23. When descending through an isothermal layer at constant CAS, what does the TAS do?
- Increase at a linear rate
  - Increase at an exponential rate
  - Remain the same
  - Decrease
24. What is  $V_{MO}$  calculated from?
- CAS
  - TAS
  - COAS
  - EAS
25. Descending from FL390 at maximum ground speed, what will the pilot be limited by?
- $V_{MO}$  initially then  $M_{MO}$  at a specified altitude
  - $M_{MO}$  initially then  $V_{MO}$  at a specified altitude
  - $V_{NE}$  initially then  $M_{MO}$  at a specified altitude
  - $V_{NO}$  initially then  $V_{NE}$  at a specified altitude
26. At constant weight, regardless of altitude, an aircraft always lifts off at a constant:
- EAS
  - TAS
  - ground speed
  - CAS



27.  $V_{FE}$  is the maximum speed that:
- the flaps can be operated
  - the flaps may be extended in the take-off configuration
  - the flaps may be extended in the landing configuration
  - the flaps may be extended in a specified configuration
28. The white arc on the ASI indicates:
- $V_{S1}$  at the lower end and  $V_{LE}$  at the upper end
  - $V_{S0}$  at the lower end and  $V_{LE}$  at the upper end
  - $V_{S0}$  at the lower end and  $V_{FE}$  at the upper end
  - $V_{S1}$  at the lower end and  $V_{FE}$  at the upper end
29. An ASI circuit consists of pressure sensors. The pitot probe measures:
- total pressure & static pressure
  - dynamic pressure
  - static pressure
  - total pressure
30. Mach number is defined as the ratio of:
- IAS to LSS
  - TAS to LSS
  - CAS to LSS
  - EAS to LSS
31. If a pitot source is blocked in an ASI, and the drain hole is blocked, but the static source is open, what will happen?
- ASI reading goes to zero
  - ASI under-reads
  - ASI over-reads
  - ASI behaves like an altimeter
32. In a turn at constant angle of bank, the rate of turn is:
- independent of weight and proportional to TAS
  - dependent on weight and inversely proportional to TAS
  - independent of weight and inversely proportional to TAS
  - dependent on weight and proportional to TAS
33. The turn indicator is a useful gyroscopic instrument. When used in association with an attitude indicator it will show:
- angular velocity about the yaw axis
  - direction of turn
  - angular velocity about true vertical axis
  - speed of turn
- 1, & 3
  - 2, & 3
  - 3, & 4
  - 1, & 2

34. If an aircraft, fitted with a DRMC, takes off on a westerly heading, in the northern hemisphere, the DRMC will indicate:
- a turn to the north
  - oscillates about west
  - no turn
  - a turn to south
35. When turning through 90° at constant attitude and bank, a classic artificial horizon indicates:
- nose up and correct angle of bank
  - attitude and bank angle are correct
  - nose up and bank angle too low
  - nose up and bank angle too high
36. The factors which will affect a turn indicator are:
- angle of bank
  - aircraft speed
  - aircraft weight
- all 3
  - 1 & 2
  - 1 & 3
  - 2 & 3
37. To obtain heading information from a gyro stabilized platform, the gyros should have:
- 1 degree of freedom and a horizontal axis
  - 1 degree of freedom and a vertical axis
  - 2 degrees of freedom and a horizontal axis
  - 2 degrees of freedom and a vertical axis
38. What are the inputs to the ADC?
- OAT
  - dynamic pressure
  - TAT
  - static pressure
  - electric power
  - pitot pressure
  - AoA
- 1, 2, 5 & 6
  - all 7
  - 3, 4 & 6
  - 3, 4, 5, 6, & 7

39. The properties of a turn indicator are:
1. one degree of freedom
  2. two degrees of freedom
  3. two springs connected to the outer gimbal
  4. spin axis in the longitudinal plane
  5. spin axis parallel to the yaw axis
  6. spin axis horizontal
- a. 1, & 6
  - b. 2, & 5
  - c. 1, & 4
  - d. 2, & 6
40. A gravity erector system corrects errors on a:
- a. DGI
  - b. artificial horizon
  - c. turn indicator
  - d. RIMC
41. In a gyromagnetic compass the flux gate transmits information to the:
- a. heading indicator
  - b. amplifier
  - c. error detector
  - d. erecting system
42.  $V_{NO}$  is the max. speed which:
- a. the pilot can fully deflect the controls
  - b. should only be exceeded in still air and with caution
  - c. should never be exceeded
  - d. must not be exceeded for flap/gear extension
43. If while level at FL270, at a constant CAS, temperature falls, what happens to the Mach No.?
- a. Decreases
  - b. Increases
  - c. Remains constant
  - d. Increases depending on whether temp >ISA or < ISA
44. If the static vent becomes blocked on an unpressurized a/c, what could you do?
- a. Open the window
  - b. Break the VSI glass
  - c. Compute altitude mathematically
  - d. Select standby pitot source
45. On a turn and slip indicator, needle to the left and ball to the right indicates:
- a. turn to the right, not enough bank
  - b. turn to the left, too much bank
  - c. turn to the left, not enough bank
  - d. turn to the right, too much bank

46. **What is density altitude?**
- Altitude in the standard atmosphere at which the prevailing density is equal to the density in the standard atmosphere
  - Pressure altitude corrected for prevailing temp
  - Temperature altitude
  - Pressure corrected
47. **A radio altimeter is:**
- ground based and measures true altitude
  - ground based and measures true height
  - a/c based and measures true altitude
  - a/c based and measures true height
48. **An a/c is travelling at 120 kt, what angle of bank would be required for a rate 1(one) turn?**
- 30°
  - 12°
  - 18°
  - 35°
49. **An a/c is travelling at 100 kt forward speed on a 3° glide slope. What is its rate of descent?**
- 500 ft/min
  - 300 ft/min
  - 250 ft/min
  - 600 ft/min
50. **If the pitot tube is leaking (and the pitot drain is blocked) in a non-pressurized a/c, the ASI will:**
- under-read
  - over-read
  - over-read in the climb, under-read in the descent
  - under-read in the climb, over-read in the descent
51. **Using a classic artificial horizon, the a/c performs a right turn through 270 degrees at a constant angle of bank and rate of turn. The indication is:**
- nose up, too much bank
  - nose up, not enough bank
  - nose up, wings level
  - bank and pitch correct
52. **In a DGI what error is caused by the gyro movement relative to the earth?**
- Earth rate
  - Transport wander
  - Real wander
  - Latitude error

53. In a right turn while taxiing, the correct indications are:
- needle left, ball right
  - needle left, ball left
  - needle right, ball right
  - needle right, ball left
54. An aircraft is taking off on a runway heading 045°, in still air, with a compass having 0° deviation. The runway is on an agonic line. What are the northerly turning errors (northern hemisphere)?
- Compass moves to less than 045°
  - Compass moves to more than 045°
  - Compass stays on 045° if wings are kept level
  - Compass remains on 045°
55. True heading can be converted into magnetic heading using a compass and:
- a map with isogonal lines
  - a map with isoclinical lines
  - a deviation card
  - a deviation curve
56. At sea level ISA, TAS:
- equals CAS
  - is greater than CAS
  - is less than CAS
  - cannot be determined
57. What will the altimeter read if the layers beneath the aircraft are all colder than standard?
- Read lower than the real altitude
  - Read higher than the real altitude
  - Read the correct altitude
  - Readings will fluctuate
58. The flux valve in an RIMC:
- is supplied with AC current
  - is fed with DC
  - is made of perm-alloy magnetic steel
  - has its own self-exciter unit
59. The indications of a Machmeter are independent of:
- temperature (OAT)
  - static pressure
  - differential static and dynamic pressure
  - dynamic pressure

60. An artificial horizon has:
- 1 degree of freedom and an horizontal axis
  - 2 degree of freedom and an horizontal axis
  - 1 degree of freedom and a vertical axis
  - 2 degree of freedom and a vertical axis
61. The rigidity of a gyro is improved by:
- increasing rpm and concentrating the mass on the periphery of the rotor
  - increasing rpm and concentrating the mass at the hub of the rotor
  - decreasing rpm and concentrating the mass on the periphery of the rotor
  - decreasing rpm and concentrating the mass at the hub of the rotor
62. What is the speed of sound at sea level ISA?
- 644 kt
  - 661 kt
  - 1059 kt
  - 583 kt
63. What is the speed of sound at 25 000 ft and -28 degrees C?
- 624 kt
  - 618 kt
  - 601 kt
  - 610 kt
64. What is the speed of sound at 30 000 ft and -40 degrees C?
- 562 kt
  - 595 kt
  - 590 kt
  - 661 kt
65. If a constant CAS is maintained in a climb, what happens to the Mach number?
- Remains constant
  - Increases
  - Decreases
  - Unable to determine without knowledge of the temperature
66. A compass swing is used to:
- align compass north with magnetic north
  - align compass north with true north
  - align magnetic north with true north
  - get true north and lubber line aligned
67. The TAT probe measures TAT by:
- $TAT = SAT + \text{kinetic heating}$
  - $TAT = SAT - \text{heating due to compressibility}$
  - $TAT = SAT - \text{kinetic heating}$
  - $TAT = SAT + \text{heating due to compressibility}$

68. If a pitot tube and drains are blocked at altitude by icing, during a descent the ASI will:
- read constant airspeed
  - under-read
  - over-read
  - show zero
69. An IRS is aligned when turned on so as to:
- calculate the computed trihedron
  - establish true and magnetic north
  - establish position relative to true north and magnetic north
  - establish magnetic north
70. The advantages of an ADC over a traditional pitot-static system (list):
- position and compressibility correction
  - reduced lag
  - ability to supply many instruments
  - ability to act as an altimeter following failure
- 1, 2 & 3
  - 1, 2 & 4
  - 2, 3 & 4
  - 1, 3 & 4
71. The frequency band used for a radio altimeter is:
- SHF
  - VHF
  - UHF
  - LF
72. What is the purpose of the latitude nut in a DGI?
- To correct for latitude error
  - To correct for transport wander
  - To correct for earth rate
  - To correct for coriolis error
73. Total Air Temp is always \_\_\_\_\_ than Static Air Temp and the difference varies with \_\_\_\_\_
- warmer, altitude
  - colder, altitude
  - warmer, TAS
  - colder, TAS
74. In a slightly banked turn, the turn needle will indicate:
- roll rate
  - rate of yaw
  - angular velocity about the vertical axis
  - rate of pitch

75. The Primary Flying Display (PFD) displays information dedicated to:
- engine data and alarms
  - flight path
  - weather radar
  - aircraft systems
76. What are the inputs to the FMS?
- Radio Aids
  - Engine Parameters
  - Air Data
  - Route Data
  - Terminal Data
  - Operating Data
- 1, 3, 4 & 6
  - 2, 3, 4, & 5
  - All of the above
  - 1, 2, 3 & 6
77. What are the upper and lower limits of the yellow arc on an ASI?
- Lower limit  $V_{LO}$  and upper limit  $V_{NE}$
  - Lower limit  $V_{LE}$  and upper limit  $V_{NE}$
  - Lower limit  $V_{NO}$  and upper limit  $V_{NE}$
  - Lower limit  $V_{LO}$  and upper limit  $V_{LE}$
78. What does the blue line on an ASI of a twin propeller engine aircraft indicate?
- $V_{YSE}$
  - $V_{NO}$
  - $V_{FE}$
  - $V_{MCA}$
79. The gravity erecting device on a vertical gyro is used on which instrument?
- Directional gyro unit
  - Turn indicator
  - Artificial horizon
  - Gyromagnetic device
80. In a VSI lag error is improved by:
- bi-metallic strip
  - two
  - use of an accelerometer system
  - return spring
81. An aircraft fitted with a DRMC is landing in a southerly direction, in the southern hemisphere. What indications will be seen on the DRMC?
- 180° turn to east
  - No apparent turn
  - Turn to west
  - Oscillation around south



82. What is the maximum drift of a gyro, due to earth rate?
- 90° per hour
  - 180° per hour
  - 15° per hour
  - 5° per hour
83. An aircraft is flying a true track of 360° from 5° south to 5° north. What is the change in apparent wander rate?
- 0° per hour
  - +3° per hour
  - 3° per hour
  - Depends upon ground speed
84. When turning through 180° at constant attitude and bank, a classic artificial horizon indicates:
- nose up and correct angle of bank
  - attitude and bank angle are correct
  - nose up and bank angle too low
  - nose up and bank angle too high
85. What is the Schuler period?
- 48 minutes
  - 84 seconds
  - 48 seconds
  - 84 minutes
86. Mach number is defined as:
- the ratio of pitot pressure to dynamic pressure
  - the ratio of static pressure to dynamic pressure
  - the ratio of dynamic pressure to static pressure
  - the ratio of static pressure to pitot pressure
87. You are flying at a constant FL290 and constant Mach number. The total temperature increases by 5°. The CAS will:
- remain approximately constant
  - increase by 10 kt
  - decrease by 10 kt
  - will increase or decrease depending on whether you are above or below ISA.
88. An aircraft turns from south-west to south-east when situated at 45°N, what heading should you roll out on if using a DRMC?
- 130°
  - 115°
  - 140°
  - 155°

89. What is SAT?
- Relative temperature measured in K
  - Differential temperature measured in K
  - Relative temperature measured in °C
  - Ambient temperature measured in °C
90. If an aircraft climbs, at constant Mach No., in ISA conditions, what happens to the TAS and the CAS?
- TAS increases and CAS increases
  - TAS remains constant and CAS decreases
  - TAS decreases and CAS increases
  - TAS decreases and CAS decreases
91. Where is the earth rate wander and the transport wander of a gyro equal to zero?
- North Pole
  - Equator
  - 45° N
  - 45° S
92. What happens when the static pressure supply, to an altimeter, becomes blocked during a descent?
- Reduces to zero
  - No effect
  - Under-reads
  - Indicates altitude at which blockage occurred
93. What happens when the static vent supplying an ASI is blocked, and the ram air inlet remains clear?
- ASI acts opposite to an altimeter
  - ASI always over-reads / reads a higher value
  - ASI always under-reads / reads a lower value
  - ASI acts like an altimeter
94. In a left turn while taxiing, the correct indications are:
- needle left, ball right.
  - needle left, ball left.
  - needle right, ball right.
  - needle right, ball left.
95.  $V_{LO}$  is defined as:
- the maximum speed at which to fly with the landing gear retracted
  - the maximum speed at which the landing gear may be retracted or extended
  - the maximum speed at which to fly with the landing gear extended
  - the minimum speed at which to fly with the landing gear extended

96.  $V_{NE}$  is defined as:
- the speed which must not be exceeded in still air, or without caution
  - the speed above which the landing gear may not be extended
  - the speed which must never be exceeded
  - the maximum speed for normal flap extension to be selected
97. In a gyro-magnetic compass, where does the torque motor get its information from?
- The flux gate
  - Error detector
  - The rotor gimbal
  - Amplifier
98. If an aircraft is descending at constant Mach number, and the total air temperature remains constant, what happens to the CAS?
- Remains constant
  - Decreases
  - Increases
  - Increases if the temperature is below standard, and decreases if the temperature is above standard
99. What are the advantages of a laser gyro compared to a conventional gyro?
- Has a longer cycle life
  - Takes longer to set up/ spin up
  - Uses more power
  - Takes longer to align
100. A Machmeter measures the ratio of:
- pitot pressure to static pressure
  - (pitot pressure minus static pressure) to static pressure
  - pitot pressure times static pressure
  - pitot pressure to (static pressure times pitot pressure)
101. Which instrument has a  $2^\circ$  rotation in the horizontal axis?
- Artificial horizon
  - Flux detector
  - Directional gyro indicator
  - Turn indicator
102. The maximum drift error sensed by an uncompensated DGI will be:
- $15^\circ$  per hour
  - $30^\circ$  per hour
  - $45^\circ$  per hour
  - $60^\circ$  per hour

103. The green arc on the ASI is used to identify which speed range?
- $V_{SO}$  to  $V_{NO}$
  - $V_{S1}$  to  $V_{FE}$
  - $V_{S1}$  to  $V_{NO}$
  - $V_{S1}$  to  $V_{LO}$
104. Pressure altitude may be defined as:
- lowest forecast regional pressure
  - pressure measured in the standard atmosphere
  - altitude indicated with QFE set on the altimeter
  - altitude indicated with QNH set on the altimeter
105. What is the effect on an altimeter reading if variations in static pressure occur near to the pressure source?
- A change in hysteresis error
  - A change in the instrument error
  - A change in the position error
  - A change in the compressibility error
106. What is the value of the angle of magnetic dip at the South Pole?
- $0^\circ$
  - $45^\circ$
  - $90^\circ$
  - $60^\circ$
107. A standby artificial horizon must have the following properties:
- a remote gyro
  - its own power supply
  - only to be used in emergency
  - its own gyro
  - one for each certified pilot
- all the above
  - 1, 3, & 5
  - 2, 3, & 4
  - 2 & 4
108. During a descent at constant CAS and temperature, the Mach No.:
- increases
  - remains constant
  - increases if SAT is greater than standard temperature and decreases if it is lower
  - decreases

109. The single most significant item which makes a servo altimeter more accurate is:
- electromagnetic pick-off
  - logarithmic scale
  - temperature compensated spring
  - multiple pointers
110. Which of the following gyro instruments has one degree of freedom?
- Artificial horizon
  - Turn indicator
  - Directional gyro
  - Slaved gyro compass
111. If a large aircraft is side slipped to starboard, and the port static vent is blocked, what will the altimeter read?
- Under-read
  - Read correctly
  - Over-read
  - Fluctuate
112. Mach number is determined from: ( $P_T$  = total pressure,  $P_S$  = static pressure)
- $(P_T + P_S) \times P_T$
  - $(P_T - P_S) \times P_S$
  - $(P_T \times P_S) \times P_T$
  - $(P_T - P_S) / P_S$
113. The right static vent is blocked, when the aircraft yaws to the right. What does the altimeter do?
- Over-read
  - under-read
  - unaffected
  - return to zero
114. If the radio altimeter fails:
- height information disappears
  - aural warning given
  - radio alt flag, red lamp, and aural warning given
  - radio alt flag and red lamp activates.
115. During a descent at a constant Mach number, there is an increase of total temperature by  $5^\circ$ . What effect does this have on CAS?
- Remains almost constant
  - Increases if SAT is more than standard and decreases if SAT is less than standard
  - Increases by 10 kt
  - Decreases by 10 kt

116.  $V_{NO}$  is defined as:
- maximum structural cruising speed
  - never exceed speed
  - manoeuvring speed
  - maximum operating speed
117. If the left static vent is blocked, and the right static vent is clear. What will the altimeter read if the aircraft maintains constant level?
- read correctly whatever the situation
  - under-read
  - if side slipping to the left, altimeter will over-read
  - if side slipping to the right, altimeter will over-read
118. An aircraft is flying at constant indicated altitude, over a cold airmass. The altimeter reading will be:
- greater than the real altitude
  - standard altitude
  - same as the real altitude
  - less than the real altitude
119. The Machmeter compares: ( $P_T$  = total pressure,  $P_S$  = static pressure)
- $(P_T - P_S)$  to  $P_T$
  - $(P_T + P_S)$  to  $P_S$
  - $(P_T - P_S)$  to  $P_S$
  - $P_T$  to  $P_S$
120. From where does the air data computer (ADC) obtain aircraft altitude?
- OAT probe
  - Dynamic – absolute ambient pressure
  - Absolute barometric sensor on aircraft fuselage
  - IRS
121. An aircraft is accelerating to take off on a runway with a QDM of  $045^\circ$ . Which way does the DRMC move, if the aircraft is in the northern hemisphere?
- Less than  $45^\circ$
  - More than  $45^\circ$
  - Correct if wings are level
  - Correct
122. When turning right onto north, through  $90^\circ$ , what heading on your DRMC should you roll out on, if the aircraft is in the northern hemisphere?
- $020^\circ$
  - $360^\circ$
  - $340^\circ$
  - $320^\circ$

123. What does a radio altimeter, for an aircraft in the landing configuration, measure?
- Height of aircraft wheels above the ground
  - Height of the aircraft above the ground
  - Altitude of the aircraft
  - Altitude of the aircraft wheels
124. Why is a servo altimeter better than a sensitive altimeter?
- It has an electromagnetic pick-off
  - It is more accurate at low level
  - It has ambient pressure in the capsule
  - It is fitted with a knocking device
125. In an altimeter what is found in: the capsule (i) and to the case (ii)?
- (i) partial vacuum      (ii) static input
  - (i) static input      (ii) partial vacuum
  - (i) pitot input      (ii) static input
  - (i) total input      (ii) ambient input
126. What principle does the radio altimeter work on?
- Pulse modulation
  - Amplitude modulation
  - Pulse modulation and carrier wave
  - Frequency modulation and carrier wave
127. What is indicated on the ASI when the static vent blocks during a descent?
- Under-reads
  - Reads correctly
  - Over-reads
  - Reads zero
128. A rate integrating gyro is used in:
- an inertial attitude unit
  - an autopilot system
  - an inertial navigation system
  - a rate of turn indicator
129. The error in a directional gyro due to the earth's rotation, at a mean latitude of  $45^{\circ}\text{N}$ , will cause the spin axis to move by:
- $10.6^{\circ}$  clockwise
  - $10.6^{\circ}$  anticlockwise
  - $7.6^{\circ}$  clockwise
  - $7.6^{\circ}$  anticlockwise
130. What are the components of a ring laser gyro?
- Mirrors and 2 cavities
  - 2 anodes and 2 cathodes
  - 2 beams of laser light
  - Horizontal gyro axis and 1 degree of freedom

131. Where on the earth's surface is the earth rate drift of a DGI equal to  $15.04^\circ$  per hour?
- $15^\circ$
  - $30^\circ$
  - $0^\circ$
  - $90^\circ$
132. If you maintain the same CAS and altitude (FL270), and the temperature increases, what happens to the Mach No.?
- Increases at an exponential rate
  - Decreases at an exponential rate
  - Remains the same
  - Increases
133. If CAS is kept constant, what happens to the Mach no.?
- as the altitude increases the Mach No. will increase
  - as the altitude increases the Mach No. will decrease
  - as the temperature increases the Mach No. will increase
  - as the temperature decreases the Mach No. will decrease
134. The pendulous type correction detector fitted to the DGI provides:
- torque on the sensitive axis
  - two torque motors on the horizontal axis
  - pendulous internal nozzle on the outer gimbal
  - one torque motor
135. An aircraft is fitted with two altimeters. One is corrected for position error, the other is not corrected for position error:
- ATC will receive erroneous information of flight level
  - at high speed the non-compensated altimeter will show a lower altitude
  - provided that the ADC is working normally, there will be no error to either altimeter
  - at high speed the non-compensated altimeter will show a higher altitude
136. Density altitude is defined as:
- the altitude of the airfield elevation corrected for lapse rate
  - the altitude reading on the altimeter which has QNH set on it
  - the altitude corresponding to the standard atmosphere compensated for ambient density
  - the altitude showing on the altimeter with the lowest regional QNH set
137. The pitot tube of an ASI gives a direct reading of:
- static pressure
  - total & static pressure
  - total pressure
  - dynamic pressure



138. When descending from FL350 to FL50 at maximum speed, the limitations which apply are:
- $V_{MO}$
  - $V_{MO}$  then  $M_{MO}$
  - $M_{MO}$  then  $V_{MO}$
  - $M_{MO}$
139. The pressure measured at the forward face of the pitot probe is:
- dynamic pressure
  - static pressure
  - total pressure
  - total pressure + static pressure
140. What has inputs from the flux valve?
- Error detector
  - Heading indicator
  - Amplifier
  - Precession motor
141. Machmeter readings are subject to which of the following errors?
- Density error
  - Setting error
  - Temperature error
  - Position/pressure error
142. Sound propagates at a speed which depends only on:
- density
  - temperature
  - temperature & pressure
  - pressure
143. What aircraft system uses a frequency of 4400 MHz?
- SSR
  - Radio altimeter
  - Weather radar
  - ATC radar
144. A low altitude radio altimeter, used in precision approaches, has the following characteristics:
- 1540 MHz to 1660 MHz range
  - pulse transmissions
  - frequency modulation
  - height range between 0 and 2500 ft
  - an accuracy of +/- 2ft between 0 and 500 ft
- 1, 4 and 5
  - 3 and 4
  - 3 and 5
  - 2, 3 and 5

145. A modern low altitude radio altimeter uses the principle of:
- pulse modulated waves, with the difference between the transmitted and received waves displayed on a circular screen
  - frequency modulated waves, where the difference between the transmitted wave and the received wave is measured
  - wave modulation, with frequency shift due to Doppler effect of the ground reflected wave being measured
  - triangular wave, with the frequency shift of the ground reflected wave being measured
146. The frequencies used in a low altitude radio altimeter are:
- 5 GHz to 6 GHz
  - 5400 MHz and 9400 MHz
  - 4200 MHz to 4400 MHz
  - 2700 MHz to 2900 MHz
147. The difference between magnetic north and true north can be derived from a:
- deviation curve
  - deviation card
  - map with isoclinic lines
  - map with isogonal lines
148. A direction gyro gets its directional information from:
- air data computer
  - direct reading magnetic compass
  - flight director
  - flux valve
149. What is the principle of operation of a VSI?
- Differential pressure across a capsule
  - Total pressure in a capsule
  - Static pressure in a capsule
  - Dynamic pressure in a capsule
150. In a remote indicating compass, what component feeds the amplifier?
- Gyro precession signal
  - Flux valve
  - Annunciator
  - Error detector
151. An aircraft turns right, through 90°, onto North, at 48N, using a direct indicating compass. The aircraft is turning at rate 2. What heading should the aircraft roll out on?
- 010°
  - 030°
  - 330°
  - 350°

152. What is the normal operating range of a low altitude radio altimeter?
- 0 to 2500 ft
  - 50ft to 2500 ft
  - 0 to 10 000 ft
  - 0 to 7500 ft
153. What is a radio altimeter used for?
- To determine aircraft height above mean sea level
  - To determine aircraft height above ground level
  - To determine pressure altitude
  - To determine aircraft altitude
154. Why must latitude and longitude be inserted into an IRS?
- To determine the aircraft position relative to the earth
  - To check the IRS position with the Flight Management System
  - To enable the levelling procedure to commence
  - To determine the accuracy of the alignment
155. An aircraft is flying a true track of  $360^\circ$  from  $5^\circ$  south to  $5^\circ$  north. What is the average apparent wander rate?
- $0^\circ$  per hour
  - $+ 5^\circ$  per hour
  - $- 5^\circ$  per hour
  - Depends upon ground speed
156. You commence a rate 2 turn from south-east to south-west, in the northern hemisphere. On what heading do you stop the turn?
- $240^\circ$
  - $255^\circ$
  - $235^\circ$
  - $205^\circ$
157. A directional gyro is valid only for a short period of time. The causes of this inaccuracy are:
- earth rotation
  - longitudinal accelerations
  - a/c motion over the earth
  - mechanical defects
  - gyro mass
- 1, 3 & 5
  - 1, 3 & 4
  - 1, 2 & 3
  - all of the above

158. An aircraft, in the southern hemisphere, is decelerating to land on a westerly heading. The direct reading magnetic compass will indicate:
- an apparent turn to north
  - an apparent turn to south
  - correctly
  - an oscillation about west
159. What is the input to a VSI?
- Static pressure
  - Differential pressure
  - Total pressure
  - Dynamic pressure
160. The component(s) used to align an inertial strapdown unit in the horizontal plane is/are:
- accelerometers and gyroscopes
  - accelerometers
  - flow inductors
  - gyroscopes
161. A ring laser gyro consists of:
- a gyro with 2 degrees of freedom
  - two moving cavities using mirrors
  - a laser split into two beams
  - two electrodes (anodes and cathodes)
162. The Directional Gyro Indicator (DGI) can:
- not align itself with magnetic north
  - automatically align itself with magnetic north
  - have 1° of freedom
  - have 2° of freedom
163. The pitot tube comprises a mast to position it away from the skin of the aircraft to:
- avoid disturbance from aerodynamic flow about the aircraft
  - position it outside the boundary layer
  - provide anti-ice protection
  - provide easy access for maintenance
164. Using a classic artificial horizon, the a/c performs a right turn through 360 degrees at a constant angle of bank and rate of turn. The indication is:
- nose up, too much bank
  - nose up, not enough bank
  - nose up, wings level
  - bank and pitch correct

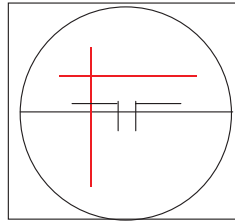
## Answers to Flight Instruments Questions

1	2	3	4	5	6	7	8	9	10	11	12
b	b	a	b	a	c	b	a	c	b	b	c
13	14	15	16	17	18	19	20	21	22	23	24
b	d	a	c	b	c	b	b	b	a	d	d
25	26	27	28	29	30	31	32	33	34	35	36
b	d	a	c	d	b	d	c	d	a	c	b
37	38	39	40	41	42	43	44	45	46	47	48
a	d	a	b	c	b	c	b	c	a	d	c
49	50	51	52	53	54	55	56	57	58	59	60
a	a	a	b	d	a	a	a	b	a	a	d
61	62	63	64	65	66	67	68	69	70	71	72
a	b	d	b	b	a	d	b	a	a	a	c
73	74	75	76	77	78	79	80	81	82	83	84
c	c	b	c	c	a	c	c	b	c	c	a
85	86	87	88	89	90	91	92	93	94	95	96
d	c	a	b	d	d	b	d	a	a	b	c
97	98	99	100	101	102	103	104	105	106	107	108
d	c	a	b	a	a	c	b	c	c	d	d
109	110	111	112	113	114	115	116	117	118	119	120
a	b	a	d	b	a	c	a	c	a	c	c
121	122	123	124	125	126	127	128	129	130	131	132
a	c	a	a	a	d	c	c	a	c	d	c
133	134	135	136	137	138	139	140	141	142	143	144
a	a	d	c	c	c	c	a	d	b	b	b
145	146	147	148	149	150	151	152	153	154	155	156
b	c	d	b	a	d	c	a	b	a	a	b
157	158	159	160	161	162	163	164				
b	a	a	a	c	d	a	d				

## Automatic Flight Questions

1. The flight director command bars on the display shown are commanding:

- fly up and left
- fly down and right
- fly down and left
- fly up and right



2. Where are the flight director modes displayed?

- PFD
- ND
- EICAM
- FD control panel

3. The autopilot is in heading select mode, and the aircraft is flying on a heading of 270°. If you change heading to 360°, the flight director command bars will:

- roll command bar goes full deflection right and then does not move until the aircraft heading is within 30° of the selected heading
- roll command bar moves to right and centres when AFDS angle of bank to intercept has been achieved
- the heading command bar will disappear and the heading hold will disengage
- roll command bar moves to the right and then progressively returns to the centre as the deviation from the selected heading reduces

4. What are the basic functions of an autopilot?

- Maintain pitch attitude
- Maintain wings level
- Altitude hold
- Heading hold
- Speed hold

- all 5
- 1 & 2
- 1, 2 & 3
- 1, 2, 3, & 4

5. At 50 feet AGL during an autoland, what happens to the glide slope signal?

- It continues to be actioned
- It is disconnected
- It is factored for range
- It is used to flare the aircraft

6. What is the wavelength of an ILS signal?

- Centimetric
- Hectometric
- Metric
- Decimetric

7. **A yaw damper indicator will indicate to the pilot:**
- yaw damper movement of rudder position
  - rudder position
  - rudder position due to pedal displacement
  - yaw damper movement on ground only
8. **The autothrottle is set to climb at a constant Mach number. If the temperature does not change, what happens to the CAS?**
- Increases
  - Decreases
  - Increases, but only if the outside air temperature decreases
  - Stays the same
9. **Autothrottle engaged mode can be checked by the pilot, using:**
- primary flight display
  - thrust control computer
  - position of throttles
  - navigation display
10. **The interception of the localizer beam by the autopilot is:**
- on a constant magnetic course
  - a mode using an interception versus range computation
  - a mode using an interception versus radio deviation law
  - on a constant heading
11. **Engagement of the autopilot is not possible when:**
- electrical supply is faulty
  - the turn control knob is not set to centre off
  - there is a synchronization fault
  - there is a fault in the attitude reference unit
- 1, 2, 3, & 4
  - 1, & 4 only
  - 1, 3, & 4
  - 2 & 4 only
12. **On which instrument are the flight director bars normally present?**
- Primary EICAS
  - ADI
  - ND
  - EHSI
13. **What happens at 50 ft whilst carrying out an autoland?**
- glide slope and localizer disconnect and aircraft continues to land
  - radio altimeter controls the rate of descent
  - radio altimeter controls the angle of attack
  - glide slope disconnects and aircraft continues descent

14. If you have selected a heading of 180° and are flying aircraft on heading of 160° to intercept the correct course, the ADI vertical bar will be central when?
- Only if aircraft is subject to 20° port drift
  - Only if aircraft is subject to 20° starboard drift
  - Cannot be centralized
  - Will only be central when flying correct attitude to intercept desired heading
15. If the autopilot is selected to VOR mode, what happens if the aircraft flies over the cone of confusion?
- Temporarily follows current heading until exiting the cone of confusion
  - VOR disengages and heading hold engages
  - The pilot must select an alternate roll mode
  - The pilot manually flies the aircraft following flight director roll commands
16. The autopilot disconnects (or the autoland is completed) at:
- 100 ft
  - decision height
  - flare
  - roll out
17. The control law in a fly-by-wire system is a relationship between:
- how the pilot's control demands are translated into control surface movements.
  - input and output at the amplifier level respectively control the deviation data
  - computer input deviation data and flap position modification
  - the versine signal between the ailerons and elevators
18. What are the autopilot minimum requirements in order to fly single pilot operations in IFR conditions or at night?
- Two axis autopilot with altitude hold and heading hold.
  - Two axis autopilot with altitude hold, heading hold, VOR tracking and alt acquire
  - Single axis autopilot with altitude hold only
  - Single axis autopilot with heading select and VS
19. When flying level in the cruise the ..... holds height and the ..... holds the speed:
- autopilot, autopilot
  - autothrottle, autothrottle
  - autothrottle, autopilot
  - autopilot, autothrottle
20. At what height during a semi-automatic landing is the autopilot disengaged?
- 100 ft
  - 45 ft
  - Decision height
  - 14 ft



21. At the missed approach point the TOGA switch on the throttles is depressed. Which of the following statements are correct?
1. Autopilot selects max. power
  2. GA power selected
  3. Aircraft automatically cleans up
  4. Autopilot flies the GA manoeuvre
  5. Pilot manually flies manoeuvre
- a. 2 & 5
  - b. 1 & 5
  - c. 1 & 4
  - d. 2 & 4
22. If a go-around is initiated from an auto-approach:
1. the autothrottle selects thrust as soon as the TOGA switch is pressed
  2. the autopilot carries out the climb
  3. the autopilot retracts flap and landing gear to reduce drag
  4. the pilot performs the climb
  5. the pilot retracts the flap and landing gear to reduce drag
- a. 1, 2 & 4
  - b. 1, 2 & 5
  - c. 1, 3 & 4
  - d. 1, 2 & 3
23. An autoland system which can continue to automatically land the aircraft after a single failure is called:
- a. fail-passive
  - b. fail-soft
  - c. fail-safe
  - d. fail-active
24. Where can the pilot look to see the autothrottle mode?
- a. PFD
  - b. Overhead panel
  - c. Throttle control panel
  - d. EICAS
25. Where can the pilot look to see the thrust limit mode?
- a. PFD
  - b. Overhead panel
  - c. Throttle control panel
  - d. Primary EICAS

26. The autopilot is engaged with no modes selected. What is the autopilot providing?
- Wing levelling
  - Altitude hold
  - Auto-stability with auto-trim
  - LNAV and VNAV
27. When is an autoland procedure complete?
- At the markers
  - At the beginning of the ground roll
  - At decision height
  - At the flare
28. During a CAT 2 approach, what is providing the height information?
- Capsule stack
  - Radio Altimeter
  - Captain's barometric altimeter
  - Central Air Data Computer
29. Autoland flare is initiated at:
- 1500 ft
  - 330 ft
  - 50 ft
  - 5 ft
30. An autopilot capable of altitude hold and heading hold is a minimum requirement for:
- single pilot operation in VMC and IMC
  - single pilot operation under IFR and at night
  - aircraft over 5700 kg
  - dual pilot operation (in IFR)
31. During a fully automatic landing the autopilot:
- and the autothrottle control the approach at least until the flare
  - and the autothrottle control the approach at least until the roll-out
  - and the autothrottle control the approach at least until decision height
  - controls the approach (at least) until the roll-out, the pilot controls the power
32. A landing is considered to be automatic when:
- autopilot flies the ILS to decision height, and then disengages
  - autothrottle maintains speed until decision height, and then disengages
  - autothrottle disengages thrust at 50 ft
  - autopilot flies the ILS until the flare
  - the flare is automatic
- 2, 3 & 5
  - 1 & 2
  - 4 & 5
  - 1 & 4

33. In an autopilot system, modes for stabilizing the a/c include which of the following?
1. Yaw damper.
  2. Pitch attitude holding.
  3. VOR axis holding.
  4. ASI & Mach hold.
  5. Horizontal wing holding.
  6. Altitude holding.
- a. 1, 2 & 4
  - b. 1, 2 & 5
  - c. 1, 5 & 6
  - d. 2, 4 & 6
34. In an autopilot system, a/c flight path modes include which of the following?
1. Pitch attitude holding.
  2. Horizontal wing holding.
  3. VOR axis holding.
  4. Inertial heading holding.
  5. ASI & Mach hold.
  6. Yaw damper.
- a. 1, 2 & 4
  - b. 1, 2 & 5
  - c. 2, 4 & 6
  - d. 3, 4 & 5
35. Autothrottle can hold which of the following?
1. Speed.
  2. Mach No.
  3. Altitude.
  4. N1/EPR.
  5. VOR capture.
  6. Vertical profile.
- a. 1, 2 & 3
  - b. 1, 2 & 4
  - c. 1, 2 & 6
  - d. 1, 3 & 5
36. An autopilot system whereby if one A/P fails cannot carry out an autoland is called fail\_\_\_\_\_:
- a. passive
  - b. safe
  - c. operational
  - d. redundant
37. In a yaw damper:
- a. ailerons are moved in proportion to Mach No.
  - b. ailerons are moved in proportion to rate of angular velocity
  - c. rudder is moved in proportion to Mach No.
  - d. rudder is moved in proportion to rate of angular velocity

38. "LOC ARMED" lights up on the annunciator, this means:
- localizer beam captured
  - localizer armed and awaiting capture
  - localizer alarm is on
  - ILS is captured
39. What is the most basic function of an autopilot?
- Altitude hold
  - Heading hold
  - Wing leveller
  - Altitude and heading hold
40. What does the autopilot pitch/rotate around?
- Centre of gravity
  - Manoeuvre point
  - Centre of pressure
  - Neutral point
41. During a semi-automatic landing:
- the A/P is disengaged at DH having followed the ILS
  - the A/T flies airspeed down to approximately 30 ft and automatically disengages
  - the A/P flies the approach and flare and roll-out
  - the A/T flies approach speed and disengages automatically at DH
42. If only a single A/P is used to climb, cruise and approach, following a failure:
- it is fail-passive with redundancy
  - it is fail-operational and will not disconnect
  - it is fail-soft and will not disconnect
  - it is fail-safe and will disconnect
43. In heading select the autopilot delivers roll commands to the controls to bank the aircraft:
- proportional to TAS, but not beyond a specified maximum.
  - Set bank of 27 degrees.
  - Set bank of 15 degrees.
  - Proportional to the deviation from the selected heading.
- 1&2
  - 2&3
  - 3&4
  - 4&1

44. Regarding autopilot and autothrottle:
1. A/P holds IAS/MACH when climbing in LVL CHG and A/T controls thrust
  2. A/P holds altitude in cruise with ALT HOLD, A/T controls IAS/Mach
  3. A/P holds pitch in descent in V/S mode, A/T controls thrust
  4. A/P holds alt in climb mode, A/T controls IAS/Mach in speed
- a. 1 & 2
  - b. 3 & 4
  - c. 1, 2 & 3
  - d. 2, 3 & 4
45. Auto-trim is fitted to an autopilot:
- a. to provide control about lateral axis
  - b. to prevent snatching on disengaging A/P
  - c. to prevent snatching on engaging A/P
  - d. to correct for Mach tuck
46. Autothrottle can hold:
1. speed
  2. flight path
  3. altitude
  4. Mach
  5. EPR / N1
  6. attitude
- a. 1, 2, 6
  - b. 1, 4, 5
  - c. 1, 2, 3, 4
  - d. 3, 4, 5
47. What is the purpose of the synchronization in an autopilot?
1. Prevents snatch on disengagement
  2. Prevents snatch on engagement
  3. Cancels rudder control inputs
  4. May not allow the autopilot to engage if unserviceable
- a. 1 & 2
  - b. 1 & 3
  - c. 2 & 4
  - d. 3 & 4
48. When operating with the autopilot in ALT hold mode what happens if the Captain's barometric altimeter pressure setting is increased?
- a. ALT hold disengages
  - b. Nothing
  - c. The aeroplane will climb
  - d. The aeroplane will descend

49. **TO/GA is engaged:**
- automatically at GS capture
  - automatically when an autopilot fails
  - by the pilot pressing a button on or near the throttles
  - by the pilot selecting flare
50. **On crossing the cone of confusion of a VOR when in VOR mode of the autopilot what will happen to the roll channel?**
- Always coupled to the selected VOR radial
  - Temporarily disconnected
  - Damped by a trim input from the lateral trim system
  - Temporarily switches to heading mode
51. **The function of auto-trim is:**
- to synchronize the longitudinal loop
  - to relieve forces on the autopilot servomotor prior to hand over
  - to react to altitude changes in ALT HOLD mode
  - to relieve forces on the control column before hand over
52. **The Mach trim system:**
- compensates for the rearward movement of the CP due to shockwave formation
  - compensates for the forward movement of the CP due to shockwave formation
  - controls the aircraft in roll
  - is operational at low subsonic speeds
53. **The Flight Director horizontal and vertical bars are up and left of aircraft symbol on the ADI; these indications are directing the pilot to:**
- increase pitch angle, turn left
  - decrease pitch angle, turn left
  - increase pitch angle, turn right
54. **What does FADEC do?**
- Engine limitation protection**
  - Automatic engine starting sequence**
  - Manual engine starting sequence**
  - Power management**
- 1 & 2
  - 4 only
  - 1 & 4
  - 1, 2, 3 & 4
55. **What does the Mach trim system use to prevent 'Mach tuck'?**
- Elevator
  - Elevator/rudder
  - Rudder
  - Elevator/aileron/rudder

56. The autosynchronization system does which of the following?
1. Prevents snatching on engagement
  2. Prevents snatching on disengagement
  3. Cancels rudder input
  4. Works in climb, cruise and descent
- a. 1&2
  - b. 2&3
  - c. 1&4
  - d. 3&4
57. When turning into a desired radial, FD bars indicate:
- a. a 45° angle of bank
  - b. a 30° angle of bank
  - c. a 15° angle of bank
  - d. correct attitude to intercept radial
58. If a pilot was to carry out a roll manoeuvre, on release of CWS what does the A/P do?
- a. Roll wing level and maintain heading only
  - b. Maintain attitude only
  - c. Maintain track and attitude only
  - d. Roll wing level and maintain MCP selected roll

## Answers to Autoflight Questions

1	2	3	4	5	6	7	8	9	10	11	12
a	a	b	b	b	c	a	b	a	d	a	b
13	14	15	16	17	18	19	20	21	22	23	24
d	d	a	d	a	a	d	c	d	b	d	a
25	26	27	28	29	30	31	32	33	34	35	36
d	c	b	b	c	b	b	c	b	d	b	a
37	38	39	40	41	42	43	44	45	46	47	48
d	b	c	a	a	d	d	a	b	b	c	b
49	50	51	52	53	54	55	56	57	58		
c	d	b	a	a	d	a	c	d	b		



## Warning & Recording Questions

1. The primary input to a basic stall warning system is:
  - a. angle of attack
  - b. IAS
  - c. slat/flap position
  - d.  $M_{NO}$
2. Where in the aircraft does EASA require the FDR to be fitted?
  - a. At the back
  - b. At the front
  - c. In the wings
  - d. In the undercarriage bay
3. An FDR fitted to an aircraft of over 5700 kg after Apr 98 must record for:
  - a. 10 hours
  - b. 25 hours
  - c. 30 minutes
  - d. 60 minutes
4. Where can TCAS be displayed?
  1. On its own screen
  2. On the EFIS
  3. Weather Radar
  4. On VSI
  - a. All 4
  - b. 1, 2 & 4
  - c. 2 & 3
  - d. 1 & 4
5. The principle that TCAS uses is:
  - a. primary radar
  - b. ATC radar
  - c. RT communications
  - d. transponders in the aircraft
6. When an intruder aircraft has no altitude reporting facility, i.e. Mode equipped with 'A' transponder only, TCAS can give:
  - a. corrective RA only
  - b. TA followed by a preventative RA
  - c. TA only
  - d. preventative RA only

7. What does an FDR record when combined with a CVR?
1. Cockpit voice
  2. Radio
  3. Public addresses from the cockpit
  4. Cabin voice
- a. 1, 2 & 3
  - b. 1 & 2
  - c. All 4
  - d. 2 & 4
8. A GPWS system requires:
- a. light & bell
  - b. aural signals only
  - c. aural signals which may be supplemented by visual signals
  - d. aural, tactile and visual signals or a combination thereof
9. The Altitude Alert system alerts the pilot:
- a. at decision height
  - b. at the selected altitude
  - c. when reference altitude equals the selected altitude
  - d. when deviating from the selected altitude
10. The Altitude Alert system:
- a. may alert by visual signals when approaching the selected altitude
  - b. activates a warning light on reaching selected altitude
  - c. engages auto-trim on reaching selected altitude
  - d. disengages auto-trim on reaching selected altitude
11. TCAS II obtains information from:
1. pressure encoding from mode S transponder
  2. radio altimeter
  3. aircraft specific configurations
  4. inertial reference unit (IRU)
- a. 1, & 2
  - b. 1, 2, & 4
  - c. 1, 2, 3, & 4
  - d. 1, 2, & 3
12. What does a CVR record?
- a. Cabin crew conversations
  - b. Cabin crew conversation on intercom
  - c. PA announcements even when not selected on flight deck
  - d. Radio conversations

13. **A stall warning system fitted to a large aircraft will always include:**
- various inputs including speed brake position, a warning module and a visual or aural warning
  - various inputs including landing gear micro switch, a warning module and an aural warning
  - various inputs including EGT, a warning module and an aural warning
  - stick-shakers and/or stick-push
14. **GPWS is active between what heights?**
- 0 ft and 2500 ft
  - 50 ft and 2450 ft
  - 0 ft and 2450 ft
  - 50 ft and 5000 ft
15. **What is the correct response to a TCAS RA?**
- Smoothly and immediately follow the climb or descent commands
  - Request permission to manoeuvre from ATC
  - Follow ATC instructions as these override TCAS RAs
  - Turn 90° and smoothly and immediately follow the climb or descent commands
16. **What symbol is used to represent an RA on a TCAS PPI?**
- Yellow circle
  - Red lozenge
  - Red square
  - Red circle
17. **Which of the following is a preventative RA?**
- Monitor vertical speed
  - Turn left
  - Traffic, traffic
  - Climb, climb now
18. **What input is there to TCAS II?**
- Mode 'A' transponder which gives TA and RAs
  - Mode 'C' transponder which co-ordinates avoidance manoeuvres
  - Mode 'C' and 'S' transponders which co-ordinate avoidance manoeuvres
  - Mode 'S' transponder which co-ordinates avoidance manoeuvres
19. **An aircraft with a MTOM greater than 5700 kg registered after 1 April 1998 requires a CVR which:**
- records the last 2 hours of flight
  - records the last 72 hours of flight
  - records the last 30 minutes of flight
  - records the last 8 hours of flight

20. What are the EASA OPS requirements for the CVR to start and stop recording?
- From before the aircraft is able to move under its own power, until it is no longer able to do so
  - From the time the first engine is started and stops 5 minutes after the last engine is shutdown
  - From the time when the first engine is started and stops 5 minutes after the APU is shutdown
  - From the time when the aircraft is first able to move under its own power until 5 minutes after it is no longer able to do so
21. An altitude alerting system must at least be capable of alerting the crew on:
- approaching selected altitude
  - excessive deviation from selected altitude
  - excessive vertical speed
  - excessive terrain closure
  - abnormal gear/flap combination
- 1 & 2
  - 1,2,3 & 4
  - 1,2,3,4 & 5
  - 1,2 & 3
22. For an aircraft that weighs more than 5700 kg and was registered after 1 April 1998, the FDR and CVR must record respectively:
- 25 hours and 1 hour
  - 25 hours and 2 hours
  - 10 hours and 2 hours
  - 10 hours and 1 hour
23. "Other traffic" which is assessed as not being a threat will be indicated by a TCAS system as:
- a solid red square
  - a solid white or cyan diamond
  - a hollow cyan diamond
  - a hollow cyan square
24. Which of the following are modes of the GPWS?
- Excessive sink rate
  - Altitude loss after T/O or go-around
  - Excessive Glide slope deviation
  - High climb rate
  - Flaps in incorrect position
  - High altitude descent
  - Stall
- All 7
  - 1, 2, 3, & 5
  - 1, 2 & 3
  - 1, 3, 5 & 7

25. What corrective action is given by TCAS?
- Turn left or right
  - Climb or descend
  - Contact ATC
  - Turn then climb or descend
26. The flight data recorder actually starts running:
- at the beginning of the T/O run
  - when the first engine is started
  - when the gear is retracted
  - when a/c lines up on runway
27. GPWS may indicate (list):
- Excessive sink rate after T/O
  - Excessive descent rate
  - Excessive closure with terrain
  - Ground proximity not in the landing configuration
  - Excessive glide slope deviation
  - Altitude call-outs
  - Bank Angle alerting
- 1, 4, 5 & 7
  - All 7
  - 1, 2 & 3
  - 1, 3, 6 & 7
28. TCAS II when fitted with mode C transponder may give:
- TA only
  - TA and RA in horizontal plane
  - TA and RA in vertical plane
  - RA only
29. According to EASA OPS when must the CVR on a 50 seat turbo prop a/c begin recording?
- Switch on to switch off
  - From lift off to when the weight on wheels switch is made on landing
  - From before the a/c is capable of moving under its own power to after the a/c is no longer capable of moving under its own power
  - At commencement of the taxi to turning off the runway
30. What is the GPWS Mode 3 audible alert?
- DON'T SINK, DON'T SINK followed by WHOOP WHOOP, PULL UP if the sink rate exceeds a certain value
  - DON'T SINK, DON'T SINK continuously
  - DON'T SINK, DON'T SINK followed immediately by WHOOP WHOOP, PULL UP
  - SINK RATE repeated each 1.5 seconds. Penetrating the second boundary generates an aural alert of WHOOP, WHOOP PULL UP

31. CVR components consist of:
1. microphone
  2. crash/fire resistant construction
  3. independent battery
  4. a flight data recorder
- a. 1 & 2
  - b. 1 & 4
  - c. 1, 2, 3 & 4
  - d. 1, 2 & 4
32. If an aircraft GPWS detects an excessive rate of descent with gear and flaps up, the alert and warning will be:
- a. DON'T SINK, DON'T SINK
  - b. TOO LOW TERRAIN, TOO LOW TERRAIN followed by TOO LOW GEAR, TOO LOW GEAR
  - c. SINK RATE followed by 'WHOOO WHOOP PULL UP
  - d. TERRAIN TERRAIN followed by 'WHOOO WHOOP PULL UP
33. The requirement to carry a GPWS concerns aircraft which are, depending on their age, weight and passenger capacity:
1. Turboprop
  2. Piston
  3. Jet
- a. 1, 2 & 3
  - b. 1, 3
  - c. 3
  - d. 1

**Answers to Warning & Recording Questions**

1	2	3	4	5	6	7	8	9	10	11	12
a	a	b	a	d	c	a	c	d	a	c	d
13	14	15	16	17	18	19	20	21	22	23	24
b	b	a	c	a	d	a	a	a	b	c	b
25	26	27	28	29	30	31	32	33			
b	b	b	c	c	b	a	c	b			

## Engine Instruments Questions

1. What type of sensor is used to measure the output of a low pressure booster pump?
  - a. Bourdon tube
  - b. Aneroid capsule
  - c. Bellows
  - d. Differential capsule
  
2. A vibration meter measures:
  - a. frequency in Hz
  - b. relative amplitude at a given frequency
  - c. period in seconds
  - d. acceleration in g
  
3. Which of the following are used to measure temperature?
  1. Thermocouple
  2. Resistance
  3. Reactance
  4. Mercury
  - a. 1, 2, 3, 4
  - b. 1, 2, & 4
  - c. 2, 3, & 4
  - d. 1, 3, & 4
  
4. A millivoltmeter measuring electromotive force between a hot and a cold junction of a thermocouple can be graduated to read temperature if:
  - a. the hot junction is kept at a constant temperature
  - b. the cold junction is maintained at 15 degrees C
  - c. the hot junction is maintained at 15 degrees C
  - d. the cold junction is maintained at a constant temperature
  
5. If both displays of an EICAS system fail what information will be displayed on the standby engine indicator?
  - a. N1, EPR, N2
  - b. N1, EPR, EGT
  - c. N2, EPR, EGT
  - d. EGT, N1, FF
  
6. A capacitive type gauging system may measure mass due to:
  - a. fuel dielectric constant being equal to that of air and proportional to density
  - b. fuel dielectric constant being equal to that of air and proportional to 1/density
  - c. fuel dielectric constant being twice that of air and proportional to density
  - d. fuel dielectric constant being twice that of air and proportional to 1/density



7. If the intake probe of an EPR system becomes blocked with ice this will cause the EPR gauge to:
- under-read during take-off
  - over-read during take-off
  - be unaffected
  - read zero
8. The power output of a turbo-propeller aircraft is measured by the amount of torque being produced. The indication can be in which of the following units?
- Newton metres
  - PSI
  - Percentage
  - Pounds feet
  - EPR
- 1, 2, 3, 4, 5
  - 1, 2, 3, 5
  - 2, 3, 4, 5
  - 1, 2, 3, 4
9. If one probe of a multi-sensor TGT system failed, the reading would:
- increase by 30 degrees C
  - decrease by 30 degrees C
  - be practically unaffected
  - fall to zero
10. During the take-off run the effect of increasing airspeed is to cause the EPR indication to:
- remain constant
  - increase
  - decrease
  - increase and subsequently decrease
11. An advisory message on the EICAS system screen would be:
- displayed in amber on the lower screen with associated caution lights and aural tones
  - displayed in red, normally on the upper screen, and requiring immediate corrective action
  - displayed in amber, normally on the upper screen with aural warnings
  - displayed in amber, normally on the upper screen, indented one space to the right
12. A supercharged aircraft is climbing at its maximum permitted boost of 8 psi (16 inHg) from sea level to its full throttle height of 10 000 feet. If sea level pressure is 29.92 inHg, when the aircraft reaches 5000 feet where the pressure is 24.72 inHg, what will be the approximate indication on the MAP gauge?
- 18 inHg
  - 33 inHg
  - 41 inHg
  - 46 inHg

13. An aircraft has a compensated capacitance fuel contents gauging system and is refuelled so that the total fuel contents are 76 000 kg at a temperature of 18 degrees C and an SG of 0.81. Whilst the aircraft is parked the temperature increases to 26 degrees C and the SG becomes 0.80. The indicated fuel contents have:
- increased by 5%
  - increased by 10%
  - decreased by 5%
  - remained the same
14. The working principle of a capacitive fuel contents gauging system is based upon:
- volume of fuel
  - changes in capacitance
  - height of fuel
  - dielectric value
15. A volumetric fuel flowmeter is different to a mass flowmeter because the mass flowmeter compensates for:
- dielectric
  - density
  - volume
  - pressure
16. Cylinder head temperature measurement works on the principle of:
- differential expansion
  - Wheatstone bridge
  - ratiometer
  - thermocouple
17. The electrical tacho generator system uses:
- single phase AC whose frequency varies with the speed of the engine delivered to a single phase synchronous motor and drag-cup
  - three phase AC whose frequency varies with the speed of the engine delivered to a three phase synchronous motor (squirrel cage) and drag-cup
  - a tacho probe and phonic wheel measuring speed and sending information to a squirrel cage motor and drag-cup
  - single phase DC whose frequency varies with speed of the engine converted to a square wave pulse delivered to a servo driven instrument
18. Where very accurate temperature indication is required the indicator used will be:
- galvanometer
  - direct reading
  - moving coil
  - ratiometer

19. Total Air Temperature (TAT) is equal to:

- a. SAT + ram rise
- b. RAT + friction rise
- c. SAT – RAT
- d. RAT + ram rise

20. The principle of the fuel-monitoring device giving the fuel burnt is:

- a. multiplying flight time by fuel consumption
- b. capacitance variation of a capacitor
- c. difference of indication according to departure value
- d. integration of instantaneous flow

21. To measure the fuel quantity on a heavy aircraft we use:

- 1. capacitor gauges
- 2. electric gauges with round floats
- 3. the indication can directly be indicated as a mass
- 4. the indication can not be indicated as a mass

The combination of all correct statements is:

- a. 1, 4
- b. 2, 3
- c. 2, 4
- d. 1, 3

22. For a capacitor gauge:

- 1. the fuel dielectric value is half that of air
- 2. the fuel dielectric value varies proportionally to the temperature of the fuel
- 3. the probes are connected in parallel
- 4. fuel dielectric value varies inversely with the fuel level
- 5. the gauge accuracy is within 2%

The correct statements are:

- a. 3, 5
- b. 2, 3
- c. 3, 4
- d. 1, 2

23. The capacitor gauge principle is based on:

- a. variation of capacitance of a capacitor with the nature of the dielectric
- b. variation of capacitance by volume measure at the probe
- c. variation of the EMF in a Wheatstone bridge
- d. variation of outflow and couple in the system

24. Among the following parameters:

EGT  
EPR  
FF  
N1  
N2  
Oil pressure  
Fuel pressure

The ones that can be used to monitor a gas turbine thrust setting are:

- a. EGT, N1, N2, oil pressure
- b. EGT, N1, FF, EPR
- c. EGT, EPR, FF, high-pressure fuel
- d. fuel pressure, N1, N2, oil pressure

25. The most significant parameters and the most important that express the thrust of a gas turbine engine are:

- a. EGT or N2
- b. N2 and FF
- c. FF and EGT
- d. N1 and EPR

26. On a modern twin spool turbofan, the main handling parameter is:

- a. the temperature upstream the turbine or EGT
- b. a rotational speed and a temperature
- c. the rotational speed of the high-pressure compressor
- d. the rotational speed of the low-pressure compressor

27. Two main indications used to evaluate a turbojet thrust are:

- a. rotational speed of the fan (N1) or the total pressure at the outlet of the low-pressure turbine
- b. fan rotational speed (N1) or total pressure at the high-pressure compressor outlet
- c. fan rotational speed (N1) or EPR
- d. high pressure turbine rotational speed or EPR

28. The measure of a torque can be made by measuring:

- a. Engine oil pressure at a fixed crown of an epicyclical reduction gear of the transmission box
- b. the amount of light through a gear linked to a transmission shaft
- c. the frequency of a phonic wheel linked to a transmission shaft
- d. the frequency difference between two phonic wheels linked to a transmission shaft

29. Among these instruments, which one uses aneroid capsules?

- a. Oil thermometer
- b. Air intake pressure sensor
- c. Oil pressure sensor
- d. Fuel pressure sensor

30. In a three phase tachometer installation:

1. the transmitter is a DC generator
2. we measure an EMF proportional to the driving speed of the transmitter
3. we measure a frequency proportional to the driving speed of the transmitter
4. the receiver is a galvanometer
5. the receiver is a synchronized motor driving a magnetic tachometer

The correct statements are:

- a. 1, 2
- b. 2, 5
- c. 1, 4
- d. 3, 5

31. The working principle of mass flowmeters mostly used nowadays, is to measure in their system:

- a. the volume and viscosity of the fuel
- b. the temperature and pressure of the fuel
- c. volume mass and dielectric value of fuel
- d. kinetic energy transmitted

32. The advantage of a ratiometer is:

- a. does not require an electrical supply
- b. does not suffer from errors due to variations of supply voltage
- c. is calibrated at sea level and will be inaccurate at high altitudes
- d. it requires an AC voltage and therefore has no commutator

33. What is a synchroscope used for?

- a. Reducing vibration
- b. Putting the propellers in phase
- c. Allowing the pilot to adjust several engines to the same rpm
- d. Viewing the underside of the aircraft during flight

34. On an EICAS display what does the yellow arc on the temperature gauge signify?

- a. Forbidden operating range
- b. Exceptional operating range
- c. Normal operating range
- d. Frequent operating range

35. An rpm gauge has a red line at the upper end of the green arc, in the middle of the green arc is a smaller red arc. What is the significance of this smaller red arc?

- a. It indicates an rpm that must not be used continuously because of the increased vibration level from the engine/propeller
- b. It is maximum continuous rpm
- c. It is the rpm at which there is an increased likelihood of oil leakage
- d. It indicates an rpm that must not be used continuously because there is insufficient cooling air for the engine

36. In an ECAM system if a caution message appears the system will:
- illuminate the page number that requires to be selected
  - display a diagrammatic view of the affected system
  - will alert the pilot by an audible warning only
  - will cause the relevant buttons to light up
37. The principles used in an electrical rpm indicating system are:
- tacho probe and phonic wheel
  - DC generator producing AC
  - 3 phase AC generator driving a 3 phase AC motor
  - single phase AC generator driving a single phase AC motor
- 1 and 4
  - 1 and 3
  - 2 and 3
  - 2 and 4
38. What does the yellow band on an EICAS generated engine gauge indicate?
- Precautionary operating range
  - Maximum operating range
  - Warning limit
  - Normal range
39. Advantage of a ratiometer type measuring circuit is:
- very accurate
  - simple
  - changes indication if voltage changes
  - no external power supply is required
40. In a turbojet thrust is measured by:
- fan speed (N1) and turbine inlet pressure
  - N1 and EPR
  - compressor outlet pressure and jet pipe pressure
  - compressor inlet pressure and combustion chamber pressure
41. How will a system failure warning be shown to the pilot in the ECAM system?
- The failure will appear as a wording on the screen
  - The master warning caption will illuminate and the pilot will manually select failure mode
  - The master warning will illuminate and the primary screen will display a check list while the secondary shows a graphical display of the problem
  - The engine parameter displays will be replaced by the Flight Mode screen
42. What is used to measure gas turbine inlet pressure?
- Bourdon tube
  - Differential capsule
  - Aneroid capsule
  - Bellows

43. What does a Bourdon tube measure?
- Temperature
  - Quantity
  - Capacitance
  - Pressure
44. What are the disadvantages of an electrical float fuel quantity measuring system?
- Attitude
  - Acceleration
  - Temperature
  - Ambient pressure
  - Needs an AC power supply
- 1, 2, 3, 4, 5
  - 1, 2, 3
  - 1, 2, 5
  - 2, 3, 4, 5
45. How can temperature be measured?
- Resistance
  - Mercury
  - Thermocouple
  - Reactance
- 1,2,3,4
  - 1,2,4
  - 1,2,3
  - 1,3,4
46. A thermocouple would normally be used to measure the temperature of the:
- turbine
  - exterior
  - cabin
  - oil
47. The Bourdon tube is used in:
- temperature probes in front of the engine
  - smoke detectors
  - pressure measurement
  - vibration detectors
48. The principle upon which flowmeters (mass flow) most commonly used today work, is to measure:
- the volume and viscosity of the fuel
  - the pressure and temperature of the fuel
  - density and dielectric constant of the fuel
  - the kinetic energy transferred

49. A small turbine placed in the flow of fuel to the burners of a gas turbine engine measures:
- volume of flow by the measurement of magnetic impulses
  - mass flow by the measurement of magnetic impulses
  - mass flow by the measurement of frequency
  - volume flow by the measurement of reactance
50. The torquemeter is an instrument:
- allowing automatic synchronization of the engines
  - giving the power available by the engine
  - giving the power from the propeller by direct reading
  - allowing the determination of the power from the propeller by using a formula which is a function of the rpm
51. Torque can be calculated in a torquemeter system by the measurement:
- of the oil pressure resisting lateral movement of the gearing in an epicyclic reduction gearbox
  - of the amount of light through a gear wheel connected to the transmission
  - of the frequency of a phonic wheel connected to the planet gears of an epicyclic gearbox
  - of the difference between 2 phonic wheels connected to the transmission
52. EPR is a:
- ratio between ambient pressure and exhaust pressure.
  - ratio between ambient pressure and fan pressure.
  - ratio between intake pressure and compressor delivery pressure.
  - ratio between exhaust pressure and intake pressure.



## Answers to Engine Instruments Questions

1	2	3	4	5	6	7	8	9	10	11	12
c	b	b	d	b	c	b	d	c	c	d	d
13	14	15	16	17	18	19	20	21	22	23	24
d	b	b	d	b	d	a	d	d	a	a	b
25	26	27	28	29	30	31	32	33	34	35	36
d	d	c	a	b	d	d	b	c	b	a	b
37	38	39	40	41	42	43	44	45	46	47	48
b	a	a	b	c	d	d	b	c	a	c	d
49	50	51	52								
a	d	a	d								

## Specimen Questions

1. **A modern radio altimeter uses the frequency band:**
  - a. VHF 30 - 300 MHz
  - b. SHF 3000 MHz - 30 GHz
  - c. UHF 300 MHz - 3 GHz
  - d. HF 3 MHz - 30 MHz
  
2. **An aircraft that is assessed as not being a threat would be indicated on a TCAS system as:**
  - a. a solid red square
  - b. a solid white or cyan diamond
  - c. a hollow white or cyan diamond
  - d. a solid yellow circle
  
3. **During descent through a block of airspace of constant temperature and while flying at a constant Mach No. will cause the CAS to:**
  - a. increase
  - b. decrease
  - c. remain constant
  - d. increase at a rate of  $1.98^\circ/1000$  ft
  
4. **The true altitude of an aircraft in flight is shown from:**
  - a. the standard atmosphere
  - b. pressure altitude
  - c. density altitude
  - d. temperature altitude
  
5. **On a turn and slip indicator, needle to the left and ball to the right indicates:**
  - a. right turn not enough bank
  - b. left turn too much bank
  - c. left turn not enough bank
  - d. right turn too much bank
  
6. **What is density altitude?**
  - a. Temperature altitude
  - b. Pressure altitude corrected for the prevailing temperature
  - c. The altitude in the International Standard Atmosphere at which the prevailing density would be found
  - d. Pressure altitude corrected for Total Air Temperature
  
7. **A radio altimeter is:**
  - a. ground based and measures true altitude
  - b. ground based and measures true height
  - c. aircraft based and measures true altitude
  - d. aircraft based and measures true height

8. Which of the following are modes of the GPWS?
- i. Excessive sink rate
  - ii. Altitude loss after take-off or go-around
  - iii. Excessive glide slope deviation
  - iv. High climb rate
  - v. Flaps in the incorrect position
  - vi. High altitude descent
  - vii. Stall
- a. i ii iii v
  - b. ii iii v vii
  - c. i ii iii vii
  - d. iii iv v vi
9. An aircraft is travelling at 120 kt, what angle of bank would be required for a rate one turn?
- a. 30°
  - b. 12°
  - c. 19°
  - d. 35°
10. An aircraft is travelling at 100 kt forward speed on a 3° glide slope. What is its rate of descent?
- a. 500 ft/min
  - b. 300 ft/min
  - c. 250 ft/min
  - d. 500 ft/sec
11. What correction is given by TCAS?
- a. Turn left or right
  - b. Climb or descend
  - c. Contact ATC on receipt of a resolution advisory
  - d. Climb or descend at 500 ft/min
12. If the total pressure sensor supply line leaks, and with the drain element blocked, in a non-pressurized aircraft this will cause the ASI to:
- a. under-read
  - b. over-read
  - c. over-read in the climb and under-read in the descent
  - d. under-read in the climb and over-read in the descent
13. Using a classic attitude indicator, an aircraft performs a turn through 270° at a constant angle of bank and rate of turn. The indication is:
- a. nose up                      bank right
  - b. nose up                      bank left
  - c. nose up                      wings level
  - d. bank and pitch correct

14. The needle and ball of a turn indicator are both to the left of the datum. This indicates:
- a left turn with too much bank
  - a left turn with too little bank
  - a right turn with too little bank
  - a right turn with too much bank
15. Under conditions determined by the International Standard Atmosphere, at MSL true airspeed is:
- greater than CAS
  - less than CAS
  - equals CAS
  - is indeterminate due to the variation in temperature
16. In what range is GPWS operative?
- 2450 - 0 ft
  - 3000 - 50 ft
  - 2450 - 50 ft
  - 3000 - 0 ft
17. Which of the following are inputs to the central processing unit of the GPWS?
- Flaps
  - Landing gear
  - Glide slope
  - Unusual attitudes
  - Radio altimeter
  - VOR
- i ii vi
  - i ii iii v
  - i ii iv v
  - i ii iii v vi
18. What is another name for fail-active?
- Fail-soft
  - Fail-operational
  - Fail-safe
  - Fail-passive
19. Why must an autopilot be synchronized when you wish to disconnect?
- To ensure fail-operational landings can continue safely
  - To allow automatic pitch trimming to reset
  - To secure against abrupt changes in aircraft attitude
  - To allow for FD coupling

20. What is used for EGT measurement?
- Helical bimetallic strips
  - Thermistors
  - Radiation pyrometry
  - Thermo EMF thermocouples
21. When accelerating on a northerly heading what does the direct reading magnetic compass indicate?
- No change
  - North
  - A turn to the west
  - A turn to the east
22. Why is there a vibration device in a pressure altimeter?
- To prevent hysteresis
  - To prevent lag in a mechanical system
  - To keep pilots happy during long flights
  - To prevent icing
  - To overcome dither
- i ii iv
  - i ii
  - ii iii v
  - i ii iii
23. What does the white arc on a temperature scale indicate?
- Never exceed
  - Maximum start and acceleration temperature
  - Normal operating temperature
  - Minimum temperature
24. The rate of turn indicator is a very useful gyroscopic instrument. When used in conjunction with the attitude indicator it provides:
- angle of bank
  - rate of turn about the yaw axis
  - rate of climb
  - rate of turn athwartships
25. With the aircraft weight constant but variations in airfield altitude, take-off will always be at a constant:
- equivalent airspeed
  - calibrated airspeed
  - ground speed
  - true airspeed
26. An inertial reference system is aligned when turned on so as to:
- calculate the computed trihedron with reference to the earth
  - establish true and magnetic north
  - establish position relative to true and magnetic north
  - establish magnetic north

27. Total air temperature is \_\_\_\_\_ than static air temperature and the difference varies with \_\_\_\_\_.
- a. warmer altitude
  - b. colder altitude
  - c. warmer CAS
  - d. colder CAS
28. True heading can be converted into magnetic heading using a compass and:
- a. a map with isogonal lines
  - b. a map with isoclinical lines
  - c. a map with isobars
  - d. a deviation card
29. An aircraft flies into a colder airmass. This will cause the altimeter to:
- a. over-read
  - b. under-read
  - c. read the correct altitude
  - d. the indication will depend on the hemisphere of operation
30. A gravity-erecting device is utilized in:
- a. an artificial horizon
  - b. a directional gyroscopic indicator
  - c. a vertical speed indicator
  - d. a turn and slip
31. The rigidity of a gyroscope can be improved by:
- a. increasing the angular momentum and concentrating the mass on the periphery of the rotor
  - b. increasing the angular momentum and concentrating the mass at the hub of the rotor
  - c. decreasing the angular momentum and concentrating the mass on the periphery of the rotor
  - d. decreasing the angular momentum and concentrating the mass at the hub of the rotor
32. The outputs of a flux valve are initially sent to:
- a. an amplifier
  - b. an error detector
  - c. a compass card
  - d. a feedback loop
33. The period of validity of an FMS data base is:
- a. 56 days
  - b. one week
  - c. 28 days
  - d. varies depending on the area of operational cover

34. **An IRS differs from an INS in that it:**
- has a longer spin-up (is not affected by vertical accelerations due to gravity)
  - has a shorter spin-up time and suffers from laser lock.
  - does not need to correct for coriolis and central acceleration)
  - does not experience Schuler errors as accelerometers are strapped down and not rotated by a feedback loop
35. **In a solid state gyroscope the purpose of the dither motor is to:**
- enhance the acceleration of the gyro at all rotational rates
  - overcome laser lock
  - compensate for transport wander
  - stabilize the laser frequencies
36. **In an IRS:**
- the accelerometers are strapped down but the platform is gyro-stabilized
  - the platform is strapped down but the accelerometers are gyro-stabilized
  - accelerometers and gyros are both gyro-stabilized
  - accelerometers and gyros are both strapped down
37. **Which of the following correctly describes the gyroscope of a rate of turn indicator?**
- 1 degree of freedom
  - 2 degrees of freedom
  - Its frame is held by two springs
  - Its spin axis is parallel to the pitch axis
  - The spin axis is parallel to the yaw axis
  - The spin axis is horizontal
- i ii
  - i iv v
  - i iii v
  - i iii vi
38. **A blockage occurs in the ram air source and the drain-hole. The ASI in a non-pressurized aircraft will:**
- read a little low
  - read a little high
  - act like an altimeter
  - freeze at zero
39. **The errors associated with the directional indicator are:**
- earth rate
  - transport wander
  - banking when pitched up
  - annual movement of the poles
  - mechanical problems
- i ii iii
  - i ii iv v
  - i ii iii iv v
  - i ii iii v

40. A rate integrating gyroscope is used in:
- inertial attitude system
  - automatic flight control systems
  - inertial navigation systems
  - rate of turn indicators
- i ii
  - i iii
  - i iii iv
  - i ii iii
41. Rate of turn is affected by:
- aircraft speed
  - angle of bank
  - aircraft weight
- i ii
  - i iii
  - ii iii
  - none of the above
42. The ability of a gyroscope to indicate aircraft heading is based on it having:
- one degree of freedom in the vertical axis
  - two degrees of freedom in the vertical axis
  - two degrees of freedom in the horizontal
  - one degree of freedom in the horizontal
43. A  $V_{MO}/M_{MO}$  alerting system contains a barometric aneroid capsule:
- which is subjected to dynamic pressure and an airspeed capsule which is subjected to static pressure
  - and an airspeed capsule subjected to static pressure
  - and an airspeed capsule subjected to dynamic pressure
  - which is subjected to static pressure and an airspeed capsule which is subjected to dynamic pressure
44. When measuring different pressures (low/med/high) which of the following has the three types of sensing devices in ascending order of pressure measurement?
- Bourdon tube
  - Bellows type
  - Aneroid capsule
- i ii iii
  - iii ii i
  - i iii ii
  - ii iii i



45. Sound is propagated at a velocity which is dependent upon:
- barometric pressure
  - density
  - static pressure
  - temperature
46. The local speed of sound at mean sea level at ISA -10°C is:
- 661 kt
  - 650 kt
  - 673 kt
  - 680 kt
47. What would the compass heading be given a true heading of 247° in an area where the variation is 8°W and a compass deviation of 11°E?
- 255°
  - 244°
  - 247°
  - 266°
48. An aircraft is flying at flight level 350 at a CAS of 290 kt and a temperature deviation of ISA -10°C. The TAS and MN will be:
- |    |     |        |      |       |
|----|-----|--------|------|-------|
| a. | TAS | 498 kt | Mach | 0.885 |
| b. | TAS | 520 kt | Mach | 0.882 |
| c. | TAS | 481 kt | Mach | 0.855 |
| d. | TAS | 507 kt | Mach | 0.86  |
49. An aircraft in the northern hemisphere lands and decelerates on a westerly heading. The compass will indicate:
- a turn north
  - no turn will be indicated
  - an oscillation
  - a turn south
50. A compass swing is used to:
- align compass north with magnetic north
  - align compass north with true north
  - align magnetic north with true north
  - get true north and the lubber line aligned
51. The angle formed between the directive force and the total magnetic force is called:
- variation
  - deviation
  - dip
  - isoclinal

52. What is the speed of sound at 30 000 ft and  $-40^{\circ}\text{C}$ ?
- 562 kt
  - 595 kt
  - 590 kt
  - 661 kt
53. If a constant CAS is maintained under normal conditions in the climb what happens to the Mach No.?
- It will decrease
  - It will remain constant
  - It will decrease in an isothermal layer
  - It will increase
54. Regarding magnetism; which of the following statements is correct?
- Lines of flux run from blue pole to red pole
  - Like poles repel
  - Unlike poles repel
  - Like poles attract
  - Unlike poles attract
- i ii v
  - i iii v
  - ii v
  - i iii iv
55. The output of a double integration N/S is:
- velocity
  - departure
  - distance
  - longitude
56. A solid state gyro is:
- a rate gyro
  - a rate sensor
  - an earth gyro
  - a tied gyro
57. The magnetic heading reference unit has a precession rate of:
- $1^{\circ}/\text{min}$
  - $2^{\circ}/\text{min}$
  - $5^{\circ}/\text{min}$
  - $3^{\circ}/\text{min}$
58. If the TAS at 40 000 ft is 450 kt the Mach No. is:
- 0.815
  - 0.783
  - 0.76
  - 0.825

59. The EADI and the EHSI of an EFIS installation are also referred to by the manufacturers as:
- primary display and navigation display respectively
  - navigation display and primary display respectively
  - EICAS and ECAM respectively
  - ECAM and EICAS respectively
60. In which of the following modes may information from the AWR be displayed?
- plan
  - expanded ILS
  - map
  - full nav
  - full ILS
  - expanded nav
  - full VOR
  - expanded VOR
  - centre map
- i ii iv vii
  - i iii vii viii ix
  - ii iii vi viii ix
  - ii iii v vii ix
61. Wind information can be displayed in an EFIS system in which of the following modes?
- plan                      map                      expanded ILS    full VOR
  - map                        centre map        plan                full ILS
  - full nav                  full ILS            map                centre map
  - full ILS                  full VOR           map                plan
62. On an EADI radio altitude is displayed:
- digitally between 2500 ft and 100 ft
  - on an analogue scale below 2500 ft
  - digitally between 2500 ft and 1000 ft and thereafter as an analogue/digital display
  - as an analogue display between 2500 ft and 1000 ft and thereafter as a digital display
63. In FMS fitted aircraft the main interface between pilot and system will be provided by:
- the automatic flight control system
  - the multi-purpose control and display unit
  - the flight control unit
  - the flight management source selector

64. In the ILS mode, one dot on the lateral deviation scale on the EHSI indicates:
- a. 1 NM
  - b. 2 NM
  - c. 1°
  - d. 2°
65. On a standard 2-dot EHSI in the en route mode each dot represents:
- a. 1 NM
  - b. 2 NM
  - c. 5 NM
  - d. 10 NM
66. Given the following information calculate the instrument error of a preflight altimeter check:
- |      |                                  |        |
|------|----------------------------------|--------|
| i.   | aerodrome elevation:             | 235 ft |
| ii.  | apron elevation:                 | 225 ft |
| iii. | height of altimeter above apron: | 20 ft  |
| iv.  | altimeter reading with QFE set:  | 40 ft  |
- a. +20 ft
  - b. +30 ft
  - c. +40 ft
  - d. +10 ft

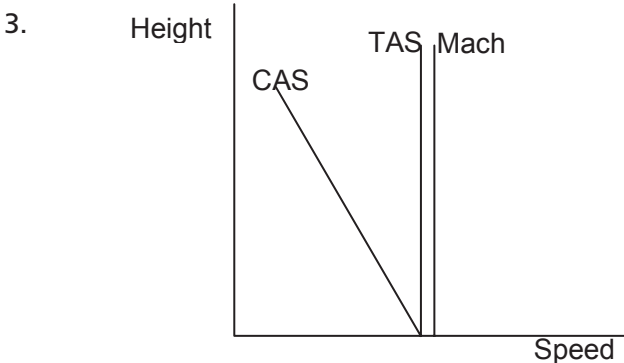
Explanations to Specimen Questions

1. A modern radio altimeter operates on the principle of a frequency modulated continuous wave in the frequency band 4200 - 4400 MHz. This is the SHF band also referred to as the centimetric (microwave) band of 3 - 30 GHz.

Answer B

2. The symbols in the answers represent the following:
  - a. solid red square - Resolution Advisory
  - b. solid white or cyan diamond (lozenge) - Proximate traffic
  - c. hollow white or cyan diamond (lozenge) - Other traffic
  - d. solid yellow circle - Traffic Advisory

Answer C



Answer A

4. True altitude is the exact vertical distance above mean sea level (AMSL). This differs from the indicated pressure if ambient conditions vary from ISA. True altitude may be calculated from pressure altitude using the navigation computer.

**Navigation Computer Method:**

Given:

- i. indicated pressure altitude -25 000 ft
- ii. ambient temperature -50°C

Method:

- i. in the altitude window set pressure altitude of 25 000 ft against the temperature of -50°C.
- ii. Against indicated altitude on the inner logarithmic circular scale read off the true altitude of 23 400 ft.

Answer B

5. In the answers the needle and ball indicate the following:
- right turn with skid (insufficient aileron, too much rudder)
  - left turn with slip (too much aileron, insufficient rudder)
  - left turn with skid (insufficient aileron, too much rudder)
  - right turn with slip (too much aileron, insufficient rudder)

Answer B

6. Density altitude is defined as "The altitude in the ISA at which the prevailing density would be found".

Density altitude may be calculated using the navigation computer or by formula.

Example:

- pressure altitude - 3000 ft
- ambient temperature - +20°C

**Navigation Computer Method**

- in the airspeed window set the pressure altitude of 3000 ft against the temperature of +20°C.
- read the density altitude over the arrow in the density altitude window. (4000 ft)

Formula Method

- density altitude = pressure altitude × (± ISA deviation × 120)  
therefore:  $3 \times (+11 \times 120)$   
= 3960 ft

**Note:** *If the ambient temperature is lower than ISA then density altitude will be lower than pressure altitude and vice versa.*

Answer C

7. Answer D
8. Answer A
9. The formula to calculate angle of bank for a rate one turn is:  
A of B = true airspeed/10 + 7°  
therefore:  $120/10 + 7^\circ = 19^\circ$

Answer C

10. To calculate the rate of descent of an aircraft in feet per minute the rule of thumb is:

5 × aircraft ground speed

therefore: 5 × 100 = 500 ft/min rate of descent

Answer A

11. TCAS I will issue a Traffic Advisory only. Manoeuvre of the aircraft is prohibited.

TCAS II will issue a Corrective Resolution Advisory instructing the pilot to take corrective action in the vertical plane only.

TCAS III will issue a Corrective Resolution Advisory instructing the pilot to take corrective action in the vertical and/or horizontal planes.

*Note: Do not confuse a "Corrective Resolution Advisory" with a "Preventative Resolution Advisory" which only provides limitations on aircraft manoeuvres as opposed to a "Corrective Resolution Advisory" which issues corrective aircraft manoeuvres.*

Answer B

12. A leak in the total (pitot) pressure line will exhaust a percentage of that pressure to atmosphere causing both the ASI and the Machmeter to under-read. The loss of pressure will cause the airspeed capsules in both instruments to under-expand.

Answer A

13. A classic attitude indicator is referring to an air driven artificial horizon. During a standard 360° the following indications will be apparent:

		<u>PITCH</u>	<u>ROLL</u>
Roll in:	360°	normal	normal
	90°	high (nose up)	too low
	180°	high (nose up)	normal
	270°	high (nose up)	too high
Roll out:	360°	normal	normal

This error is used by the application of compensation tilt.

Answer A

14. See answers to question 6

Answer A

15. This can be proved on the navigation computer. Set all parameters for ISA at mean sea level. Read off CAS on the inner scale against TAS on the outer scale. They are the same.

Answer C

16. This is based on radio altimeter height above terrain.

Answer C

17. inputs to GPWS are as follows:

- i. radio altimeter
- ii. vertical speed
- iii. ILS glide slope deviation
- iv. undercarriage position
- v. flap position
- vi. Mach number

Answer B

18. A system that can withstand at least one failure but leaves the system capable of completing the landing and roll is described as 'fail-operational'. An alternative term is 'fail-active'.

Answer B

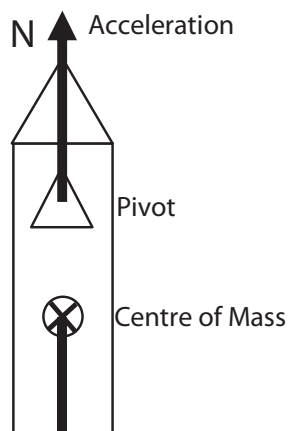
19. The autopilot synchronization system prevents 'snatching' of the flying controls on engagement of the autopilot. The auto-trim system adjusts the trim of the aircraft during automatic flight to prevent 'snatching' of the controls on disengagement of the autopilot.

Answer C

20. EGT is measured using a number of thermocouples connected in parallel to minimize the effect of failure of one of them.

Answer D

21. Newton's third law: 'to every action there is an equal and opposite reaction'. In this case the forces cancel out and the compass will continue to indicate north.



Answer B



22. Answer B

23. Answer C

24. Answer B

25. Answer B

26. A trihedron is defined as figure having three sides. Additionally, the IRS establishes true north. Magnetic indications are obtained by the application of variation, which, in the case of the Boeing 737-400 is stored in each IRS memory. The range covered for variation is from 73° North to 60° South latitude.

Answer A

27. Total Air Temperature (TAT) is warmer than Static Air Temperature due to the effect of compressibility. The faster the aeroplane flies the greater the TAT.

Answer C

28. Isoclinical lines refer to magnetic dip, isobars refer to atmospheric pressure lines on a meteorological chart and a 'deviation card' is a compass correction card positioned along side the compass in the aircraft.

Answer A

29. If the actual temperature of the column of air in which the aircraft is flying is COLDER than ISA then the True Altitude of the aircraft above mean sea level will be LOWER than the Indicated Altitude. If the actual temperature is WARMER than ISA the True Altitude will be higher.

Answer A

30. The air driven artificial horizon is also known as the 'classic attitude indicator'.

Answer A

31. Angular momentum means speed of rotation (rpm).

Answer A

32. The error detector is also known as a signal selsyn.

Answer B

33. Answer C

34. The question refers to a ring laser gyro.

Answer B

35. Answer B
36. Answer D
37. Answer C
38. Answer C
39. Answer D
40. Answer B
41. Answer A
42. Vertical axis means horizontal plane. The two are at 90° to each other. It follows, therefore, that heading information only requires one degree of freedom in the horizontal plane about the vertical axis of the gyroscope.

Answer A

43. Answer D
44. Answer B
45.  $LSS = 38.94 \times \sqrt{\text{temp in K}}$   
Or, on the CRP 5

Answer D

46. The first thing to remember is that the local speed of sound is dependent on AMBIENT temperature not ISA. In this case, applying the temperature deviation to the ISA temperature at mean sea level gives an ambient temperature of +5°C. There are two methods of calculating the local speed of sound.

The first is to use the formula in question 45 as follows:

$$LSS = 38.94 \times \sqrt{278 \text{ K}}$$

$$LSS = 649.26 \text{ kt}$$

The second method is to use the CRP 5 navigation computer as follows:

- i in the airspeed window set the ambient temperature of +5°C against the mach number index.
- ii against 10 on the inner (CAS/Mach number) scale read off the LSS on the outer (TAS) scale. 650 kt

Answer B

47.  $C \pm D = M \pm V = T$   
 $244^\circ + 11^\circ = 255^\circ - 8^\circ = 247^\circ$

Answer B

48. On the CRP 5 first calculate the TAS from the information given. Don't forget compressibility! This will give a TAS of 481 kt. Now in the airspeed window set the ambient temperature of  $-65^\circ\text{C}$  against the Mach number index. Against the TAS of 481 on the outer scale read off the Mach number on the inner scale, in this case M 0.855.

Answer C

49. During a deceleration the direct reading compass will indicate an apparent turn towards the further pole. In the northern hemisphere it will give an apparent turn towards the south whereas in the southern hemisphere it will indicate an apparent turn towards the north. Accelerations are exactly opposite indicating apparent turns towards the nearer pole.

Answer D

50. Answer A

51. Variation is the angular difference between true north and magnetic north

Deviation is the angular difference between Magnetic North and Compass North. 'Dip' is the angle formed between the horizontal component (H or directive force) of the earth's magnetic field and the total magnetic force (intensity) of the earth's magnetic field.

An isoclinal is a line joining lines on a chart of equal 'dip'. The aclinic line is a line indicating zero 'dip' and is also referred to as the magnetic equator.

Answer C

52. In this question the altitude is of no consequence. Use the same techniques as shown in question 46

Answer B

53. Owing to density error the TAS will increase dramatically in the climb even at a constant CAS. The LSS, being controlled by temperature, will also reduce. It can be seen from the formula that where  $\text{Mach No.} = \text{TAS} \div \text{LSS}$  an increase in TAS and a decrease in LSS must result in a marked increase in Mach No. A descent would mean a reversal of the speed changes seen in the climb.

Answer D

54. This question refers to the basic rules of magnetism; like poles repel, unlike poles attract and the lines of flux run from the red north-seeking pole to the blue south-seeking pole.

Answer C

55. Double integration means the second stage of integration, in this case distance along the local meridian.

Answer C

56. A solid state gyro refers to the ring laser gyro as used in inertial reference ('strapdown') systems. It is also referred to as a rate sensor.

Answer B

57. Answer B

58. The ISA temperature at 40 000 ft is  $-56.5^{\circ}\text{C}$ . Using the airspeed window of the CRP 5 set the temperature of  $-56.5^{\circ}$  against the Mach number index. Now against the TAS of 450 kt on the outer scale read off the Mach number on the inner, 0.783.

Answer B

59. Answer A

60. Answer C

61. In an EFIS system wind information can be displayed in every mode except PLAN.

Answer C

62. Answer C

63. Answer B

64. In the NAV mode 1 dot = 2 NM, in the VOR mode 1 dot =  $5^{\circ}$  and in the ILS mode 1 dot =  $1^{\circ}$

Answer C

65. Answer B

66. The apron is 10 ft below the stated aerodrome elevation, so assuming the QFE to be for the aerodrome level, an altimeter on the apron should read  $(-10)$  ft. However, the instrument is positioned in the aircraft 20 ft above the apron so it should show  $(-10) + 20 = +10$  ft. Its actual reading is +40 ft so it is over-reading by 30 ft, an instrument error of +30 ft.

Answer B

**Specimen Examination Paper****55 Questions Time 1 h 30 mins**

1. An aircraft maintaining a constant CAS and altitude is flying from a cold airmass into warmer air. The effect of the change of temperature on the speed will be:

- a. CAS will increase
- b. EAS will decrease
- c. TAS will increase
- d. TAS will decrease

1 Mark

2. Select the correct statement:

- a. EAS = CAS corrected for compressibility error
- b. EAS = IAS corrected for position error
- c. CAS = TAS corrected for density error
- d. TAS = EAS corrected for compressibility error

1 Mark

3.  $V_{LO}$  is defined as:

- a. the maximum speed at which to fly with the landing gear retracted
- b. the maximum speed at which the landing gear may be retracted or extended
- c. the maximum speed at which to fly with the landing gear extended
- d. the minimum speed at which to fly with the landing gear extended

1 Mark

4. An aircraft taking off from an airfield with QNH set in the altimeter has both static vents blocked by ice. As the aircraft climbs away the altimeter will:

- a. read the airfield elevation
- b. indicate the aircraft height AMSL
- c. read the height of the aircraft above the airfield
- d. show only a very small increase in height

1 Mark

5. In an inertial-lead VSI the source of the most pronounced error is:

- a. instrument
- b. position
- c. steep turn
- d. missed approach manoeuvre

1 Mark

6. An aircraft is descending at a constant Mach number. If the aircraft is descending through an inversion layer, the CAS will:

- a. remain constant
- b. increase
- c. decrease
- d. decrease then decrease more slowly

1 Mark

7. The combined Machmeter/ASI is subject to the following errors:

- a. position, density, instrument, compressibility, manoeuvre induced
- b. those of the machmeter only
- c. instrument, pressure and temperature only
- d. instrument and compressibility only

1 Mark

8. You are flying at a constant FL290 and constant Mach number. The total temperature increases by 5°. The CAS will:

- a. remain approximately constant
- b. increase by 10 kt
- c. decrease by 10 kt
- d. will increase or decrease depending on whether you are above or below ISA.

1 Mark

9. A factor giving an error on a direct indicating compass would be:

- a. crosswinds - particularly on east/west headings
- b. parallax due to oscillations of the compass rose
- c. acceleration on east/west headings
- d. turning through east/west headings

1 Mark

10. If an aircraft, fitted with a DRMC, takes off on a westerly heading, in the northern hemisphere, the DRMC will indicate:

- a. a turn to the north
- b. oscillates about west
- c. no turn
- d. a turn to south

1 Mark

11. To improve the horizontality of a compass, the magnet assembly is suspended from a point:

- a. on the centre line of the magnet
- b. below the centre of gravity
- c. above the centre of gravity
- d. varying with magnetic latitude

1 Mark.

12. Which of the following will affect a direct reading compass?

1. ferrous metals
2. non-ferrous metals
3. electrical equipment

- a. 1 only
- b. 1 & 3
- c. 1 & 2
- d. all 3

2 Marks

13. The rigidity (gyroscopic inertia) of a gyroscope may be increased by:

- a. increasing the number of gimbals and decreasing the number of planes of rotation
- b. increasing the speed of rotation and decreasing the mass of the rotor
- c. increasing the speed of rotation and increasing the mass of the rotor
- d. decreasing the speed of rotation and increasing the speed of the rotor.

1 Mark

14. The gravity erecting device on a vertical gyro is used on which instrument?

- a. Directional gyro unit
- b. Turn indicator
- c. Artificial horizon
- d. Gyromagnetic device

1 Mark

15. If the rpm of the rotor in a turn and slip indicator is higher than normal, the turn indicator will:

- a. over-read the correct rate of turn
- b. under-read the correct rate of turn
- c. not indicate due to the increased rigidity
- d. indicate correctly

1 Mark

16. When the pointer of a rate of turn indicator shows a steady rate of turn:

- a. the calibrated spring is exerting a force about the lateral axis equal to the rate of turn
- b. the force produced by the spring is producing a precession equal to but opposite to the direction of turn
- c. the spring is providing a force which produces a precession equal to the rate of turn (in the opposite direction to the turn)
- d. the spring is providing a force which produces a precession equal to the rate of turn (in the same direction as the turn)

2 Marks

17. With reference to the flux valve of a remote indicating compass:

- a. the flux valve is pendulously mounted and is free to turn to remain aligned with the earth magnetic field
- b. the flux valve is not subject to acceleration errors
- c. the flux valve is pendulously mounted and so it is not subject to or affected by the earth's magnetic field
- d. the flux valve is fixed to the aircraft and so turns with the aircraft to measure the angle between the aircraft and the earth's magnetic field

1 Mark

18. In a Schuler tuned INS, the largest unbounded errors are:

- a. due to acceleration errors
- b. track errors due to initial misalignment
- c. due to real wander of the platform gyroscopes
- d. created at the first stage of integration

1 Mark

19. The amber ALERT sign on an INS control and display unit:

- a. illuminates steadily for 2 minutes before reaching the next waypoint
- b. starts flashing 2 minutes before reaching the next waypoint and goes out at 30 seconds to run
- c. illuminates if power from the aircraft bus bar has been lost and the system is operating on standby battery
- d. illuminates steadily after passing a waypoint in manual mode, until the next leg is programmed in

1 Mark

20. To obtain heading information from a gyro stabilized platform, the gyros should have:

- a. 1 degree of freedom and a horizontal axis
- b. 1 degree of freedom and a vertical axis
- c. 2 degrees of freedom and a horizontal axis
- d. 2 degrees of freedom and a vertical axis

1 Mark

21. What are the advantages of an IRS compared to an INS?

- a. Reduce spin-up time and a dither motor to prevent "lock-out".
- b. Reduce spin-up time and accuracy not adversely affect by "g"
- c. Increase accuracy and a dither motor to prevent "lock-out".
- d. Insensitively to "g" and reduced wander of the gyroscope.

2 Marks



22. What errors can the Air Data Computer correct for?

1. Instrument error & ram rise
2. Compressibility & density error
3. Lag & density error
4. Position & temperature error
5. Temperature & instrument error
6. Manoeuvre error & ram rise
7. Manoeuvre & position errors

- a. 2 & 4
- b. 1 & 6
- c. 3 & 7
- d. 3 & 5

2 Marks

23. Which of the following is the FMS normal operating condition in the cruise?

- a. L NAV only
- b. V NAV only
- c. L NAV or V NAV
- d. L NAV and V NAV

1 Mark

24. Weather radar returns can be displayed in which of the following EFIS Modes?

- a. Plan      Exp ILS      Exp VOR
- b. Plan      Exp ILS      Map
- c. Map      Exp ILS      Exp VOR
- d. Map      ILS      VOR

3 Marks

25. What are the colours used on an EFIS display to show a tuned navigation aid and an airport?

- a. Green & white
- b. White & magenta
- c. Green & cyan
- d. White & yellow

2 Marks

26. WXR display is on:

- a. the captain's CRT only
- b. the co-pilot's CRT only
- c. a special screen
- d. on both the captain's and co-pilot's CRTs

1 Mark

27. **Altitude select and altitude hold are examples of:**

- a. inner loop functions in pitch
- b. manometric functions from the ADC
- c. interlocking functions
- d. outer loop functions in roll

1 Mark

28. **An autopilot delivers roll commands to the ailerons to achieve a bank angle:**

- a. proportional to TAS, but below a specified maximum
- b. set bank of 25 degrees
- c. set bank of 30 degrees
- d. proportional to the deviation from the desired heading, but not exceeding a specified maximum

1 Mark

29. **At 200 ft on an autoland:**

- a. the LOC mode is engaged in the roll channel and the G/S mode is engaged in pitch.
- b. the LOC mode is engaged in the roll channel and the FLARE mode is engaged in the pitch channel
- c. the ROLL OUT mode is engaged in the Roll channel and the G/S mode is engaged in pitch.
- d. the autothrottle is maintaining the speed and the pitch channel is maintaining the height.

1 Mark

30. **During a CAT 1 ILS approach, height is indicated by:**

- a. GPS
- b. radio altimeter
- c. marker
- d. barometric

1 Mark

31. **During an approach to autoland at 1500 feet:**

- a. off line channels are manually engaged, flare mode is armed
- b. localizer is controlling the roll channel, off line channels are automatically engaged and flare mode is armed
- c. localizer is controlling the roll channel, stabilizer is trimmed nose up and roll out is armed
- d. provided both localizer and glide slope signals are valid LAND 3 will

2 Marks

32. During an autoland the caption LAND 2 is illuminated. The system is:

- a. fail-active or fail-operational
- b. fail-passive
- c. approaching decision height
- d. requiring a crew input

1 Mark

33. If only a single A/P is used to climb, cruise and approach; following a failure:

- a. it is fail-passive with redundancy
- b. it is fail-operational and will not disconnect
- c. it is fail-soft and will not disconnect
- d. it is fail-safe and will disconnect

1 Mark

34. "LOC ARMED" lights up on the FMA part of the PFD, this means:

- a. localizer beam captured
- b. localizer beam armed and awaiting capture
- c. localizer alarm is on
- d. a/c is on localizer centre line

1 Mark

35. What is the purpose of the auto-synchronization system in an A/P?

- 1. Prevents snatching on disengagement
- 2. Prevents snatching on engagement
- 3. Cross feeds rudder and aileron inputs for co-ordination
- 4. May not allow the A/P to engage if unserviceable
- 5. Displays the control positions
- 6. Removes standing demands from the autopilot system prior to the CMD button being selected

- a. 2, 4 & 6
- b. 1, 3 & 5
- c. 2, 3 & 5
- d. 1, 4 & 6

1 Mark

36. What type of autoland system would be required for the landing to continue following a single failure below alert height?

- a. Fail-soft
- b. Fail-passive
- c. Fail-operation or fail-active
- d. Land 2 system

1 Mark

37. Which of the following apply to a yaw damper?

1. May aid the pilot in the event of asymmetric thrust after engine failure
2. Applies measured amounts of aileron to counter dutch roll
3. Increases lateral stability to stop Dutch roll
4. Is required at high altitude
5. Can automatically help in turn co-ordination
6. May deflect the aileron to counteract the natural oscillating frequency of the aircraft.

- a. 1, 4 & 6
- b. 2, 3 & 6
- c. 2, 3 & 4
- d. 1, 4 & 5

1 Mark

38. A stall warning system fitted to a large aircraft will always include:

- a. various inputs including speed brake position, a warning module and a visual or aural warning
- b. various inputs including landing gear micro switch, a warning module and an aural warning
- c. various inputs including EGT, a warning module and an aural warning
- d. stick-shakers and/or stick-push

1 Mark

39. TCAS II when fitted with mode C transponder may give:

- a. TA only.
- b. TA and RA in horizontal plane.
- c. TA and RA in vertical plane.
- d. RA only.

1 Mark

40. The Altitude Alert system alerts the pilot:

- a. at decision height
- b. at the selected altitude
- c. when reference altitude equals the selected altitude
- d. when deviating from the selected altitude

1 Mark

41. The GPWS uses inputs from:

- a. the radio altimeter, static pressure monitor, ILS receiver and the landing gear and flap position monitors
- b. the radio altimeter and the ILS receiver only
- c. the radio altimeter, ILS receiver, static pressure monitor, and the landing gear position monitor only
- d. the radio altimeter, static pressure monitor, landing gear position monitor, and the flap position monitor only

1 Mark

42. What are the components of a CVR?

1. Microphone
2. Crash/fire resistant construction
3. Independent battery
4. A flight data recorder

- a. 1 & 2
- b. 1 & 4
- c. 1, 2, 3 & 4
- d. 1, 2 & 4

2 Marks

43. What corrective action is given by TCAS?

- a. Turn left or right.
- b. Climb or descend.
- c. Contact ATC
- d. Turn then climb or descend.

1 Mark

44. What input is there to TCAS II?

- a. Mode 'A' transponder which gives TA and RAs
- b. Mode 'C' transponder which co-ordinates avoidance manoeuvres
- c. Mode 'C' and 'S' transponders which co-ordinate avoidance manoeuvres
- d. Mode 'S' transponder which co-ordinates avoidance manoeuvres

1 Mark

45. When an intruder aircraft has no altitude reporting facility, i.e. Mode equipped with 'A' transponder only, TCAS can give:

- a. corrective RA only
- b. TA followed by a Preventative RA
- c. TA only
- d. preventative RA only

1 Mark

46. Which of the following are modes of the GPWS?

1. Excessive sink rate.
2. Altitude loss after T/O or go-around.
3. Excessive Glide slope deviation.
4. High climb rate.
5. Flaps in incorrect position.
6. High altitude descent.
7. Stall.

- a. All 7
- b. 1, 2, 3, & 5
- c. 1, 2 & 3
- d. 1, 3, 5 & 7

2 Marks

47. A warning message on the EICAS system screen would be:

- a. displayed in amber on the lower screen with associated caution lights and aural tones
- b. displayed in amber, normally on the upper screen with aural warnings .
- c. displayed in red, normally on the upper screen , and requiring immediate corrective action
- d. displayed in amber, normally on the upper screen, indented one space to the right.

1 Mark

48. An aircraft equipped with digital avionics includes an ECAM system. With this centralized system, if a failure in one of the monitored systems is displayed, the crew must:

- a. cancel the warning
- b. analyse initially the failure and only respond to a level 1 warning
- c. reset the warning display after noting the failure on the left screen
- d. apply the immediate actions as directed by the checklist on the left of the two screens

2 Marks

49. An aircraft has a compensated capacitance fuel contents gauging system and is refuelled so that the total fuel contents are 76 000 kg at a temperature of 18°C and an SG of 0.81. Whilst the aircraft is parked the temperature increases to 26°C and the SG becomes 0.80. The indicated fuel contents have:

- a. increased by 10%
- b. remained the same
- c. increased by 5%
- d. decreased by 5%

1 Mark

50. **EPR is the ratio of:**

- a. the compressor outlet pressure to the compressor inlet pressure
- b. jet pipe pressure to compressor inlet pressure on a turbo-prop engine only
- c. jet pipe pressure to the compressor inlet pressure on a gas turbine engine
- d. jet pipe pressure to the compressor outlet pressure on a gas turbine engine

1 Mark

51. **If one probe of a multi-sensor EGT system became disconnected, the reading would:**

- a. increase by between 20°C to 30°C
- b. decrease by between 20°C to 30°C
- c. fall to zero
- d. be largely unaffected

1 Mark

52. **The principle of the fuel-monitoring device giving an indication of the total fuel burnt is:**

- a. multiplying flight time by fuel consumption
- b. capacitance variation of a capacitor
- c. difference of indication according to departure value
- d. integration of instantaneous flow

1 Mark

53. **The red arc in the middle of the green band of a piston engine rpm indicator signifies:**

- a. maximum rpm
- b. minimum rpm
- c. rpm at which a greater level of vibration is encountered
- d. rpm that must never be exceeded in the cruise

1 Mark

54. **Torquemeters provide a reliable measure of power output from:**

- a. a turbo-jet engine
- b. a noise suppression unit
- c. a turbo-propeller engine
- d. an APU

1 Mark

55. Which of the following types of pressure gauge would be best suited to a high pressure input?
- a. Aneroid capsule
  - b. Bourdon tube
  - c. Bellows
  - d. Dynamic probe

1 Mark





## Answers to Specimen Examination Paper

(w = weighting/marks allocated for the question)

1	2	3	4	5	6	7	8	9	10	11	12
c	a	b	a	c	b	a	a	c	a	c	b
w1	w1	w1	w1	w1	w1	w1	w1	w1	w1	w1	w2
13	14	15	16	17	18	19	20	21	22	23	24
c	c	a	d	d	c	a	a	b	a	d	c
w1	w1	w1	w2	w1	w1	w1	w1	w2	w2	w1	w3
25	26	27	28	29	30	31	32	33	34	35	36
c	d	b	d	a	d	b	b	d	b	a	c
w2	w1	w1	w1	w1	w1	w2	w1	w1	w1	w1	w1
37	38	39	40	41	42	43	44	45	46	47	48
d	b	c	d	a	a	b	d	c	b	c	d
w1	w1	w1	w1	w1	w2	w1	w1	w1	w2	w1	w2
49	50	51	52	53	54	55					
b	c	d	d	c	c	b					
w1	w1	w1	w1	w1	w1	w1					

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