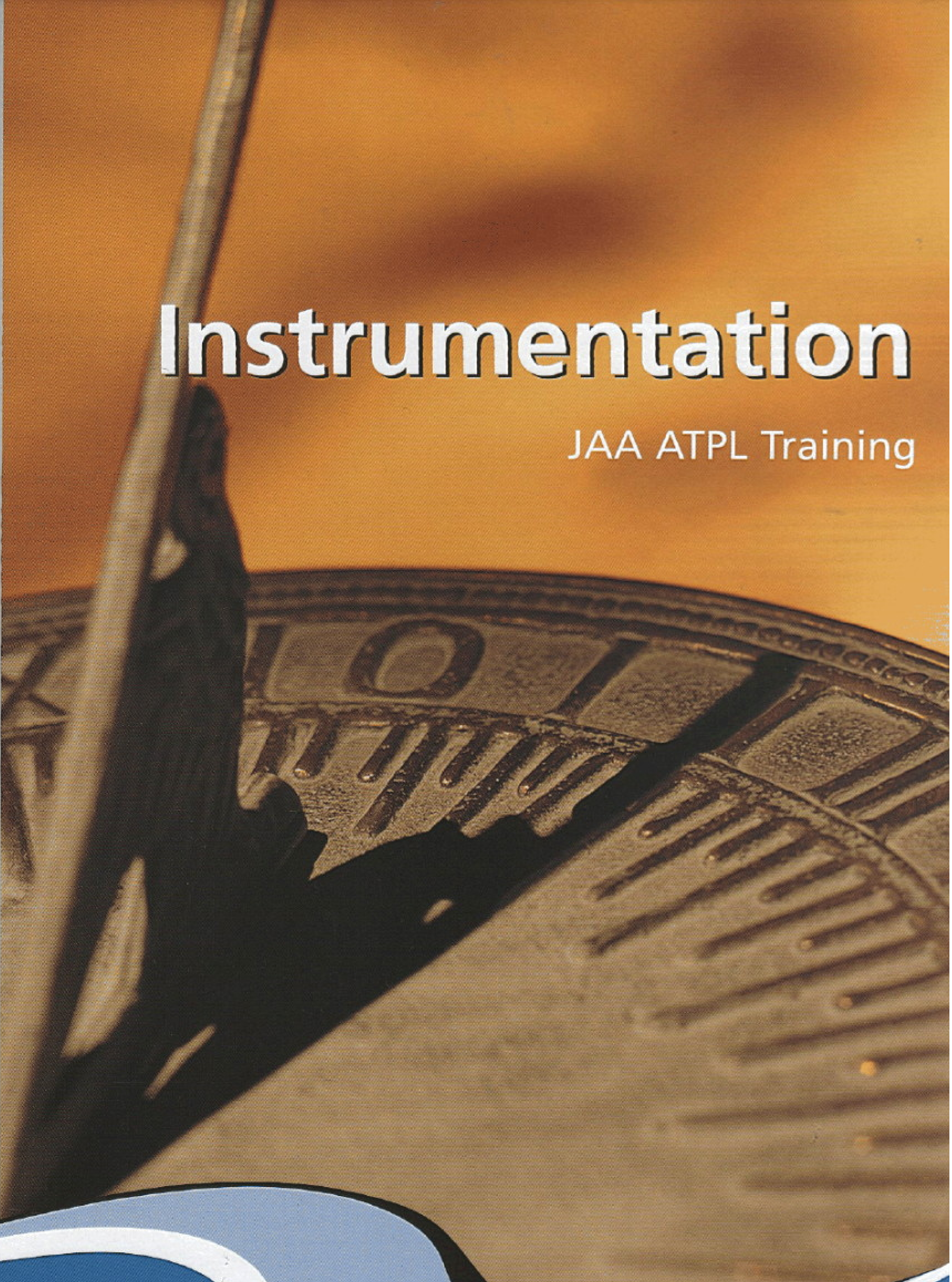

Instrumentation

JAA ATPL Training

Atlantic Flight Training Ltd

JEPPESEN

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Contact Details:

Sales and Service Department
Jeppesen GmbH
Frankfurter Strasse 233
63263 Neu-Isenburg
Germany

Tel: ++49 (0)6102 5070
E-mail: fra-services@jeppesen.com

For further information on products and services from Jeppesen, visit our web site at:
www.jeppesen.com

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PREFACE

As the world moves toward a single standard for international pilot licensing, many nations have adopted the syllabi and regulations of the "Joint Aviation Requirements-Flight Crew Licensing" (JAR-FCL), the licensing agency of the Joint Aviation Authorities (JAA).

Though training and licensing requirements of individual national aviation authorities are similar in content and scope to the JAA curriculum, individuals who wish to train for JAA licences need access to study materials which have been specifically designed to meet the requirements of the JAA licensing system. The volumes in this series aim to cover the subject matter tested in the JAA ATPL ground examinations as set forth in the ATPL training syllabus, contained in the JAA publication, "JAR-FCL 1 (Aeroplanes)".

The JAA regulations specify that all those who wish to obtain a JAA ATPL must study with a flying training organisation (FTO) which has been granted approval by a JAA-authorised national aviation authority to deliver JAA ATPL training. While the formal responsibility to prepare you for both the skill tests and the ground examinations lies with the FTO, these Jeppesen manuals will provide a comprehensive and necessary background for your formal training.

Jeppesen is acknowledged as the world's leading supplier of flight information services, and provides a full range of print and electronic flight information services, including navigation data, computerised flight planning, aviation software products, aviation weather services, maintenance information, and pilot training systems and supplies. Jeppesen counts among its customer base all US airlines and the majority of international airlines worldwide. It also serves the large general and business aviation markets. These manuals enable you to draw on Jeppesen's vast experience as an acknowledged expert in the development and publication of pilot training materials.

We at Jeppesen wish you success in your flying and training, and we are confident that your study of these manuals will be of great value in preparing for the JAA ATPL ground examinations.

The next three pages contain a list and content description of all the volumes in the ATPL series.

ATPL Series

Meteorology (JAR Ref 050)

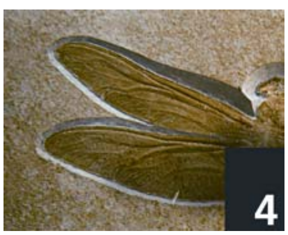
- The Atmosphere
- Wind
- Thermodynamics
- Clouds and Fog
- Precipitation
- Air Masses and Fronts
- Pressure System
- Climatology
- Flight Hazards
- Meteorological Information

General Navigation (JAR Ref 061)

- Basics of Navigation
- Magnetism
- Compasses
- Charts
- Dead Reckoning Navigation
- In-Flight Navigation
- Inertial Navigation Systems

Radio Navigation (JAR Ref 062)

- Radio Aids
- Self-contained and External-Referenced Navigation Systems
- Basic Radar Principles
- Area Navigation Systems
- Basic Radio Propagation Theory



Airframes and Systems (JAR Ref 021 01)

- Fuselage
- Windows
- Wings
- Stabilising Surfaces
- Landing Gear
- Flight Controls
- Hydraulics
- Pneumatic Systems
- Air Conditioning System
- Pressurisation
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- Engine Construction
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- Batteries
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- Automatic Flight Control Systems
- Warning and Recording Equipment
- Powerplant and System Monitoring Instruments

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- Aerofoil Airflow
- Aeroplane Airflow
- Lift Coefficient
- Total Drag
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- Stall
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- Operating Limitations
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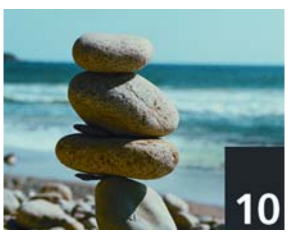
- Multi-Engine Aeroplanes – Not certified under JAR/FAR 25 (Performance Class B)
- Aeroplanes certified under JAR/FAR 25 (Performance Class A)



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- Definition and Terminology
- Limits
- Loading
- Centre of Gravity

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- ICAO ATC Flight Planning
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Chapter 1
Air Temperature Measurement

INTRODUCTION

The temperature of air under pure static conditions at the various flight levels is one of the basic parameters required to establish data that is vital to the performance monitoring of modern aeroplanes. The measurement of Static Air Temperature (SAT) by direct means is, however, not possible for all types of aeroplanes, because the measurement is potentially affected by adiabatic compression of the air at increased airspeeds. The boundary layer over the outer surface of an aeroplane flying below Mach 0.2 is very close to SAT, but at higher Mach numbers the boundary layer slows or even stops relative to the aeroplane. This in turn results in adiabatic compression, which causes the air temperature to rise well above SAT. This increase is known as **RAM Rise**, and the temperature indicated under such conditions is known as Ram Air Temperature (RAT).

RAT = SAT + RAM Rise or SAT = RAT - RAM Rise

The majority of RAM Rise is due to adiabatic compression because of the airflow coming to rest, with only a relatively small amount due to friction between the surface of the aeroplane and the high-speed airflow. The RAM Rise is always pre-calculated and is tabulated or graphed as a function of Mach number in the Operations or Flight Manual for each type of aeroplane, a sample of which is shown below.

	INDICATED MACH NUMBER										
	.30	.40	.50	.60	.70	.73	.76	.78	.80	.82	.84
IND TAT – °C	TRUE OUTSIDE AIR TEMPERATURE – DEGREES C										
70				47	39	37	35	33	31	29	27
65			49	42	35	33	30	28	26	25	23
60		49	44	37	30	28	25	24	22	21	19
55	49	45	40	33	26	24	21	19	18	16	14
50	45	40	35	28	21	19	17	15	13	11	10
45	40	35	30	23	17	15	12	11	9	7	5
40	35	30	25	19	12	10	8	6	4	3	1
35	30	26	20	14	8	6	3	1	0	– 2	– 3
30	25	21	16	10	3	1	– 1	– 3	– 5	– 6	– 7
25	20	16	11	5	– 2	– 3	– 6	– 7	– 9	–11	–12
20	15	11	6	0	– 6	– 8	–10	–12	–13	–15	–16
15	10	6	2	– 5	–11	–13	–15	–16	–18	–19	–21
10	5	1	– 3	– 9	–15	–17	–19	–21	–22	–24	–25
5	0	– 3	– 8	–14	–20	–21	–24	–25	–27	–28	–29
0	– 5	– 8	–13	–18	–24	–26	–28	–30	–31	–33	–34
– 5	–10	–13	–18	–23	–29	–31	–33	–34	–35	–37	–38
–10	–15	–18	–22	–28	–33	–35	–37	–39	–40	–41	–43
–15	–20	–23	–27	–32	–38	–39	–42	–43	–44	–46	–47
–20	–24	–27	–32	–37	–42	–44	–46	–47	–49	–50	–51
–25	–29	–32	–36	–42	–47	–49	–51	–52	–53	–55	–56
–30	–34	–37	–41	–46	–51	–53	–55	–57	–58	–59	–60
–35	–39	–42	–46	–51	–56	–58	–60	–61	–62	–63	–65
–40	–44	–47	–51	–56	–61	–62	–64	–65	–66	–68	–69

The proportion of RAM rise measured is dependent on the ability of the sensor to sense (or recover) the temperature rise. The sensitivity is expressed as a percentage and is known as the recovery factor. For example, if a sensor has a recovery factor of 0.80, it measures SAT + 80% of the RAM Rise.

A useful approximation of RAM Rise is:

$$\left(\frac{\text{TAS kt}}{100}\right)^2$$

For example, for an estimated TAS of 460 kt deduct $\left(\frac{460\text{ kt}}{100}\right)^2$ from the indicated air temperature, which is approximately equal to 21°C. If this figure is subtracted from the Indicated Outside Air Temperature (IOAT), it gives the Corrected Outside Air Temperature (COAT), which is the best determinable value of the temperature of the air through which the aeroplane is flying, and is that which is required for navigational purposes:

$$\text{COAT} = \text{IOAT} - \text{RAM Rise}$$

Alternatively written as:

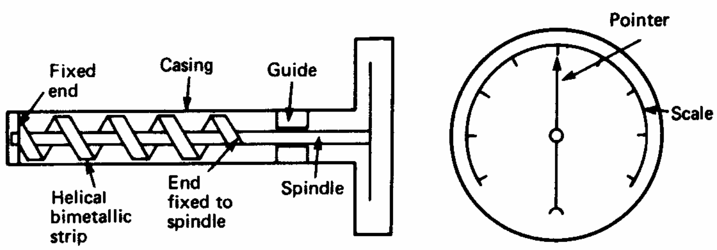
$$T_S = \frac{T_M}{1 + 0.2KM^2}$$

Where: T_S = SAT in degrees absolute.
T_M = Indicated temperature in degrees absolute.
K = Recovery Factor.
M = Mach Number.

Various types of air temperature sensors are fitted to aeroplanes, although the particular type is dependent on whether SAT or RAT is required, and the normal operating speed of the aeroplane on which it is fitted.

DIRECT READING THERMOMETER

This is the simplest type of thermometer, and it only indicates SAT. It consists of a bi-metallic element in the shape of a helix, which expands and contracts when subjected to temperature changes.

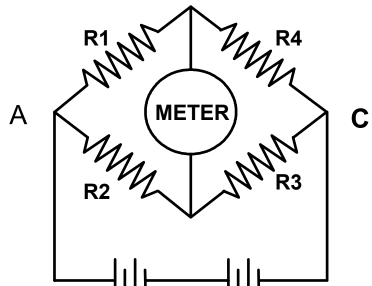


As the temperature changes, the helix winds or unwinds, and causes a pointer to rotate against a scale calibrated in degrees Celsius. This type of thermometer is normally only fitted on low-speed light aeroplanes, and passes through the fuselage or canopy with the sensitive element projecting into the airflow. The element is also shielded so that it is not directly affected by solar radiation.

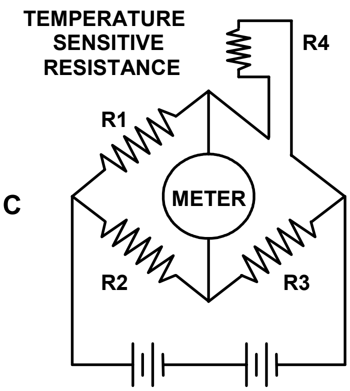
Another method of making a direct reading thermometer is to use a liquid filled bulb and capillary tube, with the bulb mounted externally in the airflow and the capillary tube transmitting the liquid pressure to a Bourdon tube. As the temperature to which the bulb is subjected changes, the liquid expands or contracts, and causes the Bourdon tube to wind or unwind. This movement of the Bourdon tube is connected to a pointer, which moves around a scale calibrated in temperature.

ELECTRICAL THERMOMETER

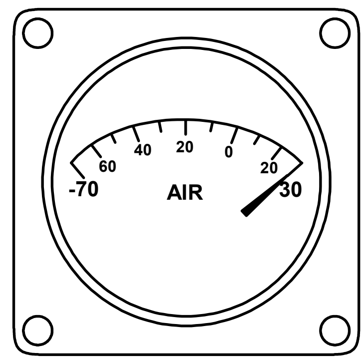
The resistance of an electrical conductor is dependent on temperature, and its magnitude changes due to thermal changes, thus altering its ability to oppose current flow when connected in a circuit. The Electrical (Wheatstone Bridge) Thermometer commonly uses the temperature/resistance technique.



WHEATSTONE BRIDGE
BASIC CIRCUIT
(1)



CIRCUIT FOR MK. 1
AND MK. 1A THERMOMETERS
(2)

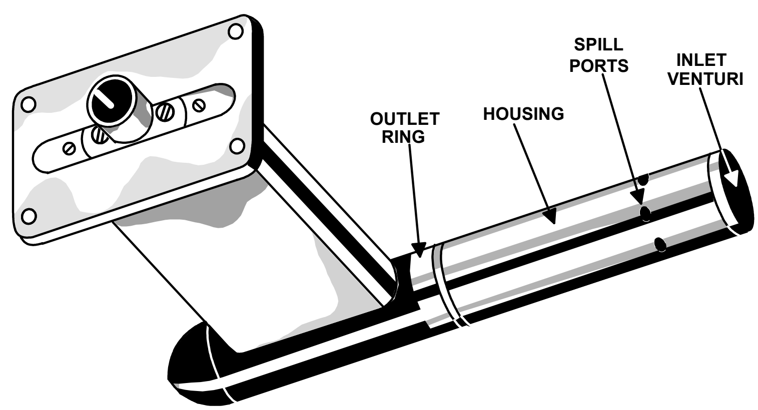


INDICATOR F ANDOR MK. 1
MK. 2 THERMOMETERS
(3)

The bridge consists of four resistance arms, which connect across a low voltage source as shown above. When the system is switched on, current flows in the circuit and divides at point A before flowing through R1 and R2 at strengths that vary as the temperature of R4 (positioned outside the fuselage) alters. At point C the currents re-unite and flow back to the voltage source. If the Bridge is balanced resistance-wise, no current flows and the moving coil galvanometer (meter) reads zero. If the temperature (resistance) of R4 changes, the bridge becomes unbalanced, and the resulting current flowing through the galvanometer registers as a change in temperature.

STATIC AIR TEMPERATURE SENSOR

The majority of temperature sensors use a platinum wire element contained either in a probe mounted in what is termed a **flush bulb configuration**, or in a specially designed probe shielded from solar radiation. A flush bulb configuration is subjected to both air temperature and skin temperature and is, therefore, less accurate than a probe type sensor.

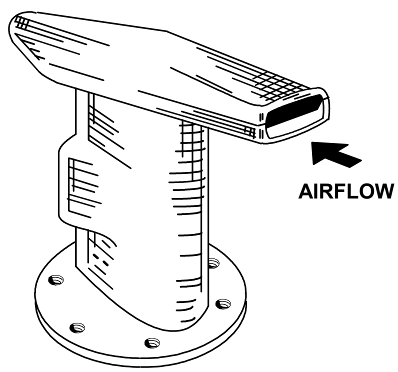


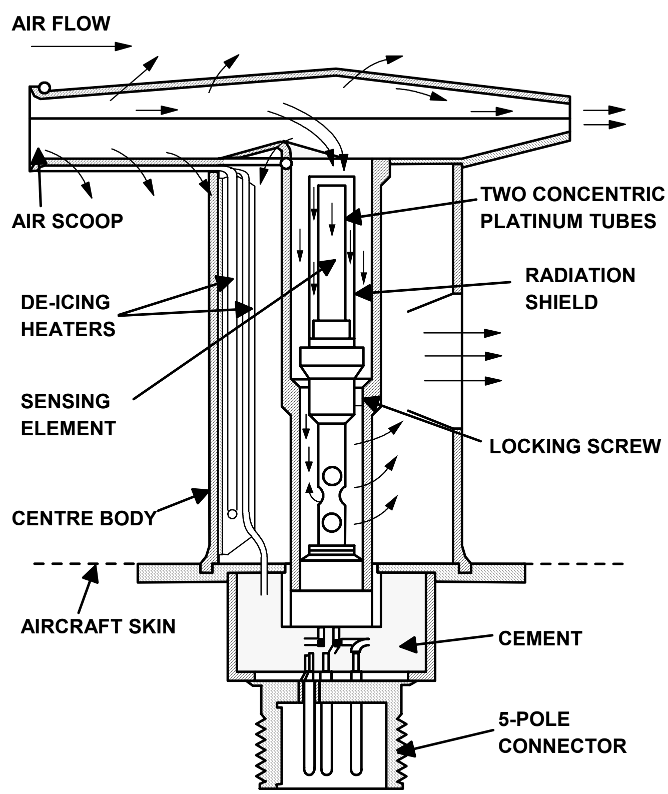
In the type of sensor shown above the probe protrudes through a hole in the aeroplane skin with the orifice facing forward into the airflow. Air enters the probe and comes to rest where a sensing element forms one leg of a Wheatstone bridge. The air flows out from the probe through the spill ports allowing a limited rate of change of air measured at the element.

Sometimes known as a total temperature head, this probe is less accurate than the total air temperature probe described below and is usually only fit to simpler and or slower aircraft where RAM rise is limited. The ability to sense RAM rise accurately is known as the recovery factor, and this type of probe has typical values of between 0.75 and 0.95.

TOTAL AIR TEMPERATURE PROBE

For aeroplanes operating at high Mach numbers, it is usual to sense and measure the maximum temperature rise possible, which is Total Air Temperature (TAT) if the recovery factor is 1, and Ram Air Temperature (RAT) if the recovery factor is less than 1. TAT or RAT is the temperature of the air when it is brought to rest (or nearly so) without the addition or removal of heat. The Rosemount probe shown below has a recovery factor of virtually 1, and is, therefore, known as a TAT probe, and is commonly used on transport-category aeroplanes.





The advantages of this type of thermometer over the flush bulb type are that it has a virtually zero time lag, and also has a recovery factor of approximately 1. This type of probe is normally connected directly to a flight deck indicator, and also to the Mach number module of an Air Data Computer (ADC).

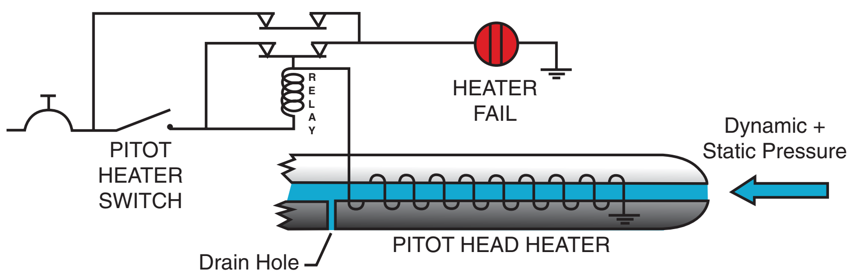
An air intake, which is mounted on top of a small streamlined strut, is secured to the aeroplane skin at a predetermined location around the nose, where it is free from any boundary layer activity. Air flows through the probe and over the sensing elements before it is expelled through a vent at the back of the strut, which allows the probe to continuously sample new air. Separation of water droplets from the air is achieved by causing the air to turn through 90° before it passes over the sensing element. Boundary layer air is drawn off by bleed holes in the casing. This layer of air is due to the pressure differential which exists across the casing. A pure platinum resistance wire, which is sealed within two concentric platinum tubes, is used to sense the temperature, and a heating element is mounted on the probe to prevent any ice forming. The heater has a minimal effect on the indicated temperature readings, with typical values of 0.9°C at Mach 0.1 and 0.15°C at Mach 1.0.

INTRODUCTION

The pitot and static systems on an aeroplane measure the total pressures created by the forward motion of the aeroplane, and the static pressure surrounding the airframe. These pressures are fed to instruments, which convert the pressure differentials into speed, altitude, and rate of change of altitude. The system is alternatively referred to as a **manometric** or **air data system**.

PITOT TUBE

The Pitot tube (pitot probe) senses the total or pitot pressure, which is the combined static and dynamic pressure of the airflow. The tube is fitted to the airframe with its opening facing directly into the airflow, and the airflow comes to rest (i.e. stagnates) inside the tube entrance.

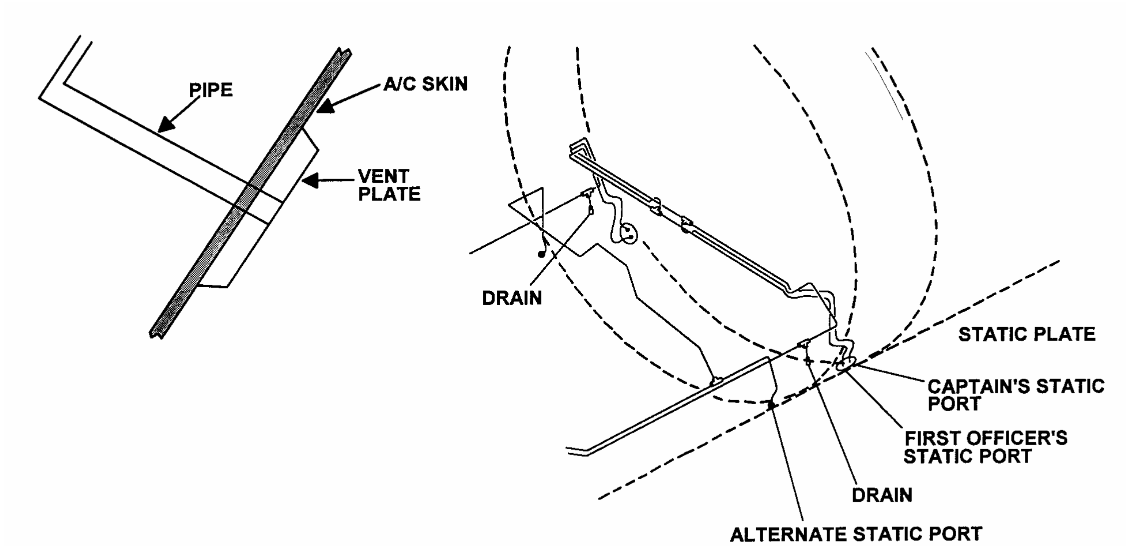


A pinhole drain allows any moisture to leak away to atmosphere without significantly affecting the sensed pressure. Because moisture can accumulate within the pipeline due to condensation, drain traps are positioned at various intervals throughout the pipe work of the aircraft, usually at low points along certain pipe runs. These drain traps contain any accumulation of water in a bowl which can be released by operating a spring loaded plunger on the underside of the bowl. After operations of these drain traps, the integrity of the pitot system requires verification by carrying out a leak test on the system. Therefore, drain traps are only operated by ground technicians during maintenance.

The probe is mounted on a part of the aeroplane where there is minimal disturbance to the airflow, and is designed to extend well forward into the airflow. These probes are typically mounted close to the nose, at the wing tips, on a pylon extending well below the wing, or at the top of the fin. The probe is also fitted with a heater, which is powered from the aeroplane electrical supply (usually 28 Volt DC or 115 Volt AC), and is switched on as required by the flight deck crew to prevent the formation of ice. An indicator light gives the operative state of the system. Some types show an amber light when switched OFF, or alternatively with the system switched ON and the heating element failed. Most transport category aeroplanes have at least two pitot tubes.

STATIC SOURCE

The ambient pressure of the air mass surrounding the aeroplane, or **static pressure**, is obtained via a static source. The static source, or **static vent**, senses the static pressure of the atmosphere, which is unaffected by the airflow. To achieve this, the source (vent) is located on a part of the aeroplane where the airflow is undisturbed by its passage, and is positioned with its entrance perpendicular to the airflow. The vent is manufactured and attached to the surface of the aeroplane so that it does not create local disturbances in the airflow. A typical design is shown below.

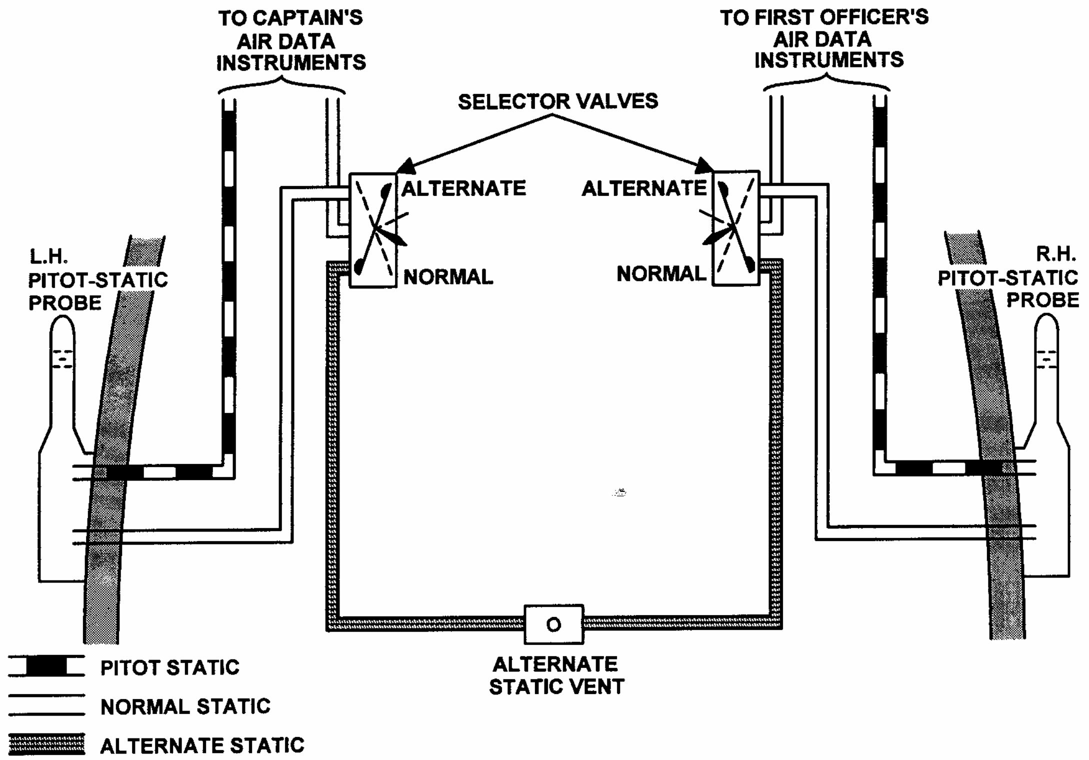


Vent pipe connections are installed with a slight downward angle to ensure adequate drainage, and as with the pitot system, drain traps are provided to contain any water condensed into the system. It is important that the vent plates are not painted as this would impair their thermal efficiency and a placard is often placed next to the static vent saying “do not paint”. The direction of the airflow around the static vent may vary as the airspeed and configuration of the aeroplane changes, and may induce errors known as position (or pressure) errors. Minimisation of these errors is achieved by carefully positioning the static vent, or by using multiple vents to average out the errors. This is known as **static balancing**, and is achieved by fitting vents on either side of the aeroplane fuselage. The purpose of this is to even out any differences of pressure that are caused by the sideways motion of the static vents, which occurs during a yaw or sideslip condition. Any residual position (pressure) errors are recorded during initial flight tests and a correction table is produced, for various airspeeds and configurations. These readings are incorporated into the Aeroplane Operating Manual (AOM).

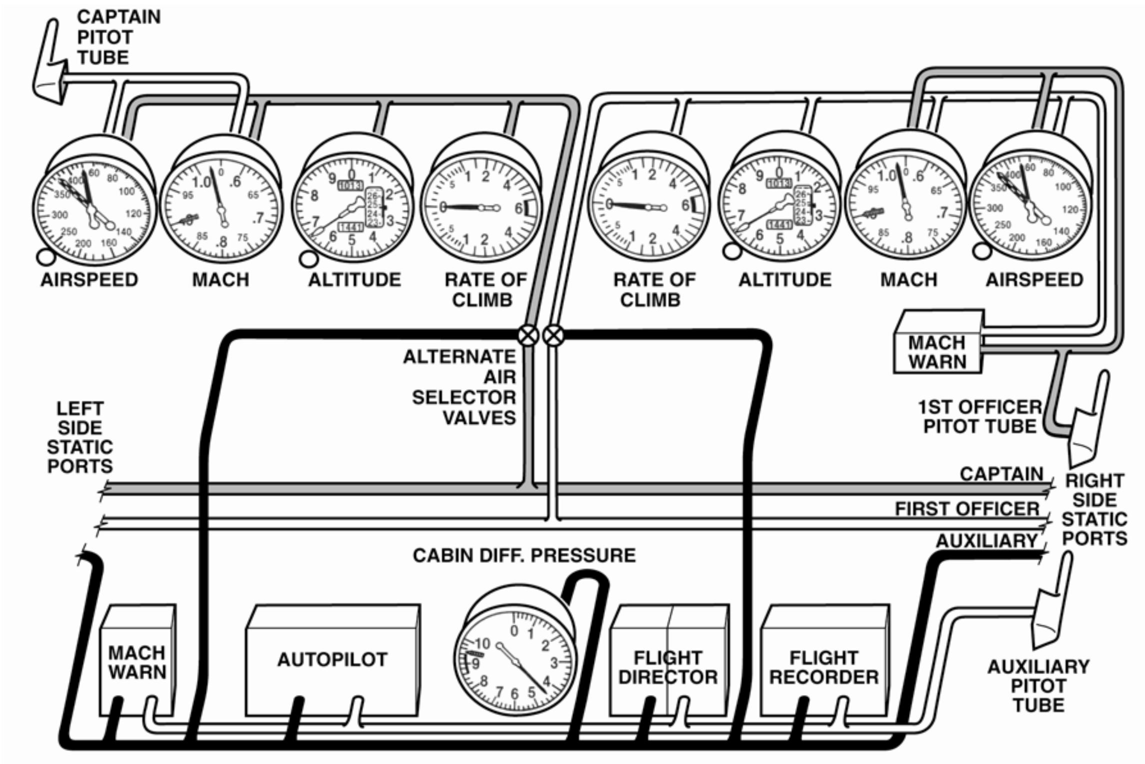
If failure of the primary pitot/static pressure source should occur, for example icing up of a pitot or pressure head due to a failed heater circuit, introduction of errors may occur in the instrument readings and other areas dependent on such pressure. The installation of a standby system in some aeroplanes acts as a safeguard against partial failure, whereby static pressure and/or pitot pressure from alternate sources is usually selected and connected into the primary system. A blockage of the pitot source only affects the ASI. A blockage of the static source affects all of the instruments, and it is thus common practice to provide an alternate static supply.

ALTERNATE STATIC SOURCE

The changeover to an alternate static source is normally achieved by selector valves located in the static lines, which are located on the flight deck, within easy reach of the flight crew. A typical internal alternate static source installation is shown below. Such a system only operates satisfactorily if the cabin is unpressurised and the air within the cabin is relatively undisturbed. When calibrating the pressure/position errors of the alternate system, the manufacturer lays down the conditions required in respect of the position of such items as windows, heating/ventilation, and doors, all of which must be observed if the system is to work correctly.



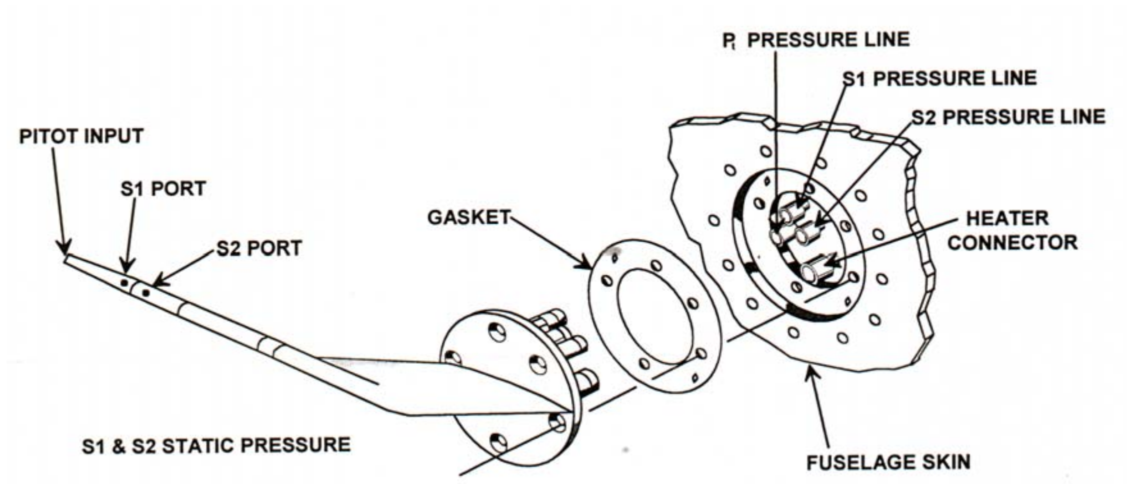
Moderate or large aeroplanes normally have a minimum of two separate static systems, and a pair of balanced static vents, as shown below, feeds each.



The second pair of static vents is normally referred to as the auxiliary static ports, whose precise location is determined during prototype development.

COMBINED PITOT-STATIC (PRESSURE) HEAD

In some light aeroplanes the complication and expense of separate pitot head(s) and static sources is often avoided by incorporating both functions into a combined pressure head. The combined head is usually mounted on a pylon below the wing toward the tip, thus reducing the position/pressure error to an acceptable level although accuracy is well below the standards achieved by systems that are more sophisticated. A typical combined pressure head is in the following illustration.



Static pressure is admitted to the pressure head through slots or holes cut into the static casing at 90° to the airflow, whilst pitot pressure enters the head via the pitot input port that faces directly into the airflow. The different pressures are fed to the flight instruments via separate, seamless, and corrosion-resistant metal pipelines. An electric heating element is also connected to the aeroplane electrical supply to prevent ice forming inside the pressure head, which may obstruct the airflow.

This sensor is particularly prone to errors during manoeuvres, when changes in pitch can result in the pitot/static sources being presented to the relative airflow in such a way that the full dynamic pressure is not sensed. More significantly, dynamic pressure effects may also intrude into the static supply.

OPERATING PROBLEMS

Any ingestion of water is normally cleared by the use of drain holes, although these are unable to cope with very heavy precipitation. If the ingested water freezes, ice blocks the pressure head. It is, therefore, fitted with a heater. Checking the heater's function is part of pre-flight checks. Blockages may also occur due to the ingestion of dirt, insects, etc, although the effect of blockages is reduced if the systems are duplicated. Where water accumulates due to heavy precipitation or to condensation over a prolonged period, drain traps are incorporated into the system at various low points along the pitot static pipelines. They take the form of clear plastic bowls with a spring loaded drain plug at the bottom, which is occasionally pushed in against the spring pressure to release the accumulated water. As there is no sure way of telling if the drain plug has fully resealed, the ground engineers carry out a pitot and/or static leak test. This is not a task that the aircrew should carry out.

PITOT/STATIC SYSTEM ERRORS

The following errors may occur in pitot/static systems:

Pressure (Position) error
This is composed of errors due to change of configuration and aircraft manoeuvring. The errors due to change of configuration are determined during aircraft flight testing and are tabulated in the Flight Manual. Manoeuvring induced errors are random in nature and are not corrected.

Changes in configuration (flaps, under-carriage, etc.)
This may cause localised turbulence around the static vents and change the pressure sensed at the static vents.

Manoeuvre induced errors
These result from dynamic changes that produce a short-term variation of pressure at the static vents.

Turbulence
This does not fall into the pressure error category; however, it can result in random pressure fluctuations at the air pressure sensors, which in turn cause fluctuating indications of the pressure instruments.

Use of alternate static vents
These usually result in a different pressure error profile. If the alternate static vent is within the non-pressurised portion of an aeroplane, the sensed static pressure will tend to be higher than it should be, depending on the fan setting and window positioning, etc.

Chapter 3
Pressure Altimeter

INTRODUCTION

The Earth is surrounded by a gaseous envelope, which is divided into several concentric layers that extend outward from the Earth's surface, up to a height of approximately 900 km (500 miles). The lowest layer is the Troposphere, and extends to an altitude of approximately 36 000 ft in temperate latitudes, whilst the outer layer is the Exosphere.

The atmosphere is held in contact with the Earth by the force of gravity, which decreases steadily with increasing distance from the Earth's surface, as does the atmospheric pressure (e.g. at the Tropopause, the pressure is approximately one quarter of its sea level value). Air density is directly proportional to atmospheric pressure and similarly reduces with increasing altitude. Another factor which affects the atmosphere is temperature, which steadily decreases with increasing altitude up to the Tropopause (36 090 ft ISA), and thereafter remains constant.

Relative to altitude, these variables are difficult to continuously measure and compensate for, so the International Civil Aviation Organisation (ICAO) formulated a table in which the values of pressure, temperature, and density vary at a prescribed rate (with altitude). This table is accepted internationally as a reference (datum) and is known as the International Standard Atmosphere (ISA), against which aeroplane performance is usually compared, and air data instruments calibrated. The following table shows how pressure, temperature, and density vary with increasing altitude in the International Standard Atmosphere (ISA).

Altitude		Pressure	Temperature	Density
(metres)	(feet)	(hPa)	(°C)	(kg/m^3)
0	0	1013.25	15	1.225
2000	6562	795	2	1.007
4000	13124	612	-11	0.819
6000	19686	472	-24	0.660
8000	26248	357	-37	0.526
10000	32810	265	-50	0.414
12000	39372	194	-56.5	0.312
14000	45934	142	-56.5	0.288
16000	52496	104	-56.5	0.166

Pressure altimeters or aneroid barometers make use of the fact that the pressure in the atmosphere decreases with increasing altitude, and are calibrated in terms of ISA to show altitude instead of barometric pressure.

PRESSURE ALTITUDE

Pressure altitude is the altitude above the standard datum 1013.25 hPa or mb (29.92 inHg).

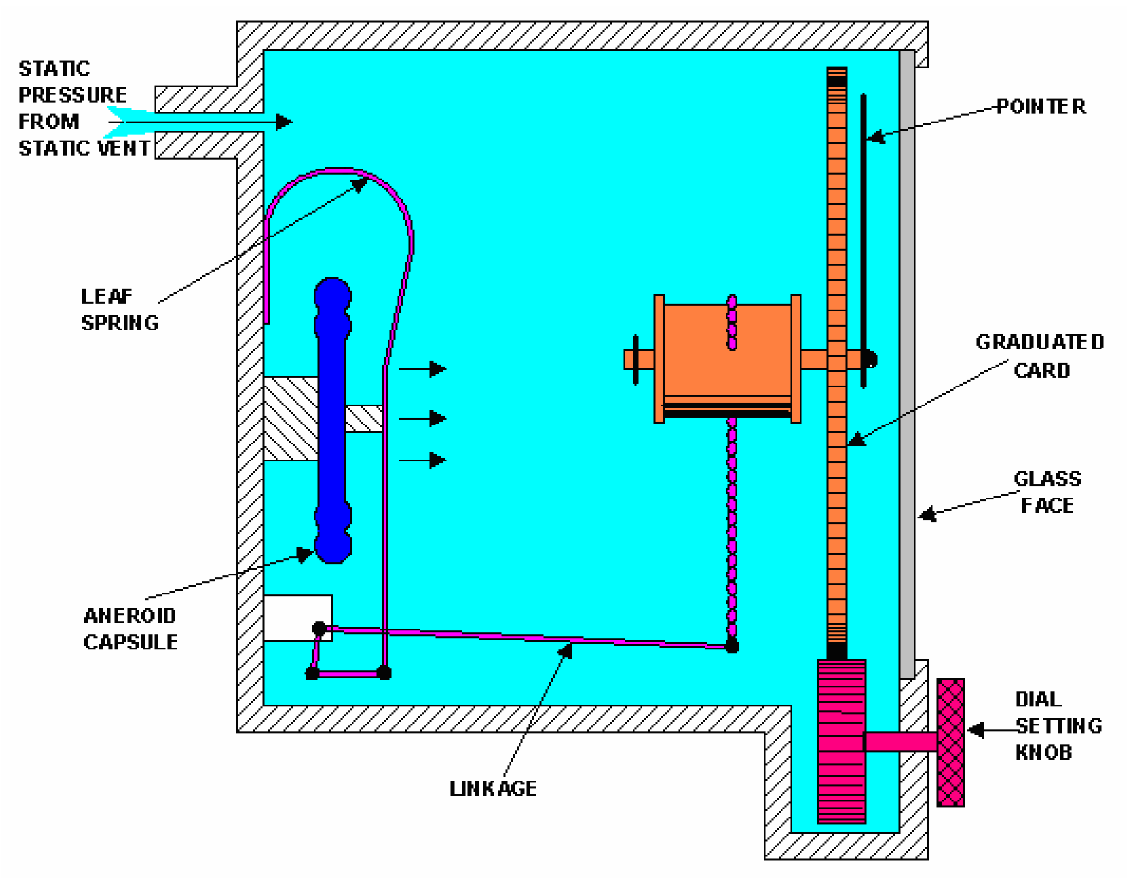
DENSITY ALTITUDE

Density altitude is pressure altitude corrected for temperature. Pressure and density are the same when conditions are standard. As the temperature rises above standard, the density of the air decreases and the density altitude increases.

THE SIMPLE ALTIMETER

The altimeter is an instrument that is designed to measure static pressure and, using the conditions of the standard atmosphere, convert that pressure into a value of altitude (e.g. if the pressure measured is 472 hPa, then a calibrated altimeter indicates 19 686 ft).

A simple or non-sensitive altimeter comprises of a partially evacuated barometric (aneroid) capsule, a leaf spring, a mechanical linkage and a pointer, as shown below.



These components are all housed in a container, which is supplied with static pressure from the static vent system. As an aeroplane climbs, the static pressure decreases, which causes the capsule to expand and drive the pointer to indicate a higher altitude. As an aeroplane descends, the capsule compresses due to the increasing static pressure, and the pointer indicates a lower altitude. The leaf spring in the instrument is designed to prevent the capsule from collapsing and acts as a balance between the capsule and the static pressure. The altimeter is normally calibrated at ISA +15°C and 1013.2mb (hPa) until it reads zero, although adjusting the datum is possible via the sub-scale dial setting knob. This allows the adjustment of the instrument for different values of mean sea level (msl), and heights above an aerodrome at which the altimeter reads zero.

DATUM SUB-SCALE SETTINGS

The setting of altimeters to datum barometric pressures forms part of the flight operating procedures, and is essential for maintaining adequate separation between aeroplanes and terrain clearance during take-off and landing. These settings are adopted universally and form part of the ICAO **Q** code of communication, which consists of three-letter groups, each having Q as the first letter. The codes normally used in relation to altimeter settings are:

QFE

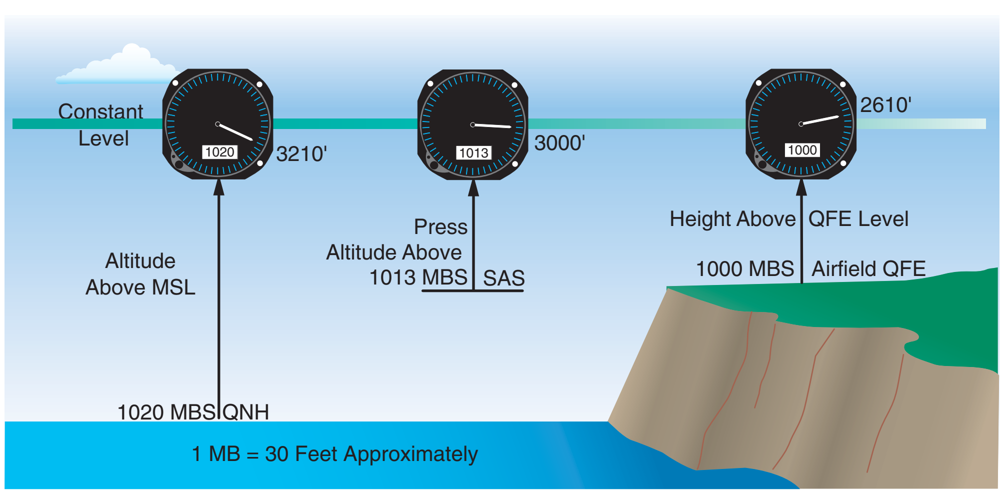
This is the pressure prevailing at an airfield, the setting of which on the altimeter sub-scale causes the altimeter to read zero on landing and take-off.

QNE

QNE is not an altimeter setting, but is the indicated height above the standard pressure setting when on the ground. This is significant at high-altitude aerodromes where QFE may be outside the range of the altimeter baro-scale. When QNE is set, the altimeter indicates the pressure altitude, which is the reported flight level. Flight levels occur at 500 ft intervals and are calculated by dividing the pressure altitude by 100 (e.g. a pressure altitude of 10 000 ft will equate to FL100).

QNH

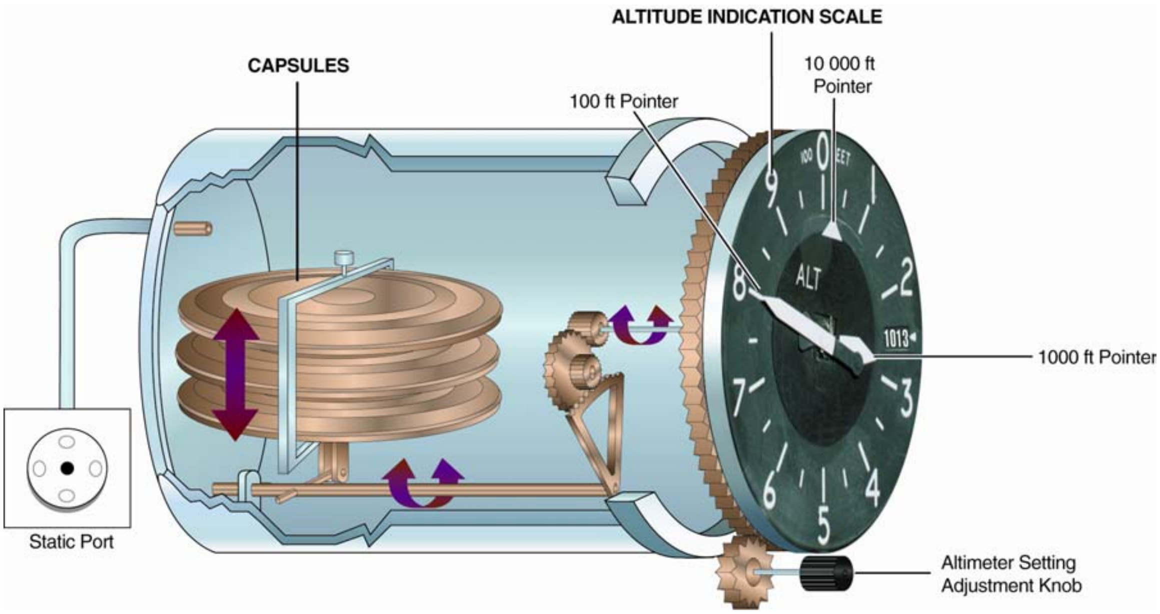
An altimeter setting equivalent to the barometric pressure measured at an aerodrome altimeter datum and corrected to sea level pressure. At the airport altimeter datum, an altimeter set to QNH indicates airport elevation.



If the aeroplane remains at the same altitude, winding on hectopascals or millibars winds on more height and vice versa.

THE SENSITIVE ALTIMETER

The sensitive altimeter is essentially the same as the simple altimeter but employs a minimum of two aneroid capsules. This provides for a more accurate measurement of pressure and provides more power to drive the mechanical linkage.



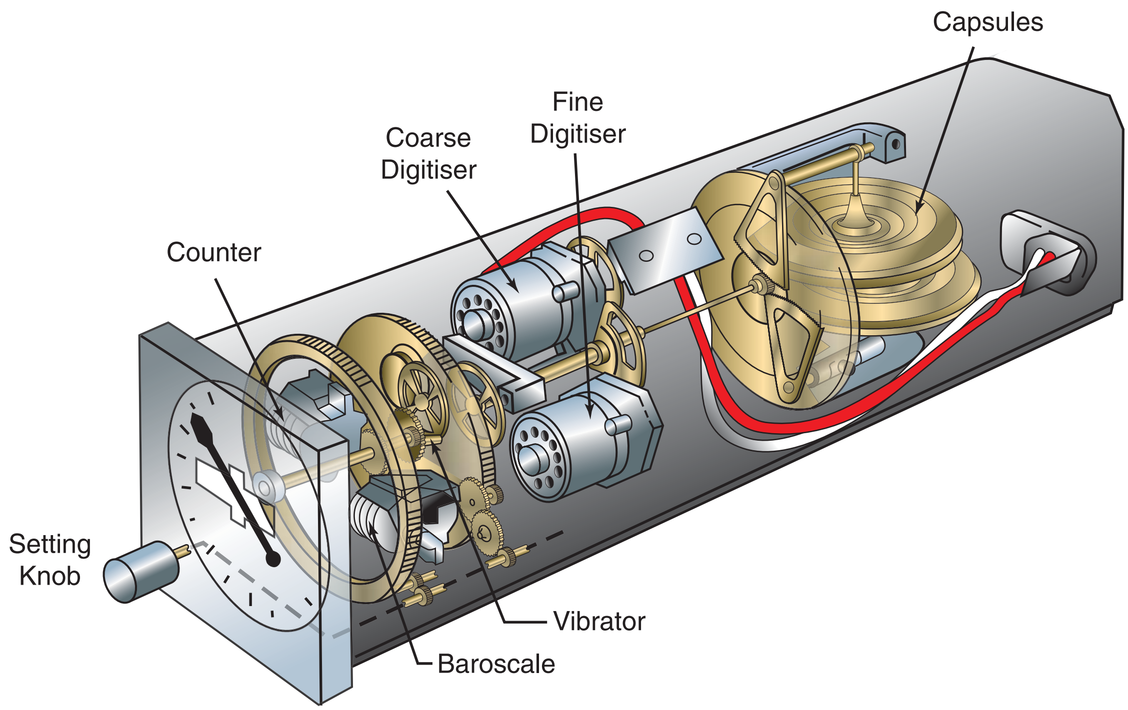
The capsules are stacked together with one face fastened down, which permits movement due to pressure changes at the other end. The movement of the capsules in response to changes in altitude (pressure) is transmitted via a suitable mechanical linkage to three pointers that display (against a graduated instrument scale) the aeroplane altitude in tens, hundreds, and thousands of feet. The whole assembly is encased in a container, which is fed with static pressure, but is otherwise completely airtight. Within the mechanical linkage, a bi-metallic insert is fitted to compensate for temperature changes that could affect the movement. As the aeroplane climbs and air pressure falls, the capsules expand. Similarly, as the aeroplane descends, the static pressure increases and the capsules contract. Since it is necessary to allow for different values of mean sea level pressure and to allow use of the altimeter in indicating altitude above the aerodrome, the altimeter is similarly provided with a means of adjusting the level at which it indicates zero feet. This is done via a barometric subscale mechanism, which adjusts the mechanical linkage and operates a set of digital counters, or calibrated dial. This is displayed in a window in the face of the altimeter, and is the datum pressure setting above which the instrument is now displaying altitude. The desired setting is again made using the knurled knob at the bottom of the instrument.

ALTIMETER DISPLAYS

A number of different types of pressure altimeters are manufactured, though they differ in detail depending on the altitude band covered, the accuracy of the instrument, and the method by which the altitude is displayed. Types of display vary from multi-needle to needle plus digital counters, with accuracy varying from 100 ft at 0 ft to 1000 ft at 35 000 ft in early models, to 35 ft at 0 ft to 600 ft at 60 000 ft in later models.

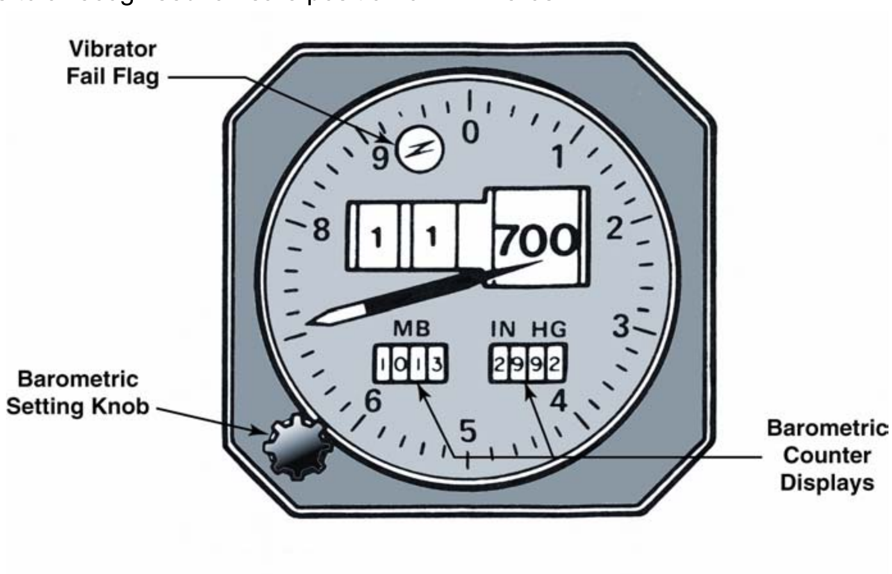
A selection of typical displays is shown above, including an instrument face showing a sector flag, barber pole display, or low-altitude warning sector. The sector flag, which has stripes in black and white, appears in a window when the altitude is 16 000 ft or lower, giving the flight crew a clear warning of approaching low altitudes during rapid descents.

The control of aeroplanes along the many air-routes near modern airports requires that an aeroplane's lateral and vertical position be monitored constantly and accurately to avoid potentially hazardous air traffic situations. To provide an automated transmission of altitude (flight level), two digitisers are normally fitted to modern altimeters. The digitisers are fitted inside the instrument case, as illustrated in the diagram on the next page, and are connected by a common gearing to the main shaft.



The angular position taken up by the rotors of each digitiser relative to its stator determines the value of a pulsed digital signal, which is produced by the assembly. The digital pulses are fed to a code converter where any ambiguity is resolved, and the total reply pulse is modified into a coded response suitable for use by an SSR transponder. The digitised altitude signal is unaffected by changes to the pressure setting on the baro-scale as it is always referenced to a datum pressure of 1013.25 hPa.

Modern altimeters are also fitted with a vibrator assembly, which is designed to reduce the initial opposition to motion of the moving parts, and to reduce any frictional lag in the system. The electrical supply additionally energises a warning flag solenoid in the digitiser-circuit-code converter, which in the event of a power supply failure, is de-energised, and allows a power failure warning flag to appear in an aperture on the dial. At the same time, the code converter also reverts to a recognised fail-safe position of ALL zeros.



DESIGN ERRORS

Altimeters suffer from the following errors:

Instrument Error

Since capsule movements require magnification, it is impossible to ignore the effect of small irregularities in the mechanism. Certain manufacturers' tolerances are accepted, and errors generally increase with altitude.

Pressure Error

Pressure errors arise because the true external static pressure is not accurately transmitted to the instrument. A false static pressure arises because of disturbed airflow near the pressure head or static vent. Pressure error is negligible at low altitudes and speeds, but becomes more significant with increasing airspeed. Correction for pressure error takes the form of a correction, which is applied to the indicated altitude, and is determined by calibration. Air Data Computers are designed to compensate for this type of error.

Time Lag

Because the response of the capsule and linkage is not instantaneous, the altimeter needle tends to lag behind whenever the altitude changes. Subsequent over-indication during descent is possibly dangerous, but is allowed in rapid descents. Time lag is virtually eliminated in the servo-assisted altimeter.

Hysterisis Error

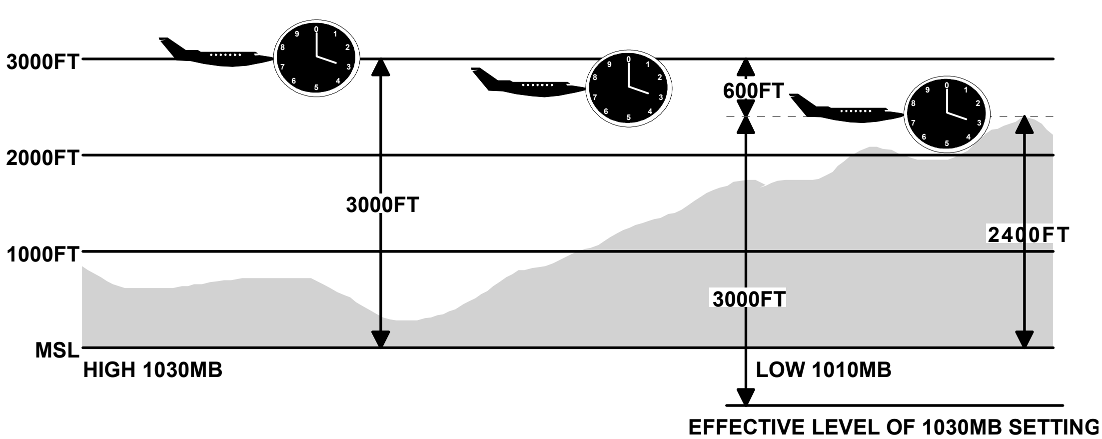
The capsules suffer from hysterisis, which causes a lag in the instrument reading during a climb or descent.

ERRORS DUE TO CALIBRATION

The calibration of the altimeter (i.e. the conversion of ambient (barometric) pressure to readings in ft) is normally based on ISA conditions. If the real atmosphere differs, the altimeter does not indicate the true vertical distance above the sub-scale datum. The most significant errors are:

Barometric Error

Barometric error occurs when the actual datum level pressure differs from that to which the subscale is set. It is caused by the changing ambient barometric pressure experienced during transit and with the passage of time. If the aeroplane flies from an area of high pressure into an area of low pressure it descends even though the altimeter reading remains constant.



The figure above illustrates the effect if the subscale is set to 1030 hPa. A subscale error of 1 hPa is equivalent to an indicated altitude error of 28 to 30 ft. The QNH has reduced to 1010 hPa, which represents an altitude change of approximately 600 ft. The subscale datum is now at a point that is effectively 600 ft below sea level, and the altimeter measures from this level.

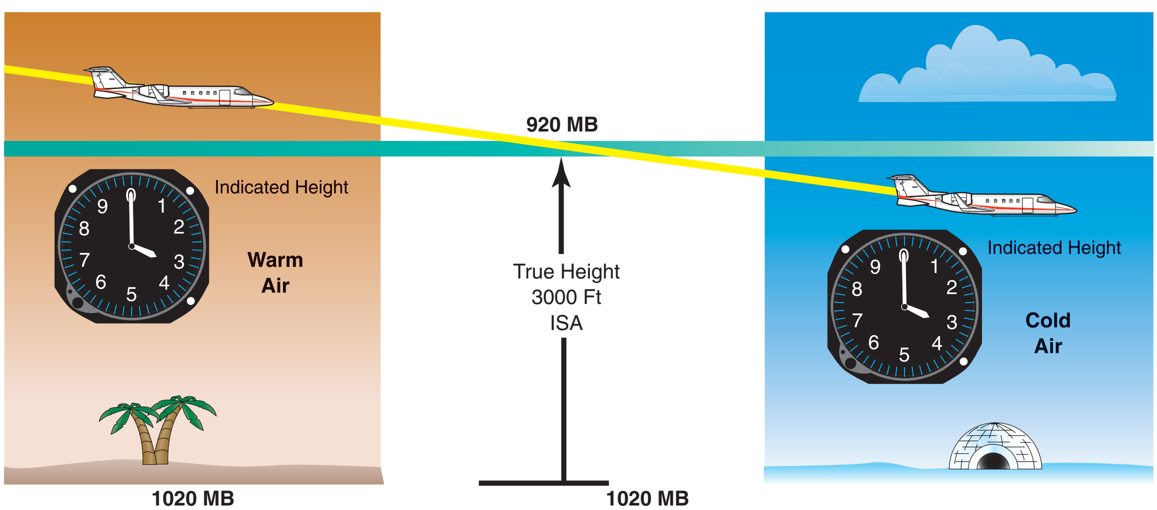
Flying into areas of low pressure is therefore potentially dangerous since the altimeter over-reads, which results in the flight crew over estimating their clearance from obstacles. Conversely, if the aeroplane flies from a low-pressure area into a high-pressure area the altimeter under-reads. When flying at low altitude it is good practice to reset the altimeter periodically to minimise the barometric error.

Orographic Error

Differences from standard may occur when air is forced to rise/descend over hills or mountains. Low pressure tends to occur in the lee of mountains with high pressure on the windward side. Additionally, vertical movement of air can result in change of temperature from ISA, which tends to induce further errors into the altimeter readings.

Temperature Error

Temperature error arises whenever the mean atmospheric conditions below the aeroplane differ from the standard atmosphere. If the actual temperature lapse rate differs from the assumed one, it then causes an incorrect indicated altitude. In general, if the air below the aeroplane is warmer than standard, the air is less dense (low pressure) and the aeroplane is higher than indicated. Conversely if the air is colder than standard, it is more dense (high pressure), and the aeroplane is lower than standard.

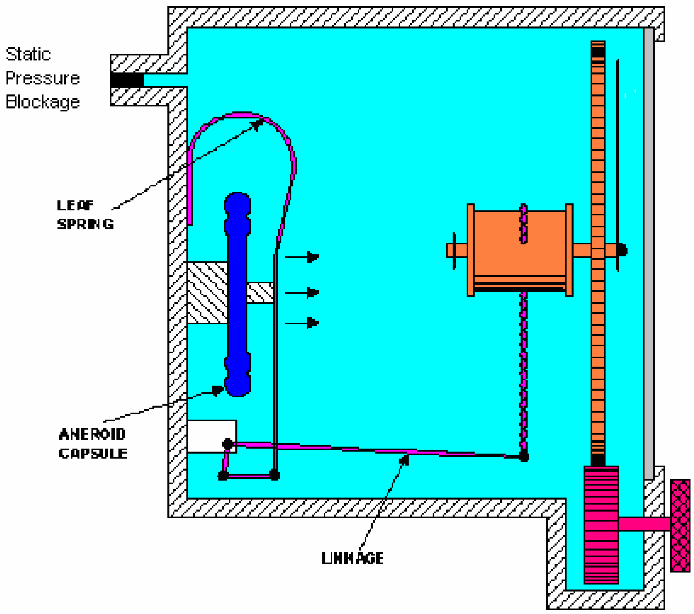


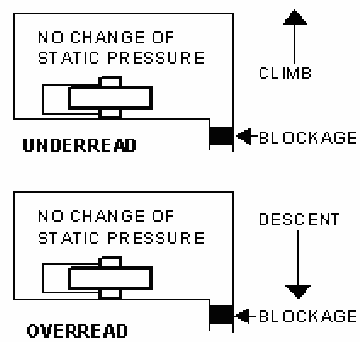
The diagram above illustrates the effect of flying from a warm atmosphere into a cold atmosphere, whilst the surface pressure or subscale setting remains constant. The correct altitude is obtained from that indicated by using the navigation computer. For 'rule of thumb' work, a temperature difference of 10°C from standard results in an error of approximately 4% of the indicated altitude.

BLOCKAGES AND LEAKAGES

If the static tube or vents become blocked, the pressure within the instrument case remains constant and the altimeter continues to indicate the altitude of the aeroplane when the blockage occurred, as shown on the next page.

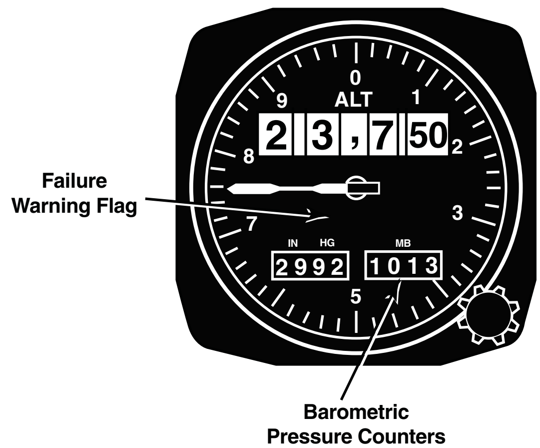
Leaks can also take place either inside or outside the pressure cabin. Within the pressure cabin, the cabin pressure altitude is shown rather than the true altitude. In some aeroplanes, an emergency source inside the fuselage is available. The static pressure inside an aeroplane is normally different from that external to the fuselage, since blowers, ventilation, etc. influence it, so that a different correction for pressure error is necessary. This correction is normally in the Aeroplane Flight Manual.





SERVO ALTIMETERS

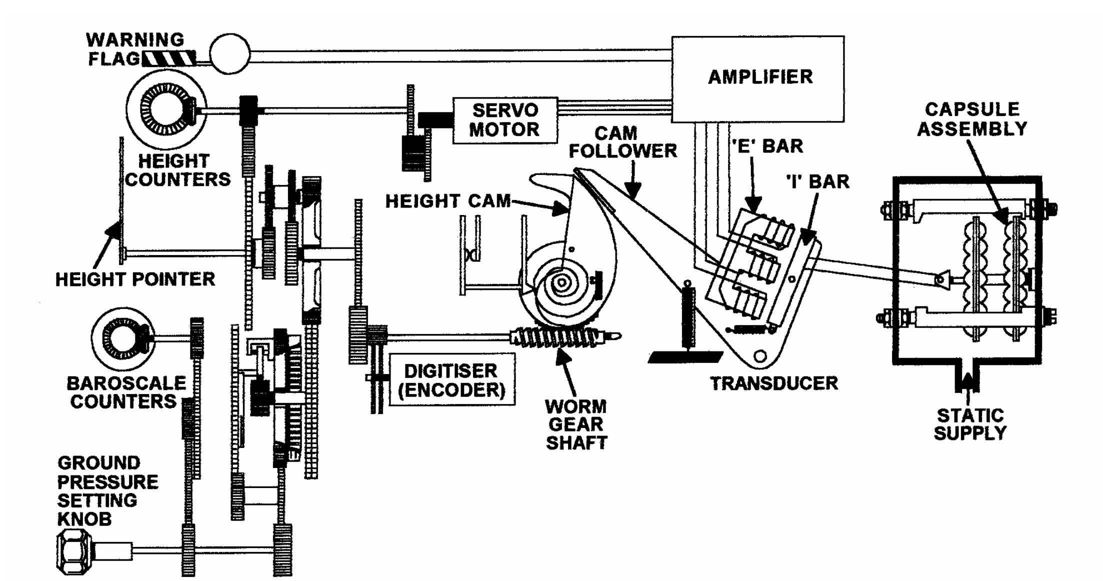
Servo-assisted altimeters use the same basic principles as sensitive altimeters, whereby pressure changes are measured using the expansion and contraction of evacuated capsules. The instrument uses an electrical servomotor to transmit the movement of conventional aneroid capsules to the instrument display. A digital counter system and a single pointer indicate the altitude. The counters are visible through four windows and show (reading from left to right) the altitude in tens of thousands, thousands, hundreds, and units of feet. The pointer moves around the dial, which is calibrated in divisions of 50 ft from 0–1000 ft, and the combined system indicates altitudes up to 100 000 ft.



The barometric subscale is a conventional drum counter type and is set by a knob on the front of the instrument. The presentation is often in millibars or inches of mercury, but in some types both settings are displayed. In this case, the displays are interconnected so that any change in millibars produces an equivalent change in inches of mercury and vice versa.

OPERATION OF A SERVO-ALTIMETER

The mechanism of a typical servo-altimeter is shown schematically below.



In this instrument the pressure sensing capsules are coupled mechanically to an electrical E and I pick-off assembly, and any movement of the aneroid capsules is transmitted through a linkage to the I bar of the E and I inductive pick-up (transducer). The amplitude of the AC voltage output from the secondary windings depends on the degree of deflection of the I bar, which is a function of pressure change. When the two air gaps become unequal, the reluctance of each circuit changes and an electrical output is generated.

The actual polarity of the output signal depends on whether the capsules expand or contract. The output signal is then amplified and used to drive a motor whose speed and direction of rotation depend on the amplitude and phase of the signal. The motor drives the gear train, which then rotates the altitude digital counters and the pointer. The motor also drives through a gearing arrangement and a cam which imparts an angular movement to a cam follower, to which the E bar of the inductive pick-off is attached. Movement of the E bar is such that it is driven until it reaches a position where the air gaps between the E and I bars once again become equal, thus completing the servo-loop. The system is very sensitive to small changes in pressure, and through the motor assembly, provides adequate torque to drive the indicating system.

The datum pressure-setting knob is linked to the cam via a gear train and worm shaft, as shown. Rotation of the knob causes the worm shaft to slide forward or backward and rotates the cam. Angular movement of the cam also alters the relationship between the E and I bars, resulting in an electrical output, which causes the counters to rotate, and drives the inductive pick-off back to its neutral position. The hPa sub-scale displays the value of the datum pressure set.

SERVO-ALTIMETER POWER FAILURE

In the event of power failure, the servo-altimeter does not function, and warning flags immediately display on the dial, indicating that the AC power supply failed. A standard sensitive altimeter is required as a standby instrument.

ALTITUDE ENCODING

The servo-altimeter has an altitude encoder (digitiser) incorporated in it to provide a coded height output, which when transmitted to a remote transponder (SSR), enables the height sensed by the capsules to be monitored on the ground, as pressure altitude. The transmitted pressure altitude is always referenced to 1013.25 mb irrespective of the actual sub-scale setting.

ADVANTAGES OF SERVO-ALTIMETERS

Servo-altimeters have the following advantages over simple and sensitive altimeters:

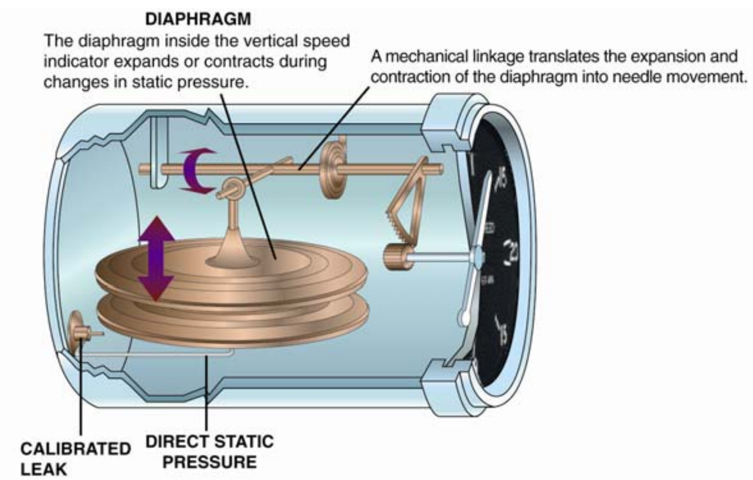
- At high altitude, very little pressure change takes place for a given change of altitude, with the result that capsule movement is considerably less than for the same change of altitude at lower levels. This factor reduces the efficiency of ordinary altimeters at high levels, whereas the servomechanism picks up a capsule movement as small as 0.0002 in per thousand ft.
- Power transmission gives greater accuracy.
- There is practically no time lag between the arrival of a new pressure in the instrument, and the positioning of the counters.
- Because it is an electrical system, it is possible to correct for pressure error, and incorporate an altitude-alerting device in the system.
- Although conventional altimeters now employ digital presentation, it is generally more common with servo-altimeters. The digital presentation reduces the possibility of misreading.
- A pointer is still available on the servo-altimeter for use at low level in assessing the rate of change of altitude.

INTRODUCTION

A Vertical Speed Indicator (VSI) is a sensitive differential pressure gauge, which records the rate of change of atmospheric pressure in terms of rate of climb or descent, when an aeroplane departs from level flight.

PRINCIPLE OF OPERATION

The principle employed is that of measuring the difference in pressure between two chambers, one of which is enclosed within the other, as shown below.



Static pressure is connected directly to the inside of the diaphragm, and indirectly connected to the area outside the diaphragm (the instrument case) via a restricted orifice (calibrated leak) that prevents the pressure outside the diaphragm from changing instantaneously.

The pressure of the atmosphere is communicated directly to the inner chamber (capsule) and through a calibrated choke or capsule case to the outer chamber. If the pressure changes, as in a climb or descent, the lag between the capsule and outer chamber results in a pressure differential across the metering unit, which is a direct measure of the aeroplane's rate of climb or descent. Movement of the capsule transmits via a mechanical linkage to a pointer, which moves against a calibrated dial on the face of the instrument, where the indications are arranged in a logarithmic scale. This allows the scale in the range 0 to 1000 ft/min to be more easily interpreted while, at the same time, allows smaller changes in the vertical speed to be registered in that range. This allows the flight crew to achieve a given flight profile more easily.

The metering unit of the VSI provides a pressure differential across the capsule case for any given rate of climb or descent, whilst compensating for variations in temperature and pressure of the atmosphere with changes in altitude. The compensation is achieved by incorporating in the metering unit both an orifice and a capillary, whose sizes are chosen so that indicator readings remain correct over a wide range of temperature and altitude conditions.

OPERATION OF THE VSI

In level flight, the pressure inside the capsule and the case will be the same, so the pointer will remain in its horizontal position, indicating a zero rate of climb. If the aeroplane climbs, the static pressure in the capsule will decrease at a quicker rate than that in the casing and the capsule will collapse slightly, and will cause the pointer to indicate a rate of climb. Conversely, if the aeroplane descends the static pressure in the capsule will increase at a higher rate than that in the casing and the capsule will expand slightly, causing the pointer to indicate a rate of descent.

ERRORS OF THE VSI

The VSI can suffer from the following errors:

LAG

When an aeroplane suddenly climbs or descends, a delay of a few seconds occurs before the pointer settles at the appropriate rate of climb or descent, which is due to the time required for the pressure differential to develop. A similar delay occurs in the pointer showing a zero rate of climb or descent when the aeroplane resumes level flight.

INSTRUMENT ERROR

This error is due to the manufacturer's tolerances. However, in service the instrument pointer is re-adjusted to the zero position using a screw adjustment.

MANOEUVRE INDUCED ERROR

Errors induced by manoeuvres or flight in turbulence can cause any pressure instrument to misread for up to 3 seconds at low altitudes and up to 10 seconds at high altitudes. The times for the VSI are often even longer. Thus, during any manoeuvre involving a change of attitude, do not place absolute reliance on the VSI.

FAULTS OF THE VSI

The following faults will have an adverse affect on the VSI reading:

BLOCKAGES

A blockage in the static line renders the instrument unserviceable, and the pointer registers zero regardless of the aeroplane's vertical speed.

BREAKAGE OR LEAKAGE IN THE STATIC LINE

A breakage or leakage in the static pressure supply line causes the static value to change as the breakage occurs (e.g. if the breakage occurs in a pressurised section of the aeroplane the VSI initially shows a high rate of descent and stabilises to give a zero indication). This reading is maintained until the aeroplane descends below the cabin altitude pressure.

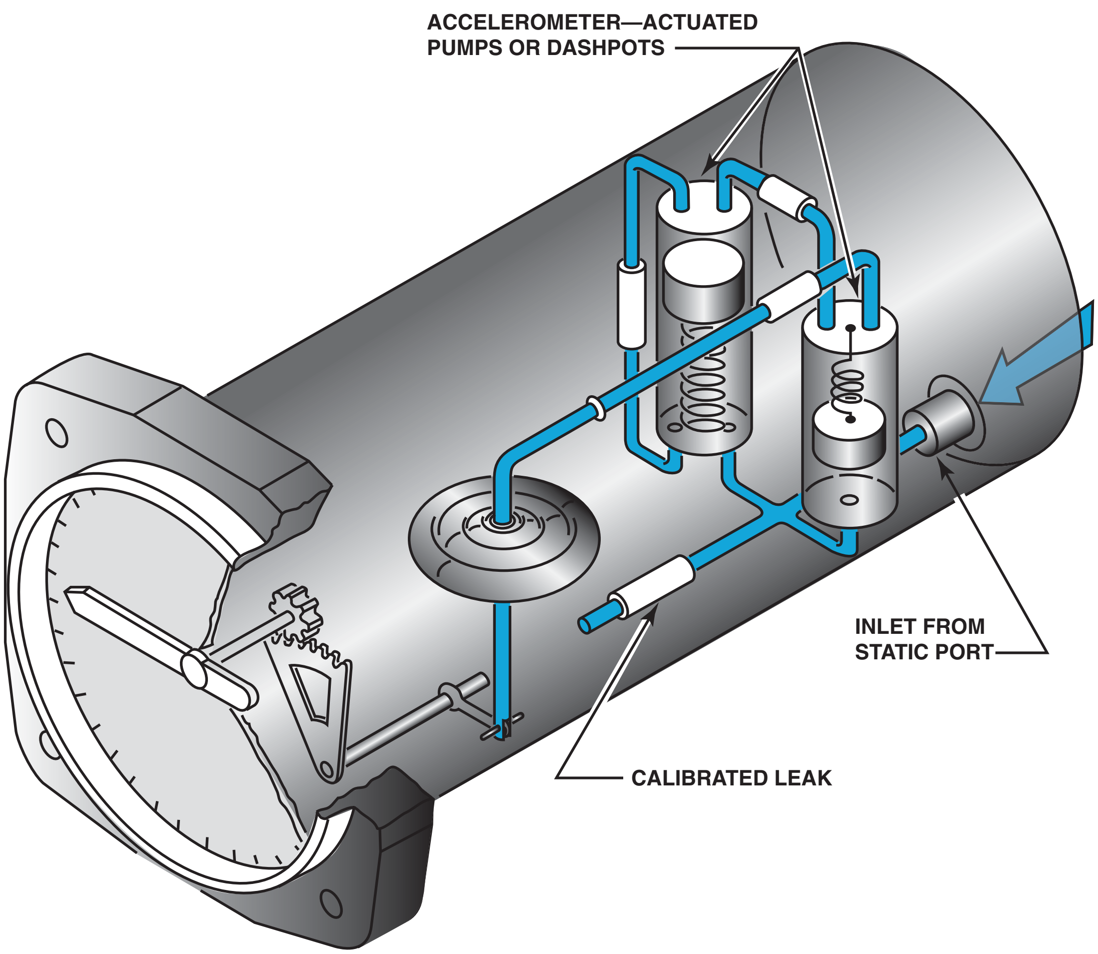
INSTANTANEOUS VERTICAL SPEED INDICATOR (IVSI)

The Instantaneous Vertical Speed Indicator (IVSI) is sometimes referred to as the Inertial Lead Vertical Speed Indicator (ILVSI). The basic construction of this instrument is shown on the next page. It consists of the same basic elements as the conventional VSI, but it is additionally fitted with an accelerometer unit that is designed to create a more rapid differential pressure effect, specifically during the initiation of climb or descent. The accelerometer comprises of two small cylinders or dashpots, which contain inertial masses in the form of pistons that are held in balance by springs and their own mass. The cylinders are connected in the capillary tube system leading to the capsule and are thus open to the static pressure source.

OPERATION OF THE IVSI / ILVSI

If a change in vertical motion is initiated, the resultant vertical acceleration (i.e. due to the change in vertical velocity along the aeroplanes vertical axis), produces a force and the pistons are displaced from their neutral position). The pistons are arranged so that one responds to nose up pitch changes (positive G) by reducing the capsule pressure, whilst the other responds to nose down pitch changes (negative G) by increasing the capsule pressure. This in turn creates an immediate pressure change inside the capsule, and produces an instantaneous movement of the indicator pointer, and in the correct sense, to the initiation of a climb or descent manoeuvre. The errors are generally the same as those affecting the conventional VSI, although the lag and induced manoeuvre errors are virtually eliminated, with the instrument providing a rapid response to changes in the aeroplanes vertical flight path. The accelerometer output will decay after only a few seconds, and the pistons will automatically return to their neutral position, by which time a steady pressure differential is established across the metering unit. The instrument will then continue to behave like a conventional VSI.

The IVSI is affected by the acceleration forces, i.e. g–forces, which act on the pistons during steep turns when the angle of bank is in excess of 40°, and may produce a false reading, known as **Turning Error**.



Chapter 5
Airspeed Indicator

INTRODUCTION

The airspeed at which an aeroplane is travelling through the air is essential to the pilot, both for the safe and efficient handling of the aeroplane and as a basic input to the navigation calculations.

PRINCIPLE OF THE AIRSPEED INDICATOR (ASI)

When an aeroplane is stationary on the ground it is subject to normal atmospheric or static pressure, which acts equally on all parts of the aeroplane structure. In flight, the aeroplane experiences an additional pressure due to the aeroplane's motion through the air, which is known as dynamic pressure, and is dependent upon the forward motion of the aeroplane and the density of the air, according to the following formula:

$$P_T = 1/2\rho V^2 + P_S$$

Where: P_T = total or pitot pressure (also known as total head pressure or stagnation pressure)
P_S = static pressure
ρ = air density
V = velocity of the aeroplane (TAS)

Re-arranging the formula, the difference between the pitot and static pressures is equal to $\frac{1}{2}\rho V^2$ (dynamic pressure). The airspeed indicator thus measures the pressure differential between the two sources, and provides a display indication graduated in units of speed.

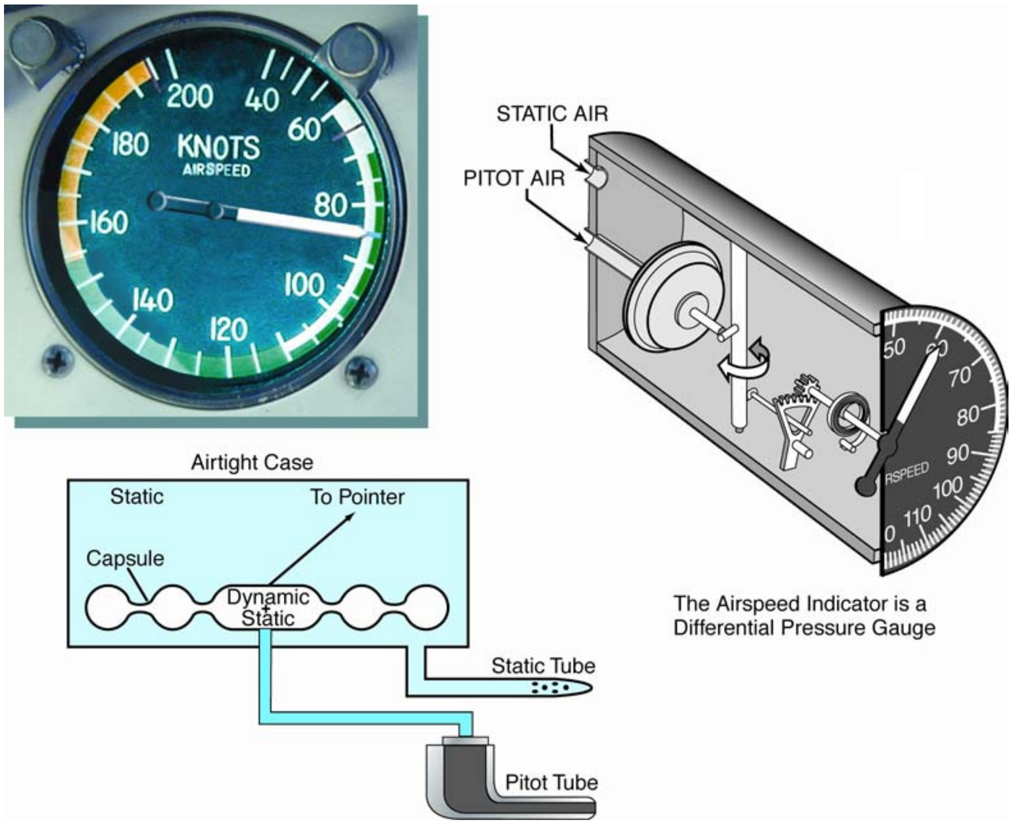
OPERATION OF A SIMPLE ASI

In the simple ASI, a capsule acting as a pressure sensitive element is mounted in an airtight case, as shown on the next page. Pitot pressure is fed into the capsule and static pressure is fed to the interior of the case, which, when the aeroplane is in motion, contains the lower pressure. A pressure difference causes the capsule to open out with any movement being proportional to the pressure differential across the capsule skin (pitot - static). A mechanical linkage is used to transfer the capsule movement to a pointer that moves over a dial, and which is normally calibrated in knots. A bi-metallic strip is also incorporated in the mechanical linkage to compensate for any expansion/contraction of the linkage caused by temperature variations.

SENSITIVE AND SERVO AIRSPEED INDICATORS

Sensitive and Servo airspeed indicators both use the same principle of operation as the simple ASI. The sensitive ASI uses a stack of two or more interlinked capsules, which are connected to two pointers via an extended gear train. This enables the instrument to respond to smaller pressure changes and thus smaller changes in airspeed. The capsule assembly has a linear pressure/deflection characteristic, which is more closely controlled than the single capsule used in the simple ASI.

The servo airspeed indicator also uses an electrical linkage rather than a mechanical linkage to position the indicator needles, which is done by amplifying the error signal.



The Airspeed Indicator is a Differential Pressure Gauge

CALIBRATION OF THE ASI

Standard datum values are used in the calibration of airspeed indicators since dynamic pressure varies with airspeed and air density. Density also varies with temperature and pressure. The values used are the sea level values of the ICAO International Standard Atmosphere (ISA).

COLOUR CODING OF THE ASI

The scale is calibrated in terms of speed, usually knots or miles per hour (MPH), but in some cases, kilometres per hour (KPH). It is essential that you know which terms are displayed on the ASI. On light aeroplanes, the dial is normally colour coded as shown on the next page, with the coloured segments indicating the following:

White arc

This arc extends from V_{SO} (stall full flap) to V_{FE} (maximum speed with flaps extended), and marks the flap operating speed range.

Green arc

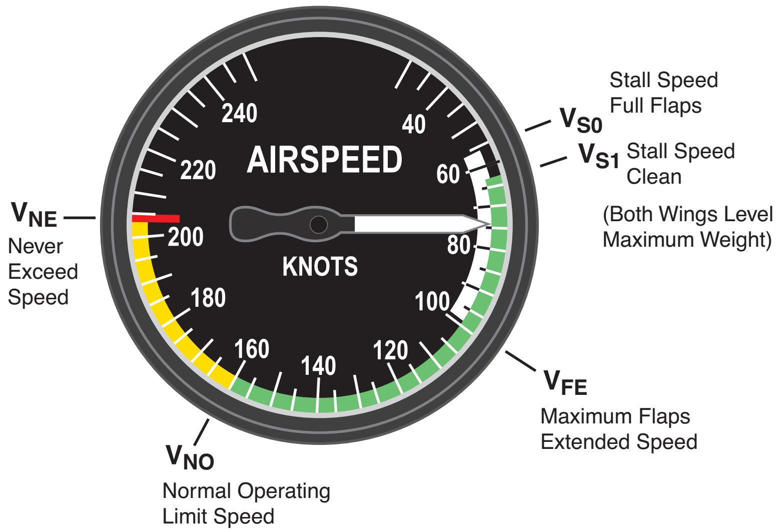
This arc extends from V_{SI} (stall clean) to V_{NO} (normal operating speed), and is the normal operating speed range.

Yellow

This arc extends from V_{NO} to V_{NE} (never exceed speed), and denotes the cautionary speed range. Operations within this speed range should not be carried out except in smooth air.

Red Radial line

This line marks V_{NE}.



Other lines are also used (e.g. a blue radial line, which is sometimes used to indicate the maximum rate of climb speed in a twin-engine aeroplane with one engine inoperative [V_{YSE}]). Some ASI's have adjustable bugs for use in setting a target speed (e.g. the threshold speed).

ASI ERRORS

The dial of the ASI is calibrated to a formula, which assumes constant air density (standard mean sea level) and no instrument defects. Any departure from these conditions, or disturbance in the pitot or static pressures applied to the instrument, results in a difference between the indicated and true airspeeds. The following sources of error exist:

Instrument Error

This error is caused by the manufacturer's permitted tolerances in the construction of the instrument. This error is determined by calibration and if found to be significant, it is recorded on a calibration card. This correction is normally combined with that for pressure error.

Pressure Error

This error arises from the movement of the air around the aeroplane and causes disturbances in the static and pitot pressure. The causes of this error are:

Position of the Pitot-Static Sensors

This can alter the pressures fed to the instrument, and is particularly so in the case of a combined pitot-static head where the dynamic pressure component may significantly affect the static supply. To minimise this source of error separate static vents are positioned well away from the pitot head, which can result in a 95% reduction in the overall pressure error. The position/pressure error is normally determined by calibration, and a pressure error card is tabulated in the Aeroplane Flight Manual. This card may also incorporate any instrument error calibrations.

Manoeuvre Induced Error

This is caused by changes in the aeroplanes attitude and/or configuration and is normally only short term. The main sources of error are normally in the static supply, but since the transient affects of manoeuvre induced error are not predictable or avoidable, the flight crew must be aware of this problem.

The pressure error changes if any of the following vary:

- Airspeed
- Angle of attack
- Configuration (flap setting, undercarriage etc)
- Position of the pitot/static sources and sideslip

Compressibility Error

The calibration formula for most airspeed indicators does not contain any compensation for the fact that the air is compressible. At low airspeeds this is insignificant, but at airspeeds over 300 KTAS, this factor becomes significant. This is especially so at high altitudes where the less dense air is easily compressed. Compressibility causes an increase in the measured value of dynamic pressure, which causes the ASI to over-read. Thus, compressibility varies with airspeed and altitude. Compensation for the error and correction is on some mechanical navigation computers, but tabulation is against altitude, temperature, and CAS in the handbooks of others.

Density Error

Dynamic pressure varies with airspeed and density of the air. In calibration, standard mean sea level pressure is used; thus, for any other condition of air density, the ASI will be in error. As altitude increases, the density decreases and the indicated airspeed (IAS), and thus equivalent airspeed (EAS) at speeds in excess of 300 KTAS, becomes progressively lower than the true airspeed (TAS). For example, at 40 000 ft the density is only ¼ of its msl value. The dynamic pressure, which is proportional to TAS^2, is ½ the msl value for the same TAS (i.e. an aeroplane flying at 400 KTAS will have an IAS of 200 knots). The following formula establishes the relationship:

$$EAS = TAS \sqrt{\frac{\rho_{\text{altitude}}}{\rho_{\text{sea level}}}}$$

For accuracy, the correction of CAS to TAS is done on a navigational computer using the ambient temperature (outside air temperature), at the required pressure altitude. A useful formula for estimating TAS is:

TAS = CAS + (1.75% of CAS per 1000 ft of altitude)

For example, for a CAS of 100 knots at 10 000 ft:

TAS = 100 + (1.75/100 x100 x 10) = 117.5 knots

The relationship between the various airspeeds is as follows:

- Airspeed Indicator Reading (ASIR) + Instrument Error Correction = Indicated Airspeed(IAS)
- IAS + Pressure Error Correction = Calibrated Airspeed (CAS)
- CAS + Compressibility Error Correction = Equivalent Airspeed (EAS)
- EAS + Density Error Correction = True Airspeed (TAS)

In practice, the combined corrections give:

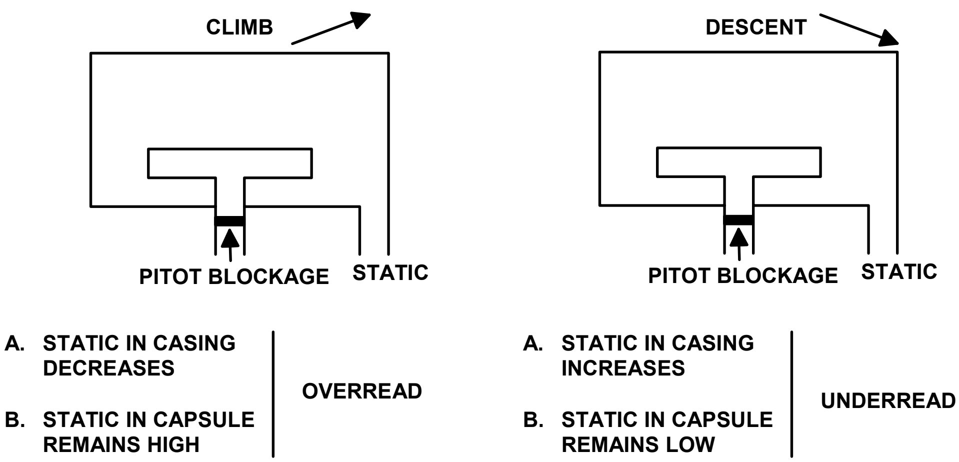
- ASIR + Instrument Error Correction + Pressure Error Correction = CAS
- CAS + Compressibility Error Correction + Density Error Correction = TAS

ASI FAULTS

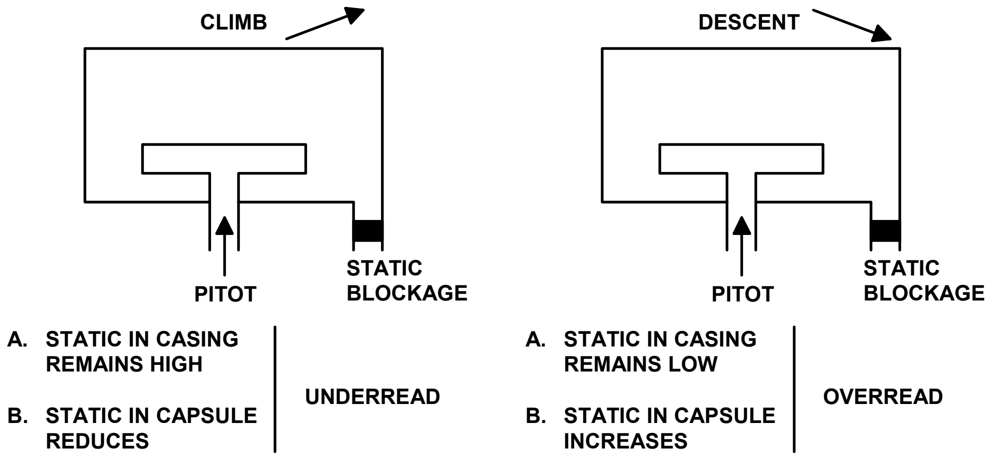
The following faults may occur in the ASI:

Blockages

A blockage of the pitot tube, as shown on the next page, possibly due to ice, causes the ASI to not respond to changes of speed in level flight. The capsule does behave as a barometer or altimeter capsule, and reacts to changes in the static pressure. If the aeroplane climbs, the ASI indicates an increase in airspeed (over-read) and if it descends, it indicates a decrease in airspeed (under-read).



If the static line is blocked, the ASI over-reads at lower altitudes, and under-reads at higher altitudes than when the line became blocked.



Leaks

A leak in the pitot system causes the ASI to under-read, whilst a leak in the static line causes the ASI to over-read in an unpressurised fuselage (cabin pressure is usually lower than the atmospheric static pressure), and under-read in a pressurised aeroplane (cabin pressure higher than static).

Whilst any under-reading of the ASI is undesirable, it is not necessarily dangerous, but over-reading of the ASI is dangerous, since a stall occurs at a higher indicated airspeed than that specified for the aeroplane.

Some modern ASI's also employ coloured flags and needles as attention getters (i.e. to indicate any electrical or transmission failure, and to draw attention to important altitude indicators).

CALCULATION OF CAS TO TAS (UP TO 300 KNOTS)

To calculate TAS from CAS using a CRP5 rotate the inner disc of the computer to align the outer edge of the airspeed window with the pressure altitude (altitude with 1013 mb set on the altimeter subscale) inside the window. Position the cursor through the CAS on the inner circular slide rule scale and read off TAS on the outer scale.

Example: If the pressure altitude is 18 000 ft, the COAT is –30°C and CAS is 170 knots, then the TAS is:

> **Solution**: In the airspeed window set 18 (pressure altitude x 1000 ft) opposite –30°C on the COAT scale. Next position the cursor through the CAS of 170 knots on the inner slide rule scale and read of the TAS of 220 knots on the outer scale.

Note: As a reminder of which way to read CAS to TAS there is a red CAS (RAS) on the inner scale between 35 and 40, and a red TAS on the outer scale between 35 and 40.

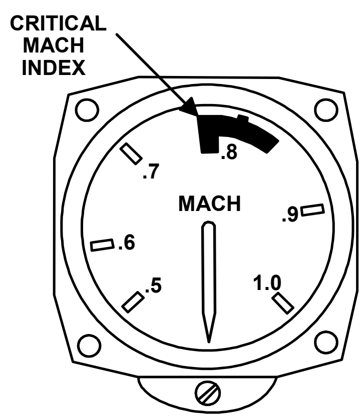
INTRODUCTION

During flight, aeroplanes emit pressure disturbances (sound waves), which radiate in all directions at the speed of sound. As the speed of an aeroplane increases, it gets progressively closer to the waves ahead of it until, at the speed of sound, the pressure waves combine into a shock wave and attach themselves to the aeroplane. The effect of the shock wave greatly increases the drag forces associated with the aeroplane, and significantly alters its stability and control characteristics. It is therefore extremely important for the flight crew to know how close to the speed of sound the aeroplane is. The instrument used to indicate this is the machmeter, which measures the ratio of the aeroplane's airspeed (TAS) to the local speed of sound, and displays it as a Mach number.

Mach No. = $^{V}/_{A}$ (Where V = TAS and A = Local Speed of Sound)

CRITICAL MACH NUMBER (M_{CRIT})

This is the speed of an aeroplane, defined in terms of the speed of the free stream airflow. In practice, the speed of the local airflow at any point around the structure varies considerably and the local airflow may reach the speed of sound when the free stream speed is much lower. The free stream speed at which any element of the local airflow reaches the speed of sound, results in local shockwaves forming on the structure. The free stream Mach number then becomes the Critical Mach number (M_{crit}). Unless specifically designed for that purpose, an aeroplane is not flown beyond M_{crit} and the value of M_{crit} is highlighted by an index mark on the face of the machmeter.



In commercial operations, it is common practice to set a lower limit, known as the **maximum operating Mach number (M_{mo})**

PRINCIPLE OF OPERATION

The speed of sound varies only with temperature. As the temperature increases, the local speed of sound increases and vice versa. The speed of sound (A) varies with the square root of the absolute temperature (° Kelvin - K) of the environment, written as ${}^{\text{New A}}/_{\text{Old A}} = {}^{\sqrt{\text{New K}}}/_{\sqrt{\text{Old K}}}$. Using the known value of the speed of sound at ISA mean sea level of 661.7 kt, where the temperature (in degrees Kelvin) is 288°, this is re-written as:

$$A = 38.94\sqrt{\text{temperature in Kelvin.}}$$

or [644 + 1.2t] where t = OAT in °C (estimation only).

N.B: ISA temp = 15° - (pressure altitude x 2° per 1000 ft)

Since Mach number is the ratio of TAS to the local speed of sound, it may also be rewritten in terms of pitot and static pressures.

TAS (V) is a function of the dynamic pressure and the local density. Dynamic pressure is the difference between pitot (P) and static (S) pressures (i.e. P-S [airspeed capsule]).

The local speed of sound (A) is also a function of temperature. According to basic gas physics, temperature is also a function of static pressure (S) and density (altitude capsule).

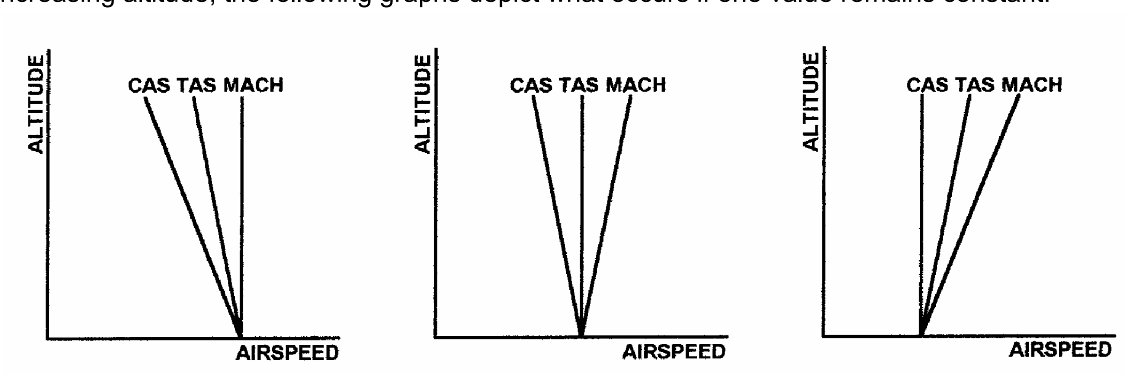
Since Mach No. = ${}^{V}/_{A}$, rewritten in a simplified form so that:

$V \propto q/\rho$	$A \propto {}^\circ K$
$\therefore \propto P\text{-}S / \rho$	$\therefore \propto S/\rho$

ρ is common to both so that:
Mach No. = ${}^{V}/_{A}$ and is **a function** of P-S / S

The machmeter is designed to measure the ratio of pitot excess pressure (the difference between pitot and static) and static pressure.

A relationship also exists between CAS, TAS, and Mach number under ISA conditions. With increasing altitude, the following graphs depict what occurs if one value remains constant:



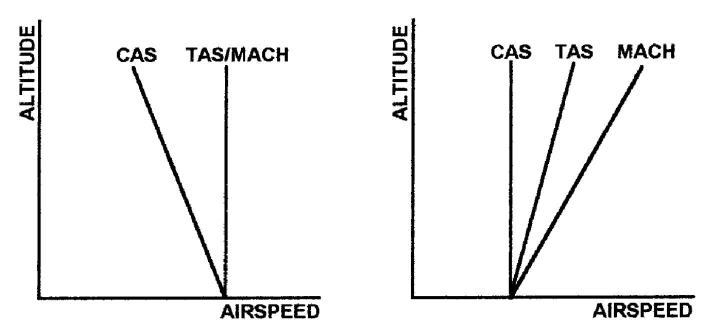
This is summarised in the following tabular format, including the effect of increasing altitude on the Local Speed of Sound (LSS).

CAS / IAS	TAS	MACH	LSS
constant	increases	increases	decreases
decreases	**constant**	increases	decreases
decreases	decreases	**constant**	decreases

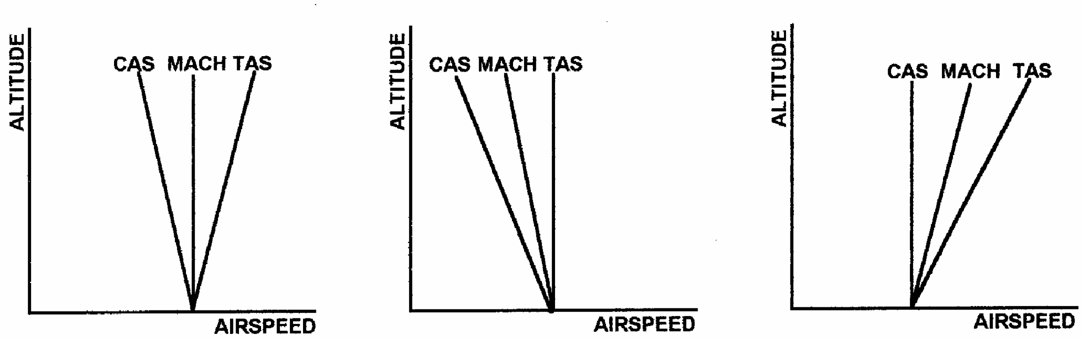
The opposite occurs with decreasing altitude under ISA conditions.

Even if the CAS or MN is unaltered intentionally, the MN may change due to a change in the OAT (e.g. high temperature to low temperature, the LSS decreases to maintain a constant MN; therefore, the TAS must decrease). For example, if two aeroplanes are travelling at the same MN, but at different flight levels, the aeroplane at the lower flight level has a higher TAS.

If one airspeed remains constant, the graphs below depict what happens to the other airspeeds if an aeroplane climbs or descends through an **isothermal layer**. At a constant Mach number, the relationship between TAS and LSS remains constant as both are equally affected by ambient temperature, although the CAS reduces during a climb and increases during a descent due to the density error. Conversely, at a constant CAS, the TAS and Mach number both increase during a climb and decrease during a descent.



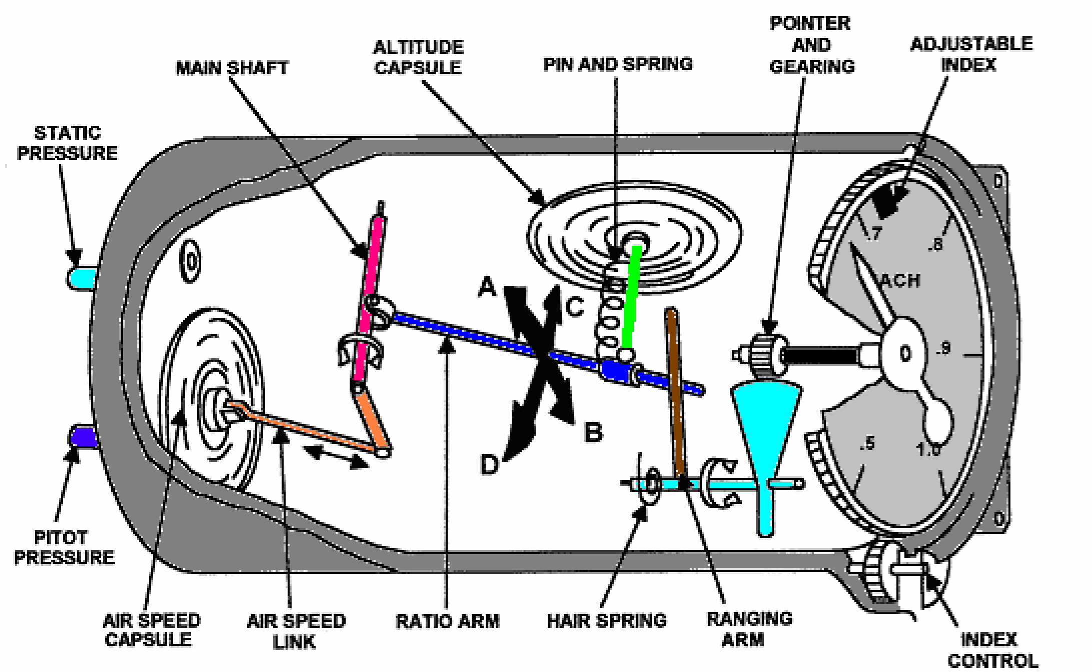
If an aeroplane climbs or descends through an **inversion layer,** the graphs below depict what happens to the other airspeeds, if one of them remains constant.



During a climb at a constant Mach number, the LSS increases due to the warmer air, so the TAS increases, but the CAS decreases due to the reduction in density. If the aeroplane climbs at a constant TAS, both the Mach number and the CAS decrease. Conversely, if the aeroplane climbs at a constant CAS, the Mach number and the TAS both increase. During a descent the reverse occurs, because the LSS decreases due to the colder air, whilst the density increases.

MACHMETER CONSTRUCTION AND OPERATION

A typical machmeter, as shown below, consists of a sealed case containing two capsule assemblies placed at 90° to each other, and a series of mechanical linkages.



The first capsule unit is an airspeed capsule, and is connected to the pitot pressure pipeline, while the interior of the instrument case is fed with static pressure. The second capsule unit is an aneroid capsule, which responds to changes in static pressure. The airspeed capsule measures the difference between pitot and static pressure and expands or contracts in response to airspeed changes. The airspeed linkage transfers movement of the capsule to the main shaft, and causes the shaft to rotate, thus moving a pivoted ratio arm in the direction A-B.

The altitude (aneroid) capsule expands or contracts, and responds to changes in altitude. Movement of the capsule transfers to the ratio arm via a spring and pin, thus causing it to move in the direction C-D.

The position of the ratio arm is therefore dependent on both pitot excess and static pressure. Movement of the ratio arm controls the ranging arm which, through the linkage and gearing. This turns the pointer, and displays the Mach number corresponding to the ratio of pitot excess pressure and static pressure. Any increase in altitude and/or airspeed results in a higher Mach number. The Critical Mach number is indicated by a specially shaped lubber mark, located over the machmeter dial. It is adjustable so that the critical Mach number for the particular type of aeroplane is displayed.

CALIBRATION

Machmeters are calibrated to a formula relating Mach number to atmospheric conditions of pressure and density.

ERRORS

As Mach number is a function of the ratio of pitot excess pressure to static pressure, only those errors in the measurement of this ratio affect the machmeter reading. The errors are:

INSTRUMENT ERRORS

Like all instruments, machmeters are subject to manufacturing tolerances, although these are extremely small.

PRESSURE ERRORS

These errors are small at the altitudes and speed ranges where machmeters are used.

DENSITY, TEMPERATURE, AND COMPRESSIBILITY ERRORS

These errors are eliminated because density changes do not alter the dynamic pressure/static pressure ratio.

BLOCKAGES AND LEAKAGES

Blockages and leaks have the same affect as they do on the ASI, but the effects are greater due to the two capsules. A blockage in the static system causes the machmeter to **over-read during a descent** and **under-read during a climb**.

ACCURACY

The accuracy of the machmeter is within +/- 0.01 M during its normal operating range, but increases to +/- 0.02 M at the limits of that range.

SERVICEABILITY CHECKS

The instrument should read zero when the aeroplane is stationary and a rough check of IAS/TAS against Mach number should be carried out using the CRP5 computer.

One way to calculate the TAS is by using the CRP5 by setting the OAT against the mach index and reading off the TAS on the outer scale against a set Mach number (MN). To determine the LSS, read the speed on the outer scale, which relates to Mach 1.0 on the inner scale.

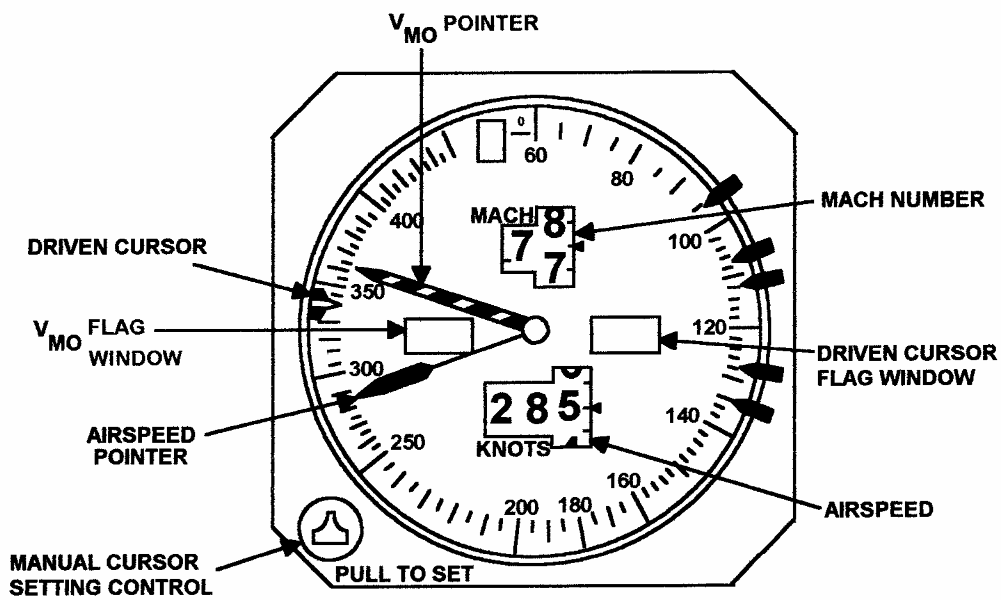
Chapter 7
Mach Airspeed Indicator

INTRODUCTION

The Mach Airspeed Indicator (MASI) combines the functions of both the airspeed indicator and the machmeter in a single instrument. As the machmeter contains an airspeed capsule, it is a relatively simple matter to incorporate the IAS output from the capsule into the display.

DISPLAY

To improve clarity, digital readouts are provided for Mach number (MN) and IAS, as well as a conventional pointer for IAS.



V_{MO} POINTER

A special feature of the MASI is the V_{MO} Pointer (Barber's Pole), which has red and white stripes. At low altitudes, the V_{MO} Pointer retains a fixed position, which indicates the maximum operating IAS permitted. As altitude increases, V_{MO} corresponds to an increasing MN until reaching M_{MO} , at which point the V_{MO} Pointer reading progressively reduces to reflect the overriding MN limit. Movement of the V_{MO} Pointer can be mechanical, although the MASI display is normally driven from the Air Data Computer (ADC).

DRIVEN CURSOR

Adjustment of the Driven cursor is adjusted either by the flight crew or automatically by the aeroplane's systems, to indicate target speeds for particular phases of flight.

BUGS

Bugs are indexes, which are set prior to flight to indicate important speeds such as $V_{1,}$ V_2, $V_{R,}$ etc.

LINKAGES

The MASI is often linked to the auto-throttle / flight management systems to provide two-way feedback. Further links to visual and audio warning devices may also be incorporated.

ERRORS

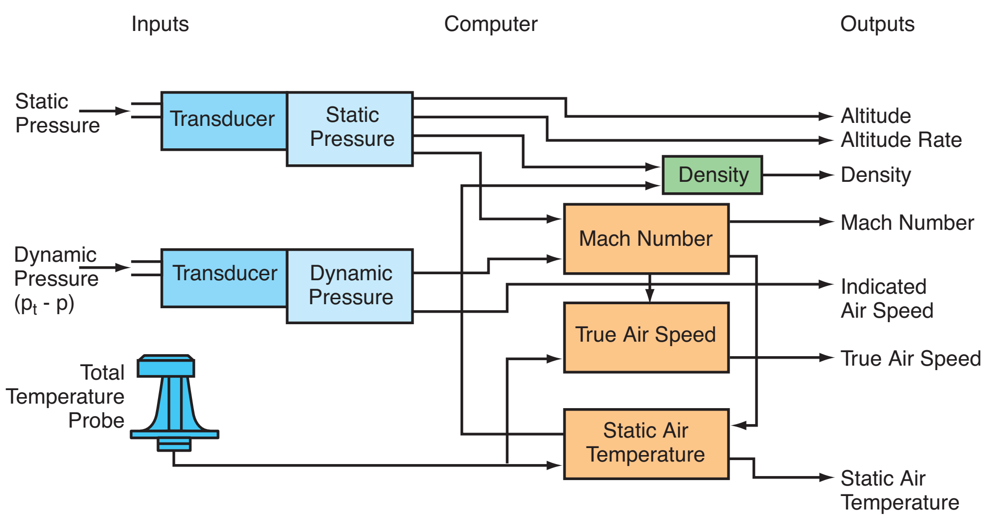
The MASI suffers from the same errors as the ASI and machmeter; namely instrument, pressure and manoeuvre errors.

INTRODUCTION

Many of the primary flight instruments on an aeroplane are dependent on pressures transmitted from the pitot/static probes through a system of pipelines before reaching the sensors in the instruments. Larger aeroplanes require longer pipes, which results in increased lag errors and a greater risk of breakage/leakage. To overcome this, and to create other benefits, most modern transport category aeroplanes use Central Air Data Computers (CADC).

THE CENTRAL AIR DATA COMPUTER

In its basic format, a CADC is an analogue device that produces electrical signals equivalent to pitot and static pressures, and Total Air Temperature (TAT), as shown below.



These signals are computed within the CADC to produce electrical output signals, which are equivalent to:

- Altitude
- CAS
- Vertical speed
- Mach number
- TAS

CADCs are powered whenever the aeroplane AC bus bars are powered, and the output signals are used to operate various flight instruments and aeroplane systems. The flight instruments supplied by the CADC are the:

- Altimeter
- Airspeed Indicator
- Vertical Speed Indicator
- Machmeter

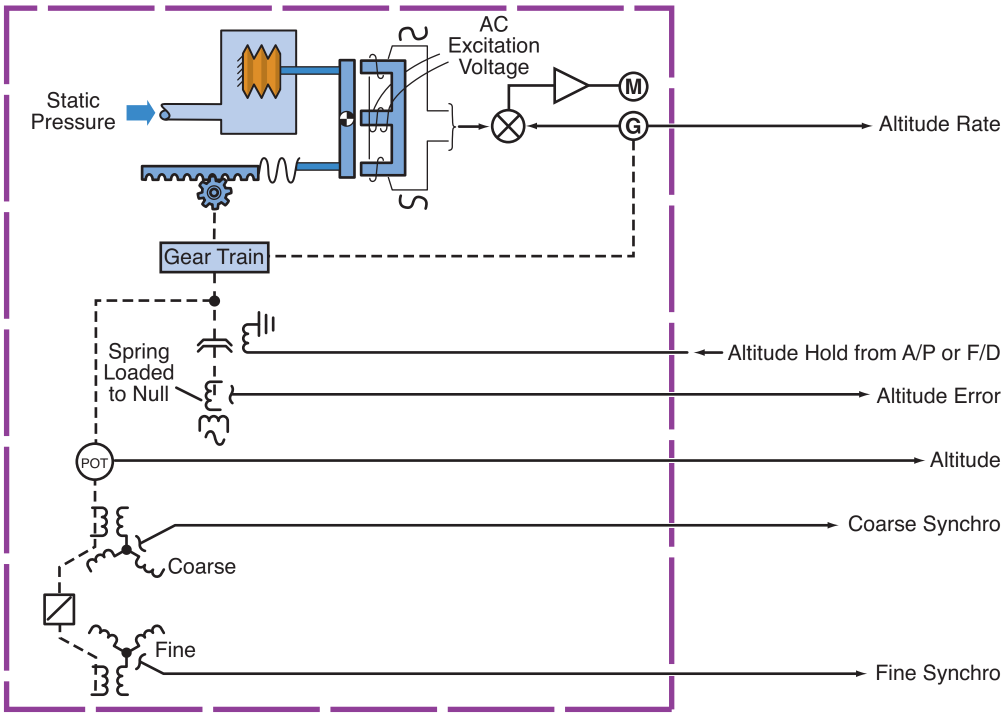
The aeroplane systems typically supplied are the:

- Flight Data recorder (FDR)
- Flight Management System (FMS)
- Automatic Flight Control System (AFCS)
- Transponder
- Ground Proximity Warning System (GPWS)
- Power Management Computer (PMC)
- Flight Director System (FDS)

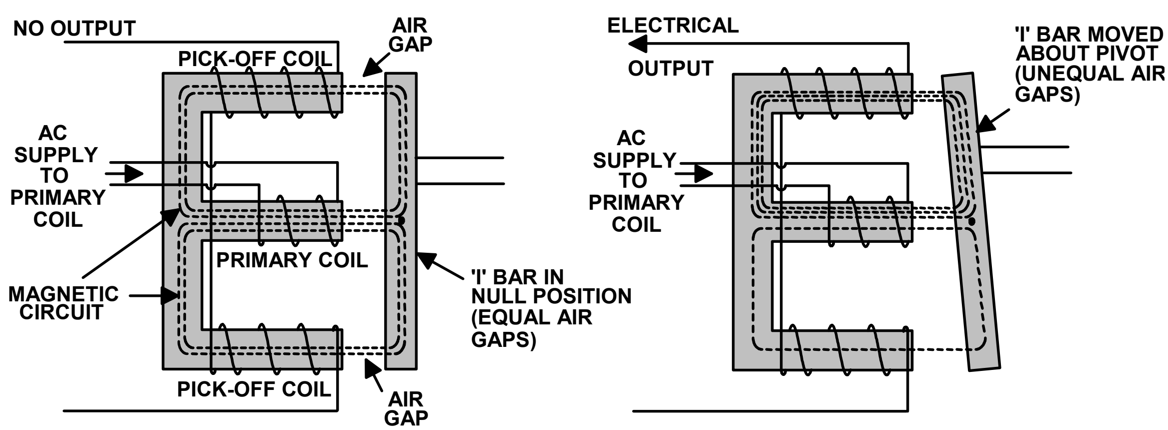
Calculated values of True Airspeed (TAS) and Total Air Temperature (TAT) or Static Air Temperature (SAT) are also normally digitally displayed on the aeroplane's instrument panel. The CADC additionally compensates automatically for both temperature and compressibility effects at high Mach numbers, thus enabling accurate instrument readings over a wide range of altitudes and airspeeds. The disadvantage of the CADC is that, unlike conventional pressure instruments, it requires power to operate, so a back-up system is provided, either via an alternate power supply, or by simple pressure instruments. Two CADCs are normally fitted, to guard against a single failure, and each computer is supplied from separate Pitot-static sources.

CONVERSION OF SENSING PRESSURES

Transducers convert the sensed pressures into electrical signals; one senses the static pressure, whilst the other senses the dynamic pressure. A typical pressure transducer utilises the expansion of a diaphragm or capsule to actuate an electrical pick-off, as shown below, for the CADC Altitude Module.



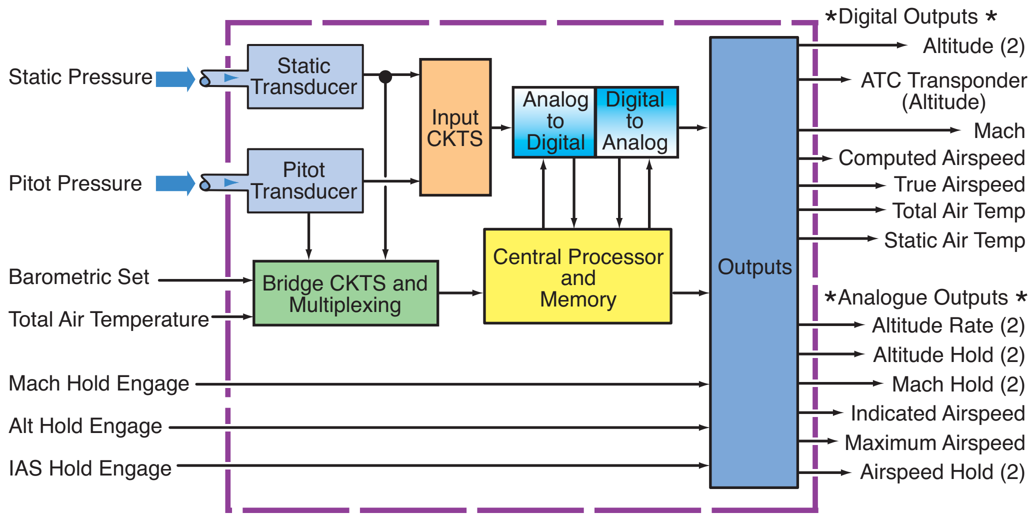
As the capsule changes shape due to a pressure change, the position of the 'I' bar relative to the 'E' bar changes. When the two air gaps are unequal, the reluctance of each circuit changes and an electrical output generates, as shown on the next page. The resultant signal is amplified and applied to a servomotor, via an output shaft, and moves the 'E' bar until the air gaps equalise. When this occurs, no further signals are fed to the amplifier, and the 'I' bar is in a null position. The output shaft, connected to a control (CX) synchro, has power applied to its rotor enabling the angular position to be measured in terms of pressure.



Some systems alternatively use piezo electric transducers, which consist of a stack of quartz discs, where each disc has a metallic pattern deposited on it. When pressure is applied to the stack, the stack flexes or bends, producing an electrical signal. The polarity of the resultant signal, dictated by the direction of flex and its overall strength, is governed by the amount that the disc flexes.

DIGITAL AIR DATA COMPUTER

Some modern aeroplanes are alternatively fitted with Digital Air Data Computers, which use digital computing and electronic circuits instead of servomotors. In this system, the analogue inputs are directly converted into digital outputs before computation, but if analogue outputs are required, they are acquired from the digital format. A typical system is shown below:



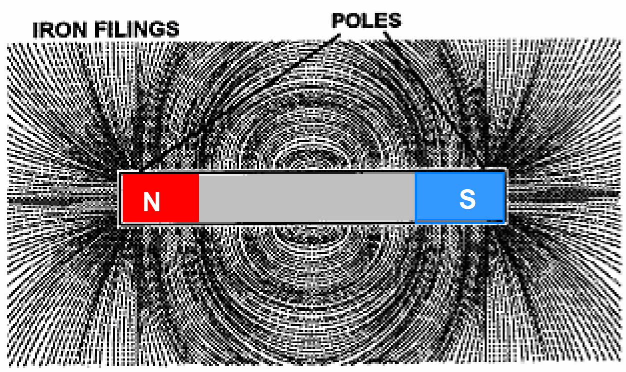
INTRODUCTION

The operating principles of direct reading compasses are based on the fundamentals of magnetism, in particular the interaction between the magnetic field of a suitably suspended magnetic element, and the Earth's magnetic field.

MAGNETIC PROPERTIES

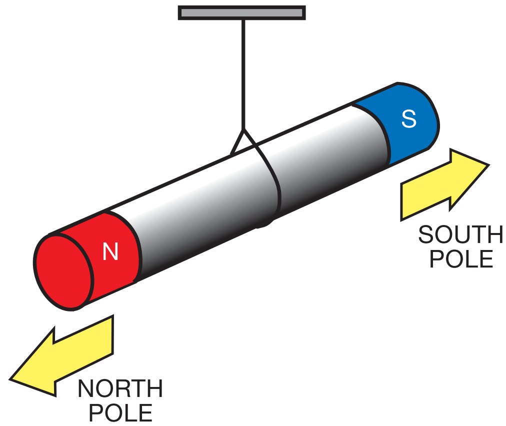
The three principle properties of a simple permanent bar magnet are that:

- It attracts other pieces of iron and steel.
- Its power of attraction is concentrated at each end of the bar.
- When suspended to move horizontally, it always comes to rest in an approximately north - south direction.

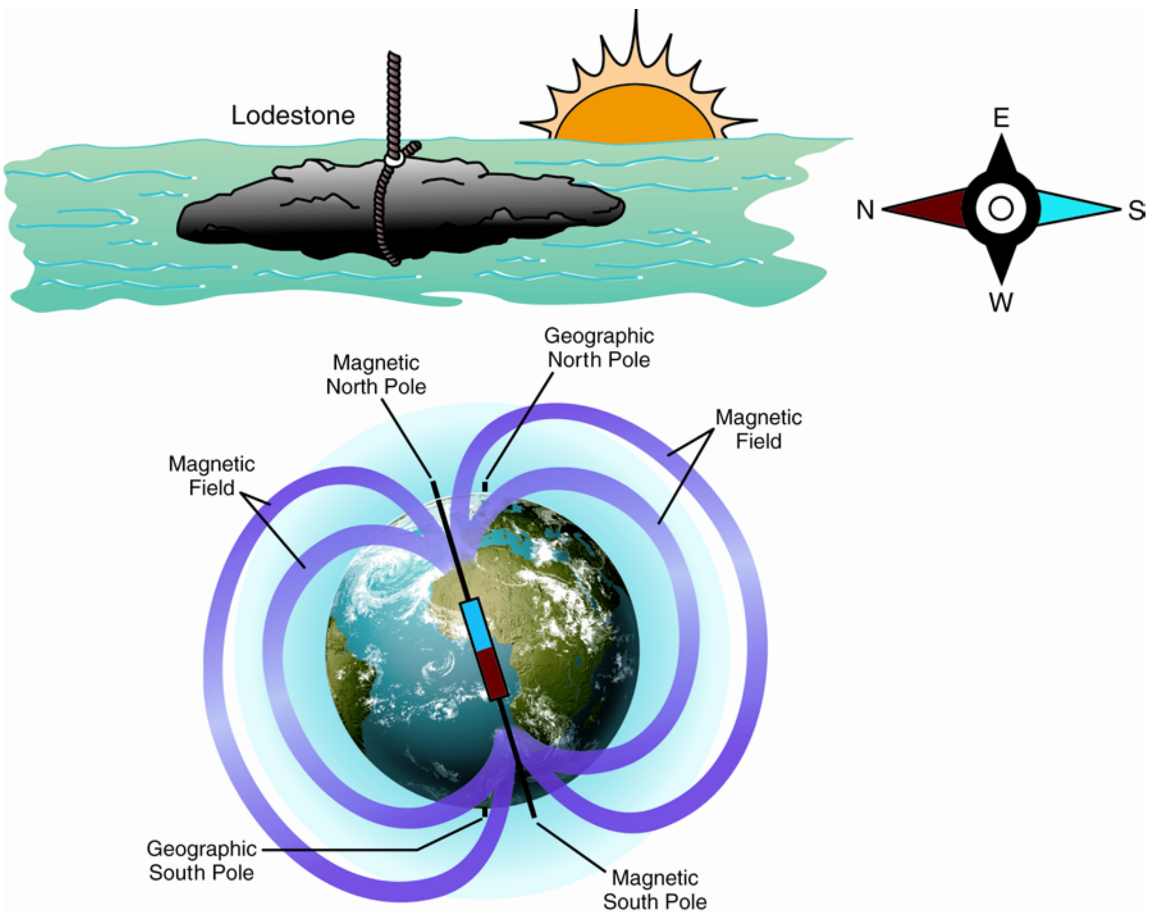


A region of influence called a magnetic field, made up of invisible lines of magnetic force, or magnetic flux, also extends outside a magnet into the surrounding space. To best demonstrate this, sprinkle iron filings on a piece of paper placed over a magnet.

This experiment illustrates that magnetism is concentrated at a magnet's extremities, called poles. Additionally, a freely suspended magnetised rod always aligns itself approximately in a north-south orientation. The end that seeks north is called the **north seeking** or **red pole**, and the end that seeks south, is called the **south seeking** or **blue pole**.



The earliest known form of magnetism was the **Lodestone**, which was a natural mineral found in Asia. If a piece of this ore is suspended horizontally by a thread, or floated on wood in water, it automatically aligns itself in a North-South direction as shown below.



This characteristic led to its use as a compass, and the name lodestone, meaning leading stone. This occurs because the Earth itself is a huge magnet with its own magnetic field.

The fields interact with each other and the lodestone aligns itself according to the fundamental laws of magnetism. Other than the Earth itself, lodestone is the only natural magnet; production of all other magnets is artificial. For example, magnetism is induced in an iron bar by stroking it with a piece of lodestone. Another type of magnet is the electromagnet, where an electric current produces the magnetic field. Magnets are also often classified by their shape, and can exist as either horseshoe, bar, or ring magnets.

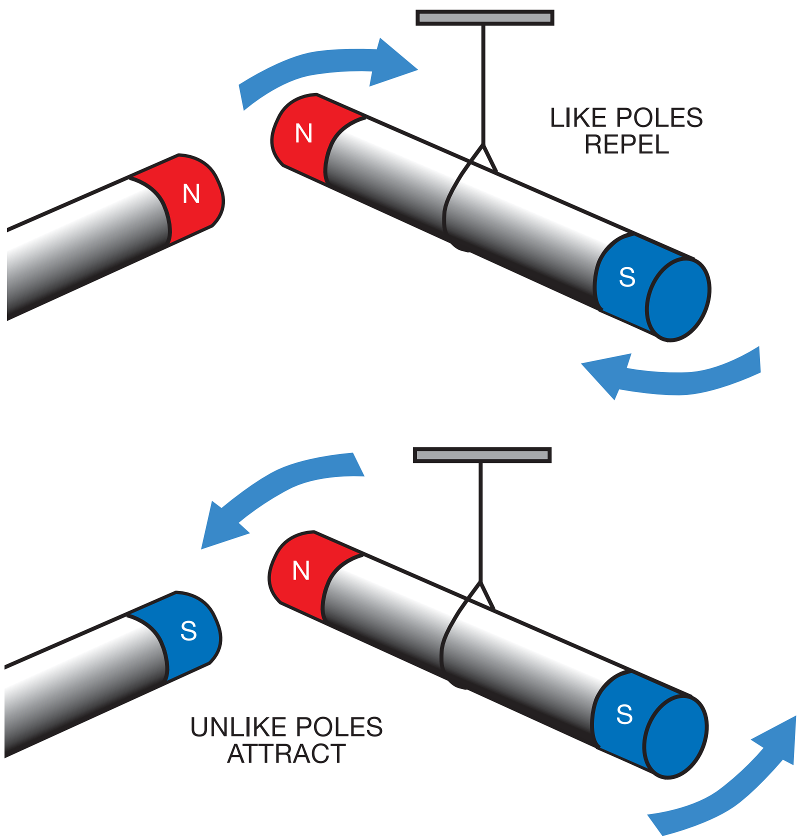
To demagnetise a magnet:

- Heat it to a temperature known as its **Curie Point**.
- Hit it with a hammer.
- Degauss it with an alternating magnetic field.

FUNDAMENTAL LAWS OF MAGNETISM

The fundamental laws are as follows:

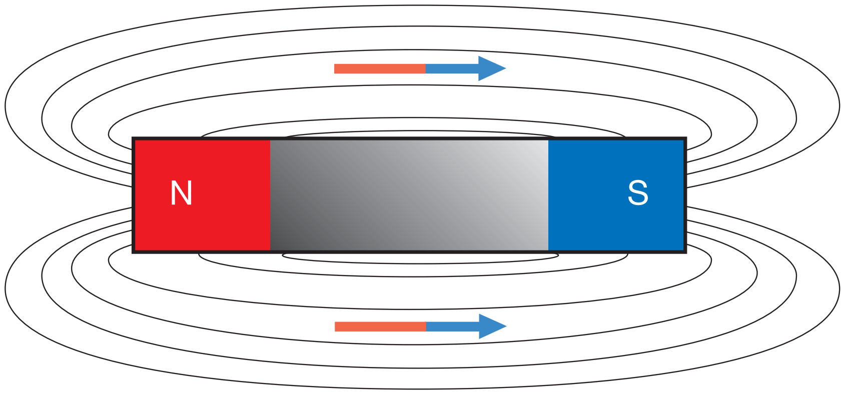
- Red or blue poles cannot exist separately.
- Like poles repel each other and unlike poles attract.



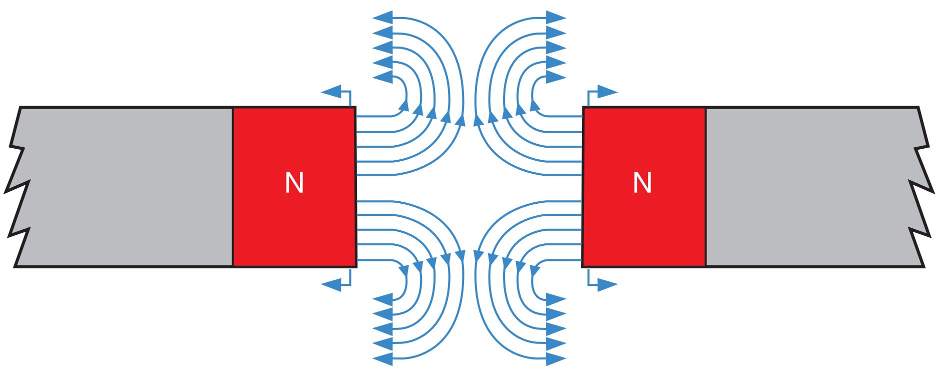
- The force of attraction or repulsion between the two magnets varies inversely as the square of the distance between them.
- A line joining the poles is called its **magnetic axis**.

CHARACTERISTICS OF LINES OF MAGNETIC FLUX

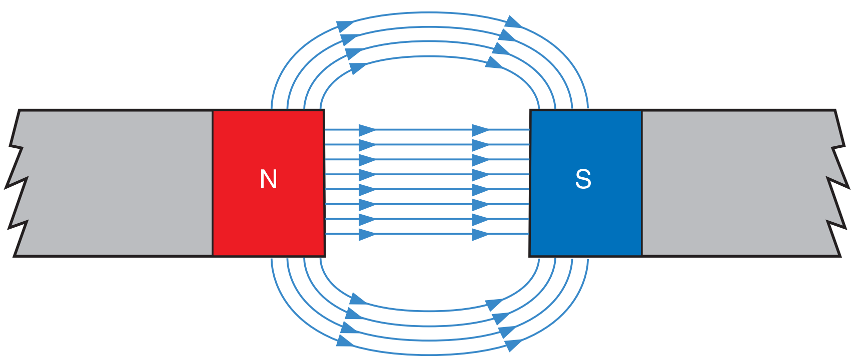
The magnetic field is the region in which the force exerted by a magnet is detected. This field consists of magnetic flux, measured in Webers (Wb), and often represented in direction and intensity by lines of flux.



- The lines of flux have direction or polarity. They flow from the north pole to the south pole outside the magnet, but flow from the south pole to the north pole within the magnet. These lines are continuous and always form complete loops.
- The lines of flux do not cross one another, like poles repel, as shown on the next page, and

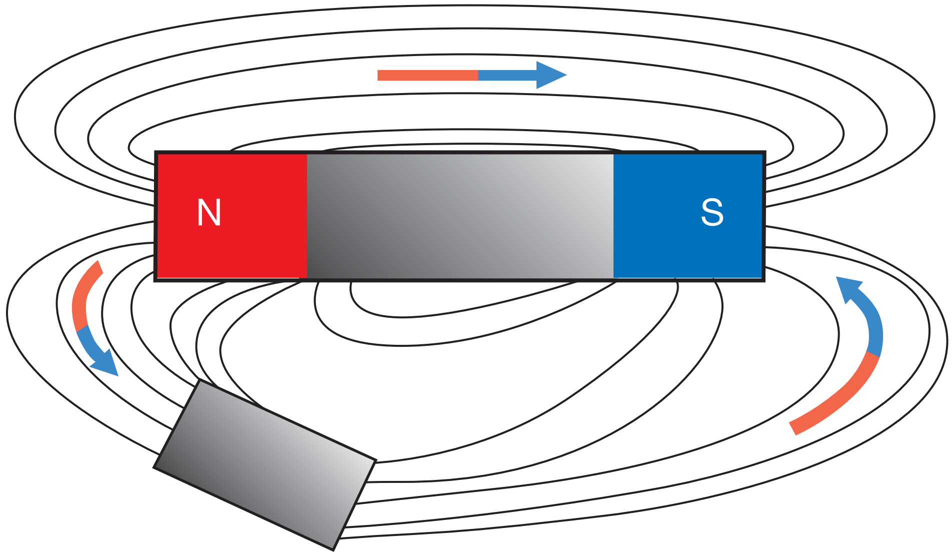


- Lines of flux tend to form the smallest possible loops (i.e. unlike poles attract).



- Establishment of magnetic flux is easier in some materials than in others. All materials, whether magnetic or not, also have a property called **reluctance** that resists the establishment of magnetic flux, and equates to the resistance found in an electrical circuit.

- Lines of magnetic flux can distort by interaction with other lines of flux, as shown below.



MAGNETIC MATERIALS

Theoretically, magnetic fields affect all materials to some extent, and are categorised as follows:

FERROMAGNETIC

This is the property of a material that enables it to become a permanent magnet (i.e. when placed in a magnetic field the material becomes magnetised). The most common ferromagnetic materials are iron, cobalt, and nickel, and alloys of these materials. Ferromagnetic materials can divide into two further categories:

Hard Iron

This material, known as a **permanent magnet**, is difficult to magnetise, but when removed from a magnetic field, it retains the magnetism for a considerable length of time, unless it is subject to a strong demagnetising force (e.g. cobalt and tungsten steel).

Soft Iron

This material, known as a **temporary magnet**, is easily magnetised, but once removed from the magnetic field easily loses its magnetism (e.g. silicon iron).

The above terms also describe the magnetic effects that occur in aeroplanes.

PARAMAGNETIC

This is the property of a material, which when placed in a magnetic field slightly attracts the lines of magnetic force, but once removed loses its magnetism. The most common materials are platinum, manganese, and aluminium.

DIAMAGNETIC

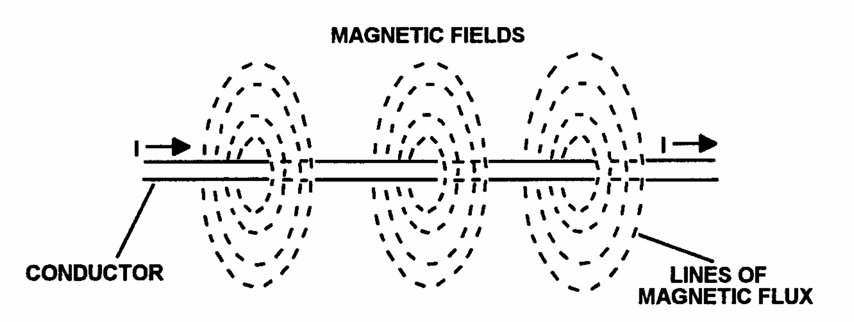
This is the property of a material, which when placed in a magnetic field slightly repels the lines of magnetic force. The most common materials are copper and bismuth.

PERMEABILITY

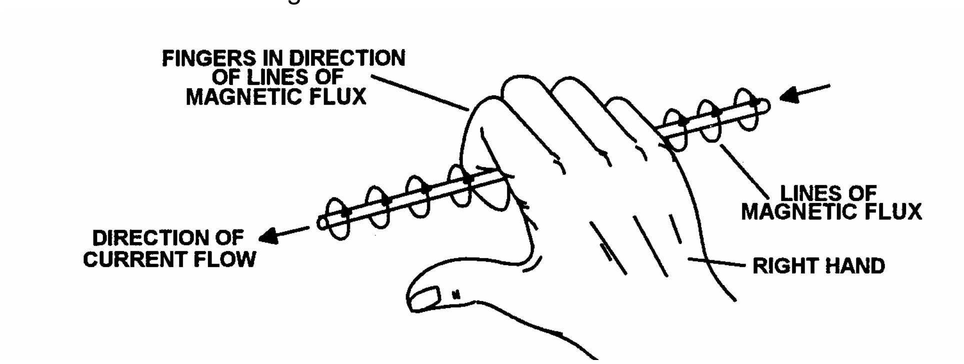
Permeability (μ) is the ease by which magnetic flux induces into a material, and is comparative to conductance in an electrical circuit. It is the ratio of B/H, where B is the induced magnetic flux, and H is the magnetising force.

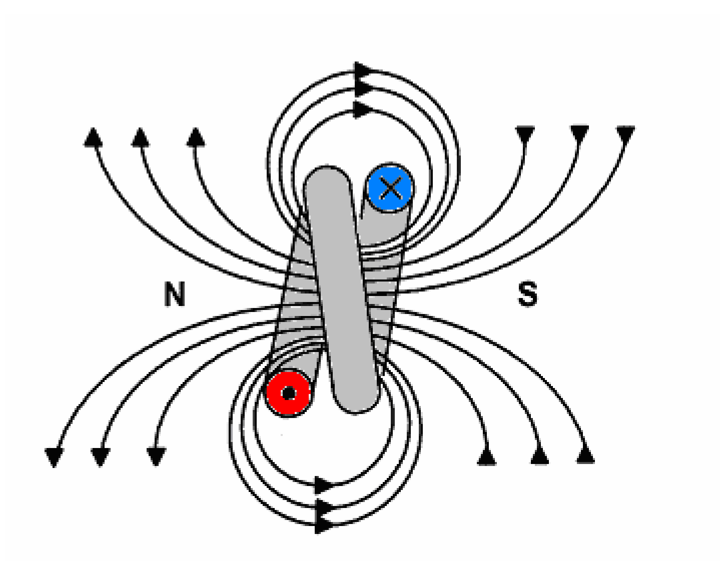
ELECTROMAGNETISM

When current flows through a conductor, a magnetic field forms around it, with the magnitude directly proportional to the amount of current flow.



The direction of the conventional current flow determines the direction of the field around the conductor. To establish the direction, use the Right Hand Grasp Rule, as shown below. By pointing your thumb in the direction of the current flow, and gripping the conductor, your fingers indicate the direction of the magnetic field.





The magnetic field produced by a straight piece of wire is relatively weak and is of no practical use. It has direction, but no north or south poles and, unless the current is extremely high, the resulting magnetic field has little useful strength. However, its magnetic characteristics greatly improve by shaping the wire into a loop.

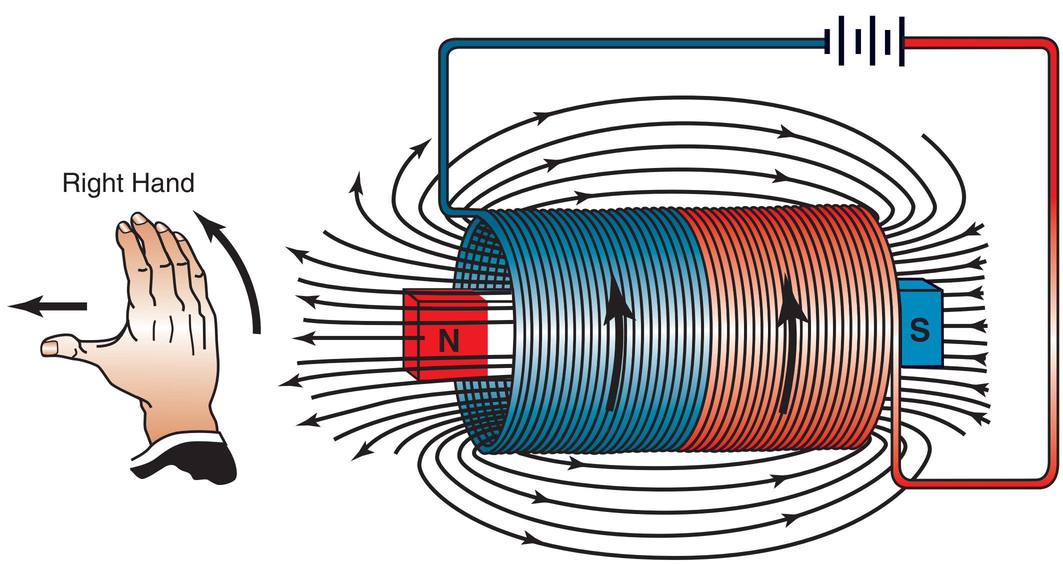
This causes:

- The lines of magnetic flux to move closer together
- The majority of the lines of magnetic flux to concentrate at the centre of the loop
- Creation of north and south poles at the ends of it, and assumption of the magnetic characteristics of a permanent magnet. Lines of magnetic flux emerge from the north pole and return via the south pole.

AN ELECTROMAGNET

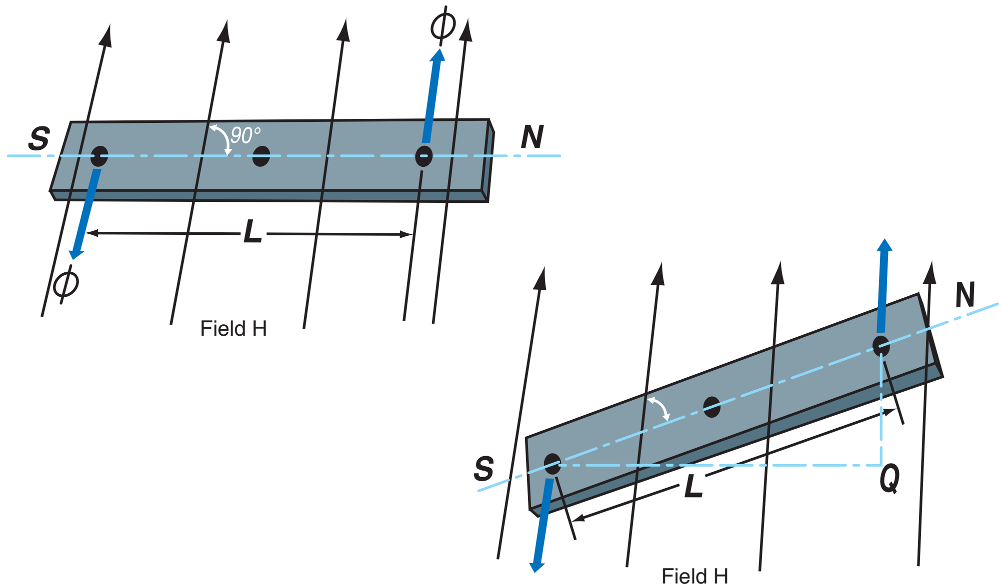
A strong magnetic field is produced if the wire is formed into a coil, which is commonly known as a solenoid, as shown on the next page. Also the larger the magnitude of the current, or the higher the number of turns of the coil, the greater the magnetic strength.

If a soft iron bar is inserted into a coil, this further increases the strength of the magnetic flux around the coil. This has the affect of concentrating the lines of magnetic flux, and the polarity of the coil can be determined if the direction of the current through the coil is known, by using the Right Hand Grasp Rule. Your thumb points in the direction of the North Pole if the fingers of your right hand are wrapped around the coil in the direction of the current flow.



MAGNETIC MOMENTS

The magnetic moment of a magnet is the tendency for it to turn, or be turned by another magnet. It is a requirement of any aeroplane compass design that the strength of the moment is such that the magnetic detection system rapidly responds to the directive force of the magnetic field.



The diagram above shows a pivoted magnet of pole strength S and length of magnetic axis L, which is positioned at right angles to a uniform magnetic field H. In this situation, the field distorts and "passes through" the magnet. The distortion resists and the field tries to align the magnet with the magnetic field. The forces applied to the magnet act in opposite directions, and the resulting magnetic moments (m = S x L) produce a couple, which swings the magnet into line with the magnetic field.

The greater the pole strength and the longer the magnetic moment, the greater the magnet's tendency is to quickly align itself with the applied field.

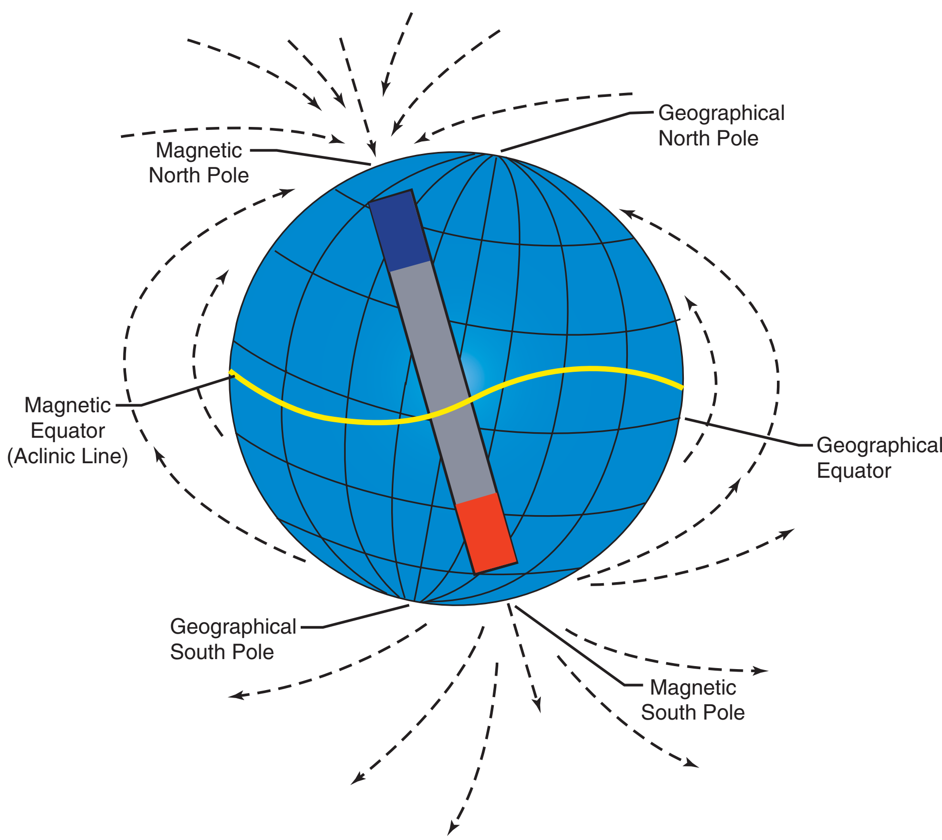
PERIOD OF OSCILLATION OF A SUSPENDED MAGNET

If a suspended magnet deflects from its position of rest in a magnetic field, the magnet is immediately subjected to a couple urging the magnet to resume its original position. When the deflecting influence is removed, the magnet swings back, and if undamped, the system continues to oscillate about its equilibrium position before coming to rest. The time taken for the magnet to swing from one extremity of oscillation to the other and back again is known as the **period** of the magnet. As the magnet approaches its aligned position, the amplitude of the oscillations gradually decreases, but the period remains the same, and is not altered by simply adjusting the amplitude. The period of a magnet depends upon its shape and size or mass (the factors that affect its moment of inertia), the magnetic moment, and the strength of the field in which it is oscillating. The period of oscillation increases if the magnet's mass is increased, and becomes shorter if its field strength increases.

Chapter 10
Terrestrial Magnetism

INTRODUCTION

The Earth is itself a huge magnet surrounded by a weak magnetic field that culminates in two internal magnetic poles situated near the North and South geographic poles. The Earth's magnetic field is similar to that produced at the surface of a short but very powerful bar magnet, and is why the magnetic poles cover relatively large geographic areas, as the lines of magnetic force spread out. This is why the lines of force are horizontal near the equator. The precise origin of the field is unknown, but for simplicity the analogy of the bar magnet at the Earth's centre is useful in visualising the general form of the Earth's magnetic field.

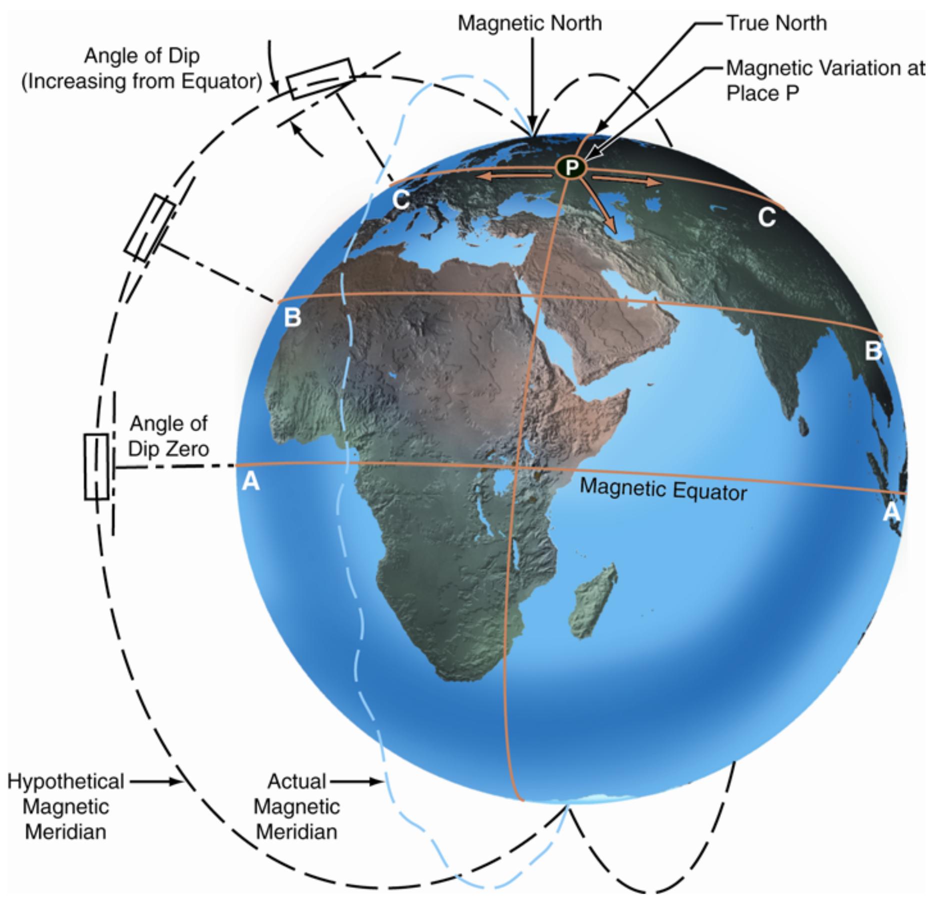


An imaginary line called a **Magnetic Meridian** joins the poles together. If a freely suspended magnetised needle is positioned at various locations within the Earth's magnetic field, it lines itself up with its red pole pointing towards the Earth’s magnetic North pole (i.e. with respect to magnetic North).

The Earth's magnetic field differs from that of an ordinary magnet in many respects because the magnetic poles themselves continually alter their position by a small amount. The magnetic field at any point on the Earth's surface is also not constant, because it is subject to both periodic and irregular changes.

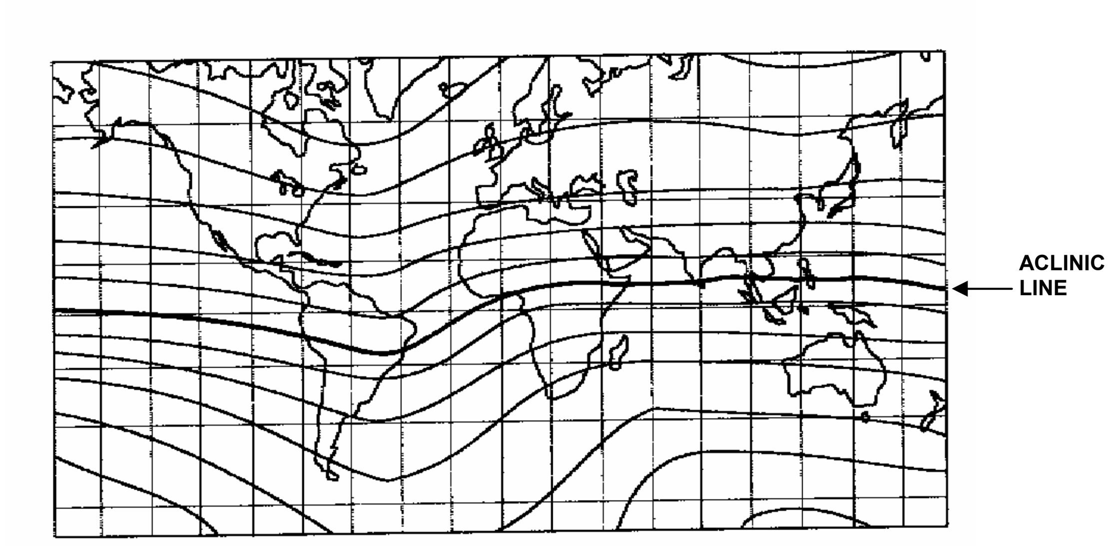
MAGNETIC DIP

A freely suspended magnetic needle settles in a definite direction at any point on the Earth's surface, by aligning itself with the magnetic meridian at that point. It does not lie parallel to the Earth's surface at all points, because the Earth's lines of magnetic flux (force) are themselves not horizontal.



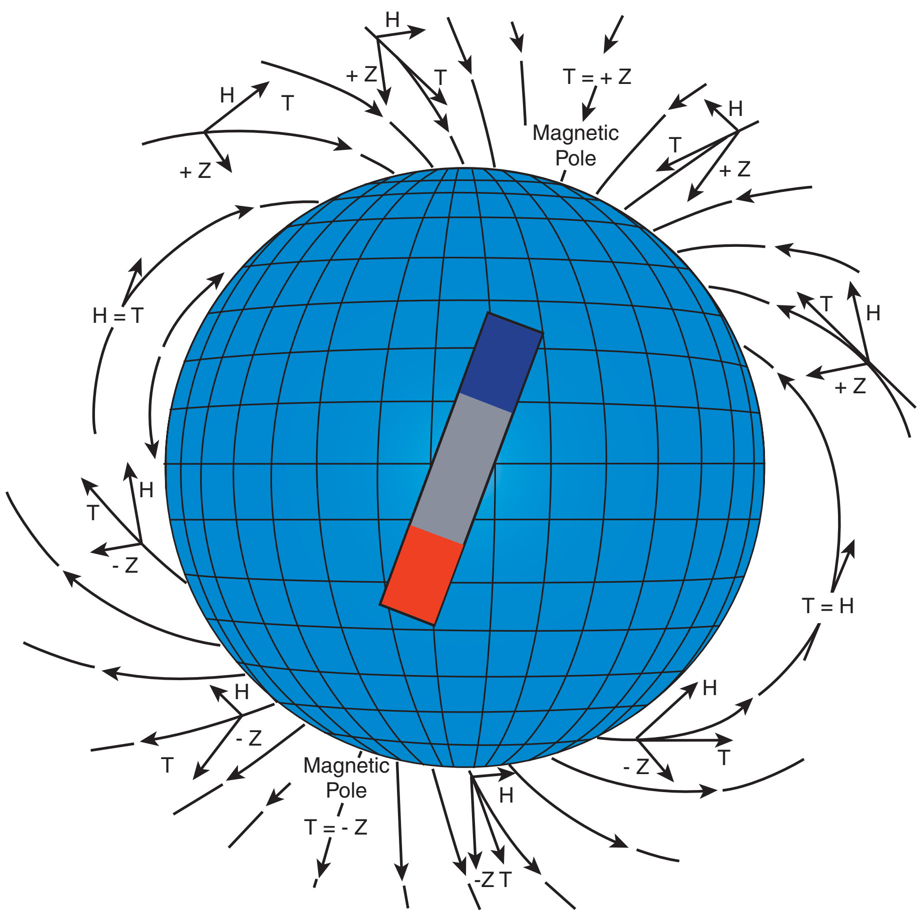
The lines of force initially emerge vertically from the South magnetic pole, and then bend over to become parallel with the Earth's surface, before descending vertically at the North magnetic pole. Thus if a magnetic needle is transported along a meridian from North to South, it initially has its red end pointing down towards the Earth. Near the magnetic equator, the needle is horizontal; and at the southern end of its travel the blue end points toward the Earth.

The angle that the lines of force make with the Earth's surface at any given place is called the **Angle of Dip** and varies from 0° at the magnetic equator, to virtually 90° at the magnetic poles. Lines drawn on the Earth’s surface joining places of equal dip are known as **Isoclinals** (BB and CC), whilst a line joining places having zero dip are an **Aclinic** lines (AA). The Aclinic Line is also the magnetic equator, which is close to the geographical equator, but is not the same line.



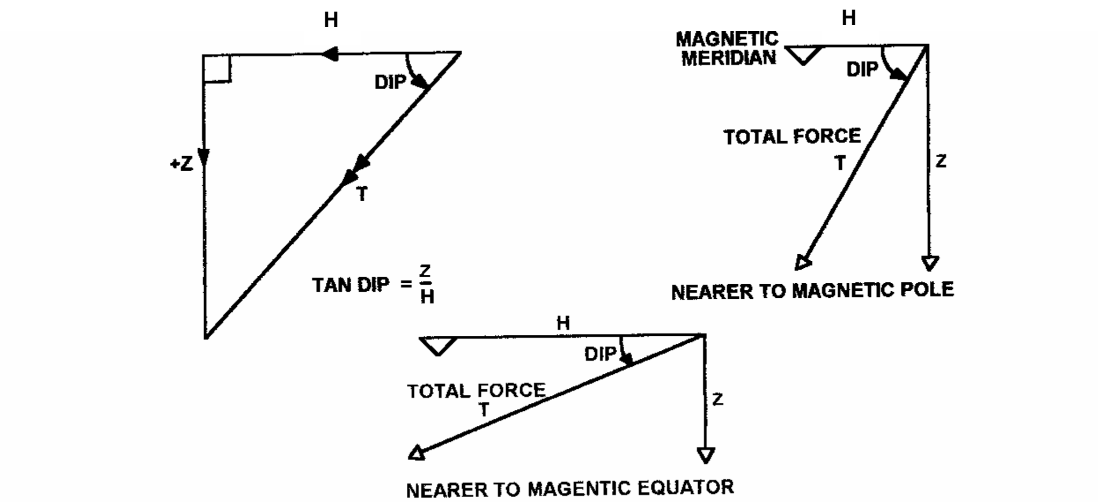
Isoclinals can also be plotted on charts of the world to show how the magnetic dip varies worldwide.

Dip is conventionally positive when the red end of a freely suspended magnetic needle is below the horizontal and negative when the blue end dips below the horizontal, as shown below.



EARTH'S TOTAL MAGNETIC FORCE

If a freely suspended magnetic needle comes to rest in the Earth's field, it does so under the influence of the total force of the Earth's magnetic field at that point. The value of this total force at any given place is not easy to measure, but needs to be known. It is usual, therefore, to resolve the total force into a horizontal component termed "H" and a vertical component termed "Z". If the value of angle of dip (θ) for the particular location is then known, the total force (T) is readily calculated. Knowledge of the horizontal component "H" and vertical component "Z" is of considerable practical value, as both are responsible for inducing magnetism into the various ferrous metal parts of the aeroplane (both hard and soft iron) which lie in their respective planes. Both components may be responsible for providing a deflecting or deviating force around the aeroplane's compass position, a force whose value requires determination and calibration, if the compass is to provide a worthwhile heading reference. The relationship between dip, horizontal, vertical and total force is shown below:



This figure shows that "H" is a maximum value at the magnetic equator and decreases in value towards the poles. Conversely, "Z" is zero at the magnetic equator and, like the value of dip, increases towards the poles.

EXAMPLES

1. If the value of H is 0.22 and the value of Z is 0.44 at a place X, then the angle of dip at this place is:

$$\text{Tan Dip} = \frac{Z}{H} = \frac{0.44}{0.22} = 2$$

$$\text{Angle of Dip} = \text{Tan}^{-1}\, 2 = 63°26'$$

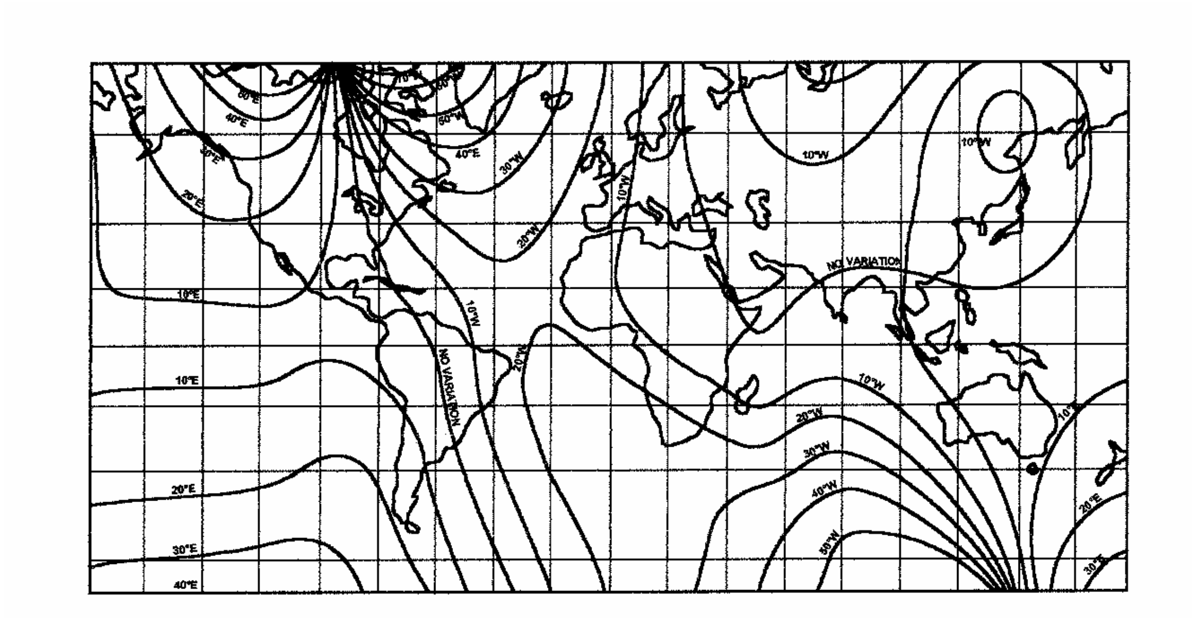
2. If the Angle of Dip = 60° and Z = 0.27, then the values of H and T are:

$$H = \frac{Z}{\text{Tan } 60°} = \frac{0.27}{1.7321} = 0.1559$$

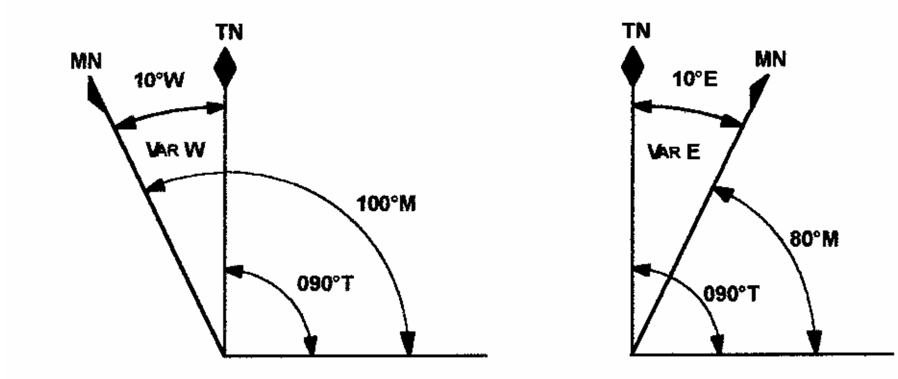
$$T = \frac{0.27}{\text{Sin } 60°} = \frac{0.27}{0.8660} = 0.3118$$

MAGNETIC VARIATION

In a similar manner, meridians and parallels, constructed with reference to the geographic poles, magnetic meridians, and parallels, may be plotted with reference to the magnetic poles. If a map is prepared showing both true and magnetic meridians, the meridians intersect each other at angles varying from 0° to 180° at different points on the Earth's surface, as shown below.



Isogonal or **Isogonic** lines are used on charts to show the amount of variation, and to join all places on the Earth’s surface having the same angle of variation, whereas a line on the chart where the variation is nil is called the **Agonic** line. When the direction of the magnetic meridian inclines to the left of the true meridian, the variation is ‘west’, when the inclination is to the right of the true meridian, the variation is ‘east’. Variation can change from 0° in areas where the magnetic meridians run parallel to the true meridian, to 180° in places located between the true and magnetic north poles. The angle between the true and magnetic meridians at any place when looking north is known as **magnetic variation**, an example of which is shown below:



At some locations on Earth, where the ferrous nature of the rock deposits disturbs the Earth's magnetic field, abnormal magnetic anomalies occur, which may cause large changes in the value of variation over very short distances.

While variation differs all over the world, it does not maintain a constant value in any one place, and the following changes, which are not constant in themselves, may occur:

- Secular change occurs over long periods, due to the changing position of the magnetic poles relative to the true poles.
- Annual change is a small seasonal fluctuation super-imposed on a secular change.
- Diurnal (daily) change, are apparently caused by electrical currents flowing in the atmosphere because of solar heating.
- Magnetic storms associated with sunspot activity. These may last from a few hours to several days, with the intensity varying from very small to very great. The effect on aeroplane compasses varies with intensity, but both variation and local values of "H", are modified whilst the "storm" lasts.

These factors also have a similar affect on the Angle of Dip.

INTRODUCTION

According to the JARs, it is a mandatory requirement that a direct reading compass is fitted as standard on all aeroplanes, and is fitted where the flight crew can easily read it. Magnetic material and electrical circuits surround the flight deck that influence the Earth's magnetic field and cause the compass needle to deflect away from the local magnetic meridian. This is known as **compass deviation** and can result in deviation either east or west of magnetic north by an amount, which depends on the aeroplane's heading and latitude. Fortunately, any deviation can be analysed and resolved into components acting along the aeroplane's major axes, and action can be taken to minimise these effects. The causes of the deviation are analysed and corrected for by carrying out a compass swing, although some residual deviation remains, and recording them on a deviation card.

TYPES OF AEROPLANE MAGNETISM

Aeroplane magnetism is classified in a similar manner to that of hard iron and soft iron (i.e. how readily the materials become magnetised).

HARD IRON MAGNETISM

This form of magnetism is of a permanent nature, and is due to the presence of iron or steel parts used in the aeroplane structure, power plants, and other equipment. The Earth's magnetic field influences the molecular structure of the ferrous parts of the aeroplane during its construction when it lies on one heading for a long period. Hammering and working of the materials also plays a major part in the magnetism of the aeroplane components, whilst they are lying in the magnetic field.

SOFT IRON MAGNETISM

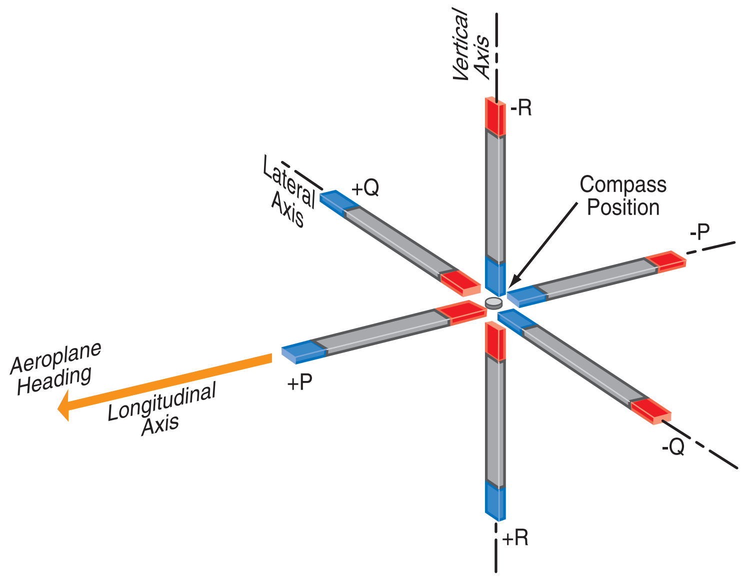
This form of magnetism is of a temporary nature and is caused by the magnetically soft metallic parts becoming magnetised due to induction by the Earth's magnetic field. The effect of this type of magnetism is dependent on the aeroplane's heading and the local Angle of Dip (magnetic latitude), and its geographical location.

Letters normally indicate the components of aeroplane magnetism, which cause deviation; capital letters indicate permanent hard iron magnetism, whilst small letters indicate induced soft iron magnetism. Positive deviations (those deflecting the compass needle to the right) are termed **easterly**, whilst negative deviations (deflection of the compass needle to the left) are termed **westerly**.

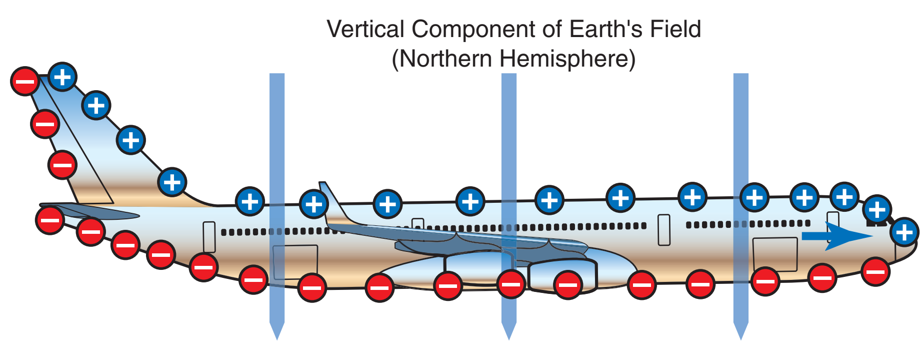
COMPONENTS OF HARD IRON MAGNETISM

The overall magnetic field created by the hard iron magnetism of the aeroplane is broken down into components which can be likened magnets lying longitudinally, laterally and vertically about the compass position, labelled as components ‘P’, ‘Q’, and ‘R’ respectively.

The magnetic strength of these components remains constant regardless of the aeroplane heading or latitude, but may vary with time due to a weakening of the magnetism in the aeroplane. By convention if the deviating effect of the aircraft’s magnetic field causes the compass needle to move clockwise and the north end of the compass to dip, it is classed as positive deviation. This situation would occur when a blue pole is ahead and to the right of the nose of the aircraft. The vertical component (R) has a minimal affect on compass heading in normal straight and level flight and is therefore not considered in the calculation of compass deviation, but its presence needs to be understood.

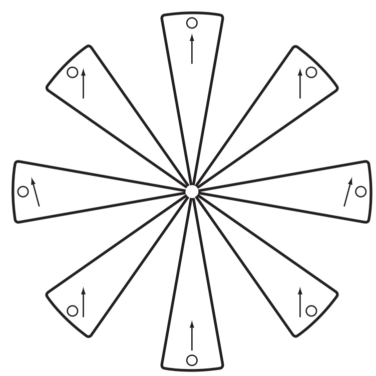


An aircraft built in the Northern Hemisphere has a vertical component of magnetism induced into it, which would be polarised blue above the airframe and red below as shown.



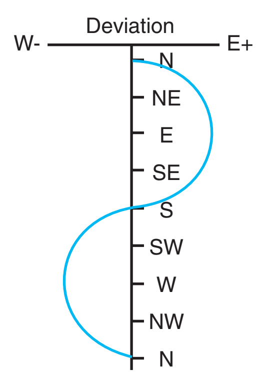
When the effect of this vertical magnetism is analysed in the horizontal plane through the compass needle, it reveals a net blue pole to the front of the aeroplane, and net red pole to the rear. Similar effects occur in the lateral vertical planes, with the overall polarity depending on the actual aeroplane design. The polarity and strength of the magnetism is not affected by the aeroplane heading.

If the aeroplane is heading north, the imaginary magnet due to component P, together with the compass needle, is in alignment with the aeroplanes fore and aft axis, and the Earth's directive force (H). P is added to, or subtracted from H, but does not cause any deviation. If the aeroplane is turned through 360°, then as the turn is commenced (ignoring compass pivot friction and liquid swirl etc) the magnet system remains attracted to the Earth's H component. Component P does continue to act along the aeroplane's fore and aft axis. It causes the compass needle to align itself in the resultant position between the directive force (H) and the deflecting force (P), which causes the needle to point a number of degrees east or west of north, depending on the polarity of P, as shown in the diagram below.



The amount of deviation increases during the turn, reaching a maximum value when travelling east and west, and zero when travelling north and south. Deviation resulting from a positive P is represented by a sine curve, as shown below.

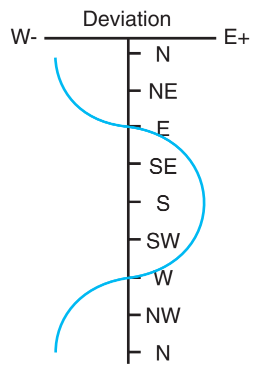
Hdg (C)	Deviation
N	No deviation; directive force increased
NE	Easterly deviation
E	Max Easterly deviation
SE	Easterly deviation
S	No deviation; directive force decreased
SW	Westerly deviation
W	Max Westerly deviation
NW	Westerly deviation



This shows that the deviation due to P is proportional to the sine of the aeroplane's Compass Heading, i.e.

$$\text{Deviation} = P \sin \text{Heading (C)}$$

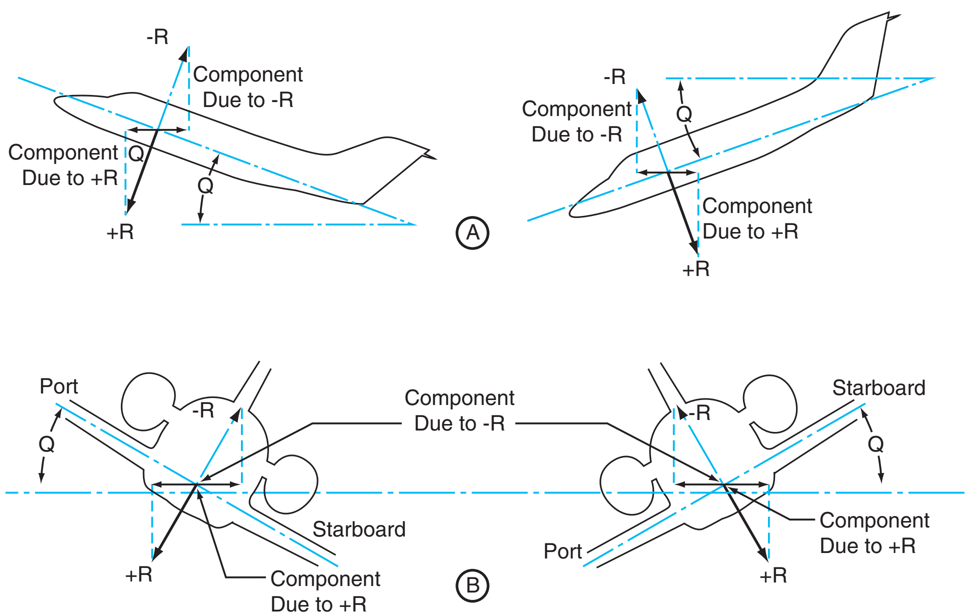
Component Q also produces a similar effect, but since it acts along the aeroplane's lateral axis (wing tip to wing tip), the deviation resulting from Q is a maximum value when travelling north and south, and zero when travelling east and west (i.e. when the component is aligned with the directive force [H]). Deviations resulting from a negative Q (blue pole to the left of the compass position) can be represented as a cosine curve, as shown below.



This curve shows that the deviation due to Q is proportional to the cosine of the aeroplane's Compass Heading, i.e.

$$\text{Deviation} = Q \cos \text{Heading (C)}$$

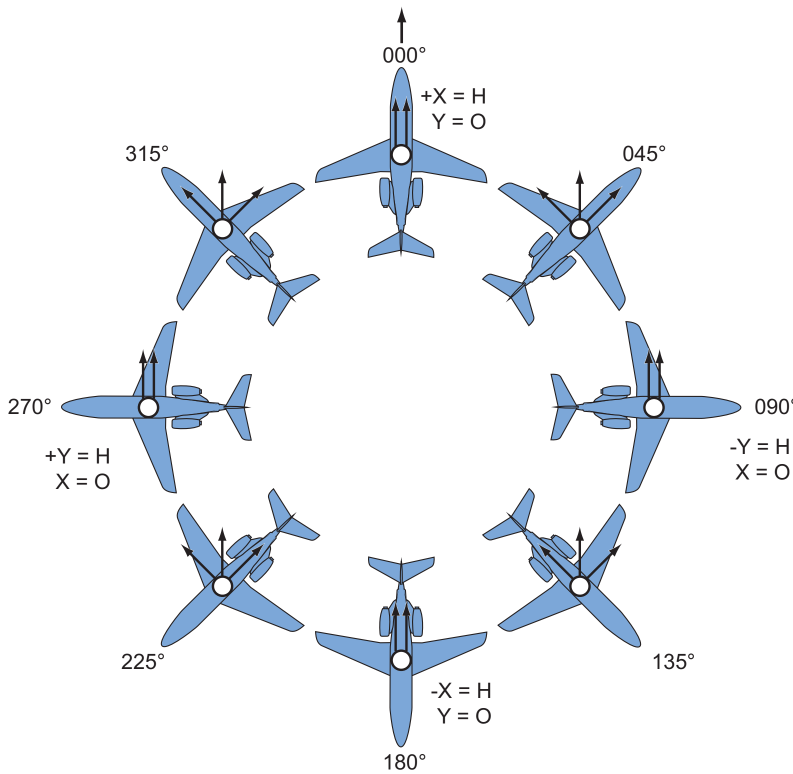
Component R acts in the vertical plane and thus has no effect on the compass system when the aeroplane is in level flight. If the aeroplane flies with its lateral or longitudinal axes away from the horizontal, it displaces component R from its vertical position, and the resulting horizontal vector of this component affects the compass system. The following diagram demonstrates this by illustrating how an element of R would affect the components P and Q.



Notably a similar situation exists with a tail wheel aeroplane when it is on the ground. The value of R may vary when the aeroplane is climbing and descending, but because the angles involved are normally small, any deviation resulting from component R is also correspondingly small. Additionally the turning and acceleration errors associated with a direct reading compass during turns makes the errors due to R of no practical significance. The effect of component R is also negligible in remote indicating compasses, since the turning errors are virtually eliminated in this instrument, because of its associated electronic circuitry.

COMPONENTS OF SOFT IRON MAGNETISM

Soft iron magnetism, which is present at the compass position, can be considered as originating from soft iron rods placed adjacent to the compass in which magnetism is induced by the Earth's magnetic field. This field has two components, H and Z, but in order to analyse the effect of soft iron, the H component must be further split into two horizontal components, X and Y. When these components are put together with the Z component of the Earth's magnetic field, they relate directly to the three principle axes of the aeroplane. The diagram below shows how the polarities and strengths of X and Y alter with a change in aeroplane heading, as the aeroplane turns relative to the direction of component H.



Component Z acts vertically through the compass, and, therefore, does not affect the directional properties of the magnet system. If the aeroplane moves to a new geographic location, then because of the change in the Earth's magnetic field strength and direction, all three components of soft iron magnetism change. The sign of Z changes only if the aeroplane changes the magnetic hemisphere in which it is operating.

The soft iron components, which mainly affect the compass are indicated conventionally by the small letters, 'c' and 'f', and are related to the Earth's field components X, Y, and Z. Out of the soft iron components, cz and fz are the most important, since their polarity remains the same, regardless of the aeroplane heading. They also act in the same manner as the hard iron components P and Q respectively. Pairs of vertical soft iron rods (VSI), which are positioned respectively fore and aft and laterally about the compass position, represent the components, cz and fz respectively. In the northern hemisphere (magnetic), the lower pole of each rod is induced with 'red' magnetism.

DETERMINATION OF DEVIATION COEFFICIENTS

In order to minimise the effects of hard and soft iron magnetism on the aeroplane's compass, it is necessary to determine the deviations caused by the components of aeroplane magnetism on various headings. The value of any deviations can then be analysed into the **coefficients of deviation**. Five coefficients exist, namely A, B, C, D and E, of which D and E are purely due to soft iron, and are not covered in this manual. The remaining coefficients are important to aeroplane magnetism and are as follows:

Coefficient A

This is usually constant on all headings and is caused by the misalignment of the aeroplane compass. To calculate this coefficient, find the average of the algebraic sum of the deviations resulting from a number of equally spaced compass headings. Readings are typically taken on the four cardinal and four quadrantal headings, thus:

$$\text{Coefficient A} = \frac{\text{Deviation on N + NE + E + SE + S + SW + W + NW}}{8}$$

Coefficient B

This is the result of the resultant deviation caused by the presence of hard iron P and soft iron cz, with the deviation being a maximum, when heading east or west. To calculate this coefficient, divide the algebraic difference between the deviations on the compass headings east and west by two, thus:

$$\text{Coefficient B} = \frac{\text{Deviation on east - Deviation on west}}{2}$$

For any given heading, coefficient B may also be expressed as:

$$\text{Deviation} = \text{B x sin heading}$$

Coefficient C

This is the result of deviations caused by hard iron Q and soft iron fz, with the deviation being a maximum when heading north and south. To calculate this coefficient, divide the algebraic difference between the deviations on compass heading north and south by two, thus:

$$\text{Coefficient C} = \frac{\text{Deviation on north - Deviation on south}}{2}$$

For any given heading coefficient, C may also be expressed as:

$$\text{Deviation} = \text{C x cos heading}$$

The total deviation on an uncorrected compass for any given aeroplane compass heading may be expressed as:

$$\text{Total deviation} = \text{A + B sin heading + C cos heading}$$

MINIMUM DEVIATION

If Coefficient A is not present, minimum deviation occurs on the heading where the value of B sin heading + C cos heading is minimum, thus:

$$\text{Tan heading} = \frac{C}{B}$$

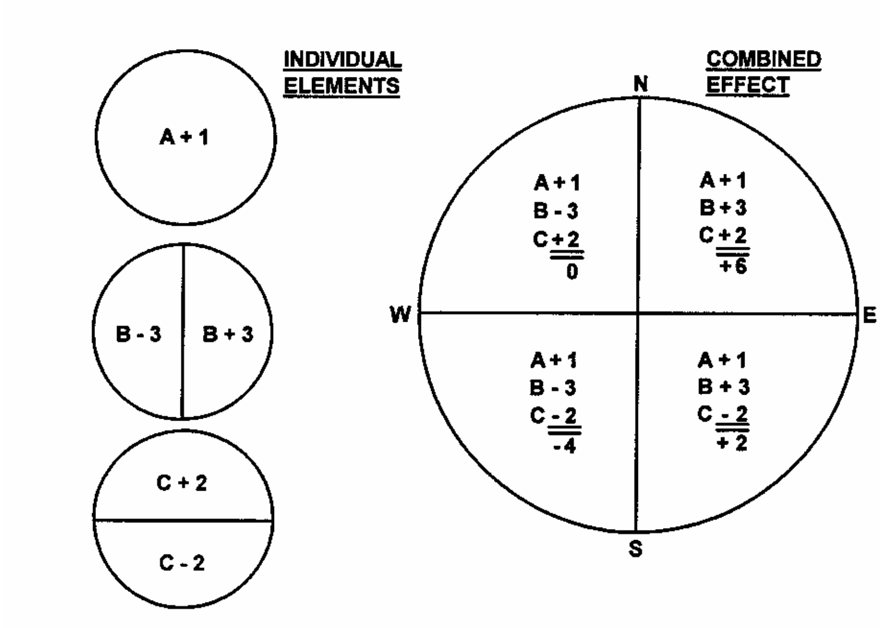
This heading is at right angles to that for maximum deviation, but if Coefficient A is present, determine the minimum deviation parameters by a compass swing.

MAXIMUM DEVIATION

If the deviation due to Coefficient A is constant on all headings, maximum deviation occurs when the value of B sin heading + C cos heading is a maximum, thus:

$$\text{Tan heading} = \frac{B}{C}$$

The value of the tangent derived for maximum deviation corresponds to two reciprocal headings in opposing quadrants, thus to determine the correct heading it is necessary to construct a swing circle, as shown on the next page.



The above diagram illustrates the condition where coefficients A = +1, B = +3 and C = +2. Maximum deviation thus occurs in the north-east quadrant. The actual value is:

$$\text{Tan } \frac{B}{C} = \frac{3}{2} = 1.5 \text{ which corresponds to a maximum deviation heading of } 056°$$

Substituting these values in the formula for maximum deviation gives:

$$A + B \sin \text{heading} + C \cos \text{heading} = 1 + 3 \sin 56° + 2 \cos 56° = +4.6°$$

JOINT AIRWORTHINESS REQUIREMENTS (JAR) LIMITS

JAR (25) for large aeroplanes requires that a compass residual deviation card (placard), showing the calibration of the magnetic compass (Direction Indicator) in level flight with the engines running, must be installed on or near the instrument. The placard must show each calibration reading in terms of magnetic heading of the aeroplane in no greater than 45° steps. Furthermore, the compass, after compensation, may not have a deviation in normal level flight of greater than 10° on any heading.

The following Joint Airworthiness Requirements (JARs) govern the positioning of the compass:

- The distance between a compass and any other item of equipment containing magnetic material shall be such that the piece of equipment does not result in a deviation of greater than 1°, nor shall the combined effect of any such equipment exceed 2°. The same ruling also applies to installed electrical equipment and any associated wiring when the systems power up.
- Any movement of the flight controls or undercarriage should not result in a change in deviation of greater than 1°.
- The effect of the aeroplanes permanent and induced magnetism, as given by coefficients B and C, together with any associated soft iron components shall not exceed:

Coefficient	Direct Reading Compass (degrees)	Remote Reading Compass (degrees)
B	15	5
C	15	5

Note: a). After correction the greatest deviation on any heading shall be 3° for direct reading compasses and 1° for remote indicating compasses.

b). Emergency standby compasses and non-mandatory compasses need not fully comply with JAR, but evidence of satisfactory installation is required.

COMPASS SWINGING

This is a special calibration procedure, carried out to determine the amount by which the compass readings are affected by hard and soft iron magnetism. This process enables determination of the deviations, calculation of the respective coefficients, and compensation for deviations. Carry out compass swinging:

- On acceptance of the new aeroplane from the manufacturer.
- When a new compass is fitted.
- Periodically, normally every three months.
- Following a major inspection.
- Following a change of magnetic material in the aeroplane.
- If the aeroplane is moved permanently or semi-permanently to another airfield involving a large change of magnetic latitude.
- Following a lightning strike or prolonged flying in heavy static.
- After standing on one heading for more than four weeks.
- When carrying ferrous (magnetic) freight.
- Whenever specified in the maintenance schedule.
- For the issue of a Certificate of Airworthiness (C of A).
- At any time when the compass or residual deviation recorded on the compass card are in doubt.

THE COMPASS SWING PROCEDURE

There are a number of methods to carry out a compass swing, but the most common method is to use an engineer with a landing or datum compass mounted on a tripod. The specialised compass is positioned either in front of, or behind the aeroplane, and is aligned with the aeroplanes fore and aft axis. An experienced compass adjuster normally carries out this process, which is split into two phases; the correcting, and the check swing, as follows:

- Ensure that the compass is serviceable.
- Ensure the removal of all equipment not normally carried in flight from the aeroplane.
- Ensure that all equipment normally carried in flight is correctly stowed.
- Take the aeroplane to a swing site (at least 50 m from other aeroplane and 100 m from a hangar).
- Ensure that the flying controls are in their normal flying position, the engines are running, and the radio and electrical equipment are switched on.
- Position the aeroplane on a heading of south (M) and note the deviation (i.e. the difference between the datum compass and the aeroplane compass readings).
- Position the aeroplane on a heading of west (M) and note the deviation.
- Position the aeroplane on a heading of north (M) and note the deviation. Calculate coefficient C and apply it directly to the compass reading.
- Insert the compass corrector key in the micro adjuster box, and turn the key until the compass needle shows the corrected reading. Remove the key.
- Position the aeroplane on a heading of east (M) and note the deviation. Calculate coefficient B and correct for B in the same manner as for coefficient C.
- The correcting swing is now complete.
- Carry out a check swing on eight headings, starting on southeast (M), and note the deviation on each heading.
- Calculate coefficient A on completion of the check swing and apply to the compass reading. Loosen the compass, or, for remote indicating instruments, the detector head retaining screws, and rotate the device until the compass needle indicates the correct heading. Re-tighten the retaining screws.
- Having applied coefficient A algebraically to all deviations found during the check swing, plot the (remaining) deviations, and make out a compass deviation card for placing in the aeroplane.

AN EXAMPLE OF A COMPASS SWING

The readings taken during a typical compass swing are as follows:

The Correcting Swing

<table>
<tr><th></th><th>Datum Compass Heading (M) (degrees)</th><th>Aeroplane Compass Heading (C) (degrees)</th><th>Deviation (degrees)</th><th></th></tr>
<tr><td>S</td><td>182</td><td>180</td><td>+2</td><td></td></tr>
<tr><td>W</td><td>274</td><td>270</td><td>+4</td><td></td></tr>
<tr><td>N</td><td>000</td><td>354</td><td>+6</td><td></td></tr>
<tr><td colspan="4">Coefficient C = $\frac{+6-(+2)}{2}$ = +2</td><td>Make the compass read 356°</td></tr>
<tr><td>E</td><td>090</td><td>090</td><td>0</td><td></td></tr>
<tr><td colspan="4">Coefficient B = $\frac{0-(+4)}{2}$ = -2</td><td>Make the compass read 088°</td></tr>
</table>

The Check Swing

Datum Compass (M) (degrees)	Aeroplane Compass(C) (degrees)	Deviation (degrees)	Residual Deviation Following 'A'
136	131	+ 5	+ 2
183	181	+ 2	- 1
225	221	+ 4	+ 1
270	268	+ 2	- 1
313	308	+ 5	+ 2
000	358	+ 2	- 1
047	044	+ 3	0
092	090	+ 2	- 1

$$\text{Coefficient A} = \frac{25}{8} = +3$$

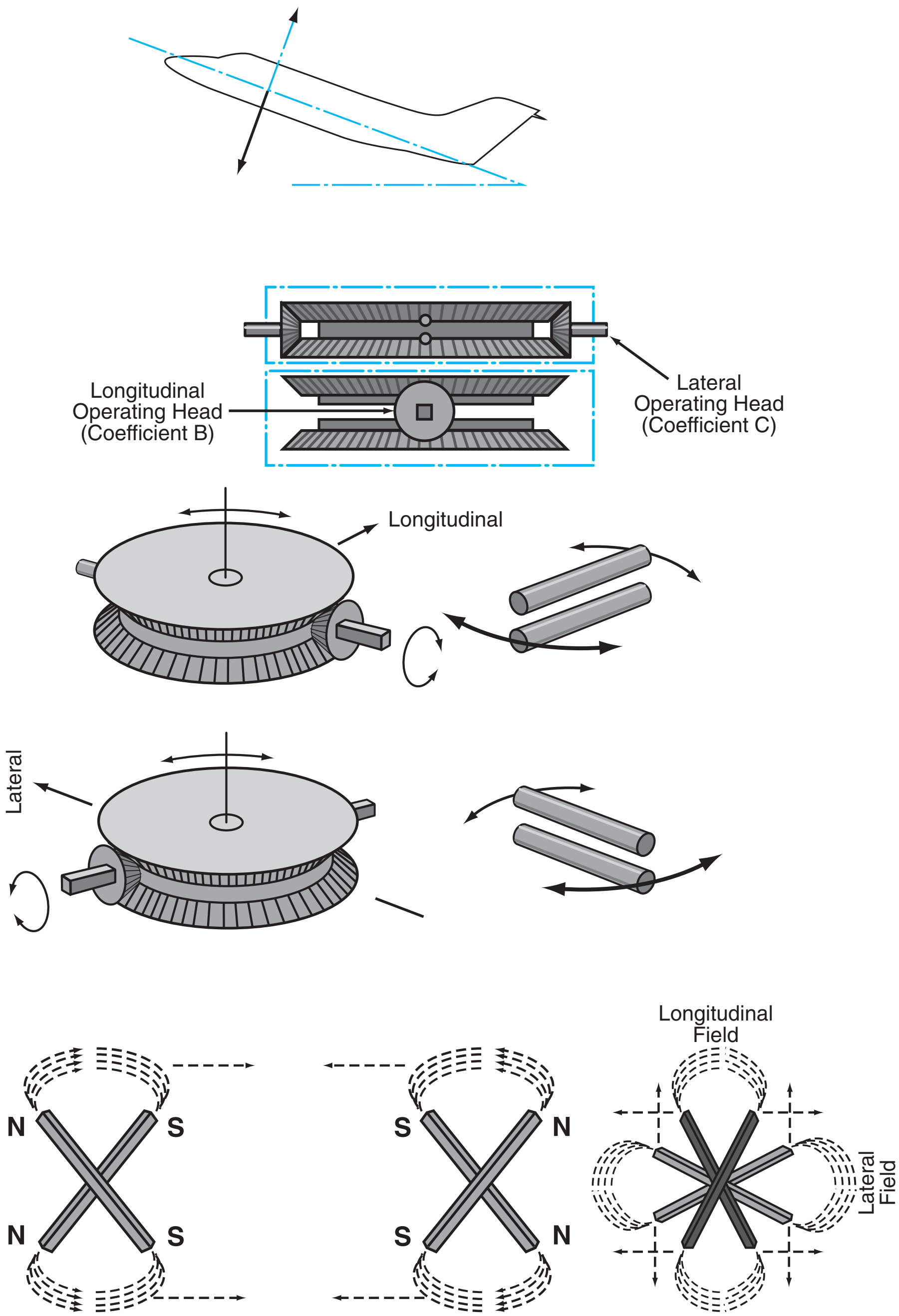
The resultant readings enable the production of a deviation card, showing residual deviations against magnetic headings, which is placed in the aeroplane adjacent to the compass position.

DEVIATION COMPENSATION DEVICES

Following the compass swing procedure, the resultant coefficients C, B and A, are used to correct or offset the compass needle by an amount in degrees equivalent to the existing deviation using one of the following methods:

MECHANICAL COMPENSATION

The majority of these devices consist of two pairs of magnets, which are fitted in a bevel gear assembly made of a non-magnetic material, and are mounted above each other, as shown below. This device is known as a **micro-adjuster**, and it ensures that when the magnets are in their neutral position one pair is parallel to the aeroplanes fore and aft axis to compensate for any Coefficient C corrections, whilst the other pair lies athwartships to compensate for any Coefficient B corrections. Using the compass correction key enables a small pinion to be turned, which in turn rotates one pair of bevel gears.



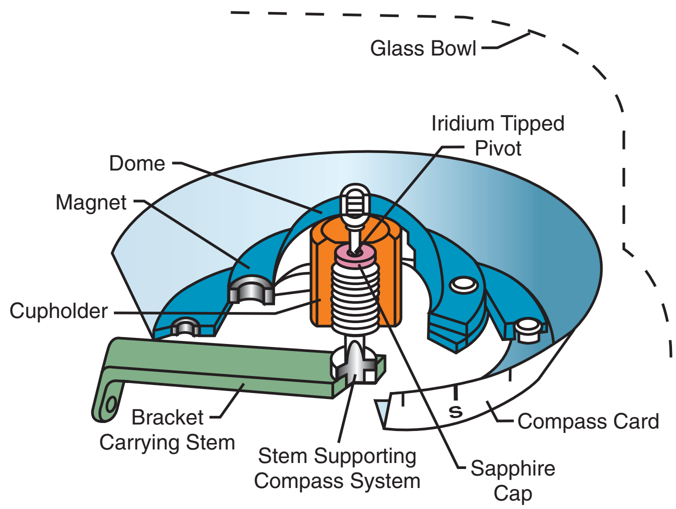
Each pair of magnets displace from their neutral position, as shown in the diagram above, and deflect the compass needle to correct for Coefficient B or C, depending on which pair of magnets are used.

ELECTRICAL COMPENSATION

This type is for use in the remote indicating compass and uses two variable potentiometers, connected to the coils of the flux detector unit. The potentiometers correspond to the Coefficient B and C magnets of a mechanical compensator and when moved with respect to calibrated dials, insert small DC signals into the flux detector coils. The resulting magnetic fields produced by these signals are sufficient to oppose those causing deviations and correspondingly modify the output from the detector head via the synchronous transmission link. This in turn drives the gyro and hence the heading indicator to display the corrected readings.

INTRODUCTION

The Direct Reading Magnetic Compass (DRC) is based on a simple magnetic needle, and points toward the northern end of the Earth's magnetic field. It is also installed in an instrument of dimensions and weight that makes it suitable for use in aeroplanes. Under JAR 25, it is a mandatory requirement that all modern civil transport aeroplanes carry a direct reading non-stabilised magnetic compass as a standby direction indicator. The most commonly found direct reading compass is the ‘E’ type, which is illustrated below.



PROPERTIES OF A DIRECT READING COMPASS

For a direct reading compass to function efficiently, the magnetic element must possess the following properties:

HORIZONTALITY

This ensures that the magnet system remains as nearly horizontal as possible, thereby sensing only the horizontal or directive component of the Earth's magnetic field. This is achieved by making the magnet system pendulous, by mounting the magnet, below the needle pivot, as shown in the diagram above. The magnet system when freely suspended in the Earth's magnetic field tends to align itself with the direction of that field (i.e. align itself in the direction of the total field (T), where T is the resultant of the Earth's horizontal (H), and vertical (Z) fields). If the system tilts, the C of G moves out from beneath the pivot and introduces a righting force upon the magnet system, which tends to oppose and reduce the overall 'Z' component. The compass takes up a position along the resultant of the two forces, 'H' and the reduced effect of 'Z', thus minimising the effect of dip. In temperate latitudes, the final inclination of the needle is approximately 2° to 3° to the horizontal, but this inclination increases when flying nearer the poles, such that, by about 70° north or south, the compass is virtually useless. The displacement of the C of G is purely a function of the system's pendulosity, and is not a mechanical adjustment, so it works in either hemisphere, without further adjustment.

SENSITIVITY

This ensures that the DRC is capable of operating effectively down to low 'H' values, and is achieved by increasing the pole strengths of the magnet used, so that it remains firmly aligned with the local magnetic meridian. Sensitivity is also aided by keeping pivot friction to a minimum by using an iridium-tipped pivot, which is free to move in a sapphire jewelled cup. The compass bowl is additionally filled with a liquid, which reduces the overall effective weight of the magnet system, and helps lubricate the pivot.

APERIODICITY

This ensures that the oscillation of the sensitive element about a new heading, following a turn, is minimised (i.e. a **deadbeat return** characteristic). If a suspended magnet is deflected from its position of rest and then released, it tends to oscillate around the correct direction for some time before stabilising. This is obviously undesirable, as it could, at worst, lead to the pilot chasing the needle. The compass needle comes to rest with minimal oscillation, achieved by:

1. Filling the bowl with methyl alcohol or a silicon fluid, and fitting damping filaments to the magnet system

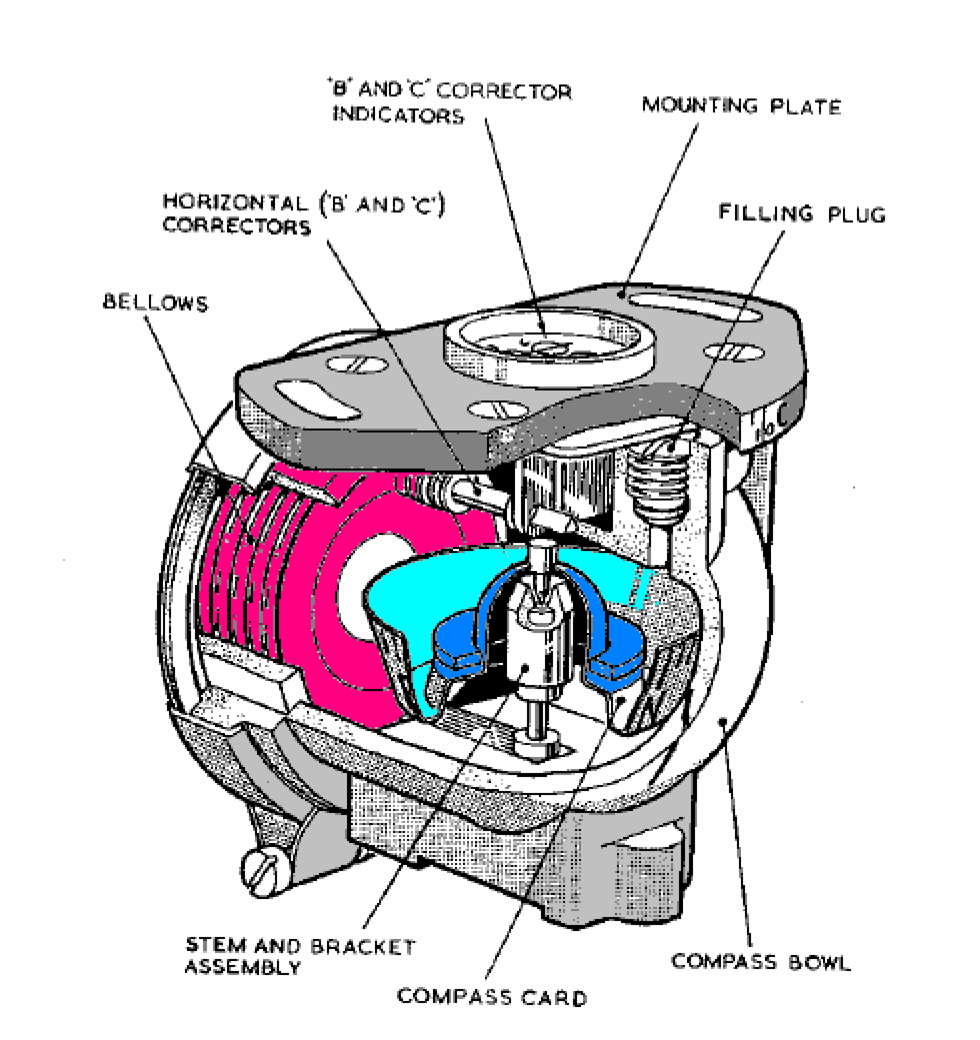
2. Keeping the lever arm of the magnet system as short as possible, but keeping its strength high. This has the effect of maximising its directional force, whilst reducing its moment of inertia

3. Using the fluid to reduce the apparent weight of the system

4. Concentrating the weight as close to the pivot point as possible, to reduce further the turning moment

'E' TYPE COMPASS

The majority of standby compasses in use today are of the card type, an example of which is shown below.



The compass consists of a single circular cobalt steel magnet; attached is a light metal compass card, mounted so that it rotates as close as possible to the inner face of the bowl, thus keeping parallax errors to a minimum during reading. The card is graduated with white markings on a black background, every 10°, with any intermediate indications estimated by interpolation. Figures are also shown every 30° and the cardinal points are marked with appropriate letters, N, S, E, and W. A white, vertical lubber line is engraved on the inner face of the bowl, against which the actual heading is observed.

The system is suspended on an iridium-tipped pivot that revolves in a sapphire jewelled cup, which is in turn secured to a central stem, and is firmly attached by a bracket to the base of the bowl. This gives the magnet system freedom of movement of $\pm$ 20° from the horizontal, and 360° in azimuth.

The bowl is moulded from plastic, and is painted on the outside with black enamel, except for a small area at the front through which the vertical card can be seen. This part of the bowl is also moulded so that it has a magnifying effect on the compass card.

The bowl is filled with a silicone fluid, which has no detrimental affect on the plastic bowl, and because its properties are not significantly affected by its temperature/viscosity. The liquid used in the compass bowl is also transparent and has a high resistance to corrosion. It must also not discolour during its use. Furthermore, a bellows-type expansion chamber is located at the rear of the bowl that compensates for changes in liquid volume due to any variation in temperature. This ensures that the liquid neither bursts a seal, nor contracts, leaving vacuum bubbles in the fluid.

One disadvantage of using a liquid in the compass bowl is that, in a prolonged turn, it tends to turn with the aeroplane, thus taking the magnet system with it, and affecting the compass readings. This is known as **liquid swirl**, which is minimised by providing a good clearance between the damping wires and the sides of the compass bowl. Liquid swirl also tends to delay the immediate settling of the system when a new compass heading is selected.

The effects of deviation co-efficient B and C are compensated for by permanent magnet corrector assemblies, secured to the compass mounting plate.

PRE-FLIGHT CHECKS

Prior to flight, the flight crew carries out the following checks:

- Check the security of the compass.
- Carry out a visual check for signs of any external damage.
- Check that the liquid is free from bubbles, discoloration, and sediment.
- Check that the compass illumination system is serviceable.
- Test for pivot friction by deflecting the magnet system through 10-15° each way, and note the readings on return, which should be within 2° of each other.

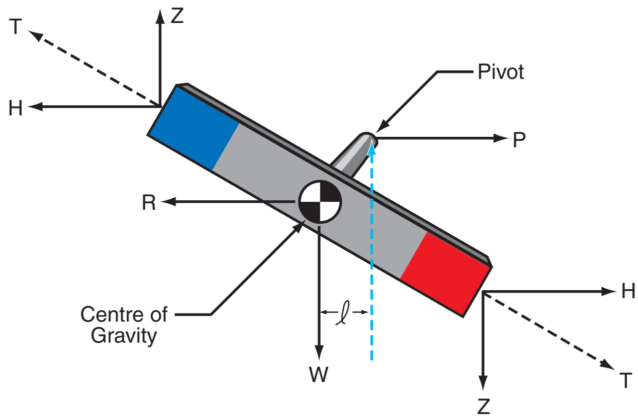
PRINCIPLE OF A PENDULUM

Consider a plain pendulum that is freely suspended in the aeroplane fuselage. If the aeroplane maintains a constant direction and speed, the pendulum remains at rest, but if the aeroplane turns, accelerates, or decelerates, the pendulum is displaced from its true vertical position. This occurs because the inertia of the pendulum causes the centre of gravity to lag behind the pendulum pivot, thus deflecting it away from its normal vertical position, directly beneath its point of suspension.

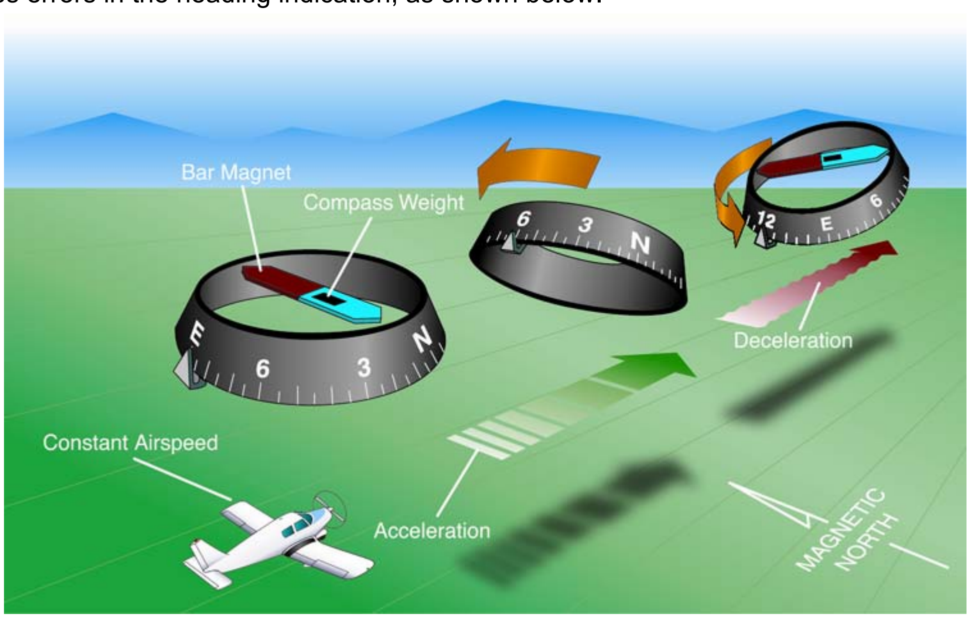
The magnet system (in the compass) is pendulous, so any acceleration or deceleration in flight will similarly result in a displacement of the C of G away from its normal position. This establishes a torque about the vertical axis of the compass, and unless the compass is on the magnetic equator, where the Earth's field vertical component 'Z' is zero, it is subject to dip.

ACCELERATION ERRORS

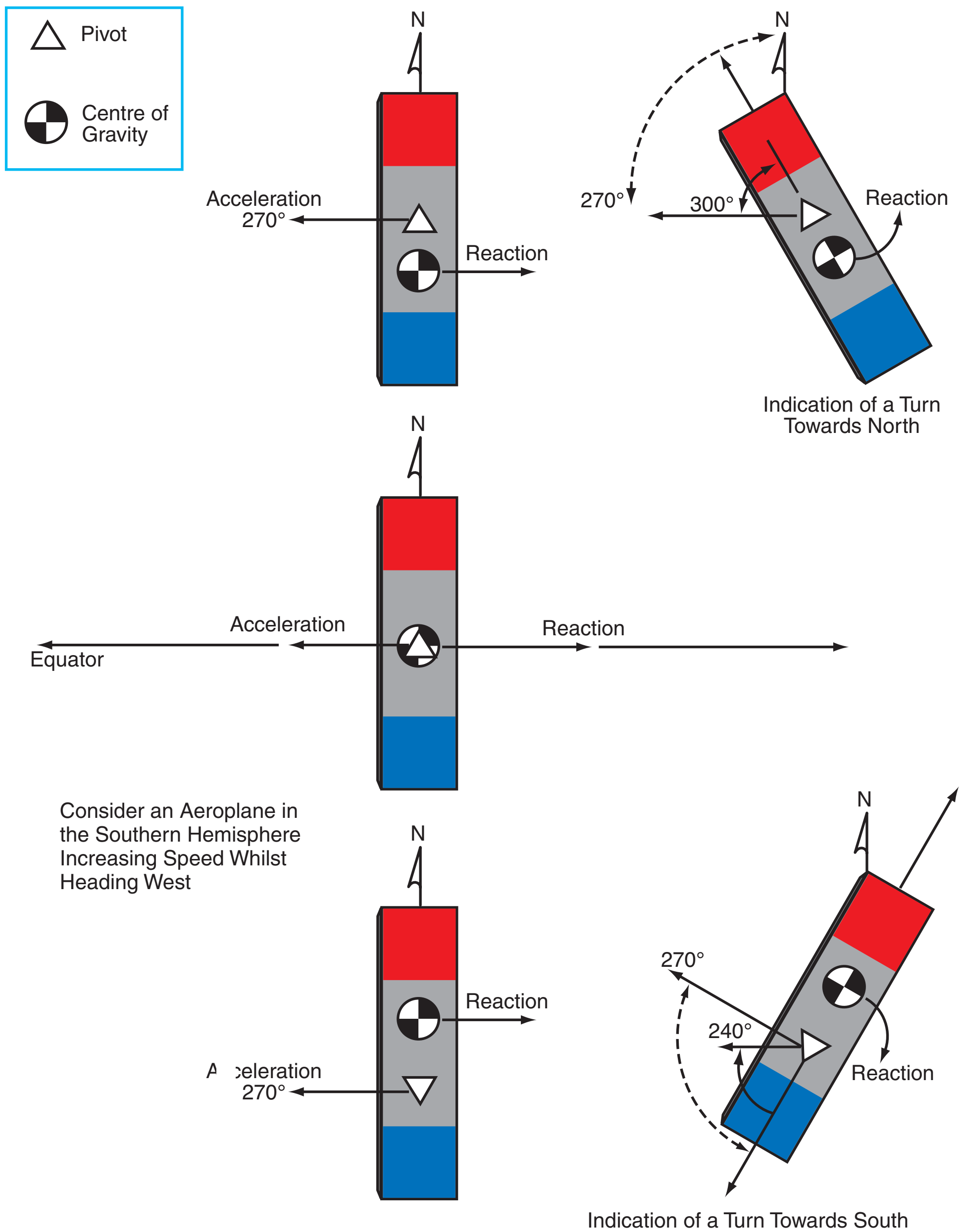
The force applied by an aeroplane when accelerating or decelerating on a fixed heading applies to the magnet system at the pivot, which is the magnet's only connection with the rest of the instrument. The reaction to this force is equal and opposite, and acts through the C of G of the magnet system, which is below and offset from the pivot (except at the magnetic equator), as shown below. The two forces constitute a couple which, dependent on the aeroplane's heading, causes the magnet system to alter its angle of dip (i.e. attempt to restore the magnet to its horizontal position or to rotate it in azimuth).



The diagram above shows how the forces affect a magnet system when an aeroplane is accelerating on a northerly heading. The resulting acceleration force is similarly applied to the magnet system at the pivot, whilst an equal and opposite reaction 'R' acts through the C of G, which is below, but offset from the pivot. The resultant couple causes the northern end of the magnet system to dip further, thus increasing the angle of dip without any rotation in azimuth. This occurs because the pivot 'P', and C of G, are both in the plane of the local magnetic meridian. Conversely, if the aeroplane decelerates when flying in a northerly direction, the resultant couple tilts the magnet system down at its southern end. The opposite occurs when accelerating/decelerating in a northerly direction along the magnetic meridian in the Southern Hemisphere. If the aeroplane is flying in either hemisphere, any changes in speed on headings other than northerly or southerly, also results in azimuth rotation of the magnet system, and produces errors in the heading indication, as shown below.



Acceleration errors are also caused by the vertical component of the Earth's magnetic field, which occurs because of the magnet systems pendulous mounting, and causes the compass card to tilt during changes of speed. This deflection causes a further error, which is most apparent on easterly and westerly headings. When an aeroplane is operated in the Northern Hemisphere and accelerates on either of these headings, the resulting error causes the magnet system to rotate, and the compass to indicate a turn to the north. Conversely, if an aeroplane decelerates on either of these headings, the resulting error causes the magnet system to rotate, and the compass to indicate a turn to the south.



These indications are reversed in the Southern Hemisphere. If the aeroplane decelerates when flying in a westerly direction, the action and reaction of ‘P’ and ‘R’ respectively, have the opposite effect, and cause the assembly to turn in the opposite direction, with all of the forces again turning in the same direction.

The errors due to acceleration and deceleration are summarised in the following table:

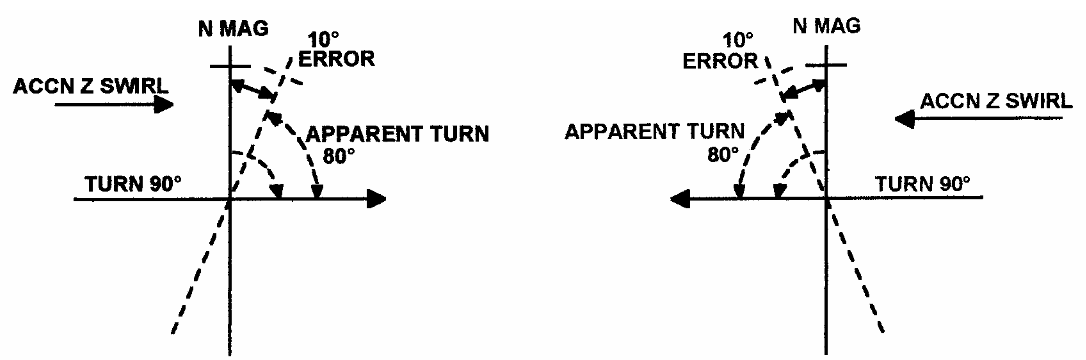
Heading	Speed	Needle Turns	Visual effects
East	Increase	Clockwise	Apparent turn to north
West	Increase	Anti-clockwise	Apparent turn to north
East	Decrease	Anti-clockwise	Apparent turn to south
West	Decrease	Clockwise	Apparent turn to south

Note:

1. In the Southern Hemisphere, the errors are in the opposite sense.
2. Similar errors can occur in turbulent flight conditions.
3. No errors occur at the magnetic equator, as the value of ‘Z’ is zero and hence the pivot point and C of G are co-incident with each other.

TURNING ERRORS

During a turn, the compass pivot is carried along the same curved path as the aeroplane. The centre of gravity (of the magnet system), offset from the pivot, is used to counter the effect of ‘Z’, and is thus subject to centrifugal acceleration. Furthermore, in a correctly banked turn the magnet system tends to maintain a position parallel to the athwartships (wingtip-to-wingtip) axis of the aeroplane, and is tilted in relation to the Earth's magnetic field. This places the pivot and C of G out of alignment with the local magnetic meridian. The magnet system is subject to a component of ‘Z’, and this causes it, when turning through North in the Northern Hemisphere, to rotate in the same direction as the turn. This further increases the turning error, and causes the compass to under-indicate, as shown below.



TURNS W-N-E AND E-N-W ALL ERRORS WITH TURN HENCE UNDER INDICATION

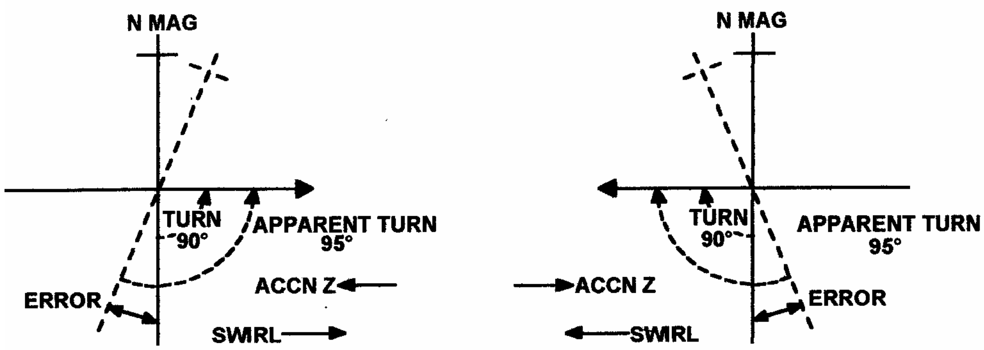
The magnitude and direction of the turning error is thus dependent on the aeroplane’s heading, its angle of bank (the degree of tilt of the magnet system), and the local value of ‘Z’ (dip). The turning error is a maximum value on northerly/southerly headings, and is particularly significant within 35° of these headings.

If an aeroplane turns east, as soon as the turn commences, the magnet system's C of G is subject to a centrifugal acceleration, and causes the system to rotate in the same direction as the turn. This, in turn, tilts the magnet system, and allows the Earth's vertical component 'Z' to exert a pull on the northern end, which causes further rotation of the system. The same effect occurs if the heading change is from north to west in the Northern Hemisphere.

The speed of rotation of the system is a function of the aeroplane's bank angle and rate of turn. Because of these factors, the compass may register the following indications:

- A turn in the correct sense, but smaller than that carried out when the magnet system turns at a slower rate than the aeroplane
- No turn when the magnet system turns at the same rate as the aeroplane
- A turn in the opposite sense because the magnet system turns at a faster rate than the aeroplane

When turning from a southerly heading in the Northern Hemisphere onto an easterly or westerly heading, the rotation of the system and indications registered by the compass are the same as when turning from north, except that the compass over-indicates the turn.



TURNS W-S-E AND E-S-W ACCN Z NOW AGAINST TURN
SWIRL WITH TURN ALWAYS

The effects of turning through North and South in the Northern Hemisphere are summarised in the following table:

Turn Direction	Needle Movement	Visual Effect	Liquid Swirl	Corrective Action
Through North	Same as aeroplane	Under Indication	Adds to Error	Turn less than needle shows
Through South	Opposite to aeroplane	Over Indication	Reduces Error	Turn more than needle shows

The liquid in the bowl not only provides damping, but it also tends to turn with, and in the same direction as the turn. Referred to as Liquid Swirl, this motion either adds to, or subtracts from, the overall needle error, which is dependent on its relative movement.

In the Southern Hemisphere, the south magnetic pole dominates, and in counter-acting its downward pull on the compass magnet system, the C of G moves to the northern side of the pivot. The errors are in the opposite sense. If an aeroplane turns from a northerly heading onto an easterly heading, the centrifugal acceleration acting on the C of G causes the needle to rotate more rapidly in the opposite direction to the turn, thus indicating a turn in the correct sense but of greater magnitude than that actually carried out. The turn is over-indicated. Turning from a southerly heading onto an easterly or westerly heading in the Southern Hemisphere, because of its C of G, which is still north of the compass pivot, results in the same effect as turning through north in the Northern Hemisphere.

Note: 1. In the Southern Hemisphere, the errors are opposite to those occurring in the Northern Hemisphere.

2. The Northerly turning error is greater than southerly, as liquid swirl is additive to the compass magnet system movement.

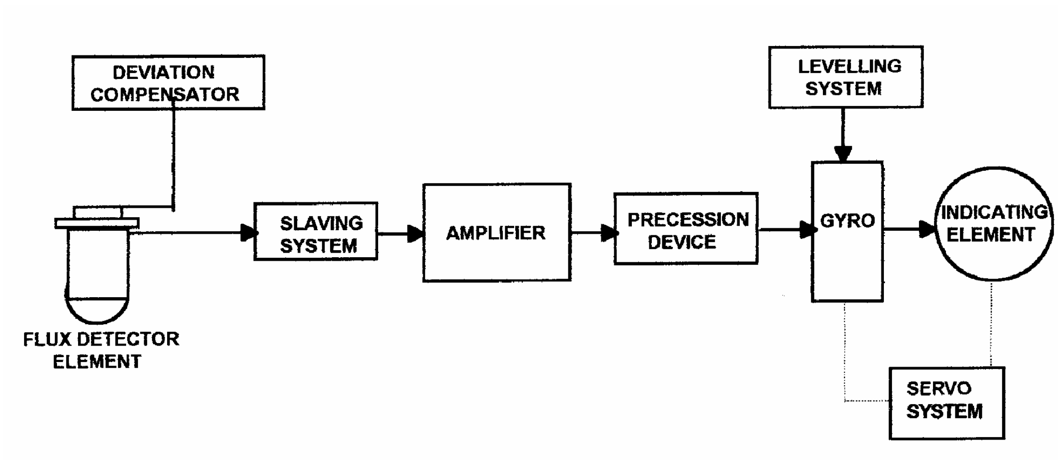
Chapter 13
Remote Indicating Compass

INTRODUCTION

A **remote indicating compass (RIC)** or **gyro magnetic compass (GMC)** combines the use of a gyroscope with the Earth's lines of magnetic flux to give the best of both a **Direction Indicator (DI)** and a **Direct Reading Compass (DRC)**. This type of compass essentially consists of a magnetic compass whose indications are stabilised via a gyroscopic element, thus minimising the affects of turning and acceleration errors. The magnetic part of the system simply senses the Earth's meridian, and does not align itself with it, although it is subject to aeroplane accelerations. Changing magnetic fields or normal acceleration forces does not affect the gyroscopic element, but it is subject to precessional forces caused by friction etc. This type of compass thus integrates the heading indication provided by the magnetic compass with the directional properties of the gyroscope to minimise the overall errors, in order to provide accurate indications of an aeroplanes heading.

RIC ARCHITECTURE

The main components of a typical RIC system are shown in the following diagram.

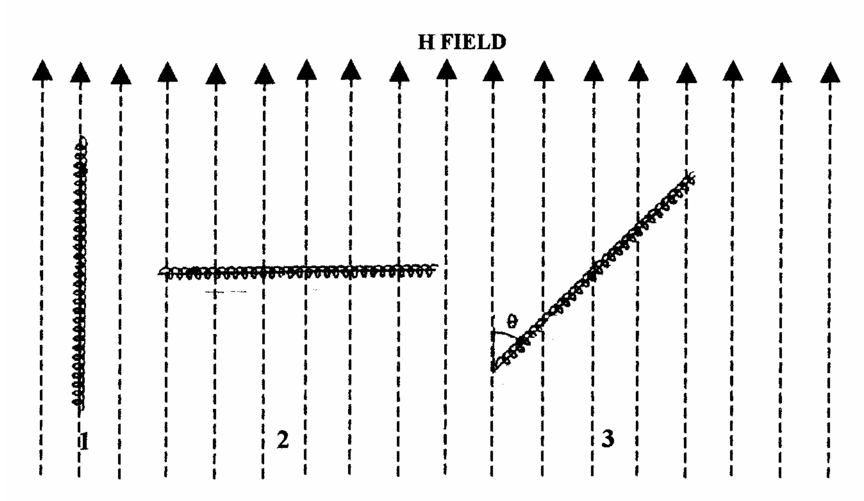


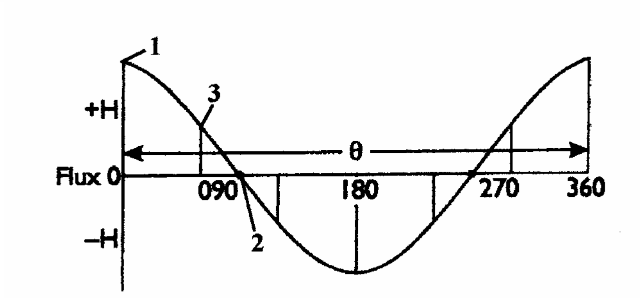
PRINCIPLE OF A FLUX DETECTOR ELEMENT

Unlike the detector element of the simple magnetic compass, the element used in a remote-indicating compass is fixed-in-azimuth, senses the effect of the Earth's magnetic field as an electro-magnetically induced voltage, and operates as follows:

If a highly permeable magnetic bar or coil is exposed to the Earth's field, it acquires a magnetic flux, which is solely dependent on the magnetic latitude at which the system is operating. The amount of flux induced is determined by the strength of the Earth's horizontal 'H' component, and the direction of the permeable element relative to the direction of this component.

The diagram below shows the amount of flux induced in a single coil when placed at different orientations to the Earth's magnetic (H) field. If the coil is placed with its longitudinal axis parallel to the H field, the maximum magnetic flux passes through the coil. If the coil is alternatively rotated through 90°, so that it is at right angles to the field, it produces zero magnetic flux, and if the coil was rotated through a further 90°, it re-aligns itself with the H field, but this time in the reverse direction. In this position, it again produces maximum flux, but is in the opposite algebraic sense. The coil thus shows a cosine relationship (zero flux at 90° and maximum flux at 0°) between the field direction and the coil alignment.

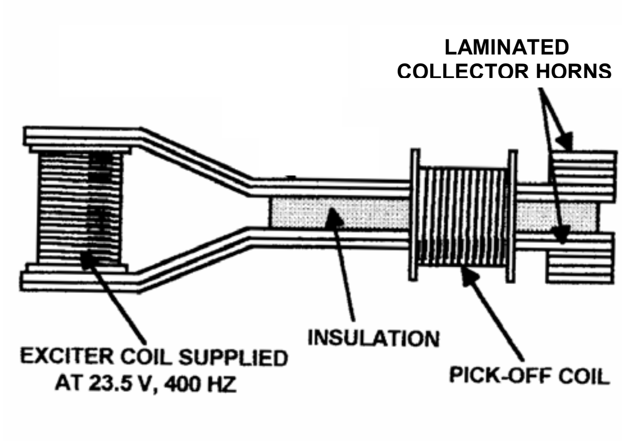




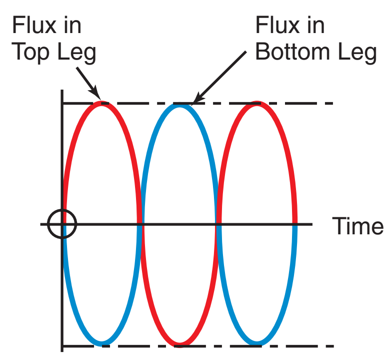
For example if an aeroplane is on a heading of 060° (M), the flux intensity is H cos 60°. Similarly, the flux intensity due to the Earth's magnetic field on a heading of 120° (M) is again H cos 60°, but the polarity of the flux has reversed, since cos 120° is negative. Conversely, on a heading of 300° (M) the induced flux is the same sign and value as for a heading of 060° (M).

A simple system is thus impracticable, because in order to determine the magnetic heading it is first necessary to measure the magnetic flux in the coil, which is difficult to establish, and second, it is subject to an ambiguity in heading, which requires resolution.

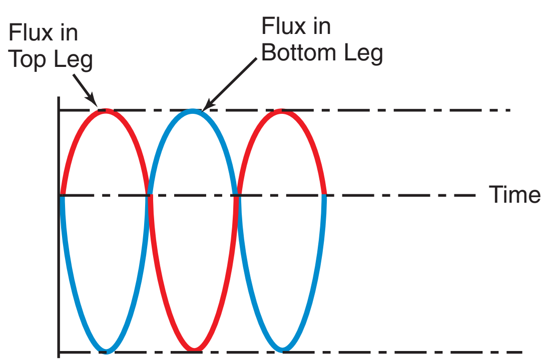
If, according to Faraday, "there is a change of flux linked with a circuit, an EMF will be induced in that circuit", then the flux could be easily converted into a measurable electrical current. For an aeroplane, however, at any given position and direction, if a single coil was used the flux produced would be of constant value. It is therefore necessary to convert the steady flux into a changing one, so that a current representing the actual heading would flow. This is achieved in the flux detector unit via a device called a **Flux Valve**.



A flux valve consists of two identical bars or spokes of highly permeable (easily magnetised and de-magnetised) material, mounted on a common hub. The hub is wound with a coil, known as the **Primary Coil**, and is connected in series to an AC single-phase power source of 23.5volts, at a frequency of 400 Hz. A pick-off or secondary coil is wound around both bars and registers the rate of change of flux in the permeable material. The effect of passing an alternating current through the primary coil has the following affect on the amount of flux produced in each leg:



The amplitude of the flux produced in each leg is identical, although they are 180° out of phase with each other, so that the algebraic sum of the fluxes, or the total flux, equals zero. This is because at any instant of time the two bars produce flux of equal and opposite (sign) intensity. In practice this situation never occurs, since a bar placed horizontally in the Earth's magnetic field is always subject to an 'H' component (unless the aeroplane is near the north or south magnetic pole). This component produces a steady flux in both bars, which when added to their individual fluxes; bias the system by an amount equal to 'H', as shown below.

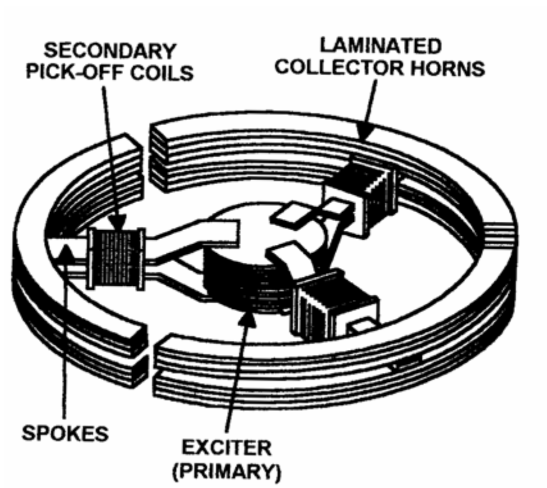


The algebraic sum of the fluxes in each leg is no longer equal to zero, and the resultant amplitude is directly proportional to the aeroplane's heading.

The changing flux in the bars results in an EMF or voltage induced in the pick-up or secondary coil, which is proportional to the 'H' component that acts along the axis of the flux valve.

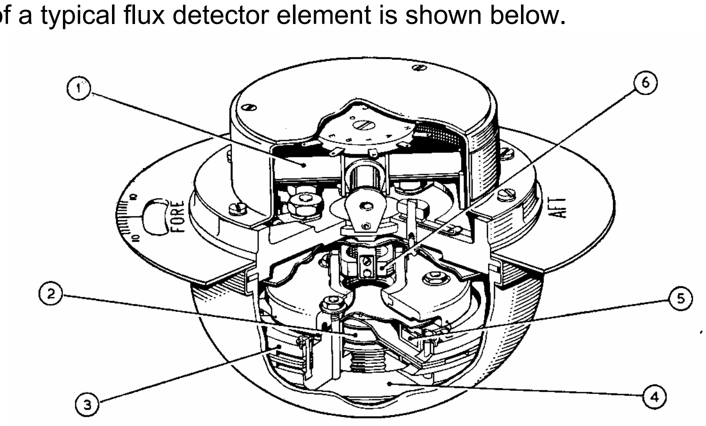
The single flux valve, however, has ambiguity over four headings, although two of these have different algebraic signs, which is resolved in the detector by using three spokes (flux valves), positioned 120° apart, as shown on the next page. In this arrangement, a laminated collector horn is located at the outer end of each flux valve to concentrate the lines of the Earth's magnetic force along the parent spoke. This increases the overall sensitivity of the detector head and increases the magnitude of the induced voltage in the secondary coil.

It is still possible to align the compass with a 180° error, but the instrument detects this, and immediately starts to precess the gyro unit to the correct heading.



FLUX DETECTOR UNIT

The construction of a typical flux detector element is shown below.

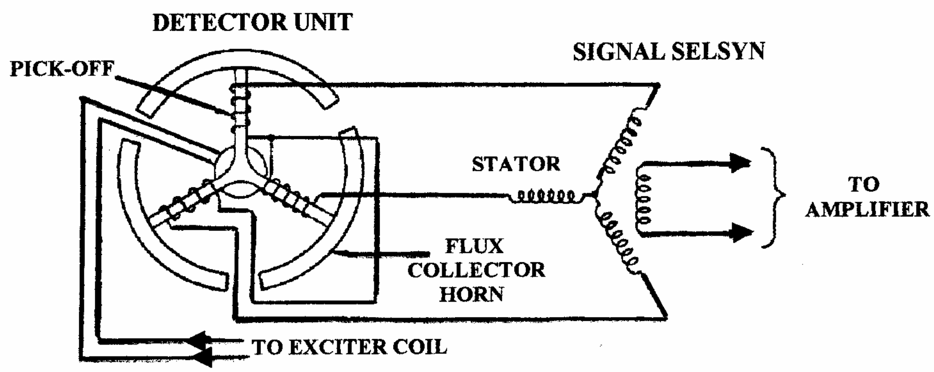


1. ELECTRO-MAGNETIC DEVIATION COMPENSATOR
2. EXCITER COIL
3. FLUX VALVE
4. PENDULOUS WEIGHT
5. PICK-OFF COIL
6. UNIVERSAL JOINT

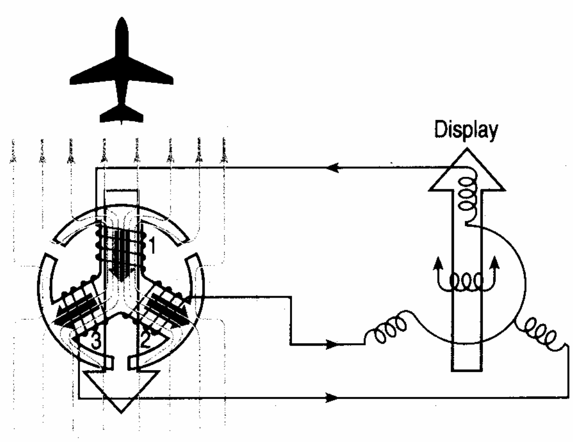
The spokes and coil assemblies are pendulously suspended from a universal **Hooke** joint. This permits limited freedom in pitch and roll, and ensures that the magnetic element senses the maximum value of 'H'. Unlike a Direct Reading Compass, the magnetic element has no freedom in azimuth. The unit's case is hermetically sealed and partially filled with fluid to damp out any element oscillation. The complete unit is normally fitted in the wing or fin tip, so that it is separated from electronic circuits and aeroplane magnetism near the flight deck, which may cause compass deviation. The complete unit is secured to the aeroplane by means of a flange, which contains three screw slots, one of which has calibration marks, and allows the removal of any coefficient A error. A compensating device is also fitted at the top the detector unit casing, which enables the removal of any coefficient B and C errors.

OPERATION OF THE REMOTE INDICATING COMPASS SYSTEM

The directional reference established by the remotely located detector unit electronically transmits to another location in the aeroplane via a signal selsyn (synchro receiver) unit, as shown on the next page. It is for use in monitoring the action of a gyro, or simply displayed on an indicator as a value of aeroplane heading. Any monitoring is carried out through a series of selsyn (synchro) units, which in turn synchronises the movements of the individual system elements.

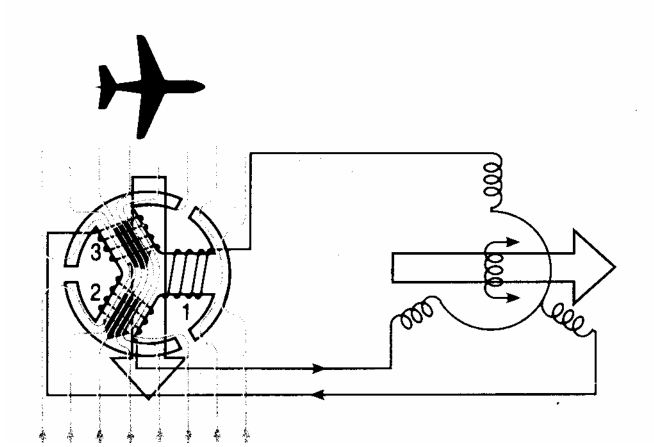


If the flux detector is positioned on a steady heading, say 000°, as shown below, then a maximum voltage signal is induced in the pick-off coil (secondary winding) 1, whilst coils 2 and 3 have voltages of half strength and opposing phases induced in them.

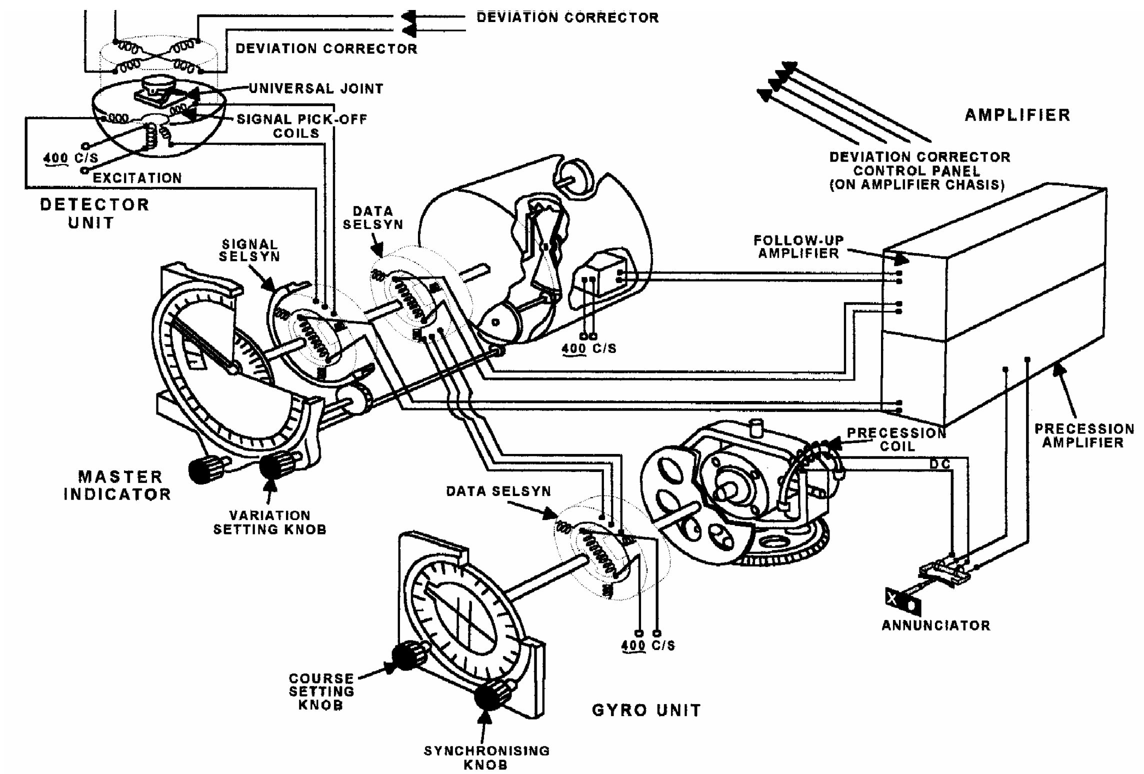


These signals are then fed to the corresponding legs of the stator of a signal selsyn, where the reproduction of the voltages occurs, and combines to establish a resultant field across the centre of the stator. The resultant is in exact alignment with the Earth's magnetic field that is passing through the detector unit. If the position of the rotor of the signal selsyn is at right angles to the resultant, no induction of voltage occurs in the windings, and it is in a ‘null’ position. The monitored directional gyro is similarly aligned with the Earth’s resultant field vector, so the heading indicator also shows 000°.

If the aeroplane turns through 90°, the disposition of the flux valve pick-off coils are as shown below.



No signal voltage is induced in coil 1, whilst coils 2 and 3 have increased voltages, with the voltage in 3 opposite in phase to 2. The resultant voltage across the signal selsyn stator rotates through 90°, and is directly related to the aeroplane’s magnetic heading. Assuming that the selsyn and gyro are still in their original positions, the resultant is now in line with the selsyn rotor, and a maximum voltage is induced in the rotor. The resulting voltage error signal is then fed to a precession amplifier, where it is phase-detected, and amplified, before it is passed to a torque motor fitted around the Directional Gyro’s (DG) horizontal axis.



The motor applies a torque, which then precesses the gyro around the vertical axis, and rotates the data selsyn rotor, attached to the drive shaft of the Gyro Magnetic Compass. This induces a voltage in the stator, which is duplicated in the data selsyn stator in the Master Indicator unit. If the rotor is out of alignment with the resultant magnetic field, a voltage error signal is induced in the rotor, fed to a Follow-up Amplifier, amplified before passing to a servomotor, and mechanically coupled to the data and signal selsyn rotors. Both rotors and the dial of the Master Heading Indicator rotate until the correct heading is indicated. The rotors are additionally coupled, such that when rotation is complete, both rotors lie in the null position of the fields produced in their individual stators (i.e. at right angles to the resultant of the field induced in the selsyn stators, and hence no current flows).

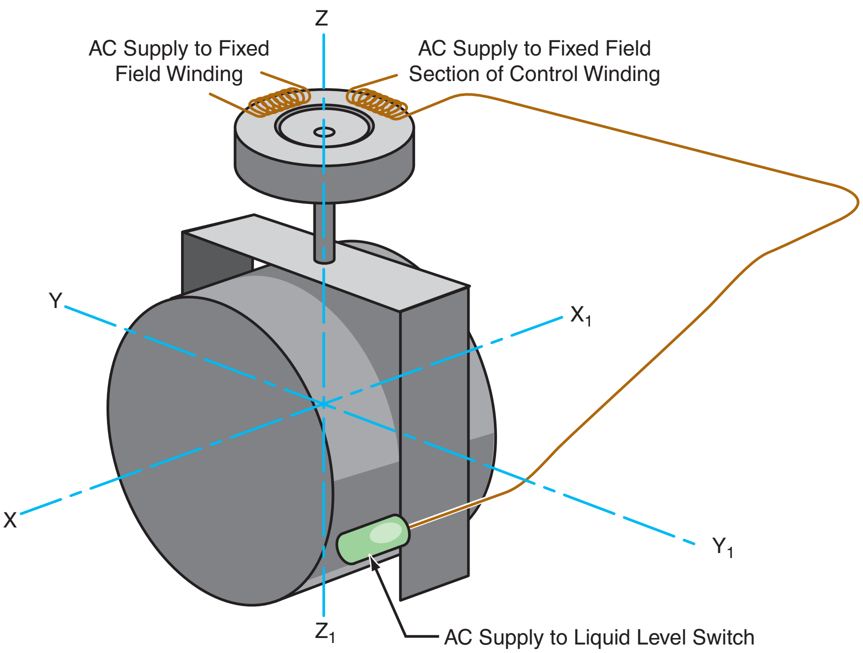
The servomotor drives a tach-generator that supplies feedback signals to the Follow-up Amplifier, and damp out any oscillations in the system. Heading information is also transmitted to other locations in the aeroplane through the installation of additional selsyns (servo-transmitters) in the master gyro unit, and the Master Indicator.

In this example, the system is again in an equilibrium position, and the indicator shows that the aeroplane has turned through 90°.

This system also uses the signal from the detector unit to stabilise the DG and keep it aligned with the magnetic field sensed by it. Additionally, since the data selsyn rotor is mounted on the shaft of the Directional Gyro (DG), if the gyro drifts or is misaligned, an error signal is created in the rotor.

GYROSCOPE ELEMENT

In addition to the use of efficient synchro transmitter/receiver systems, it is also essential to employ a gyroscope that maintains its spin axis in a horizontal position at all times. A gyro erection mechanism is therefore essential. This consists of a torque motor, mounted horizontally on top of the outer gimbal, with its stators fixed to the gimbal and its rotor attached to the gyro casing, as shown below.



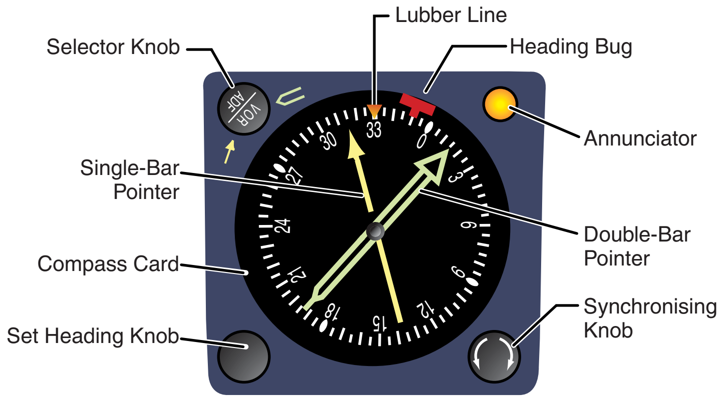
The torque motor switch is normally of the liquid level type, and is mounted on the gyro rotor housing, or inner gimbal, to move with it.

When the gyro axis is horizontal, the liquid switch is open and no current flows to the levelling torque motor. When the axis tilts, the liquid completes the contact between the switch centre electrode and an outer electrode, providing power in one direction or another to the torque motor. The direction of current decides the direction of torque. The direction of the current determines the direction of the torque applied, and precesses the gyro axis back to the horizontal, and then the liquid switch is broken.

Depending on the type of compass system, the directional gyroscope element may be contained in a panel-mounted indicator, or it may be an independent master gyro, located in a remote part of the aeroplane. Systems adopting the master gyro are now the most commonly used, because in serving as a centralised heading source, they also provide for more efficient transmission of the data to flight director and automatic flight control systems, with which they are now closely linked.

HEADING INDICATOR

An example of a basic type of dial indicator, used with modern gyro magnetic compasses, is shown on the next page. In addition to displaying magnetic headings, this indicator is also capable of showing the magnetic bearing to the aeroplane with respect to ground stations on which radio navigation systems are sited; ADF (Automatic Direction Finding) and VOR (very high frequency omni-directional Range). For this reason, the indicator is normally referred to as a **Radio Magnetic Indicator (RMI)**, as shown on the next page.



On this instrument, the flight crew is able to set a desired heading using the ‘set heading’ knob, which is mechanically coupled to a heading bug, so that any rotation of the knob causes the bug to move with respect to the compass card. For turning under automatically controlled flight, rotation of the ‘set heading’ knob also positions the rotor of a data selsyn, which then supplies twin commands to the autopilot system. On modern equipment the main heading reference is on the EFIS and/or HSI.

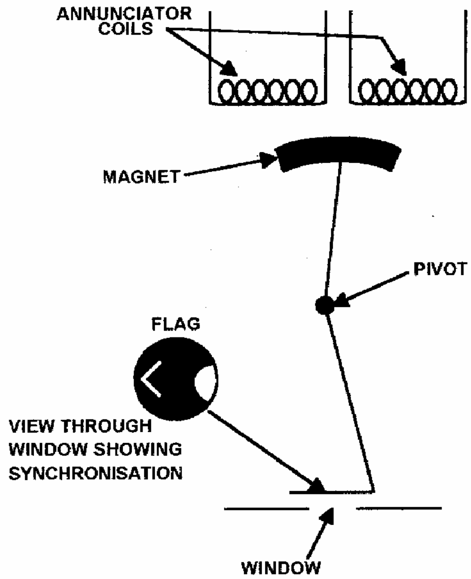
MODES OF OPERATION

All gyro compass systems provide for the selection of two modes of operation: **SLAVED**, in which the gyro is monitored by the detector element, and **FREE GYRO**, in which the gyro is isolated from the detector unit and functions as a straightforward directional gyroscope. The latter operating mode is selected when a malfunction in the monitoring mode occurs or the aeroplane is flying in latitudes where the value of ‘H’ is too small to use as a reliable reference.

SYNCHRONISING INDICATORS

The function of the synchronising indicator (annunciator) is to indicate to the user that the gyro is synchronised with the directional reference sensed by the detector unit. The synchronisation indicator may be integrated with the heading indicator, or may be a separate unit mounted on the aeroplane’s instrument panel. The annunciator is activated by monitoring signals from the detector head to the gyro slaving torque motor, and is therefore connected into the gyro slaving circuit.

A typical annunciator is shown on the next page. It consists of a small flag marked with a dot and a cross, which is visible through a window in one corner of the heading indicator (if so mounted). A small magnet is located at the other end of the shaft, and positioned adjacent to two soft iron cored coils, which are connected in series with the precession circuit. When the gyro is out of synchronisation with the detector head, a current flows through the coils, and attracts the magnet in one direction or the other such, that either a dot or a cross shows in the annunciator window. If the system is correctly synchronised, the annunciator window is clear of an image; but in practice the flag will tend to move slowly from dot to cross, and back again, which indicates that the system is working correctly.



MANUAL SYNCHRONISATION

When initially applying electrical power to a compass system operating in the 'slaved' mode, the gyroscope may be out of alignment from the detector head by a large amount. The system consequently starts to synchronise, but as the rate of precession is normally low (1° to 2° per minute), some time may elapse before achieving synchronisation. In order to speed up this process, a manual synchronisation system is always incorporated.

A manual synchronisation knob, whose face is clearly marked with a dot and a cross, is sited on the heading indicator. This knob is mechanically coupled to the stator of the gyro data selsyn, and when it is pushed in and rotated in the direction indicated by the annunciator, the stator turns, which induces an error voltage into its rotor. This signal then feeds to the Follow-up amplifier and servomotor, which in turn drives the signal selsyn rotor and gyro, via the Precession amplifier and precession torque motor, into synchronisation with the detector head. At the same time, the signal selsyn rotor drives into a null position, and no error signals are present when the system is fully synchronised.

OPERATION OF AN RIC IN A TURN

If an aeroplane enters a turn, the gyroscope maintains its direction with reference to a fixed point (rigidity) and the aeroplane turns around the gyro. The rotor of the data selsyn located in the gyro unit similarly rotates, and error signals generate in the stator, which pass to, and reflected in the stator of the data selsyn located in the Master Indicator. The rotor of the data selsyn now no longer lays in the null of the induced field, and a voltage generates, which passes via the Follow-up amplifier to the servomotor. The servomotor then drives the face of the indicator round, so that the compass card keeps pace with the turn, and at the same time drives the rotors of the data and signal selsyns to keep pace with the turn. During this time the detector unit, which is fixed in azimuth, is turning in the Earth's magnetic field, to induce the flux in each spoke of the detector unit, which also continuously changes. This results in a rotating field produced in the stator of the signal selsyn, which normally results in a change in flux detected by the rotor of the signal selsyn, which in turn passes an error signal to the precession circuit. The rotor of the signal selsyn is however already rotating under the influence of the servomotor. The speed and direction of rotation of the rotor therefore matches that of the stator field, and thus no error signal is present for transmission to the precession circuit, so no gyro precession occurs.

When the aeroplane resumes straight and level flight, rotation of the data selsyn rotor in the gyro unit ceases. The field between the stators remains constant, and no current flows in the servo-loop. The heading indicator display also stops rotating and the system returns to its previous synchronised condition.

In a steep and prolonged turn, a slight de-synchronisation may occur due to the introduction of a small component of 'Z', while the detector head is out of the horizontal for a protracted period. However, on coming out of the turn, the compass card rapidly resumes the correct heading through the normal precession process. Apart from this small error, the system is virtually clear of turning and acceleration errors.

ADVANTAGES OF A REMOTE INDICATING GYRO MAGNETIC COMPASS

The advantages of a gyro magnetic compass over a DI or direct reading instrument are:

- The DI suffers from real and apparent drift and needs resetting in flight. In addition, when resetting to the magnetic compass, the aeroplane must be flown straight and level, whereas the detector unit constantly monitors the gyro magnetic compass.
- The detector unit can be installed in a remote part of the aeroplane, well away from electrical circuits and other influences due to airframe magnetism.
- The flux valve technique used in the detector unit senses the Earth's magnetic meridian rather than seeking it, which makes the system more sensitive to small components of 'H'.
- The unit provides a heading reference to higher magnetic latitudes than the direct reading magnetic compass.
- Turning and acceleration errors are minimised because:

    1. The detector unit is fixed in azimuth.
    2. The precession signals aligning the compass are kept to a low value (typically 5°/minute)
    3. There is a roll cut out switch that isolates the precession system during turns at bank angles of greater than 10°.

- The compass may be detached from the detector unit by a simple switch selection to work as a DI, so a normal DI is not required.
- The system can be readily used to monitor other equipment (e.g. autopilot, Doppler, RMI, etc).
- Repeaters can also be made available to as many crew stations or systems as desired.

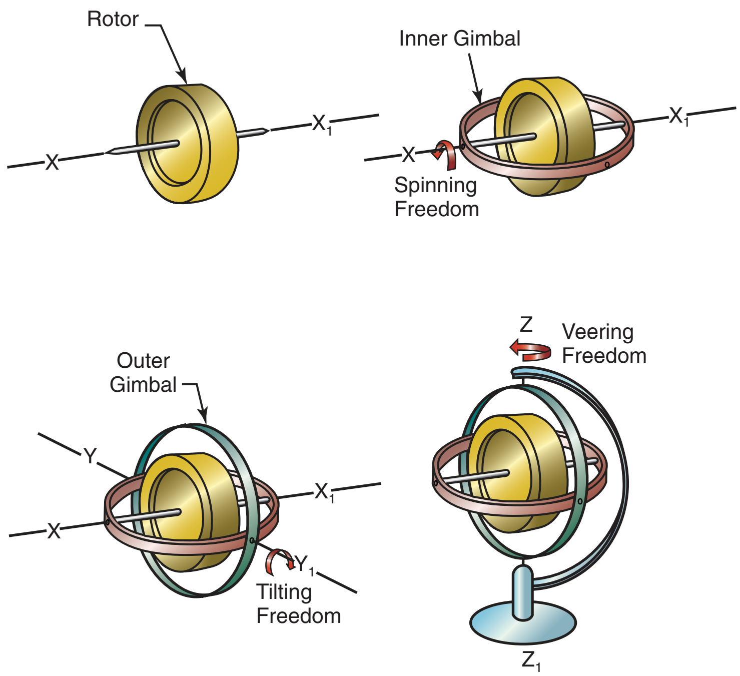
DISADVANTAGES OF THE REMOTE INDICATING GYRO MAGNETIC COMPASS

The disadvantages of a gyro magnetic compass over a DI or direct reading instrument are that it is:

- much heavier than a direct reading compass.
- much more expensive.
- electrical in operation and therefore susceptible to electrical failure.
- much more complicated than a DI or a direct reading compass.

INTRODUCTION

A gyroscopic system is one where a universally mounted heavy metal wheel or rotor has three planes of freedom, as shown in the diagram below.



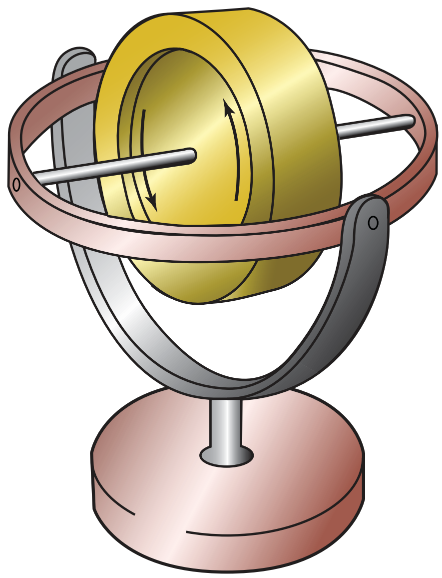
The aligned axes provide:

- **Spinning Freedom** Rotation about the spin axis (XX_1)
- **Tilting Freedom** Rotation of the spin axis in the vertical plane (YY_1)
- **Veering Freedom** Rotation of the spin axis in the horizontal plane (ZZ_1)

Freedom of movement within the three planes is obtained by mounting the rotor in two concentrically pivoted rings, called **inner** and **outer gimbal rings**. The gimbal system mounts so that with the gyro in its normal operating position, all of its axes run mutually at right angles to each other, and intersect at the centre of the rotor. A gyro with its axis of rotation in the horizontal plane is a **horizontal gyro**, and a gyro with its axis of rotation in the vertical plane is a **vertical gyro**. The gyro's plane of rotation contains the sensitive axes, rotation about which causes the gyro to precess (described later).

PRINCIPLE OF CONSTRUCTION

A gyroscope consists of a weighted wheel or rotor, which spins at high speed (8000-24 000 rpm) and is mounted in a series of hinged mounting rings, called **gimbals**, as shown on the next page. A gyro has 3 axes of freedom, one of which is its spin axis, and is able to move relative to the mounting base around one or both of the remaining axes. Ignoring the spin axis, one degree of freedom exists when the gyro can rotate around only one axis, and two degrees of freedom exist if it is free to move around both axes. A gyro with two degrees of freedom is known as a **Free** or **Space** Gyro.



GYROSCOPIC PROPERTIES

When the rotor spins at high speed, the device becomes a true gyroscope, and possesses the following fundamental properties:

RIGIDITY IN SPACE

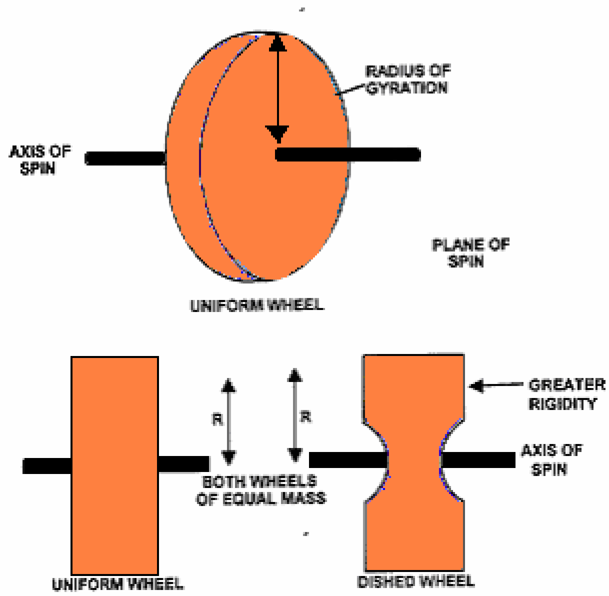
The gyro tries to remain pointing in the same direction or position in space, even when its mounting base is tilted or rotated (i.e. the gyro has a tendency to maintain its plane of spin, dependent on its speed, mass, and radius about which the mass is displaced [Newton's 1st Law]). The greater the rigidity the more difficult it is to move the rotor away from its plane of spin, unless an external force acts on it. For example, if a spinning bicycle wheel was to fall over its spin axis, it must be rotated through 90°, and unless an external force is applied, it does not do so as long as it maintains a reasonable speed.

$$\text{Rigidity (R)} = \frac{S \times I}{F}$$

where: I = Moment of Inertia

F = External Force

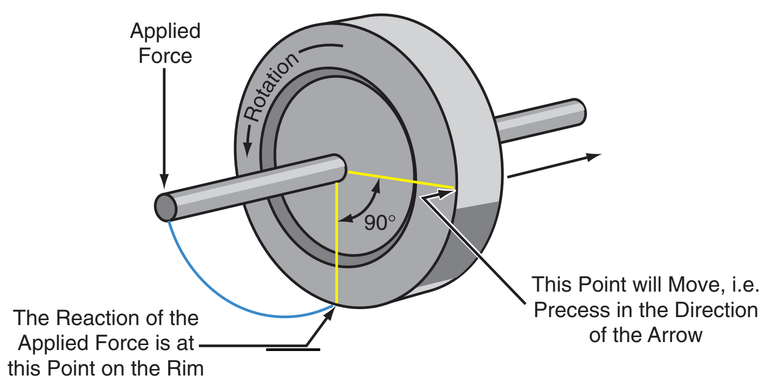
S = Speed of Wheel (rpm)



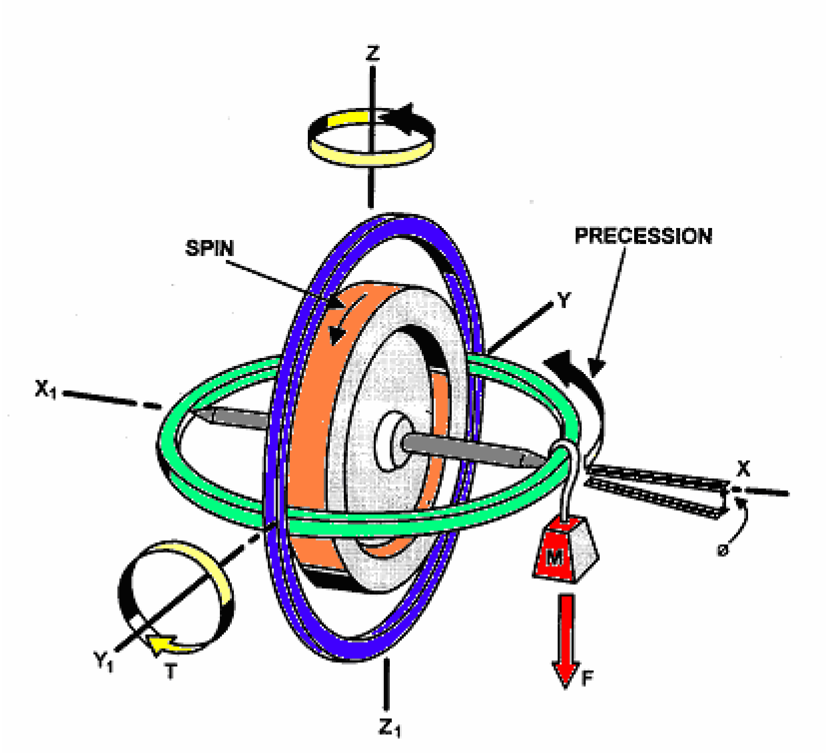
Rigidity is proportional to the rotor's rpm, and its moment of inertia, which increases if the rotor has a large radius, with the bulk of its mass distributed around its rim. This is opposite to that in the DRC where the mass is placed as close to the pivot point axis as possible to prevent aperiodicity.

PRECESSION

When an external force is applied to the spinning rotor via the gimbal assembly, the gyro does not move in the direction of the force, but in a direction perpendicular to that of the applied force. For example, if a force is applied to the spin axis, the gyro does not move in the direction of the applied force, but instead rotates due to a force applied 90° later in the plane of rotation, as shown below.



Similarly, if a downward push force is applied to the inner gimbal of a gyro system, the gyro precesses about its outer gimbal pivot point.



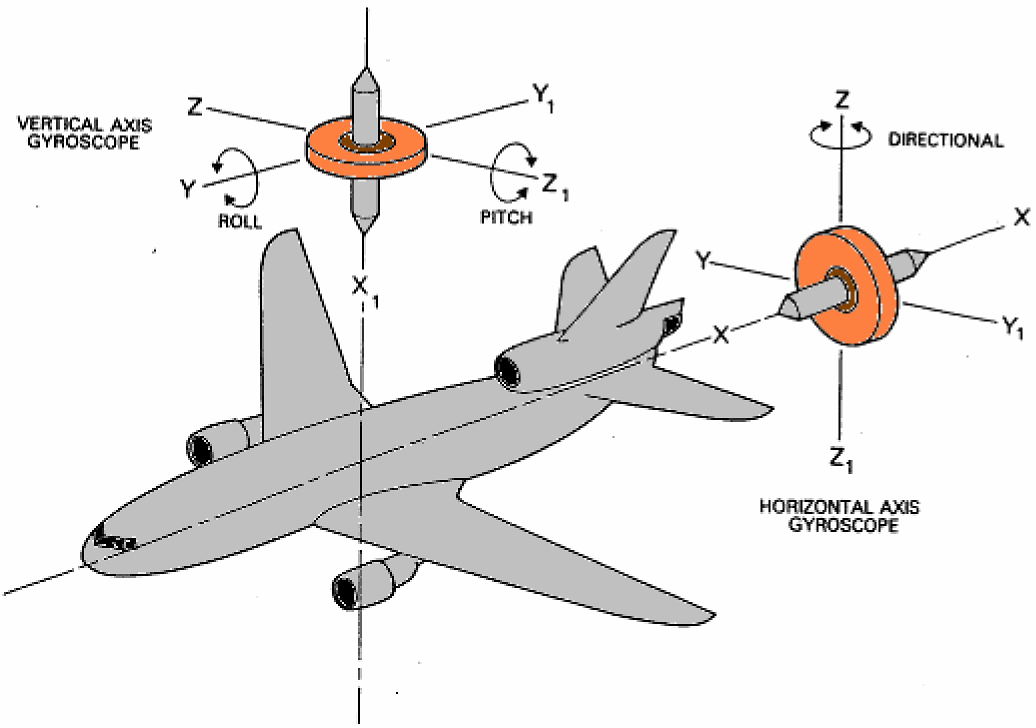
The strength and direction of the applied force, the moment of inertia of the rotor, and the angular velocity of the rotor, all affect the amount of precession. It follows that, the larger the force the greater the rate of precession, and the higher the rigidity, the lower the rate of precession. Thus in order to precess a gyro its rigidity must first be overcome.

$$\text{Precession} = \frac{\text{Applied Force}}{\text{Rigidity}}$$

Precession continues while applying the force, until the plane of rotation is in line with the direction of the applied force. At that point, the force no longer disturbs the plane of rotation, so that there is no further resistance to the force, and thus no further precession. The axis about which a force or torque is applied to a gyro is termed the **input axis**, and the one about which precession takes place is termed the **output axis**. Most gyros rotate at a constant rpm, particularly where precession plays a part in the operation of a system or indicator. If the direction of the rotor or applied force reverses, then the direction in which precession occurs also reverses.

TYPES OF GYROSCOPES

The number of degrees of freedom permitted by each type of gyroscope determines its usage, but for use in aeroplanes, they must exhibit two essential reference datums. The first is a reference against which pitch and roll attitude changes are detected, and the second is a directional reference against which change about the vertical axis is detected.



The following types of gyroscopes exist:

SPACE (OR FREE) GYRO

This is a gyro having freedom to move in all three planes. It consists of two concentrically pivoted rings, called **inner** and **outer gimbal rings**. The three planes relate to the three axes of the aeroplane (i.e. fore and aft or roll axis, lateral or pitch axis, and the normal or yaw axis). Furthermore, there is no means of external control over this type of gyro, a feature that distinguishes it from a tied or Earth gyro. This type of gyro would have no practical use in an aeroplane instrument where the gyro is required to be set to, and maintain a given direction.

TIED (OR DISPLACEMENT) GYRO

This type of gyro is a **space gyro**, which has a means of external control, and has freedom of movement about all three planes. This type is for use as a directional gyro (e.g. in the Direction Indicator [DI]).

EARTH GYRO

This type of gyro is a **tied gyro**, where the controlling force is the gravity of the Earth. This type is for use in gyro horizon, or artificial horizon instruments.

RATE GYRO

This is a gyro having one plane of freedom only, its plane of rotation being 90° removed from its plane of freedom. This type of gyro is used in measuring the rate of turn, and employs restraining springs (e.g. in the turn and balance indicator or turn co-ordinator).

RATE INTEGRATING GYRO

This type of gyro is similar to the rate gyro, having a single degree of freedom, except it uses the viscosity of a fluid (viscous restraint) to damp the precessional rotation about its output axis, instead of restraining springs. The main function of this type of gyro is to detect turning about its input axis by precessing about its output axis. This type is used on inertial navigation stabilised platforms.

SOLID STATE (RING LASER) GYRO

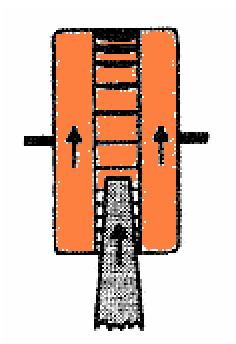
These are not gyros in the true sense, but they behave like gyros, and sense the angular rate of motion about a single axis. They consist of a solid block of temperature stable glass within which there is a cavity or laser path, filled with a lasing medium such as helium-neon. Some are triangular in shape (Honeywell), whilst others have four sides (Litton). They both have small tunnels drilled in them, with reflecting mirrors sited at each corner. Two beams of high-energy laser light pass in opposite directions around the sealed cavity and initially travel at the same speed. Any rotation of the gyro in the laser plane results in a change in the path lengths of each beam, and a control element measures the resultant frequency shift of the beams. The frequency differential is directly proportional to the angular turning rate.

POWER SOURCES FOR GYROSCOPES

Conventional gyroscopes in aeroplanes are either air (vacuum) driven, or electrically driven. In some aeroplanes, all of the gyros are either vacuum or electrically driven, whilst in others the vacuum (suction) system provides the power necessary to run the attitude and heading indicators, whilst an electrical system runs the slip and turn indicator. The electrically driven gyroscopic instruments use alternating or direct current for power.

AIR DRIVEN GYROSCOPES

This type is widely used on small aeroplanes, and is found on some large aeroplanes in order to power stand-by, or emergency instruments. In the air driven gyro, the gyro system is contained in an airtight case, the air having been removed from the casing via a suction pump or venturi, thus creating a partial vacuum inside. The vacuum source is normally an engine driven vacuum pump, which is controlled to a value between 4.5-5.0 inches of Hg, but in very simple aeroplanes, this may be achieved using a venturi tube attached to the outside of the fuselage. Air under atmospheric pressure enters the instrument casing via a filter, and flows through a shroud, which encases the rotor.



The shroud acts as the inner gimbal and has a small opening in it that directs the airflow fed to it via an inlet port onto buckets cut in the rotor periphery. The shroud also has an exhaust port to allow air to escape to the outer case. The pipeline from the inner gimbal is fed through the inner gimbal axis along the outer gimbal and through its axis to a filter system, which covers a hole in the outer case.

When the vacuum is applied to the system, the pressure in the outer case drops. Replacement air enters the system from outside the case and passes through the filter before being directed onto the rotor buckets. The rotor rotates at approximately 10 000 rpm under normal operation. The air escapes from the rotor shroud, although it can be used to provide a controlling force (tie) to the gyro.

ELECTRICAL GYROS

The majority of gyroscopes used in aeroplanes today are electrically driven, and normally use AC current, although a 24-volt DC supply feeds some. The AC powered gyro is preferable, since it avoids the use of commutators and brush gear, which require frequent servicing. The gyroscope itself is a 3-phase squirrel induction motor, constructed to obtain the maximum gyroscopic effect. To achieve this, the rotor does not rotate inside the field coils, as it does in conventional motors, but instead is positioned around the outside of the field coils. This method of construction ensures that the rotor mass is concentrated as near its periphery as possible, thus increasing gyro rigidity. A 115 volt, 3-phase, 400 hertz AC system feeds the gyro stator, and the rotor rotates at approximately 24 000 rpm. AC gyros are capable of higher rotational speeds than the DC ones, and are therefore the favoured option in instruments where high rigidity is required. DC powered gyros can also be powered from the aeroplane emergency electrical supply.

THE DISADVANTAGES AND ADVANTAGES OF AIR DRIVEN GYROS

The disadvantages of air driven gyros are that:

- Full rigidity is not reached until the rotor speed has built up. Suction driven gyroscopes normally take 4-5 minutes after starting the aeroplane engine to attain the correct operating speed. The indications provided by the instrument are however usable after 2 minutes. Conversely, venturi driven systems do not start to spin up until the aeroplane begins its take-off run, and cloud penetration must be delayed for a few minutes to reach the operating rpm.
- The speed of the rotor depends on the mass of air flowing through the system (mass flow). As the aeroplane climbs, the air density falls and the mass flow reduces. The rotor speed thus reduces and gyro rigidity deteriorates. The other thing about mass flow is that it requires a clear unimpeded flow of air. If the filters on the inlet line are blocked or partially blocked, this also affects the gyro rigidity.
- A major drawback of the air driven gyro is the need to provide airtight joints where the inlet pipes pass through the inner and outer gimbal axes, which severely limits the degree of freedom around these axes.
- Ingested dust or moisture causes corrosion and bearing wear.

The advantages of air driven gyros are that:

- They are cheap.
- They are easy to maintain.
- In the event of an emergency, they can operate without electrical power.

THE DISADVANTAGES AND ADVANTAGES OF ELECTRICALLY DRIVEN GYROS

The disadvantages of electrically driven gyros are that they:

- Depend on their power supply, so standby air driven alternatives are normally fitted.
- Tend to be more expensive than air driven gyros.

The advantages of electrically driven gyros are that:

- Higher rigidity is possible.
- Their operating rpm is more consistent.
- Their performance is not affected by altitude.
- Information can be transmitted easily to other systems.
- They have more freedom to rotate around their axes.
- The instrument case is completely sealed, which excludes dirt, and also prevents heating/cooling effects by allowing the components to be maintained at a constant temperature, if required.

GYRO WANDER

Any deviation of the gyro spin axis from its set direction is known as **gyro wander**, and is classified as follows:

REAL WANDER

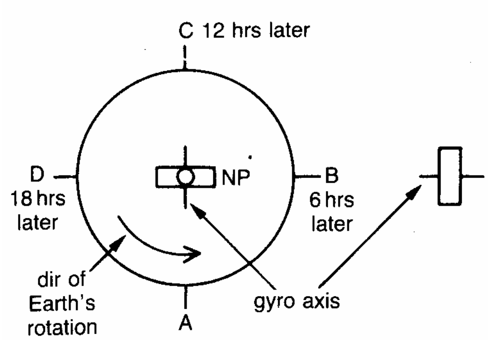
Any physical deviation of the gyro spin axis is called **real wander**. A gyro should not wander away from its preset direction, but various forces act on the rotating mass of a gyro and cause it to precess. For example bearing friction, which is always present at the spin axis. If this friction is symmetrical, it merely slows down the rotor, but if it is asymmetrical, it causes the gyro to precess. Similarly, any friction in the gimbal bearings causes the gyro to precess. Wear on the gyro may result in movement of the C of G, which may also result in a precessing force. Such errors are not constant or predictable, and cannot be calibrated, nor can corrections be applied to nullify this error.

APPARENT WANDER

In this case, the gyro spin axis does not physically wander away from its pre-set direction, but to an observer it appears to have changed its direction. This is because the gyro maintains its direction with respect to a fixed point in space, whereas the observer rotates with the Earth, thus with the passage of time the gyro appears to have changed direction, with reference to an Earth datum. Apparent wander, is also made up of horizontal components called **drift**, and vertical components called **topple**. The rate of drift and topple depends upon the latitude and can vary from zero to a maximum of 15.04° per hour (the rate at which the Earth rotates). Depending on whether a gyro has a vertical or horizontal spin axis, the rotation of the Earth also has a different effect.

HORIZONTAL AXIS GYRO

The diagram below shows a horizontal spin axis gyro positioned at the North Pole.



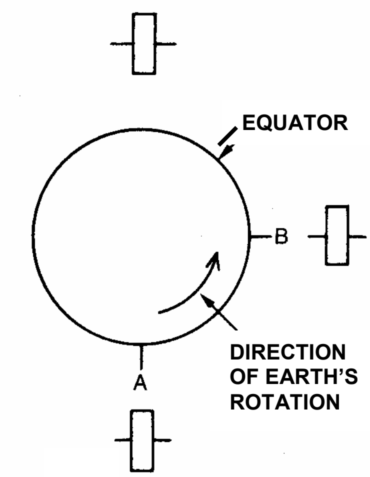
It shows an observer initially at position A, where the gyro is set so that its spin axis is directly in line with him. Six hours later, the Earth having rotated through 90°, the observer now views the gyro from position B. The observer does not however appreciate his own motion, and the gyro spin axis appears to have moved clockwise in the horizontal plane through 90°. Twelve hours later the gyro spin axis appears to have moved through 180°, and finally after twenty-four hours, with the observer back in the original position, the gyro spin axis again appears as it was when first aligned. The apparent motion in the horizontal plane is known as **gyro drift**.

If a horizontal spin axis gyro has its axis aligned in a north/south direction along the equator, during the Earth rotation, the gyro spin axis continues to remain aligned with the local meridian. This occurs because at the equator, all of the meridians are parallel to one another, and a gyro aligned with a meridian thus remains with that meridian over a 24-hour period. This means that the gyros neither drift nor topple when aligned in this manner.

If the horizontal spin axis gyro is positioned at the poles, it drifts through 360° in 24 hours (maximum drift). For example, the rate of drift at the poles is the same as the angular velocity of the Earth, at 15.04° per hour, whilst at the equator the same gyro with its spin axis aligned with the local meridian has zero drift due to Earth rotation.

Drift at intermediate latitudes = 15.04° x sin Latitude° per hour.

The diagram below shows a horizontal spin axis gyro with its spin axis aligned in an east/west direction along the equator, when observed at point A.



In this case, after six hours duration, and the Earth having rotated through 90°, the observer again views the gyro from position B, where it appears to have turned into a vertical axis gyro. This apparent change in its vertical plane is **topple**, and is the maximum value at the equator, but zero at the poles, due to the Earth's rotation.

Topple at intermediate latitudes = 15.04° x cos Latitude° per hour.

VERTICAL AXIS GYRO

Diagram (a) below shows a vertical spin axis gyro, which is positioned at the North Pole, where the rotation of the Earth beneath the gyro spin axis has no effect on the gyro (i.e. the gyro axis appears to neither drift nor topple).

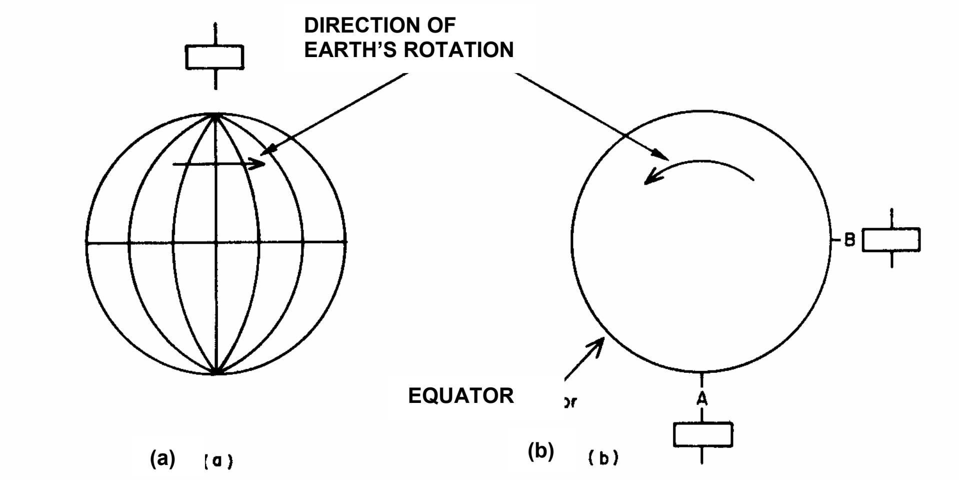


Diagram (b) above shows a gyro at the equator, with its spin axis vertical to the observer, when viewed at position A. After six hours, and the Earth having rotated through 90°, the observer at point B will view the gyro as a horizontal axis gyro. After another six hours, the spin axis again appears vertical. This apparent change in the gyros vertical axis is known as **gyro topple**. At the equator gyro topple is 360° in 24 hours, whilst at the poles it is zero.

From the above, notice that the vertical gyro suffers from topple but does not suffer from drift:

Drift = Vertical gyro unaffected

Topple = 15.04° cos latitude° per hour

Gyro drift and topple may be summarised as:

Horizontal axis gyro:	at the poles:	maximum drift and no topple
	at the equator:	no drift and maximum topple
Vertical axis gyro:	at the poles:	no drift and no topple
	at the equator:	no drift and maximum topple

TRANSPORT WANDER

This is an additional form of apparent topple/drift, which principally occurs when the gyro is placed on a platform, such as an aeroplane, that is flying in an east or west direction. The gyro carries in space in the same way as the Earth does, and this results in **transport wander**.

Transport Drift = Rate of change of longitude° per hour x sin latitude° per hour
Transport Topple = Rate of change of longitude° per hour x cos latitude° per hour

EXAMPLES OF GYRO WANDER

EG 1: If a horizontal spin axis gyro is set with its axis aligned in an east/west direction at latitude 45°N, the attitude of its spin axis after 3 hours is:

Since the gyro axis is aligned in an east/west direction at an intermediate latitude, the gyro will both drift and topple.

Drift = 15.04° sin latitude° per hour = 3 x 15.04° x sin 45° = <u>31.9°</u>

Topple = 15.04° cos latitude° per hour = 3 x 15.04° x cos 45° = <u>31.9°</u>

Note: In the Northern Hemisphere the gyro axis drifts clockwise, and anti-clockwise in the Southern Hemisphere

The spin axis will thus be aligned at 090° + 31.9° = <u>121.9° / 301.9°</u>

The eastern end of the spin axis appears to rise by 31.9° from the horizontal, and the western end is similarly depressed.

EG. 2: If the rate of change of longitude during a flight is 25° in one hour, at latitude 50°N, the amount of transport drift present is:

Transport Drift = Rate of change of longitude° per hour x sin latitude° per hour

Transport drift = 25° x sin 50° = <u>19.15°</u>

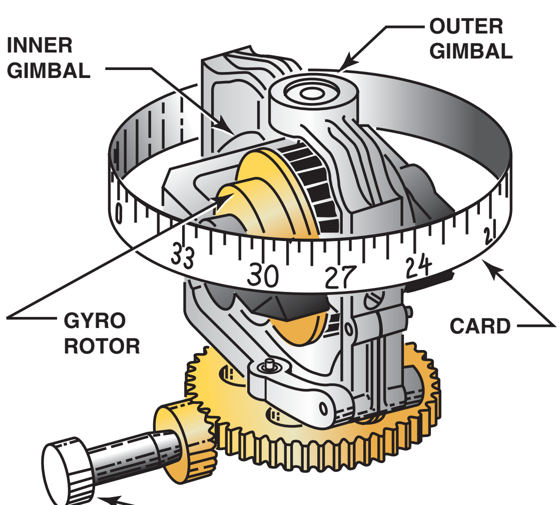
Chapter 15 Direction Indicator

INTRODUCTION

The **Direction Gyro Indicator (DI)** uses a horizontal axis tied gyro, which possesses freedom in three planes, and uses the gyroscope's property of rigidity, to stabilise an azimuth scale. It is manually aligned with the direct reading magnetic compass and, in light aeroplane, provides a stabilised directional reference for maintaining and turning accurately on to a heading. The DI is non-magnetic and is not subject to turning and acceleration errors, dip, or magnetic disturbances. The DI provides an accurate dead-beat indication of heading, and shows any deviation from the set heading instantaneously. The DI is also not north seeking, so it must be provided with a directional datum from an outside source, normally taken from the direct reading magnetic compass. It is essential to check the DI indications at regular intervals, against the direct reading magnetic compass, because after the initial synchronisation the gyro may wander, particularly after aerobatics. The DI complements but does not replace the DRC.

BASIC DESCRIPTION OF THE DIRECTION INDICATOR

The instrument is either air driven or electrically driven. In the air driven version, the instrument consists of an air-driven horizontal axis gyro, rotating at approximately 10 000 rpm.

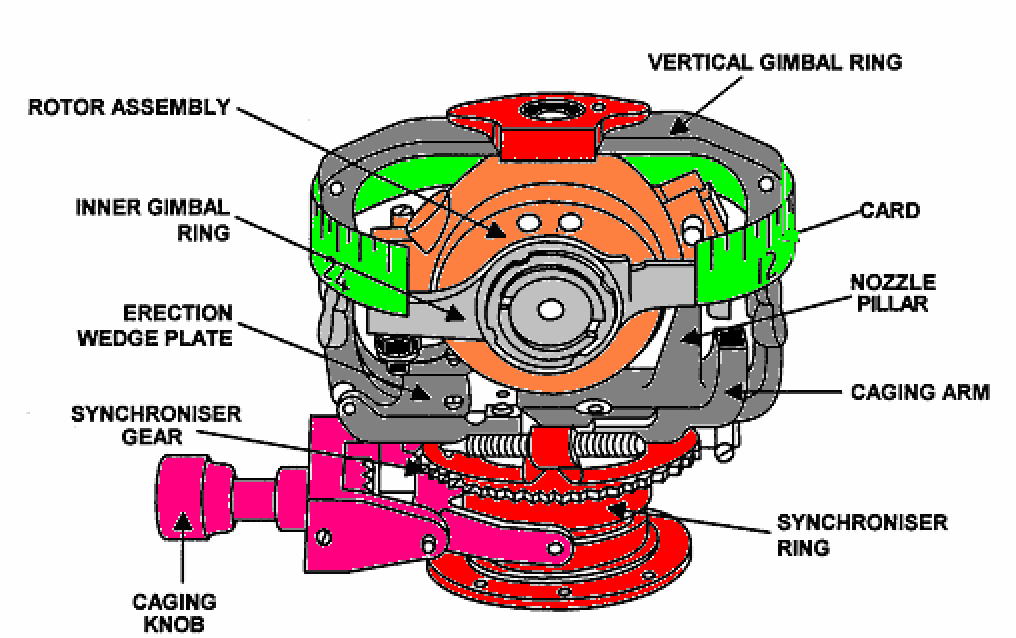


The azimuth scale is graduated from 0° to 360° in 5° divisions, with main graduations every 10°, and figures every 30°. The scale is read against a vertical lubber line.

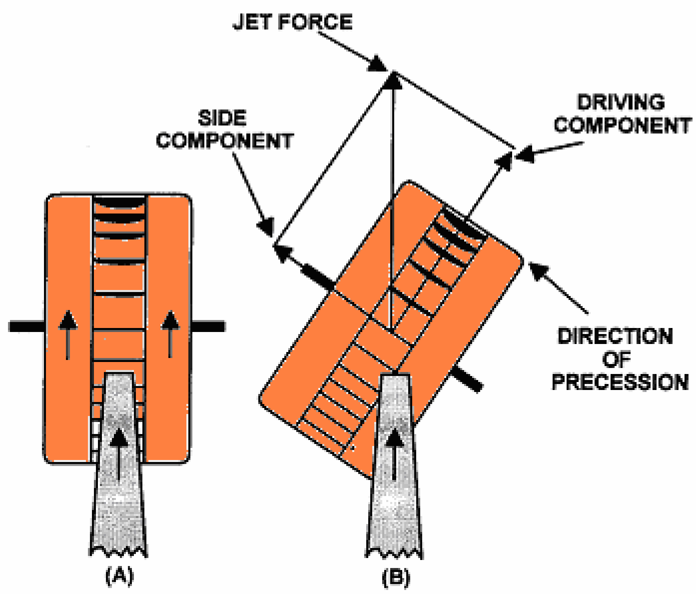
Below the window is sited a knob, which is used to cage the gyro, and to rotate the gyro assembly when setting a given heading.

OPERATION OF THE DIRECTION INDICATOR

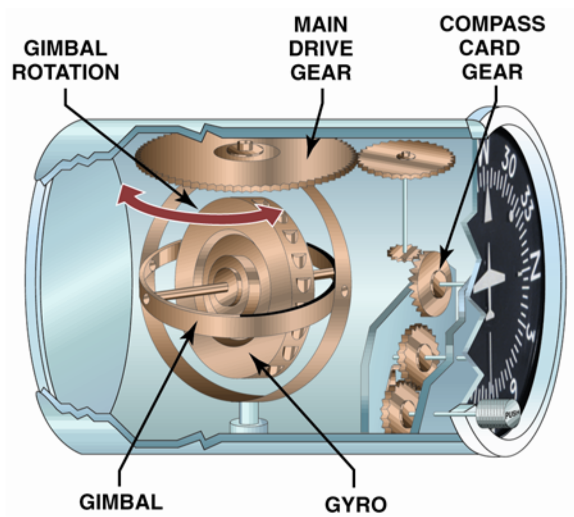
The diagram below shows the internal mechanism of a typical air driven version of the instrument.



The rotor spins about its horizontal axis, and is supported in bearings in the inner gimbal ring, which is free to rotate about the horizontal axis through 110° (i.e. 55° either side of its central position, when it is uncaged). The inner gimbal ring, supported in bearings in the vertical outer gimbal ring, is free to rotate in azimuth through 360° about the vertical axis. A nozzle, sited in the outer gimbal, directs a jet of air onto buckets cut in the rotor periphery. The action of the air ensures that the rotor reaches its operating rpm after about five minutes, when the vacuum pump develops full suction. The air jet also maintains the rotor spinning in the aircraft's vertical plane, as shown below. If the gyro topples, a component of the jet force acts at right angles to the rotor, and produces a precessing force, which erects the gyro.



In this version the indicator scale is attached to the outer gimbal, but on newer models, the synchronising gear ring also drives a sequence of gears, which connect the movements of the gyro around its vertical axis, onto a vertical scale, as shown below.



The gyro is initially erected using a caging mechanism, which manually places the gyro spin axis in its horizontal plane. The mechanism consists of a bevel pinion and a caging arm, which are both directly controlled by a caging or setting knob sited on the front of the instrument. By pushing the knob in, the bevel pinion engages with the synchroniser gear ring, and it allows the scale to adjust in azimuth by rotating the caging knob. At the same time the caging arm is raised, which locks the inner gimbal ring in its horizontal plane, and prevents the gimbal ring and rotor from toppling during resetting. The caging knob is pulled out to uncage the gyroscope, by disengaging the gears, and allowing the caging arm to drop, thus releasing the inner gimbal ring. The inner gimbal should also be locked in its horizontal position during aerobatics, to prevent severe loads being transmitted to the rotor bearings.

In the electrical version, the rotor is part of an AC 'squirrel cage' induction motor, and rotates at approximately 24 000 rpm. Initial erection is again by use of the caging device, but thereafter the gyro is tied so that it maintains its spin axis horizontal to the Earth's surface.

ERRORS ASSOCIATED WITH THE AIR DRIVEN DIRECTION INDICATOR

The air driven Direction Indicator is subject to the following errors:

REAL DRIFT AND TOPPLE

The gyro is subject to both of these errors, caused by mechanical imperfections such as a slight imbalance of the rotor/gimbal system, bearing friction and mechanical or electrical latitude correction.

APPARENT DRIFT

Earth rate (due to Earth rotation) and transport wander (due to gyro transportation in an east/west direction) cause this error. A fixed rate of compensation is calculated for the latitude of operation of the aeroplane for apparent wander, and may be applied by means of balancing nuts, attached to the gimbals. The rate of compensation, but no compensation for transport wander, is made apart from periodic resetting of the instrument to the magnetic compass heading. The drift rate can be as much as 15.04°/hour; and for this reason, the instrument requires realignment with the magnetic heading shown on the direct reading compass approximately every 15 minutes, whilst the aeroplane is in level and unaccelerated flight. Any topple experienced by the gyro is corrected for by the air jet erection system.

GIMBALLING ERROR

Any misalignment between the aeroplane axes and the navigation system axes causes this error. This error only exists during manoeuvres (i.e. during climbing, descending, or banking) but once level flight resumes, disappears.

USE OF THE DIRECTION INDICATOR (DI)

On small basic aeroplanes and some older aeroplanes, the direction indicator is the primary heading reference used to maintain the required heading, although great care must be exercised when using this instrument, due to the various drift errors that exist.

Prior to departure the DI requires alignment with the magnetic compass using the heading set knob. Also during the pre-flight process, and during taxi for take-off, the two readings require periodic comparison to ensure that the gyro is not showing large drift rates. Once in flight the DI should be again checked and reset against the magnetic compass approximately every 15 or 20 minutes, with the aeroplane in a steady level flight condition.

ADVANCED USE OF THE DIRECTION INDICATOR

In areas where the Earth's magnetic field does not provide a stable heading reference, it is necessary to base any headings on an unmonitored DI. This is a highly sophisticated instrument, where any friction effects are reduced to a minimum, although the instrument still suffers from drift errors. These errors therefore need to be calculated before they can be compensated for.

SAMPLE CALCULATION

A DI has its latitude compensatory device set to correct for operations at latitude 45°N, and is fitted to an aeroplane flying eastward along the parallel 60°N, at a groundspeed of 240 kt. The expected drift rate (°/hour) if the gyro is free from random drift errors is:

- Real drift (gyro drift) from the fixed latitude correction = 15.04° x sin latitude°/hour.
- Apparent drift (Earth's rotation) at the latitude of operation = 15.04° x sin latitude°/hour (- for a decrease in the Northern Hemisphere, or + for an increase in the Southern Hemisphere).
- Transport drift in an Easterly direction = departure x tan mid latitude°/hour.

$$= \frac{\text{Groundspeed}}{60} \text{ x tan mid latitude°/hour}$$

Note: (- if easterly tracking, or + if westerly tracking in the Northern Hemisphere)

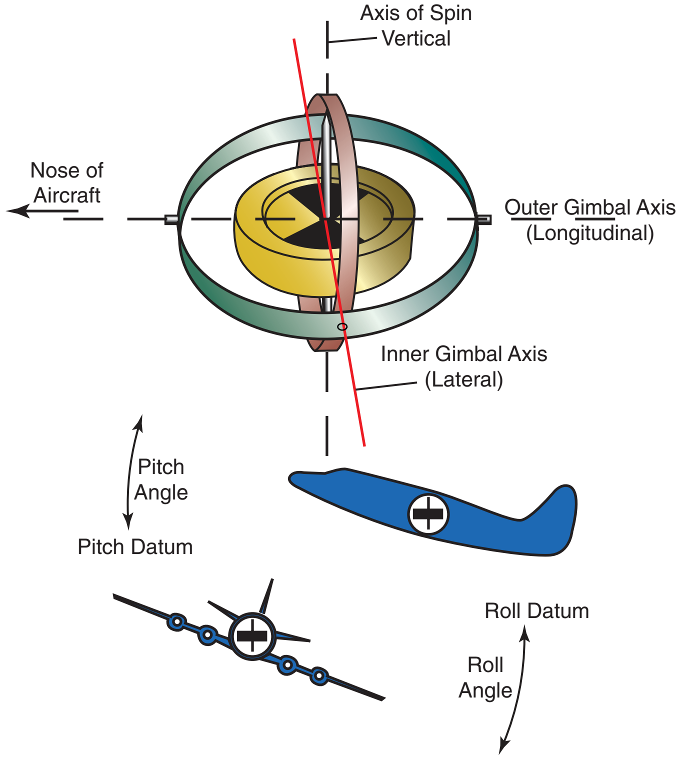
Solution:

a) Apparent Drift	= 15.04° x sin 60°	= -13.03°/hour (decrease)
b) Transport Drift (convergence)	$= \frac{240}{60}$ x tan 60°	= -6.93°/hour (east)
Total Drift = Gyro Drift + Transport Drift	= -13.03° + -6.93°	= -19.96°/hour (decrease)
Latitude correction (real drift)	= 15.04° x sin 45°	= +10.63°/hour (increase)
The Total Anticipated Drift		**= -9.33°/hour (decrease)**

By comparing the actual DI heading with the expected one (derived from these calculations), the pilot can then use this information to establish the real (random) drift error that exists.

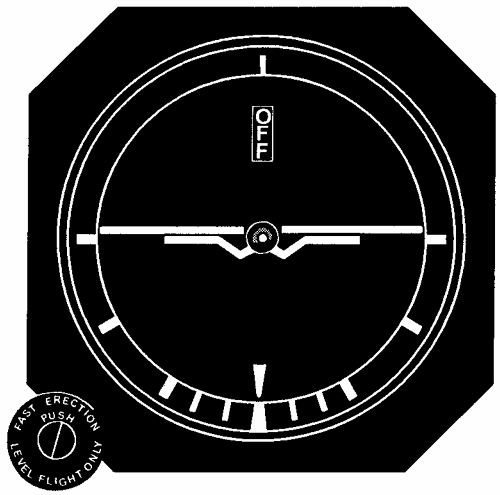
INTRODUCTION

The artificial horizon (gyro horizon) comprises of a vertical spin axis Earth gyro having freedom of movement in all three planes, and indicates the aeroplane attitude relative to its pitch (lateral) and roll (longitudinal) axes, which is essential when a natural horizon is unavailable (e.g. when flying in cloud).

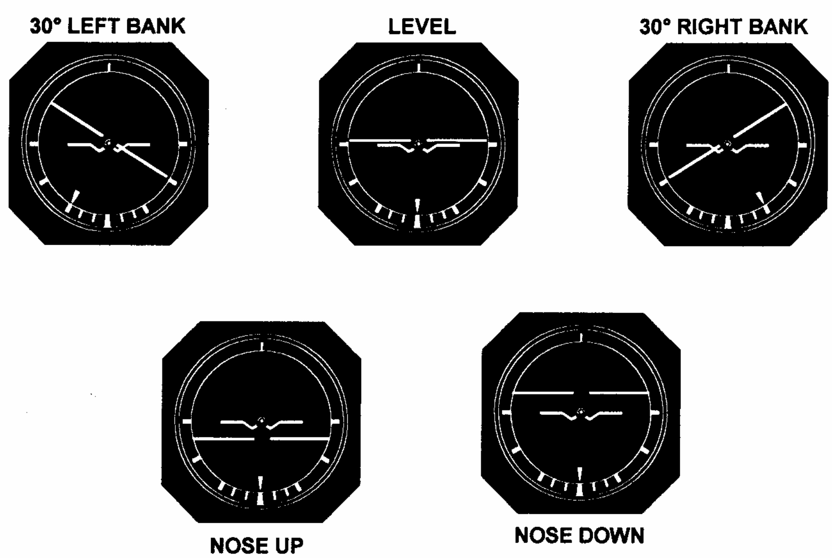


The instrument is either air or electrically driven, although the principal of operation is identical. The gyro spin axis is maintained vertical with reference to the centre of the Earth, and a bar positioned at 90° to the spin axis represents the local horizon. A symbol representing a miniature model aeroplane is fixed to the instrument case, and represents the rear view of the true aeroplane, which on some instruments is adjustable to suit the pilot's own eye level, and the particular aeroplane pitch trim setting. Shown on the next page is a typical artificial horizon display.

In flight, the aeroplane's movement about its pitch or roll axis is indicated instantaneously by movement of the case relative to a horizon (natural horizon) bar, which is held in the local horizontal by gyro rigidity. The position of the model aeroplane relative to the bar represents the attitude of the aeroplane to the natural horizon, whilst the position of a pointer relative to a fixed scale represents the aeroplane's angle of bank.



Dive and climb are indicated by the model aeroplane moving up and down with respect to the horizon bar, whilst the angle of bank is indicated by the model aeroplane appearing to bank in relation to the horizon bar. The indications expected during various flight attitudes are shown below.



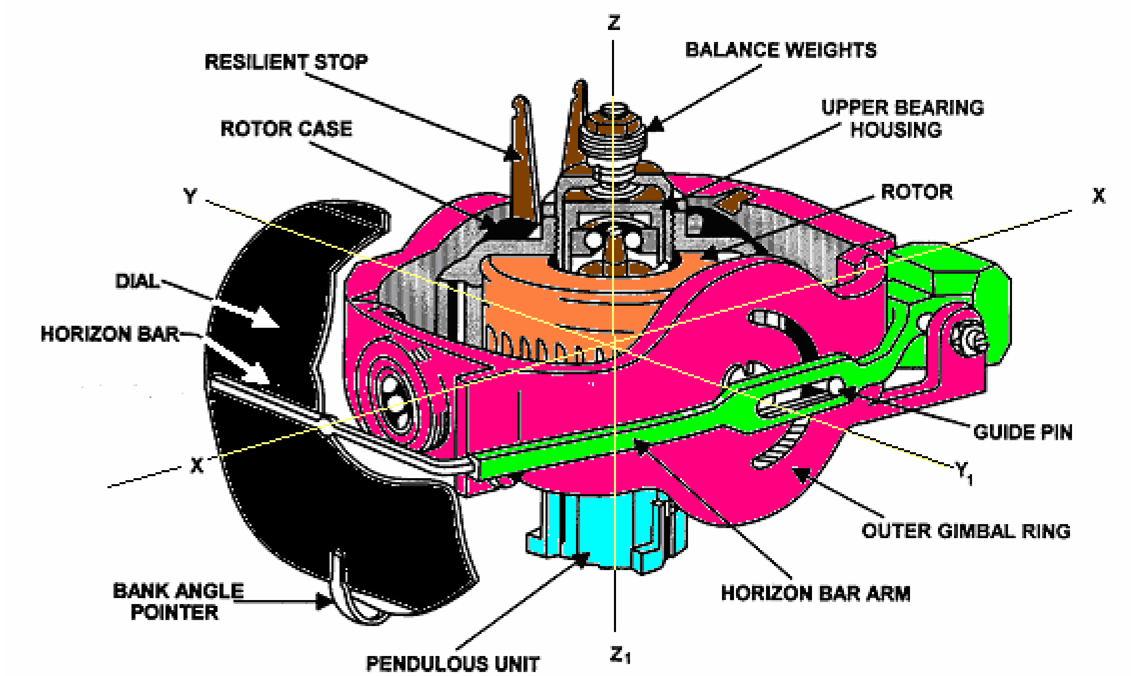
The exact angle of bank is indicated by a pointer at the bottom of the instrument, and provides a direct indication of any change of attitude, without any involvement of lag.

AIR DRIVEN (CLASSIC) ARTIFICIAL HORIZON

CONSTRUCTION

Shown below is a schematic view of an air driven artificial horizon. This type is commonly used in light aeroplanes, and as a standby instrument in commercial aeroplanes. Operated by a vacuum pump, it evacuates the air from the instrument case and gyro housing (inner gimbal). This creates a depression within the instrument, and the surrounding atmosphere enters the instrument through a filtered inlet. The air then passes through channels to jets mounted within the inner gimbal, which direct air onto buckets cut into the periphery of the rotor, and cause the rotor to rotate at approximately 13 000 rpm, in an anti-clockwise direction when viewed from above. The air then evacuates through a pendulous unit, mounted below the rotor casing, via four ports, and is controlled by two pairs of linked pendulous vanes, providing a mechanism for maintaining the gyro spin axis in its vertical plane.

The rotor spins about a vertical axis (Z-Z_1) and mounts in bearings within a sealed case, which forms the inner gimbal.



The inner gimbal, mounted in bearings within a rectangular shaped outer gimbal, is free to rotate 55° either side of its horizontal position, about the lateral axis (Y-Y_1). This enables the determination of the aeroplane's pitch attitude, which directly indicates by movement of the horizon bar. The horizon bar arm is actuated by a guide pin, which protrudes from the gyro stabilised rotor housing (inner gimbal), and moves in a curved guide slot in the outer gimbal.

The outer gimbal is mounted in an air tight instrument case, with its pivots along the fore and aft axis (X-X_1), and is free to rotate through 110° either side of its central position, in order to determine the roll attitude of the aeroplane. A background plate representing the sky, fixed to the front end of the outer gimbal, carries a bank pointer, which registers against a bank-angle scale. Movement in both cases, limited by resilient stops, prevents any internal damage to the instrument.

The instrument is gyro stabilised, and arranged so that when the gyro is erect, the horizon bar is horizontal with reference to the Earth's surface, and the angle of bank pointer is in its centre position, showing the gyro to be vertically erect with reference to the Earth's surface.

Bank indication, given by an index on the sky plate, reads against a scale printed on the glass face of the instrument. When the aeroplane banks, the rotor, inner gimbal, and outer gimbal remain rigid in their level position, whilst the instrument case and printed scale move with the aeroplane; thus, the position of the sky plate index indicates the aeroplane's bank angle against the scale.

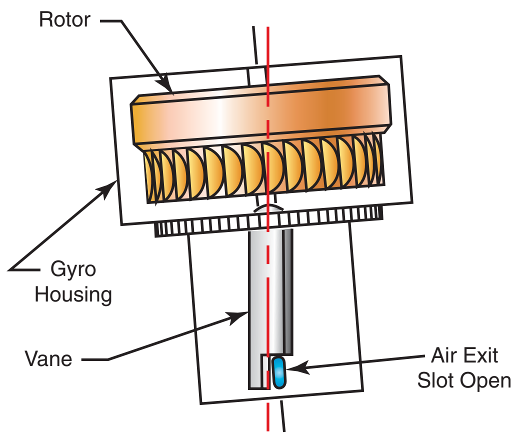
OPERATION

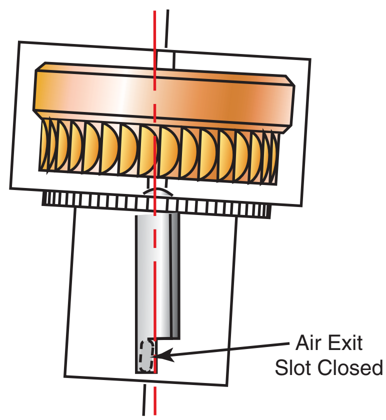
During level flight, the aeroplane's vertical axis is parallel to the rotor spin axis, with the guide pin in the centre of the slot in the outer gimbal, and the horizon bar centralised. During a climb or descent, the rotor, and hence inner gimbal, remains rigid with reference to the local vertical, whilst the outer gimbal and instrument case move with the aeroplane, and turn about the $Y\text{-}Y_1$ axis. When the aeroplane starts to climb, the rear of the instrument case and outer gimbal follow the nose of the aeroplane and rise up. This causes the guide pin to move relative to the inner gimbal, thus displacing in the slot in the outer gimbal, and placing the horizon bar below the model aeroplane, giving a relative indication of a climb. Conversely, when the aeroplane starts to descend the rear of the instrument case and outer gimbal depress with the nose of the aeroplane. The movement of the guide pin causes the horizon bar to move above the model aeroplane, thus providing a relative indication of a dive.

During a roll manoeuvre, the instrument case and model aeroplane rotate about the fore and aft axis ($X\text{-}X_1$), but the gyro assembly, including the inner gimbal, outer gimbal, and horizon bar, remains level. The model aeroplane turns in relation to the horizon bar, and provides an indication of bank.

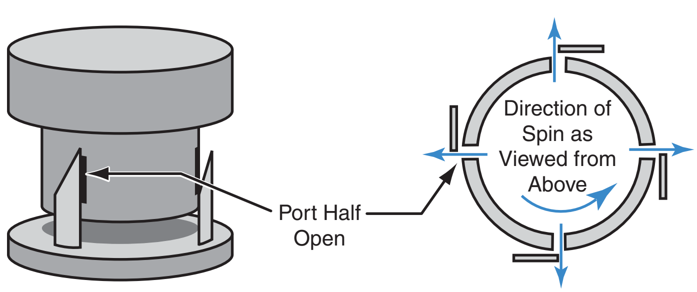
ERECTION SYSTEM

The air driven instrument incorporates a mechanical pendulous vane unit, which erects the gyroscope into its vertical position, and maintains its spin axis in that position during its operation. Shown below is a typical unit:

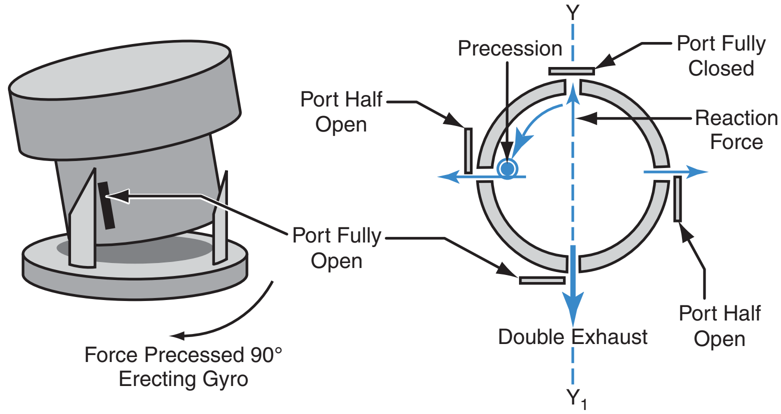




The unit, fastened to the underside of the rotor housing, consists of four knife-edged, pendulously suspended vanes, fixed in diametrically opposed pairs on two shafts supported in the unit body. One shaft is parallel to the pitch axis (Y-Y), whilst the other is parallel to the roll axis (X-X), of the gyroscope. In the sides of the unit body are four small, elongated ports, one located under each vane. Suction air, after spinning the gyro rotor, is exhausted through the ports, and the reaction of these diametrically opposed streams of air applies a force to the unit body. The vanes, under the influence of gravity, always hang in the vertical position, and govern the amount of airflow from the ports. They also control the forces applied to the gyroscope by the exhaust air reaction forces. When the gyroscope is in its vertical position, the knife-edge of each vane equally bisects each port, thus making all four port openings of equal dimension.



The air reactions are similarly equal, and the resultant forces about each axis are in balance. If the spin axis is however displaced from its vertical position, as shown below, the pair of vanes positioned on the Y-Y, axis remain vertical, thus fully opening one port whilst the diametrically opposing port is fully closed.



The increased reaction force produced by the air expelled from the fully open port results in a torque applied to the gyro body in the direction of the arrow, and thus according to the law of precession, the unit rotates about the pitch axis (Y-Y_1). The spin axis is returned to its local vertical or erect position when the vanes again equally bisect the ports, which results in equal reaction forces.

ERRORS

The air driven artificial horizon suffers from both acceleration and turning errors, and for the purpose of explanation it is assumed that the gyro rotor rotates in an anti-clockwise direction when viewed from above.

Acceleration Error

This error, known as **take-off error**, is most noticeable during the take-off phase of flight, and is caused by the pendulous unit and its associated vanes. The pendulous unit makes the rotor housing (inner gimbal) bottom-heavy, so that when the aeroplane accelerates, a force due to the unit's inertia, which is effective at the bottom of the rotor system, acts in the direction of the flight crew. The resulting force is precessed through 90° in an anti-clockwise direction, and lifts up the right-hand side of the outer gimbal. This causes the sky-plate, attached to the outer gimbal, to rotate anti-clockwise, and indicates a false turn to the right against the bank angle index. Additionally, during acceleration, both of the laterally (left and right) mounted side vanes are thrown aft, which results in the right-hand side port fully opening and the left-hand side port fully closing. This in turn produces a reaction force on the right-hand side, which when precessed through 90°, lifts the inner gimbal, and indicates a false climb. A classic artificial horizon will thus indicate a false climbing turn to the right during the take-off phase of flight.

Turning Error

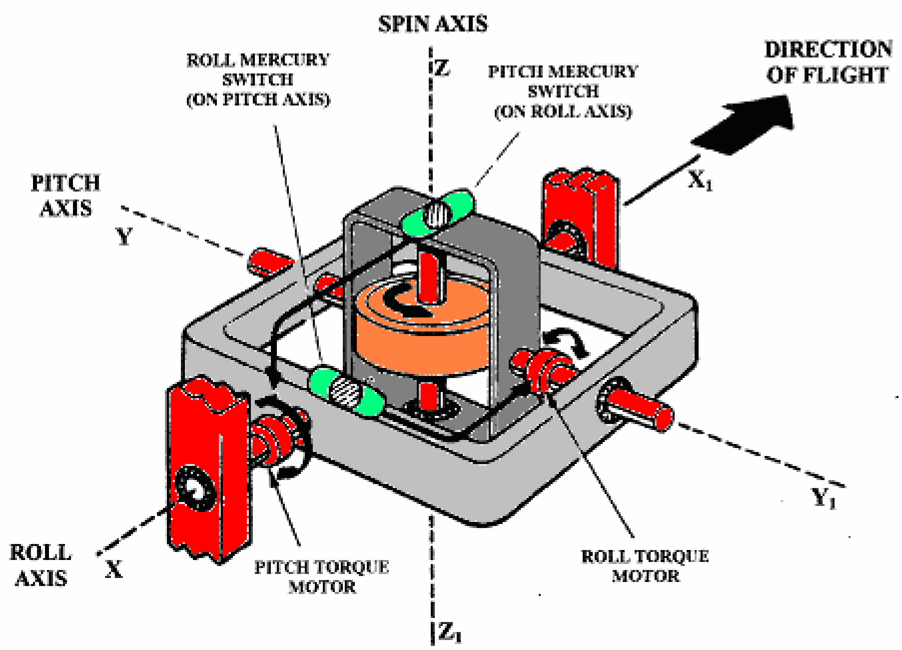
During a turn, the longitudinally (fore and aft) mounted vanes on the air-driven artificial horizon displace due to the centrifugal force acting on the pendulous unit. This causes one port to open, whilst the opposing port closes, and a reaction force sets up along the fore and aft axis of the aeroplane (X-X_1). After precessing the force through 90°, it tends to lift the outer gimbal on the left or right hand side depending on the direction of the turn, resulting in a false bank indication, or **Erection Error**. It follows that during a left or right turn the instrument under-indicates the bank angle.

Additionally, the centrifugal force causes the pendulous unit to swing outward in the opposite direction to that of the turn, causing the inner gimbal to give a false indication of climb or descent, known as **pendulosity Error**. During a left turn, the classic artificial horizon indicates a false descent, and during a right turn indicates a false climb. These two forces act together, and during a 360° turn, the error reaches a maximum value at 180°, and returns to zero when the turn is complete. In modern gyroscopes, the axis of rotation is slightly offset from its true vertical to counter these errors, although this is only valid for one particular rate of turn and airspeed. The scales, similarly offset, do not affect the indications during straight and level flight.

ELECTRICALLY DRIVEN ARTIFICIAL HORIZON

CONSTRUCTION

An example of an electrically driven artificial horizon is shown below.

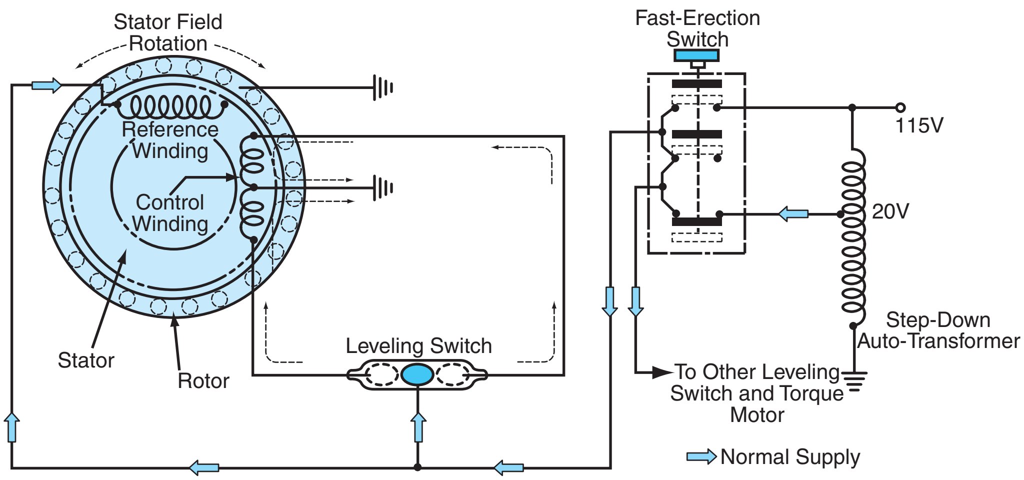


It consists of the same basic components as the vacuum-driven type, except that the vertical spin axis gyroscope is a squirrel-cage induction motor. Unlike conventional induction motors, where the rotor normally revolves inside the stator, in order to make the motor small enough to fit within the space available in a modern miniaturised instrument, the rotor is designed so that it rotates in bearings outside the stator. This ensures that the mass of the rotor is concentrated as near to the periphery as possible, thus ensuring maximum inertia, and adequate rigidity.

The squirrel-cage motor design is not only used in the artificial horizon, but is also used in other instruments that employ electrical gyroscopes. The motor assembly is carried in a housing that forms the inner gimbal, and is supported in bearings in the outer gimbal, which in turn is supported in bearings in the front and rear casing of the instrument. The horizon bar assembly is in two parts, and like the air driven version is similarly pivoted at the rear of the outer gimbal. The instrument is fitted with a torque motor erection system, which maintains the gyro in its vertical axis. The electrical motor rotates the rotor at approximately 22 500 rpm. If the power supply fails, it is indicated by a solenoid-actuated OFF flag, which appears in the face of the indicator.

TORQUE MOTOR AND LEVELLING SWITCH SYSTEM

The torque motor electrical system consists of two torque control motors, which are independently operated by mercury levelling switches; one is mounted parallel to the longitudinal axis, and the other is one is mounted parallel to the lateral axis. The lateral switch detects displacement of the gyroscope in roll, and is connected to the torque motor mounted across the pitch axis, whilst the longitudinal switch detects displacements in pitch, and is coupled to a torque motor mounted across the roll axis. Each levelling switch consists of a sealed glass tube containing three electrodes and a small quantity of mercury.



When the gyro is running in its normal operating position, the mercury in the levelling switches lays in the centre of the tubes, and is only in contact with the centre electrode, whilst the two outer electrodes, connected to their respective torque motors, remain open as shown above.

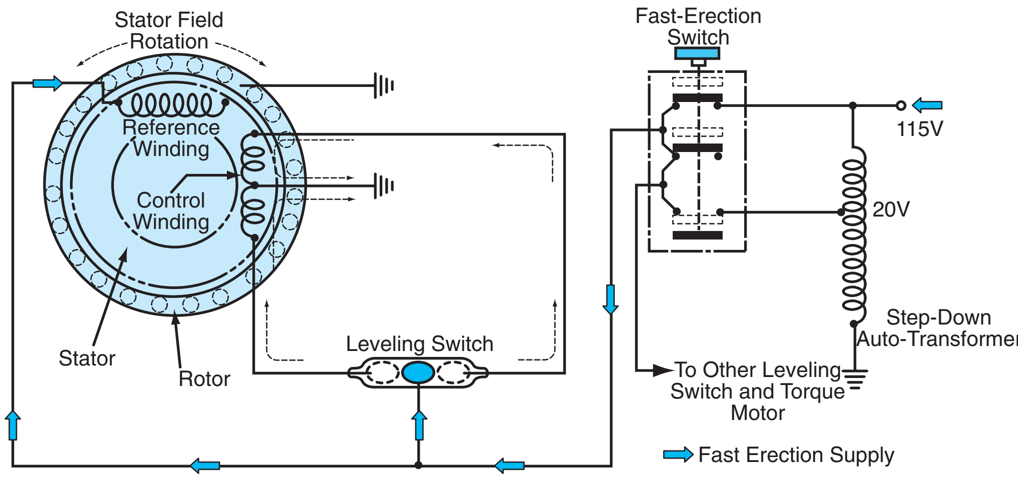
The autotransformer in the system reduces the voltage to a nominal value (20V), which is then fed to the centre electrode of the switches, so that no current flows to the torque motors when the system is level. If the gyro is displaced (e.g. about the pitch axis [Y-Y_1]), the pitch-levelling switch displaces, and the mercury rolls in the direction of pitch to make contact with one of the outer electrodes. This results in the electrical circuit to the laterally mounted pitch torque motor completing, energising the motor, and causing it to apply a torque force. According to the law of precession, a subsequent force acts on the gyro about the pitch axis, returning the gyro spin axis back to its vertical position. The pitch levelling mercury switch is now returned to its normal operating position, with the mercury element now only in contact with the central electrode, thereby removing the electrical supply from the pitch torque motor.

FAST ERECTION

On many electrically operated gyro horizons that employ the torque motor method of erection, there is also a roll cut out switch, designed to prevent false erection signals sent to the erection torque motors during a prolonged turn. In this arrangement, if the gyro rotor spin axis becomes more than 10° misaligned from the vertical, a commutator switch, fitted to the outer gimbal ring, interrupts the current flow between the mercury switches and the torque motors. If the gyro is switched off, and subsequently toppled, this device also prevents automatic erection from taking place.

In order to overcome the problem and to bring the gyro to its operational state as quickly as possible, a fast erection system is provided. This system is designed to bypass the roll cut out switch, and to apply a higher than normal voltage to the erection motors.

Shown below is a typical system, where only one mercury switch and one torque motor, are shown to emphasise how the system operates.



If the gyroscope exceeds the appropriate limits of movement from its vertical position, it is important that the gyro is brought back to normal as quickly as possible, which is achieved by pushing in the fast-erection switch. In this position the torque motors are supplied directly with 115 volts, which increases the torque motor output and hence produces greater torque. This results in the erection rate increasing from the normal 5° per minute, to between 120° and 180° per minute, depending on the particular design. In order to prevent the torque motor overheating, it is important that the fast-erection switch not be used continuously for longer than 15 seconds, nor is it used when the gyro is in its vertical position.

ERRORS

The electrically driven artificial horizon like the air driven derivative similarly suffers from both acceleration and turning errors.

Acceleration Error

In the case of the electrically driven gyro horizon, the inner gimbal does not have a pendulous erection unit hanging below it as in the case of the air driven or classic version, and is therefore not subject to the apparent turn component of acceleration error. However, the mercury in the longitudinally mounted switch hangs back and completes the circuit to the pitch torque motor, and causes the instrument to indicate only a **false climb**, and not an apparent climbing turn to the right, as in the case of the air driven variant.

Turning Error

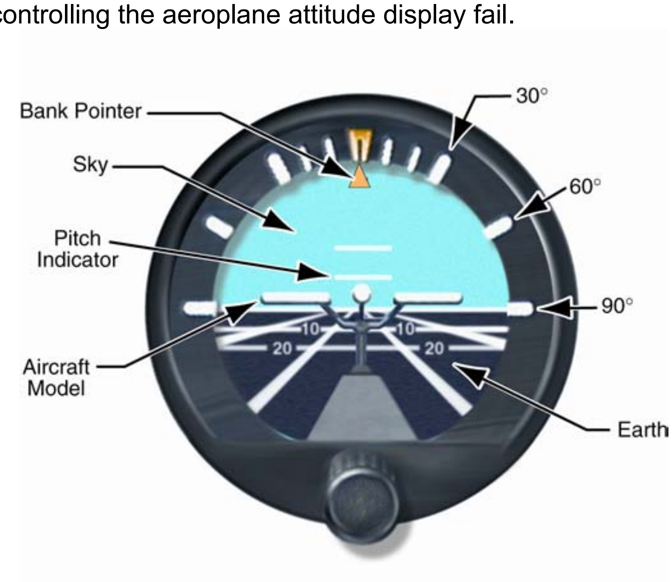
The effect on an electrically driven gyro is to displace the mercury in the lateral mercury switch, completing the circuit via one or the other of the outer electrodes to the roll torque motor, causing the instrument to indicate only a **false bank**, and not a false indication of turn and climb or descent, as in the case of the air driven variant.

REMOTE VERTICAL GYRO

On many modern aeroplanes, the attitude indicator is fed from a remote vertical gyro unit, normally sited in the avionics bay. This gyro works in the same way as the electrical gyro just described except it is not linked directly to a presentation. Pitch and bank data is fed to the remote (panel mounted) indicator by means of an electrical synchro transmission system. The same attitude information is also fed to the autopilot so that it can use the same data as the flight crew. The biggest advantage of using the remote vertical gyro is that it can provide greater degrees of freedom, and the indicator is constructed to present all attitudes with virtually unlimited freedom.

STANDBY ATTITUDE INDICATOR

Many modern aeroplanes employ integrated flight systems, which include indicators that can display not only pitch and roll attitude data from a remotely located vertical axis gyroscope, but also associated guidance data from radio navigation systems. In these systems there is no longer a need for a separate artificial horizon to be fitted, but one is fitted in order to satisfy the airworthiness requirements as a standby attitude indicator. This provides the necessary indication should the circuits controlling the aeroplane attitude display fail.



An example of the face of a typical standby indicator is shown above. This instrument uses an internal gyroscope, which is electrically operated and is powered during normal operation by the aeroplane's 115V 3-phase supply. If the normal power supplies fail, a static inverter provides 28V DC from the battery busbar, and automatically supplies the standby artificial horizon. Power from this source is always available, so attitude indications are continually displayed.

In place of the conventional stabilised horizon bar method of displaying pitch and roll, a stabilised spherical element is adopted as the reference against an aeroplane symbol. The upper half of the element is coloured blue (sky) to display climb attitudes, while the lower half is black, to display descending attitudes. Each half is graduated in 10° increments up to 80° climb and 60° descent. A pointer and scale indicates the bank angle in the normal manner. The indicator also has a pitch-trim adjustment and a fast-erection facility. When the knob is rotated in its IN position, the aeroplane symbol may be positioned through ±5° variable pitch trim. Pulling the knob out, and holding it, alternatively energises a fast-erection circuit.

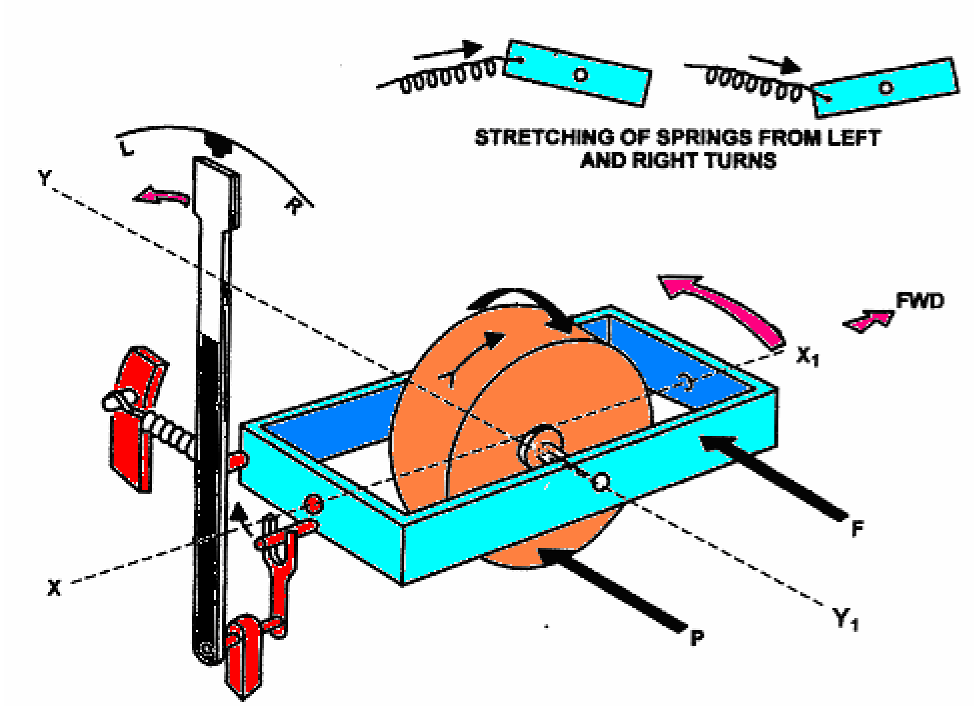
INTRODUCTION

The turn and balance indicator, previously known as the turn and slip indicator, is essentially two instruments in one casing, which provide separate indications on a common dial. A turn indicator displays the rate of, and direction of turn, using gyroscopic principles; and a balance indicator to show whether the aeroplane is performing a balanced or unbalanced turn (skidding or side-slipping). Shown below is the dial presentation of a typical turn and balance indicator.

TURN INDICATOR

CONSTRUCTION AND PRINCIPLE OF OPERATION

The turn indicator comprises of a horizontal spin axis gyro, supported in a gimbal ring, and is mounted with its plane of rotation acting along the fore and aft or roll axis (X – X_1) of the aeroplane.



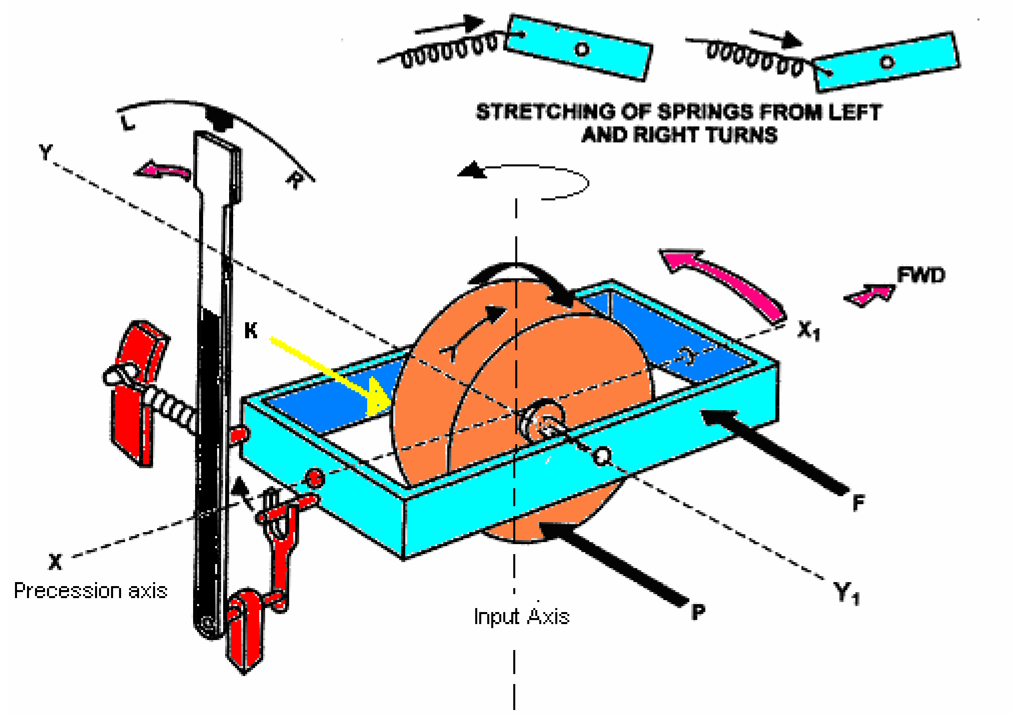
It uses a rate gyro, and has freedom of movement in the rolling plane only. The rotor is either electrically driven, and includes a power failure warning flag, or is air driven. Both types of drive are structured to produce a low rotor speed of approximately 9000 rpm, because in level flight, the gyro axis is maintained in its horizontal position by an adjustable spiral spring.

The spring attaches between the gimbal and the instrument case. A pointer is also attached to the gimbal, and moves over a scale showing the aeroplane's rate of turn, which is positioned adjacent to the zero datum mark, when the gyro is in its horizontal position (i.e. when the aeroplane is in level flight). A damping device, usually a piston cushioned by air in a cylinder, is fitted to the gimbal to ensure that the instrument reacts smoothly to changes in the rate of turn, and at the same time reacts to a definite turn rate without pointer oscillation.

When the aeroplane turns, the gyro precesses, thus tilting the rotor and gimbal ring until the precessing force matches the tension of the spring. At this point, the precession ceases, and the gyro remains inclined for the duration of the turn, giving an indication of the actual rate of turn, shown by the pointer's position on the scale. When the aeroplane stops turning the gyro returns to its original horizontal position under the action of the spring.

OPERATION

For example, when an aeroplane enters a left turn, the gyro axis, which is rigid, opposes the turn, and a force is experienced about the vertical input axis.



The gimbal ring also turns with the aeroplane, but the resultant turning moment resists due to the rigidity of the gyroscope, which precesses about the longitudinal ($X – X_1$) axis. During a left turn, a force applies at the front pivot of the gimbal ring, which is the same as applying a force at point F on the rotor rim. Due to primary precession, a subsequent force acts 90° later in the plane of rotation (i.e. at point P), causing the gimbal ring to tilt about the fore and aft axis. The pointer, connected to the gimbal ring, also moves, and in doing so indicates the direction of turn via reverse gearing. The rate of turn can also be established, since the force exerted by the spring is directly proportional to the amount of gimbal deflection.

During a left turn, the gyroscope, in precessing, stretches the spring until the force it exerts prevents further deflection of the gyro. As the gimbal ring deflects under the influence of force P, the stretched spring exerts a downward force where attached to the gimbal. This equates to a force pressing on the left-hand lower part of the gyro rotor (i.e. opposite to force P), and when precessed through 90°, produces a rotational force about the input axis, acting at point K on the rim. Force K acts in the same direction as the original turning force, F, known as Secondary Precession. When the rate of turn is established, force F reaches a constant value, and when force K reaches the same value, (i.e. the forces applied are equal and opposite), the gyro is unable to tilt any further). Force F is due to the rigidity of the gyro, and force K is a precessing force. The angle of tilt is therefore entirely dependent on the magnitude of force F, whilst the rate of turn is a function of gyro tilt.

The scale showing the rate of turn is calibrated in what are termed **standard rates** and, although seldom marked on the instrument, are classified by the numbers 1 to 4, corresponding to turn rates of 180°, 360°, 540°, and 720° per minute respectively. On commercial aeroplanes, the scale is normally only graduated to indicate rate one turns, since turns in excess of this rate, are not normally performed in these types of aeroplanes. This is because the majority of passengers do not like to experience the acceleration forces imposed during tighter turns, and this subjects the airframe to unnecessary high load factors.

ERRORS

The turn indicator does not suffer from apparent wander because the spring prevents topple in the vertical plane, and drift in the horizontal plane is impossible due to the instruments construction. Mechanical or real wander is also normally negligible, providing that the spring tension is correctly adjusted.

Erroneous indications may however be caused if the rotor speed fluctuates too far from its normal operating rpm. If the instrument case of an air-driven gyro is not airtight, air draws into the case via the leaks, resulting in a loss of efficiency. This results in a reduction in the rotor speed and the pointer indicates a lesser rate of turn. Similarly, if the speed is too high, the pointer indicates a higher rate of turn than that flown. The most likely fault is if rotor speed falls below the design rpm, which results in both the gyro rigidity and the precessional forces reducing. Of these, the reduction in the precessional forces is the most important as they are no longer able to overcome the spring tension to the same degree. The turn indicator will, therefore, under read. In effect, the following rule is easy to remember and summarises this:

Under speed of the rotor under indicates the rate of turn.

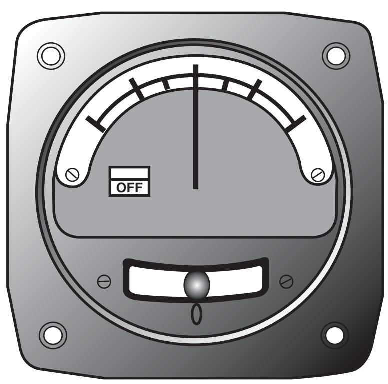
PRE-FLIGHT CHECK

If the indicator is air driven approximately five minutes should be allowed for the rotor to reach its operating rpm prior to taxiing. With the aeroplane still stationary on the ground the turn pointer should be aligned with the zero datum, but during taxiing for take-off the pointer should respond accordingly to left and right turns. In most light aeroplanes applying hand pressure to one corner of the flight instrument panel also enables the turn indicator to be checked. This is because the panel is normally fitted on shockproof mountings, and any movement results in the turn pointer indicating a momentary rate of turn.

THE BALANCE INDICATOR

CONSTRUCTION AND OPERATION

This part of the instrument uses a mechanical method to indicate that an aeroplane is correctly banked for a given rate of turn. It uses the force of gravity, which acts upon a black ball in a liquid filled glass tube, and maintains it in its true vertical position whilst the aeroplane is in straight and level flight, as shown below.



The liquid acts as a damping medium for the ball, and two expansion chambers are concealed behind the dial, to cater for temperature changes. The back of the tube is painted on the outside with fluorescent paint to provide a contrasting background for the black ball, and the whole assembly is firmly secured to the back of the dial by a bracket. The ball itself has weight and is thus affected by aeroplane manoeuvres.

If the ball remains in the centre, the turn is balanced, and no slip or skid is present, as shown in diagram (A) below.

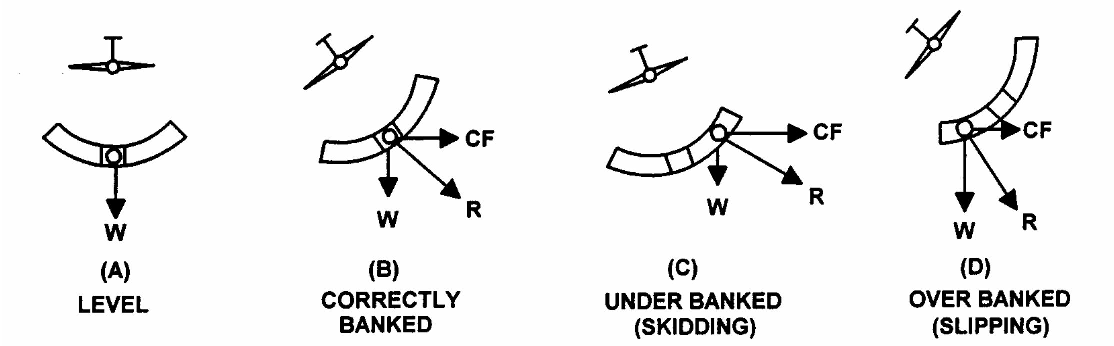


Diagram (B) shows the aeroplane making a left turn at a certain angle of bank. During this manoeuvre the indicator case and scale both move with the aeroplane. The ball is additionally subject to a centrifugal reaction, since the aeroplane is in a turn, which forces the ball away from the centre of the turn. If the turn is however carried out with the correct angle of bank, the two forces are in balance, and the ball remains in the zero position. Any increase in airspeed during the turn increases both the bank angle and centrifugal force. The ball continues to remain in line with the resultant of the two forces, as long as the bank angle is correctly maintained.

If the angle of bank for a particular rate of turn is incorrect, for example, the aeroplane is under banking, as shown in diagram (C), the aeroplane tends to skid out of the turn. This occurs because the centrifugal force predominates, and the ball displaces away from zero toward the outside of the turn. By comparison, if the aeroplane is alternatively over-banked (i.e. the angle of bank is excessive for the rate of turn, as shown in diagram [D]), the aeroplane tends to slip into the turn, since the force of gravity now predominates, and the ball moves away from its zero position toward the inside of the turn. If the aeroplane skids or sideslips, the turn is unbalanced, and if the ball remains in the centre, the turn is balanced.

LIMITATIONS AND ERRORS

The balance indicator has no operational limitations, and is also not subject to any errors.

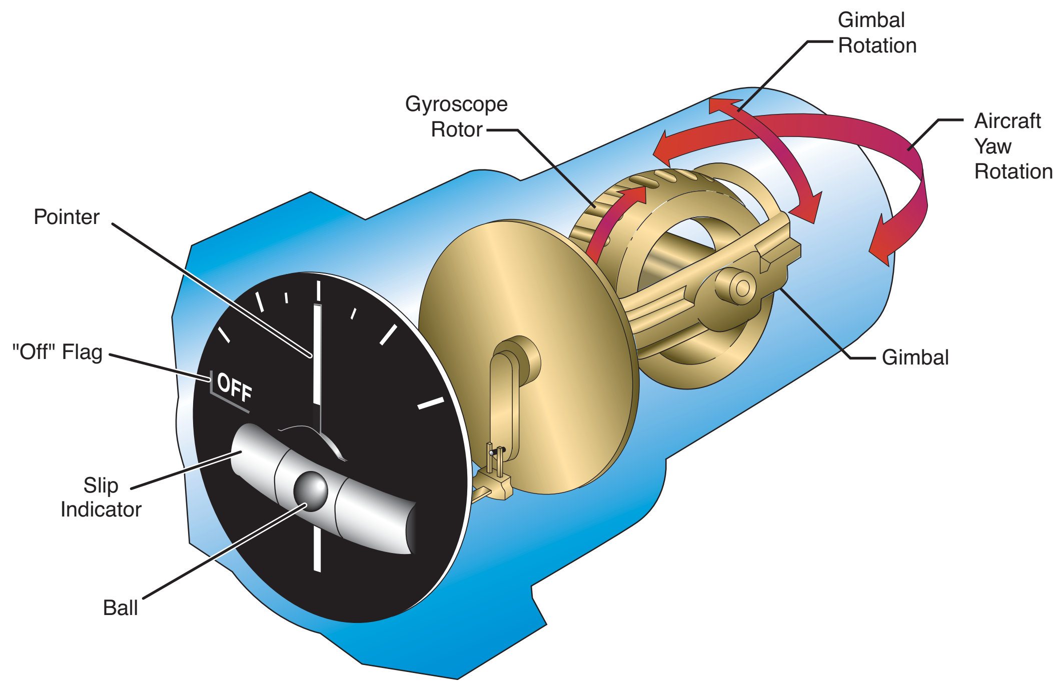
PRE-FLIGHT CHECK

With the aeroplane on level ground the ball should be in its central (zero) position, but during any turns when taxiing, the ball registers a skid.

ELECTRICALLY DRIVEN TURN AND BALANCE INDICATORS

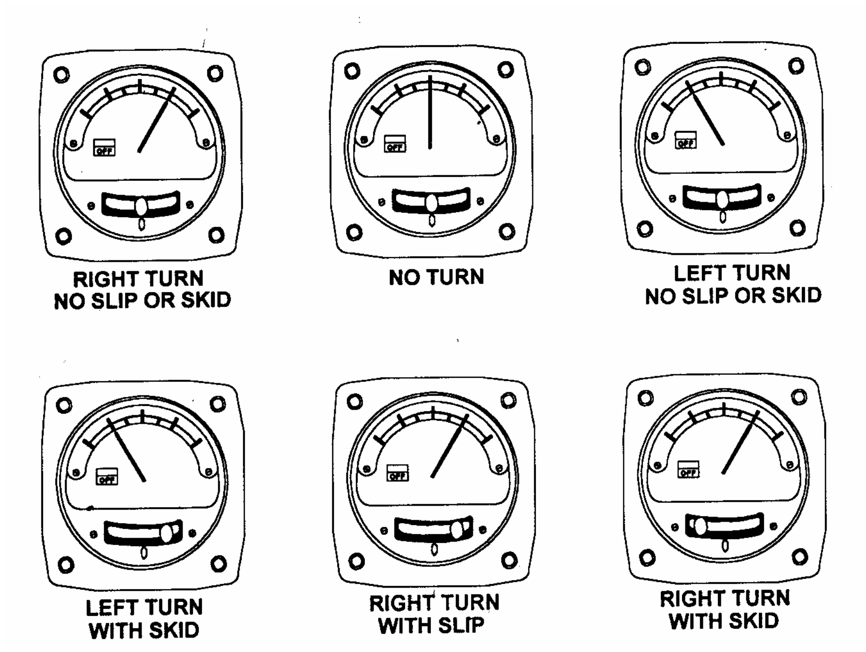
The internal mechanism of a typical electrical driven variant is similar to that of an air driven variant, as shown on the next page.

In this type it is important prior to flight to ensure that the OFF flag has disappeared from view, and during taxiing, the needle should indicate a turn in the correct direction, and the ball should indicate a skid. The flag comes into view if the rotor is not at its operating rpm (i.e. due to a power failure), and the instrument is unreliable.



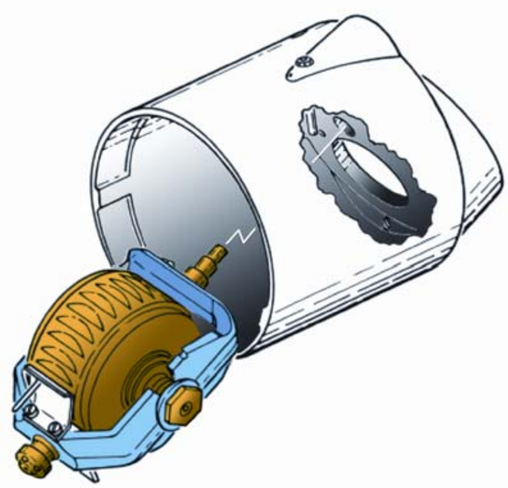
TYPICAL INDICATIONS ON A TURN AND BALANCE INDICATOR

The diagrams below show the indications that a typical turn and balance indicator are likely to show during different types of turn.



INTRODUCTION

A turn co-ordinator is a development of the turn and balance indicator and is for use in place of such instruments in a number of small, general aviation aeroplanes. The primary difference is in the position of the precession axis of the rate gyroscope. In this instrument, the gimbal is spring-restrained and mounted with its axis at approximately 30° with respect to the aeroplane's fore and aft axis, as shown below.



This has the affect of making the gyroscope sensitive to movements about the aeroplane's roll and yaw axes (i.e. to banking as well as turning).

PRINCIPLE OF OPERATION

The turn co-ordinator integrates both the rate of roll and the rate of turn together. It shows the pilot what the aeroplane is actually doing, and not what it has done by indicating the two rates on a display like the one shown:

The aeroplane symbol on the turn co-ordinator moves in the direction of turn or roll, which is unlike the artificial horizon, where the symbol is fixed to the instrument case and the horizon bar moves. "NO PITCH INFORMATION" normally appears across the indicator scale, in order to avoid confusion in pitch control, due to the instrument's similarity to an artificial horizon.

REMEMBER - THIS INSTRUMENT IS NOT AN ATTITUDE INDICATOR

If the wing of the aeroplane is lowered, even very slightly, the turn co-ordinator immediately shows a deviation from straight and level flight, whereas the turn and balance indicator shows nothing until yaw is present. The turn co-ordinator therefore anticipates a change from straight and level flight, whilst a turn and balance indicator measures a deviation, thus enabling the pilot to anticipate the resulting turn. The pilot can then simply control the turn at the required rate, indicated by the alignment of the aeroplane with the outer scale.

The gyroscope is usually powered by a DC motor, and rotates at 6000 rpm, although an AC motor, which is supplied from a solid-state inverter that is housed within the instrument, powers some turn co-ordinators. The inclusion of Silicon fluid or a graphite plunger in a glass tube is also often used in the instrument design, to assist in the damping out of any gyroscopic movements.

Chapter 19 Inertial Navigation System

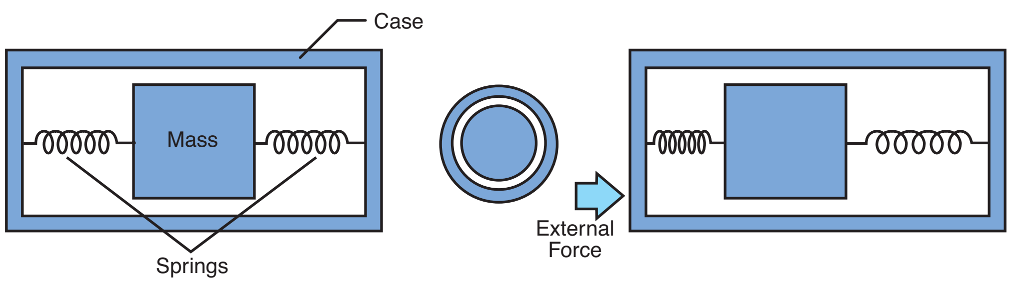
INTRODUCTION

An Inertial Navigation System (INS) provides the aeroplane's velocity and position by continuously measuring and integrating its acceleration. This system relies on no external references, is unaffected by weather, and can operate during the day or night. All corrections associated with the movement of the Earth, and transportation over the Earth's surface applies automatically. The products of an INS are position (latitude/longitude), speed (kt), distance (nautical miles), and other navigational information. The quality of information is dependent on the accuracy of the initial (input) data, and the precision with which the system aligns to True North.

An Inertial Navigation System is based on the measurement of acceleration in a known direction, or along a sensitive (input axis), which is detected and measured by an accelerometer. The output from the accelerometer is then integrated, first to provide the velocity along the sensitive axis, and second to obtain the distance along the same axis. An INS uses the process of integration because the acceleration is rarely a constant value.

For navigation in a horizontal plane, two accelerometers are needed, which are normally aligned with True North and True East. The accelerometers are placed with their sensitive axes at 90° to each other, and these axes must be maintained in the local horizontal in order to avoid any contamination due to gravity. To keep this reference valid, the accelerometers are mounted on a gyro-stabilised platform, which maintains the correct orientation during all aeroplane manoeuvres.

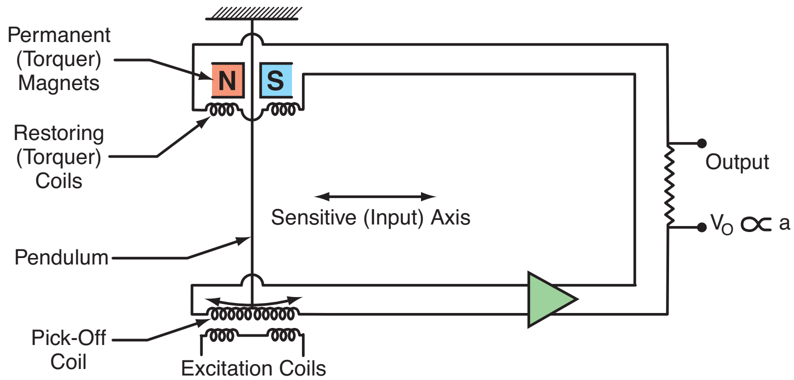
THE PRINCIPLE AND CONSTRUCTION OF AN ACCELEROMETER



The principle of an accelerometer is the measurement of the inertial force that displaces a mass when acted on by an external force (acceleration). In its simplest form, an accelerometer consists of a mass, suspended in a cylindrical casing in such a way that it can move relative to the case when the case (aeroplane) accelerates, as shown below.

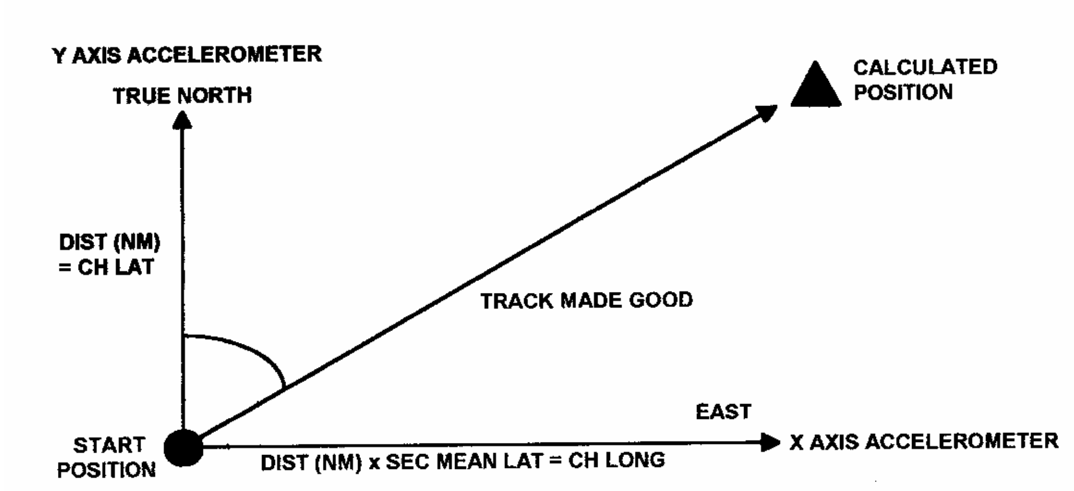
In this system, the final position of the mass is dictated by retaining springs, so that when the aeroplane is travelling at a constant velocity, the mass is positioned in the centre of the cylinder, but when the aeroplane accelerates, the mass is displaced according to the magnitude of the force exerted on the aeroplane. The final position of the mass is controlled by the pull of the springs, and the displacement of the mass is proportional to the acceleration.

The most common type of accelerometer used in the INS is the **Force Balance Accelerometer** that is based on the angular displacement of a pendulum, which rotates about a pivot point, as shown on the next page, when the aeroplane accelerates.



This type of accelerometer consists of a weighted pendulum, which is prevented from moving from its central position by two force balance (restoring) magnetic coils that act on a permanent magnet. With the outer case (aeroplane) at rest and horizontal, or moving at a constant velocity, the pendulum is in its central position, and no pick-off current flows. If the aeroplane accelerates or decelerates, the inertia of the pendulum causes it to lag behind, either to the left or right of its datum, which is detected by the pick-off coils. The resulting current, fed to the restorer coils, is of sufficient magnitude to return the pendulum to its central position. The strength and direction of the current in the coils required to resist this tendency gives a direct indication of the acceleration/deceleration applied along the accelerometer sensitive axis. This provides a measure of the rate of change of speed with time, known as its output. Conversely, any acceleration sensed perpendicular to this axis has no effect on the accelerometer.

The output signal from the restoring coils is relayed to an integrator, where it is multiplied by time to give a measure of the rate of change of distance with time (i.e. Velocity). The resulting output signal is then relayed to a second integrator where it is again multiplied by time to give the distance travelled by the aeroplane. A single accelerometer is used to determine how far an aeroplane has travelled in a definite direction for a given time. By using two accelerometers, one aligned along the North/South axis, and the other along the East/West axis, it is possible to establish the new position of an aeroplane. The INS computer processes the outputs from the accelerometers, and based on a known starting position of latitude and longitude, is able to calculate the distances travelled along each axis. For display purposes, the distances are converted into changes of latitude/longitude, and are applied to give the aeroplane's new latitude/longitude position. Changes of latitude (North/South direction) in minutes of arc equate directly to the distance covered, but due to convergence, the distance covered in the East/West direction is multiplied by the secant of the mean latitude to derive the change of longitude.



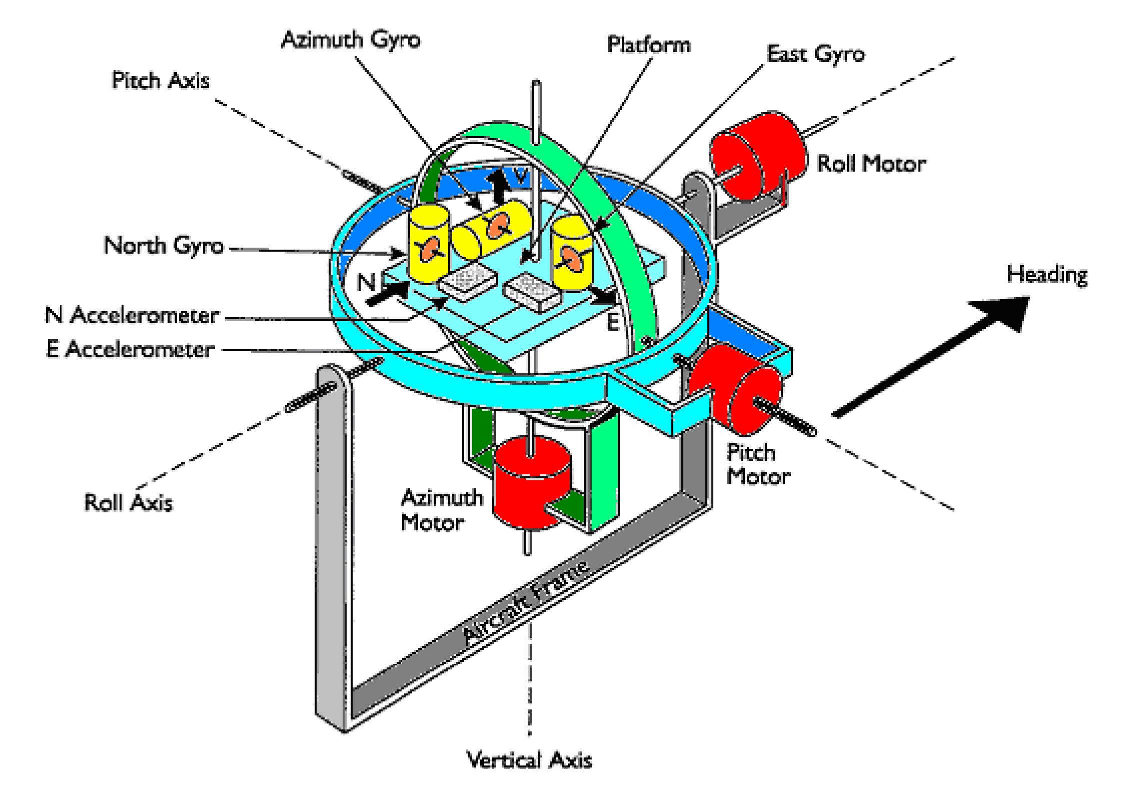
PERFORMANCE

Accelerometers used in INS applications should meet the following requirements:

Sensitivity Threshold	detect accelerations in order of 1 x 10^{-6}g
Sensitivity Range	accurate over the range –10g to +10g
Input/Output	tolerance of 0.01%
Scaling Factor	amplification of restorer current of about 5 ma/g
Zero Stability (Null Uncertainty)	gives a zero output when the input is zero, but instrument error may result in an output when the input is zero. The null position should be within +/-1 x 10^{-4}g
Small and Light	
Shock Loading	must be able to withstand a shock loading of 60g, and have a low response to vibration

OPERATION OF A GYRO-STABILISED PLATFORM

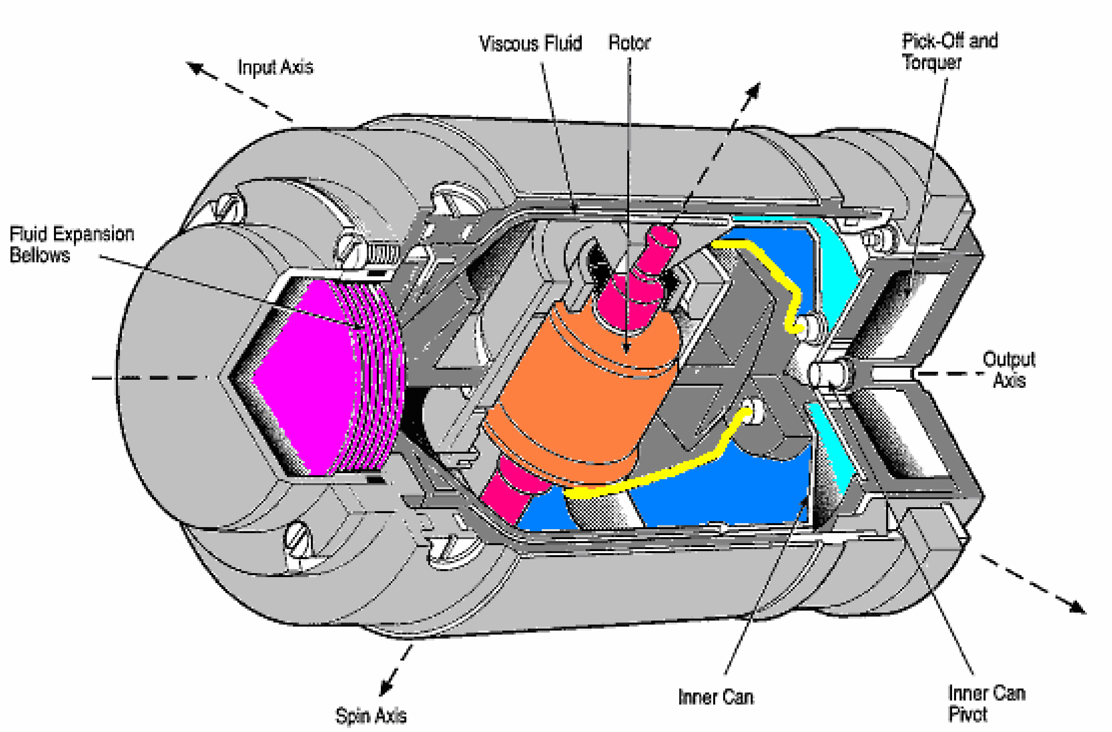
For navigation with respect to the Earth, the accelerometers in an INS are kept in their local horizontal position, and aligned with True North. This is achieved by mounting the accelerometers in the North and East directions on a North aligned gyro-stabilised platform, which isolates the accelerometers irrespective of the aeroplane's manoeuvres, or changes of aeroplane direction. Rate integrating gyros are for use as sensors to detect any departure of the platform from its level and desired alignment. These gyros are extremely sensitive and operate on the same principle as the gyro in a turn indicator.



Three gyros are normally used, one to detect rotation about the North datum, the second to detect rotation about the East datum, and the third to detect rotation about the vertical axis (i.e. they sense manoeuvres of the aeroplane in pitch, roll, and change of heading [yaw]). The gyros, which precess, detect any rotational movement of the platform, and in doing so generate a correction signal that causes an electric current to flow to the appropriate torque motor, and thus returns the platform to its original orientation with respect to the Earth. This effect is virtually instantaneous, and keeps the platform in a stable condition.

To enable the rate integrating gyros to achieve the requisite high degree of accuracy, the rotor and gimbal assemblies are immersed in a fluid, which tends to reduce any gimbal friction.

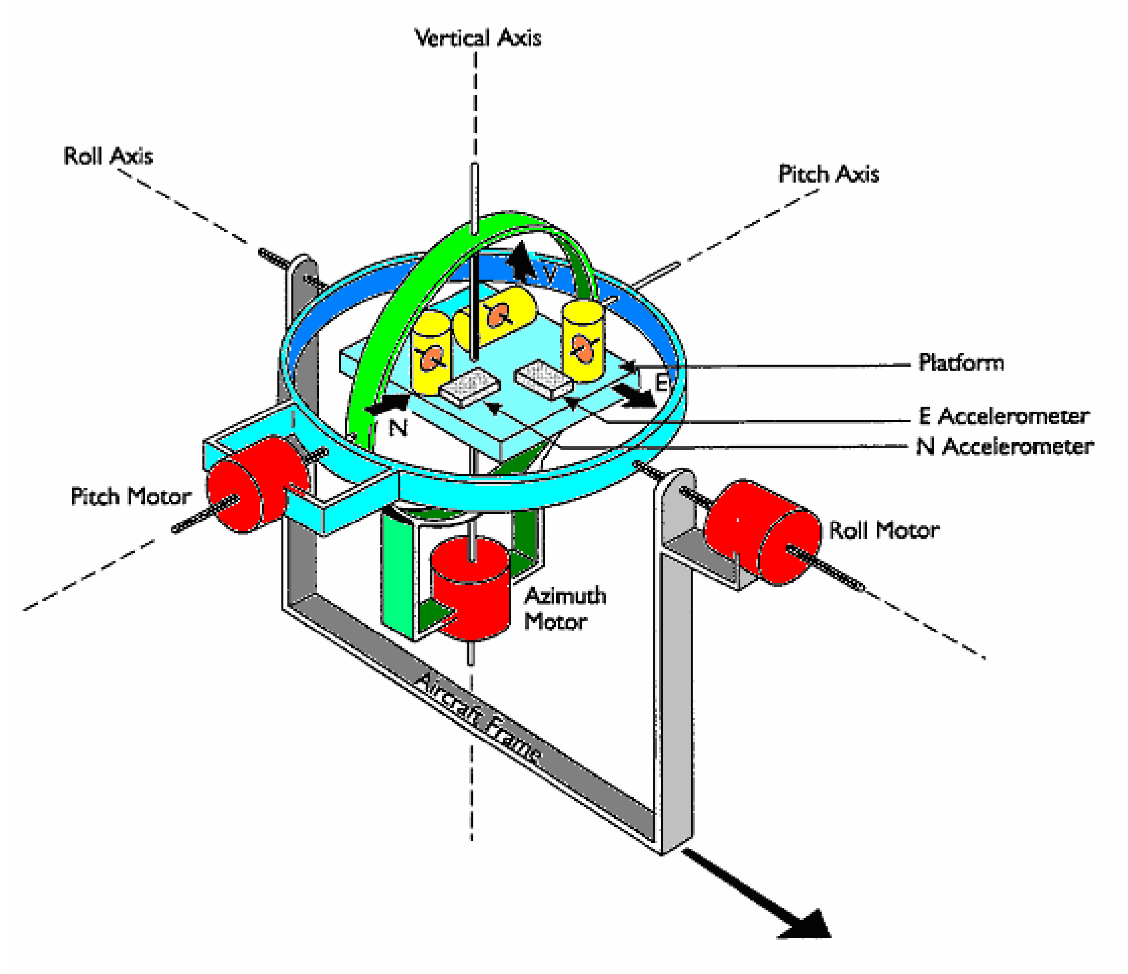
Shown below is a typical gyro:



Any torque (rotation) about the gyro's input (sensitive) axis causes the inner can to precess about its output axis (i.e. relative motion between the inner and outer cans takes place). Pick-off coils sense this movement, and the resultant output signal is proportional to the input turning rate. To avoid any temperature errors, the whole unit is closely temperature controlled.

If the platform is arranged as shown on the previous page, when the aeroplane is heading in a northerly direction, the North gyro is sensitive to roll, and the east gyro is sensitive to movements in the pitch axis. Any yaw is detected by the azimuth gyro, and all three of the rate integration gyros act to turn their respective torque motors to maintain the platform's alignment.

If the platform is alternatively arranged with the aeroplane heading in an easterly direction, as shown on the next page, the East gyro senses roll and the North gyro senses pitch. For all intermediate headings, the simultaneous action of the rate gyros/torque motors are computed, and the appropriate corrections applied. In summary, the platform isolates the accelerometers from the angular rotations of the aeroplane, and maintains the platform in a fixed orientation relative to the Earth. The assembly of accelerometers, rate integration gyros, torque motors, and the gimbal system are all mounted together on a platform, which is known as the 'stable element'.



SETTING-UP PROCEDURES

The accuracy of an INS is dependent on the alignment of the platform in azimuth and the attitude of the stable element. The platform must be horizontal (level) and aligned to the selected heading datum, normally true north, which must be maintained throughout the flight. It is important that the levelling and alignment processes be carried out on the ground with the aeroplane stationary.

Since the gyros and accelerometers used in an INS are normally fluid filled, it is necessary to first bring the containing fluid up to its correct operating temperature before the platform is aligned. The first stage in the sequence is thus a warm-up period, where the gyros are run up to their operating speeds and the fluid is temperature controlled. This phase normally takes 3-4 minutes. Whilst this is taking place, enter the current position. The alignment sequence can only begin when this sequence is complete.

LEVELLING

Coarse levelling is achieved by driving the pitch and roll gimbals until they are at 90° to each other, and aligned within 1°-2° in only a few seconds. The platform is roughly levelled using either the airframe as a reference, or by using the outputs from gravity switches to operate torque motors. **Fine levelling** follows this process where the Earth's component due to gravity is sensed by the accelerometers, and the resultant outputs are used to drive the appropriate torque motors until zero acceleration is achieved, which takes approximately 1-2 minutes to complete.

ALIGNMENT

Coarse azimuth alignment is achieved by turning the platform until the heading output agrees with the aeroplane's best-known true heading, and is carried out during the coarse levelling process. **gyro compassing**, or **fine alignment**, automatically initiates once the platform levels. If the platform is not accurately aligned with True North (in azimuth), the East/West gyro senses the rotation of the Earth. If the accelerometer is lying with its sensitive axis exactly in an East/West direction, the Earth's rotation has no effect on the gyro. If the alignment is not precise when the INS is first switched on, the East gyro detects a component of the Earth's rotation, which is used to torque the azimuth gyro until the platform is aligned, and the East/West output is reduced to zero. Depending on the degree of misalignment, this process can take up to 10 minutes, and the aeroplane should not be moved until it is completed.

> **Note:** The magnitude of the earth rate, which is affecting the East/West gyroscope is dependent on 15.04 x cos. lat°/hr, thus if the aeroplane is operating at high latitudes, this component verges toward zero, making any alignment to True North virtually impossible. The effect of latitude on the fine alignment process limits the initial alignment of the platform to mid-latitudes and equatorial regions, so the usefulness of the North aligned system is limited.

LEVELLING AND ALIGNMENT

For a conventional gyro system, the process of levelling and alignment takes approximately 15 to 20 minutes, although this time varies from equipment to equipment. It is important that the aeroplane does not move during this process, because the resultant accelerations upset the system, preventing the platform from aligning.

This process also requires corrections to be applied, which are dependent on the system being 'told' the accurate value of the present latitude and longitude. The alignment process is unsuccessful if the wrong latitude is initially entered into the system; if the error is small, it leads to poor alignment and degraded accuracy. If the latitude error is large, the platform is unable to align.

If the longitude is however wrongly inserted during the initialisation process, it does not adversely affect the alignment process, but will affect the accuracy of any subsequent position. This is because the INS is a direct reading system, and if you start with a wrong position all subsequent positions are wrong.

The inter-relationship between levelling and alignment is complex, and any slight discrepancy in one directly affects the other. It is, therefore, important that from the moment fine levelling is complete, the necessary correction applies to keep the platform horizontal with respect to the Earth. This is because the Earth is continuously rotating, and with a gyro-stabilised device, the gyros try to maintain spatial, rather than terrestrial rigidity. The platform is thus 'tilted' as the Earth moves round to maintain its position horizontal to the Earth's surface.

CORRECTIONS

Accelerometers and gyros have sensitive axes, which extend infinitely in straight lines (i.e. they operate with respect to inertial space). The Earth is however not like that, because the local vertical axes are not constant, since the Earth is a curved surface, and it rotates. It is necessary to make corrections for earth rate and transport wander, and the accelerations caused by the Earth's rotation. Any control gyro is rigid in space, and, in order to maintain an Earth reference, it must thus be corrected for both earth rate and transport wander. It is required to make further corrections for Coriolis (sideways movement caused by Earth rotation except at the equator) and centripetal acceleration. The latter is caused by rotating the platform to maintain its alignment with the local vertical reference frame.

The stabilised platform possesses the same characteristics as a single rigid gyro, and therefore suffers from drift and topple due to earth rate and transport wander as follows:

Apparent Wander:

1) **Apparent (earth rate) drift:** The azimuth gyro must be torque compensated for earth drift, which allows for the familiar 15.04 x sin. lat°/hr.

2) **Apparent (earth rate) topple:** The North gyro must be torque compensated at a rate of 15.04 x cos lat°/hr.

Transport Wander:

1) **Transport wander drift:** Transport wander causes misalignment of the gyro input (sensitive) axis at a rate, which varies directly with speed (along the sensitive axis) and latitude. For a correctly aligned platform, the speed in an East/West direction is the first integral of easterly acceleration (i.e. the output of the east accelerometer). Latitude is also calculated by the platform and, with these two values, the INS computer calculates and applies the correction for transport wander drift.

2) **Transport wander topple:** A stabilised platform, transported across the surface of the Earth, appears to topple in both the East/West and North/South planes. To keep the platform locally horizontal, transport wander corrections are applied to the pitch/roll torque motors by the appropriate amount.

Acceleration Corrections

To maintain earth orientation, the platform is slewed to maintain north alignment, and toppled to maintain in the horizontal to the Earth's surface. During this process, the accelerometers are subject to centripetal and Coriolis effects, which produce false outputs, and the INS computer must correct for these as follows:

CORIOLIS

This is caused by the sideways force, which affects the output of both the North/South and East/West accelerometers, caused by the rotation of the Earth about its axis. An aeroplane following an Earth-referenced track will follow a curved path in space, and the resultant small error is computed. The necessary corrections are applied to the outputs of the accelerometers.

CENTRIPETAL ACCELERATION

A body moving at a constant speed in a circle (such as an aeroplane flying over the surface of the Earth where the centre of the Earth is the centre of the circle) has a constant acceleration toward the centre of the Earth. This acceleration affects the accelerometers on an inertial platform and corrections to compensate for this movement must be made, and applied to the outputs of the accelerometers. The corrections made to the gyros and the accelerometers in an INS, are summarised in the following table.

AXIS	GYROS		ACCELEROMETERS	
	EARTH RATE	***TRANSPORT WANDER***	***CENTRIPTAL FORCE***	***CORIOLIS EFFECT***
NORTH	$\Omega \cos \lambda$	$\frac{U}{R}$	$-\frac{U^2 \tan\lambda}{R}$	$-2\,\Omega\, U \sin \lambda$
EAST	Nil	$-\frac{V}{R}$	$\frac{U\,V \tan\lambda}{R}$	$2\,\Omega\, V \sin \lambda$
AZIMUTH/ VERTICAL	$\Omega \sin \lambda$	$\frac{U \tan\lambda}{R}$	$\frac{U^2+V^2}{R}$	$2\,\Omega\, U \cos \lambda$

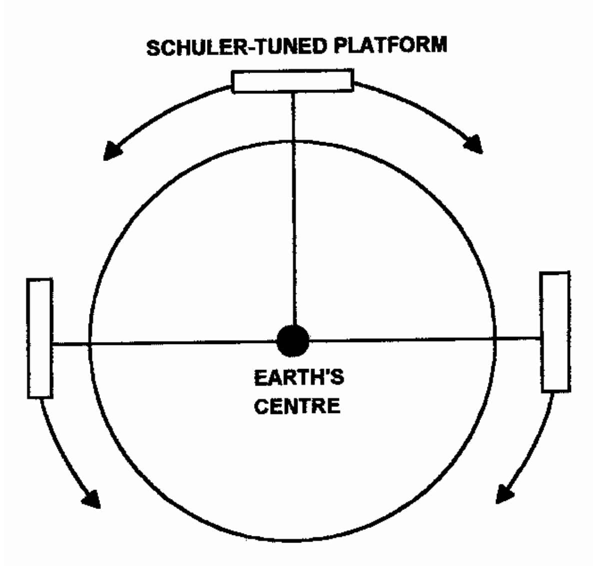
V = Velocity North
U = Velocity East
λ = Latitude
R = Radius of Earth
Ω = Rotation of Earth (15.04° / hour)

WANDER AZIMUTH SYSTEM

The effect of slew on the accelerometers increases in magnitude toward the poles, to the extent that a North aligned INS does not function properly at high latitudes. This is overcome by not physically maintaining the system aligned to True North following the initial alignment process. The alignment of the horizontal accelerometers is determined by calculation from the sensed tilt outputs of the two gyros having horizontal sensitive axes. Using this technique, the computer keeps a record of the platform azimuth alignment corrections and adds them to the initial sensed alignment mathematically. The platform does not require physical rotation and so the system becomes usable at high latitudes. In this system, the INS computer calculates the extent by which the platform has wandered from its North aligned position, and accordingly corrects the outputs from the accelerometers.

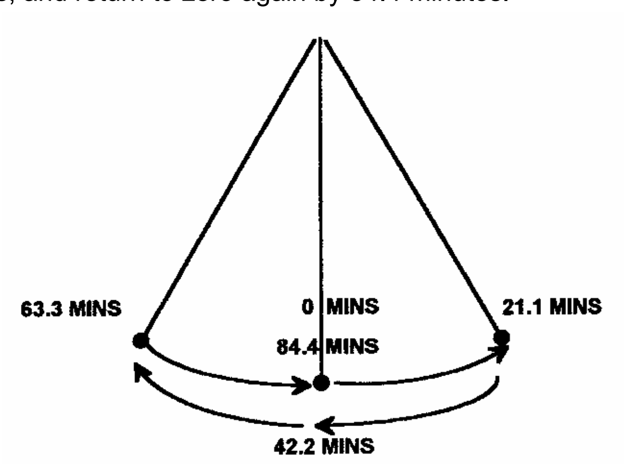
THE SCHULER TUNED PLATFORM

When a pendulous gyroscopic mass, such as an INS platform, accelerates over the Earth, it tends to oscillate, and as a result, the outputs from the sensors become inaccurate. Dr Maximillian Schuler solved the problem by demonstrating that no oscillation occurs if the platform exhibits the characteristics of an **Earth pendulum**.



Assume that the length of the pendulum is equal to the radius of the Earth, and its bob (plumb weight) is positioned at the Earth's centre with its point of suspension at the Earth's surface. If the suspension point of such a pendulum accelerates over the Earth's surface, its inertia coupled with the force of gravity combines to hold the bob stationary at the centre of the Earth, and the shaft of the pendulum remains vertical throughout. If the bob of an Earth pendulum were disturbed, as is the case if the aeroplane was the suspension point, it oscillates with a period of 84.4 minutes, as shown on the next page.

Any errors experienced, as a direct result of Schuler's oscillation, average out to zero over the 84.4 minute cycle. For example if a ground speed error of 5 kt occurs, the error builds up to 5 kt during the first 21.1 minutes, reduce to zero by 42.2 minutes, increase to 5 kt in the opposite sense by 63.3 minutes, and return to zero again by 84.4 minutes.



In practical terms, this means that the output from the INS is correct three times every Schuler Period; once when the period starts and then again at the end. It is also correct in the middle, at 42.2 minutes. At 21.1 minutes, the error is a maximum high value and at 63.3, it is at a maximum low value. For example if an aeroplane is travelling at a real ground speed of 350 kt and is subject to a bounded error of 5 kt when the platform is slightly disturbed. The following table illustrates how the ground speed varies during a typical Schuler period.

Period (min)	0	21.1	42.2	63.3	84.4
INS G/S (kt)	350	355	350	345	350

ERRORS

The INS has the following errors associated with it:

BOUNDED ERRORS

These errors build up to a maximum positive and negative values, and return to zero within each 84.4 minutes Schuler cycle. The main causes of these errors are:

a. platform tilt, due to initial misalignment
b. inaccurate measurement of acceleration by accelerometers
c. integrator errors in the first integration stage

This error thus averages out over the period.

UNBOUNDED ERRORS

The following errors exist:

CUMULATIVE TRACK ERRORS

These errors arise from the misalignment of the accelerometers in the horizontal plane, and the main causes of these errors are:

1. Initial azimuth misalignment of the platform.
2. Wander of the azimuth gyro.

CUMULATIVE DISTANCE ERROR

These errors give rise to cumulative errors in the recording of the total distance travelled. The main causes of this error are:

1. Wander in the levelling gyros.

   Note: wander causes a Schuler oscillation of the platform, but the mean recorded value of the distance travelled is increasingly divergent from the actual distance travelled.

2. Integrator errors in the second stage of integration.

The sensitivity of any INS system is due to inaccuracies in the manufacture of the rate integrating gyros and despite tight tolerances, less than 0.01°/hr is normal, thus real wander is the main culprit of unbounded errors.

INHERENT ERRORS

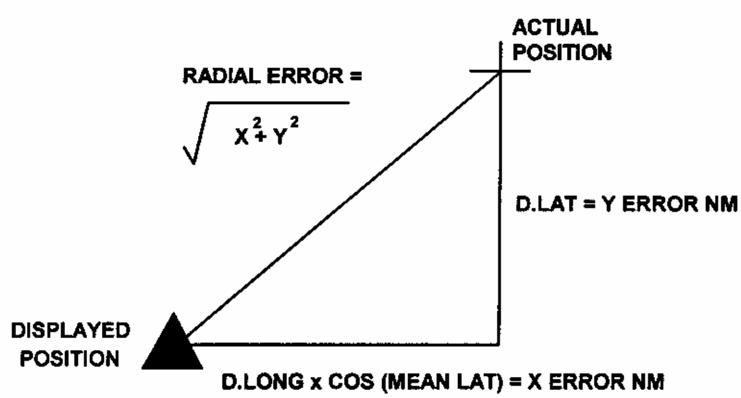
The irregular shape and composition of the Earth, together with the movement of the Earth through space, and other factors provide further possible sources of error. Such errors vary from system to system, depending upon the balance achieved between accuracy on one hand and the simplicity of design, reliability, ease of construction and cost of production on the other. The main cause of this error is:

RADIAL ERROR

This error provides an indication of the continuing serviceability of the INS, and is derived from the following:

$$\text{Radial Error} = \frac{\text{Distance of final ramp position from INS position (NM/hour)}}{\text{Elapsed time in hours}}$$

When calculating the distance between two positions, consider the latitude. It is also good practice to monitor the performance of the INS at the end of each flight. Comparing the INS display with the ramp position does this, where the radial error is established using Pythagoras, which when divided by the recorded time in the navigation mode, gives the average error.



Overall, minimal inbuilt errors exist in the INS, and according to the manufacturers, 1 nautical mile per hour is the maximum error that exists. This slightly increases during prolonged flights.

THE ADVANTAGES AND DISADVANTAGES OF AN INS

The following advantages exist:

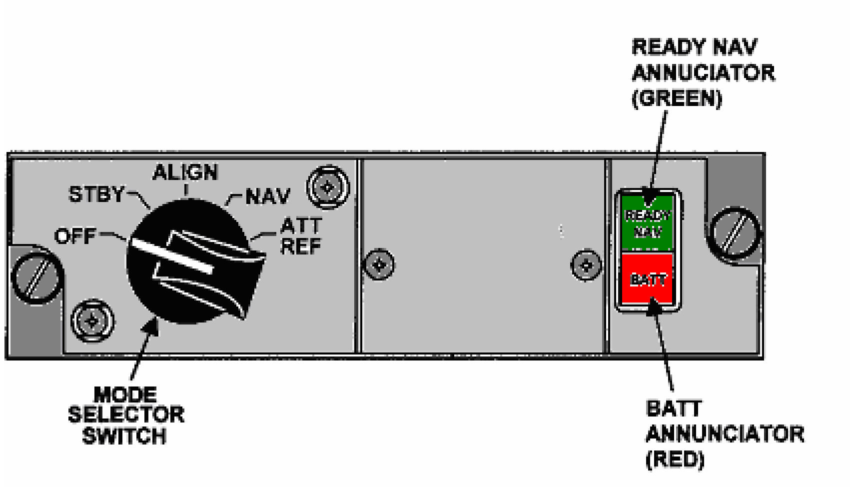
- Indications of position and velocity are instantaneous and continuous.
- Self contained; independent of ground stations.
- Navigation information is obtainable at all latitudes and in all weather conditions.
- Operation is independent of aeroplane manoeuvres.
- Given TAS, the Waypoint/Velocity can be calculated and displayed on a continuous basis.
- If correctly levelled and aligned, any inaccuracies are considered minor as far as civil air transport aeroplanes are concerned.
- Apart from the over-riding necessity for accuracy during pre-flight data insertions, there is no possibility of human error.

The following disadvantages exist:

- Position and velocity information degrades with time.
- The system is not cheap, and is difficult to maintain and service.
- Initial alignment is simple enough in moderate latitudes when stationary, but is difficult above 75° latitude, and should not be carried out in flight.

MODE SELECTOR PANEL

The functions and information displayed in the INS are controlled through a mode select panel (MSP), as shown below:



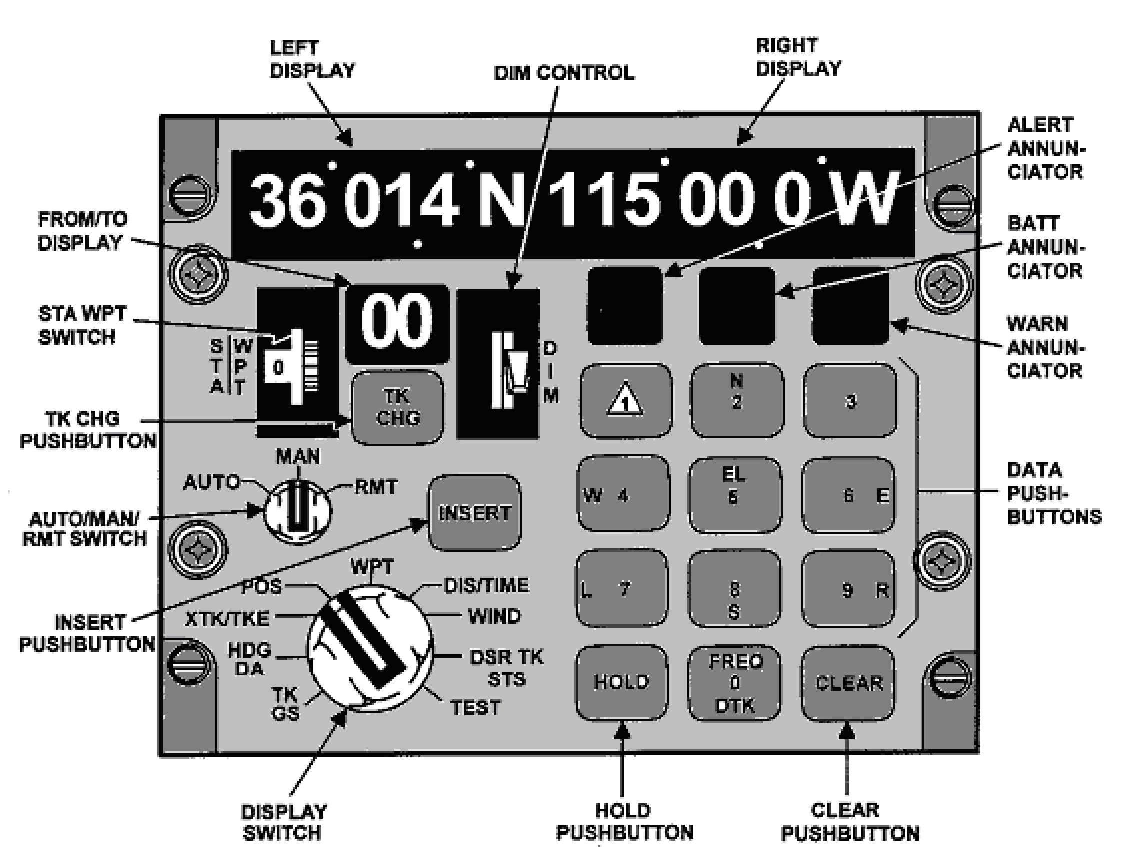
The functions provided are:

SELECTION	MEANING
OFF	Power Off.
STBY (Standby)	Power ON Power is supplied to the system, the gyros spin up, and the temperature of the oil is stabilised. The start position is normally inserted as a latitude /longitude, which is accurate to the nearest 1/10th of minute of arc.
ALIGN	Automatic alignment begins, and the platform is levelled and aligned. When the Gyro-compassing is complete, the READY NAV light (green) illuminates and indicates that the NAV mode can be selected. The aeroplane must not be moved during this process, although the aeroplane can continue to be loaded as the INS is not affected by vibration, gusts etc.
NAV	When selecting this mode, the aeroplane is ready to move. The NAV annunciator light extinguishes. This mode may be heavily indented to prevent accidental movement. This mode also requires the TAS to be fed from the ADC.

SELECTION	MEANING
ATT REF	This selects the attitude reference mode. In this mode, the INS provides pitch, roll, and platform heading outputs only. The Control Display Unit (CDU) does not display this information and selecting this mode disables the navigational capability of the INS for the remainder of the flight. This is selected if the Navigation mode fails, thus providing continual data to the EFIS and AFCS.
BATT	The illumination of this light (red) indicates that the AC power supply has failed and that the INS is running directly off the aeroplane battery (DC). This light remains on until the battery power is insufficient for the INS unit to function. A further annunciator light also illuminates on the Control Display Unit (CDU).

CONTROL DISPLAY UNIT

The diagram below shows the principle controls and displays on the Control Display Unit (CDU).



The functions are as follows:

SELECTION	FUNCTION
TK/GS	**Track and Ground speed** Track is given to the nearest tenth of a degree in the left-hand window, and the ground speed to the nearest knot displays in the right-hand window.
HDG/DA	**Heading and Drift Angle** True heading to the nearest tenth of a degree is given in the left-hand window. The drift angle is displayed in the right-hand window and is preceded by L (left drift) or R (right drift).
XTK/TKE	**Cross Track Distance and Track Angle Error** The cross track distance is the perpendicular displacement of the aeroplane from the direct Great Circle track between two selected waypoints, and is shown to the nearest tenth of a nautical mile in the left hand window. The track angle error is the angle between the current track made good, and the desired great circle track between the selected waypoints. The L or R preceding this value indicates that the actual track is to the left or right of the desired track.
POS	**Latitude and longitude of present position** Latitude displays in the left-hand window to the nearest tenth of a minute of arc, and Longitude displays in the right-hand window, to the nearest tenth of a minute of arc.
ALERT Annunciator	Comes on steady to warn the operator that the aeroplane has 2 minutes to run to the next waypoint, and then goes out when changing to the next track. If manual track changing is selected, instead of the alert light going out, it starts to flash for a new track to be inserted.
WARNING Annunciator	Illuminates when there is a system malfunction.
BATTERY Annunciator	Illuminates when operating on internal power

Chapter 20
Inertial Reference System

INTRODUCTION

Gyro-stabilised platforms are generally costly, heavy, and require a lengthy alignment procedure. With the introduction of relatively low cost, high-performance digital computers, these systems have been steadily replaced by mathematical computer software models, which modify the output signals from accelerometers that are strapped directly to the airframe. Referred to as an **inertial reference system (IRS)**, it works on the same fundamental principles as the INS, and has the following functions:

- It measures vector accelerations.
- It determines the horizontal components of these accelerations.
- It integrates the above to obtain vector velocities and distances.
- It adds the above results to a start position, to obtain the present position.

The fundamental difference between the INS and the IRS is that the latter is a **strap-down system**. The IRS senses the aeroplane's displacement about three axes to provide:

- Primary attitude
- True and magnetic headings
- Vertical speed
- Aeroplane position relative to the Earth
- Accelerations and angular rates
- Wind velocity and direction
- Ground speed

Each IRS consists of three laser gyros, three accelerometers, power supplies, a microprocessor, built in test equipment (BITE), and output circuitry. Three completely independent IRSs are normally installed on an aeroplane, and each receives barometric altitude, altitude rate, and TAS data from the Central Air Data Computer (CADC). Coupled with the gyro and accelerometer data the aeroplane's vertical speed can be determined, and the wind parameters calculated.

DESCRIPTION OF THE STRAP-DOWN SYSTEM

The strap-down system dispenses with the gimbal mounted stable element and instead uses solid-state ring laser gyros (RLG). These gyros are not required to stabilise the accelerometers, as in the case of an INS, but provide aeroplane orientation, by determining the rate of rotation around each of the aeroplane axes. The orientation data is used to process (modify) the accelerometer outputs to represent those, which, under the same conditions, are the expected outputs from the accelerometers, if they were positioned along the North, East and vertical axes. The transform matrix (a **quaternion**), is generated by digital computation, and gives the analytical equivalent of a gimballed system.

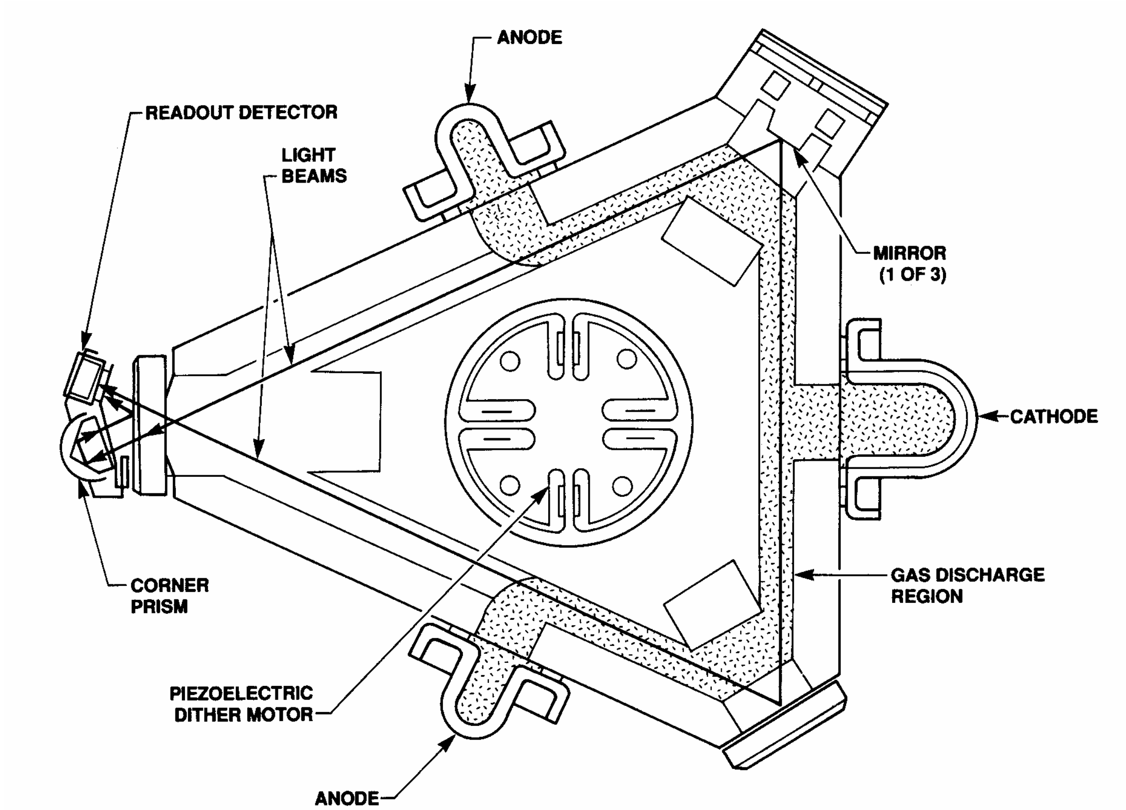
The sensitive axes of both the accelerometers and the RLGs are strapped, or fixed at 90° to each other, on the airframe along the aeroplane's principle axes. There is thus no isolation from the aeroplane's movements, and the outputs represent linear accelerations (accelerometers) and angular rates (RLGs), with respect to the aeroplane's axes.

SOLID STATE GYROS

Solid-state gyros have no moving parts, but can achieve a significantly higher level of accuracy and serviceability than a conventional gyro. Two types of solid-state gyro are currently in use in commercial aviation applications; the Ring Laser Gyro (RLG), and the Fibre Optic Gyro (FOG), which operate on similar principles.

RING LASER GYRO

Unlike conventional gyros that maintain a level attitude by a series of gimbals, the RLG is fixed in orientation to the aeroplane axes. Any change in orientation, because of an aeroplane manoeuvre, is sensed by measuring the frequencies of two contra-rotating beams of light within the gyro.



In the IRS a triad of RLGs (orthogonal axes), with their sensitive axes positioned mutually perpendicular, is utilised. A block diagram of one of these is shown above. The example shown has a triangular path of laser light, whose path length is normally 24, 32 or 45 cm. Other models use a square path (i.e. one more mirror). The RLG is produced from a block of a very stable glass ceramic compound, which has an extremely low coefficient of expansion. The triangular cavity contains a mixture of helium and neon gases at low pressure through which a current is passed. The gas (or plasma) is ionised by the voltage, which causes helium atoms to collide with, and transfer energy to, the neon atoms. This raises the neon to an inversion state, and the spontaneous return of neon to a lower energy level produces photons, which then react with other excited neon atoms. This action is repeated at speed and creates a cascade of photons,

throughout the cavity (i.e. a sustained oscillation, and the laser beam is pulsed around the cavity by the mirrors at each corner).

The laser beam is forced to travel in both directions around the cavity. Thus, for a stationary block, the travelled paths are identical, and the frequencies of the two beams are the same at any sampling point. If the block is rotated, the effective path lengths differ; one increases, and the other decreases. Sampling at any point gives different frequencies, and if the frequency change is processed, it gives both an angular change, and a rate of angular change.

By processing the difference in frequency between the two-pulsed light paths, the RLG is used as both a displacement, and as a rate gyro. There is a limit of rotation rate below which the RLG does not function because of minute imperfections (instrument error) in the mirrors. Consequently, one laser beam can 'lock-in' to the other, and therefore no frequency change is detected. If this occurs, the RLG ceases to be a gyro, which is equivalent to gimbal-lock in a conventional gyro system. Using an AC piezo-electric motor, which operates at a frequency of approximately 350 Hz, and gently vibrates or 'dithers' the complete block, prevents 'lock-in' of the two laser beams. The outputs of the RLG are digital, not mechanical, and the reliability and accuracy should exceed those of a conventional gyro by a factor of several times.

FIBRE OPTIC GYROS

Like the RLG, the FOG comprises of a triad of gyros, positioned mutually perpendicular to each other, and similarly three accelerometers. The FOG senses the phase shift proportional to angular rate in counter-directional light beams travelling through an optical fibre. FOGs are dimensionally similar to RLGs, although the FOG benefits from less weight and is overall cheaper. The FOG is however not quite as rugged, nor as efficient as the RLG.

ADVANTAGES AND DISADVANTAGES OF RLGS

RLGs suffer from the following advantages and disadvantages:

The advantages of RLGs are:

- High reliability
- Very low 'g' sensitivity
- No run-up (warm-up) time
- Digital output
- High accuracy
- Low power requirement
- Low life-cycle cost

The disadvantage of RLGs is that they are initially expensive to buy.

ALIGNMENT OF THE INERTIAL REFERENCE SYSTEM

Although the assembly is 'bolted' to the aeroplane frame, an Inertial Reference System (IRS) or RLG INS, still needs alignment to an Earth reference. Instead of levelling and aligning as in the case of a stable platform, the speed and flexibility of a digital computer allows the calculation and compilation of a transform. The transform is a mathematical solution as to where the horizontal and True North lie with respect to the triad of RLGs and accelerometers. Full alignment takes less than 10 minutes, at the end of which an offset to each output of the RLGs, and accelerometers is established. This consequently allows the local horizontal and True North references to be determined. The initial calculated values are applicable at that place, on that heading, at that time. The Earth certainly moves on, and if the aeroplane moves as well, the vital references must be safeguarded. This is achieved by making sure that the NAV mode is engaged. The complexities of 3-D motion (i.e. the interactions of pitch, roll, and yaw, require an extensive mathematical and trigonometrical juggle to be quickly conducted).

This process is the reverse of the techniques used in a conventional INS, where instead of creating a reference from a gimballed system, a reference is created from data taken from a completely different set of values. If the aeroplane heading was not altered since the IRS was last used, then a rapid alignment taking approximately 30 seconds is possible.

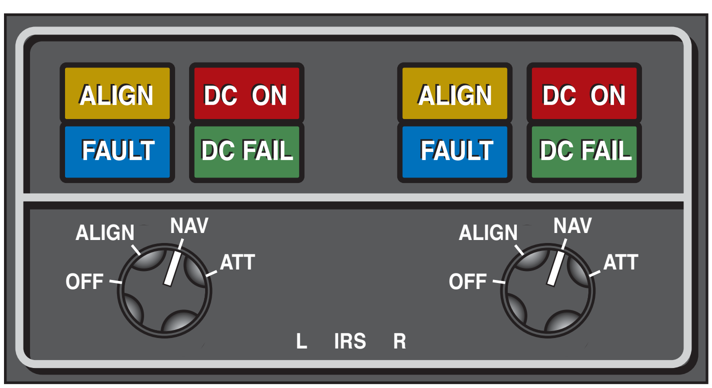
PERFORMANCE

The performance of an IRS (RLG INS) is generally slightly better than that of a conventional INS, the principal advantage being reliability. The system has the following performance criteria:

a. Position accuracy - 2nm/hr
b. Pitch/roll - 0.05°
c. Heading (T) - 0.40°
d. Groundspeed - ±8 kt
e. Vertical velocity - 30'/second
f. Angular rates - 0.1°/second
g. Acceleration - 0.01g
h. 95% probability, assuming no update with other navigation sources

THE CONTROL, DISPLAY AND OUTPUT FROM AN IRS

Control and Display of an IRS is very similar to that of the conventional INS, and a typical master switch unit (MSU) is shown below.



The outputs are on a display panel and give the information selected by the crew, with any changes made using a menu driven keypad.

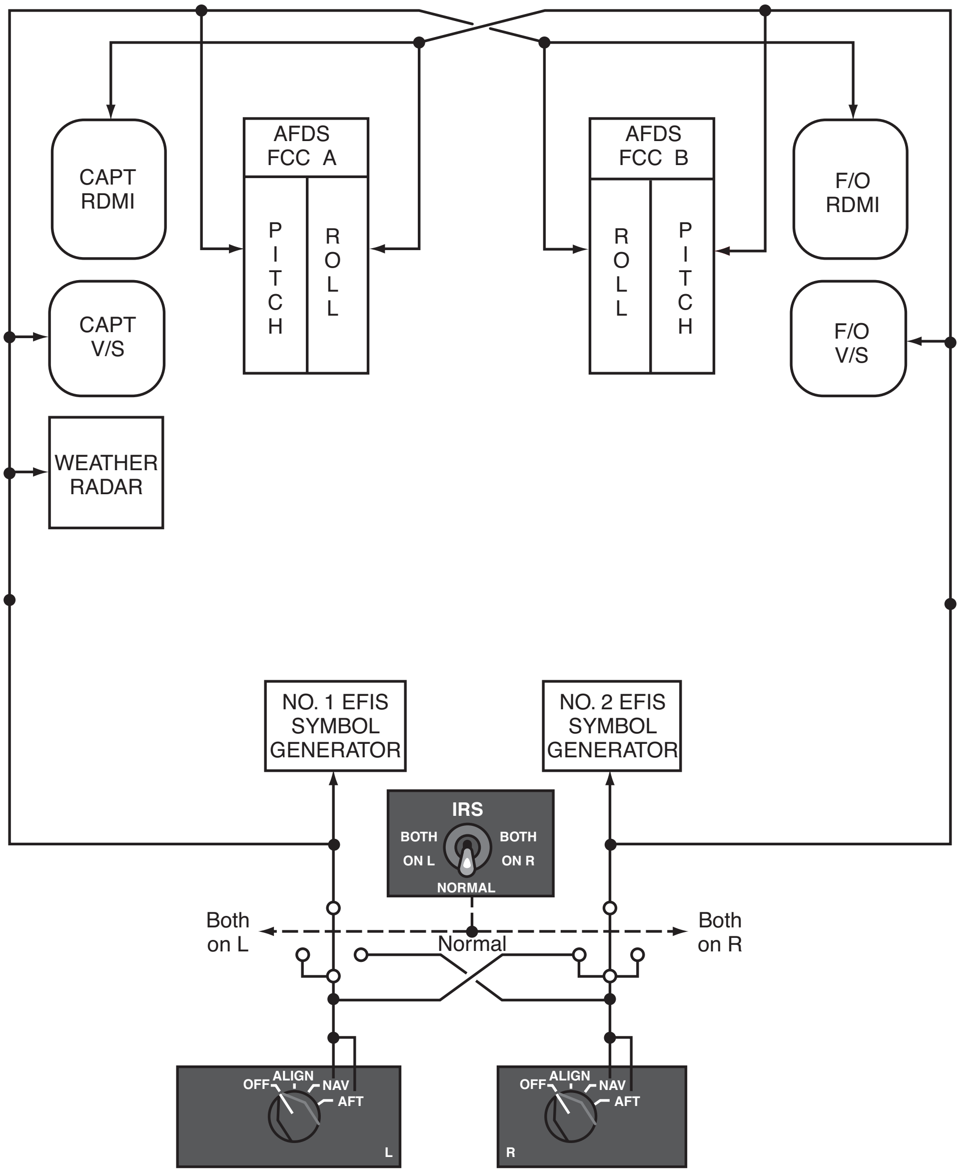
DESCRIPTION OF A TYPICAL IRS

In a typical system, magnetic variation between latitudes 73°N and 60°S is stored in the IRS memory, and data corresponding to the present position combines with the aeroplanes true heading to determine its magnetic heading. The IRS is the aeroplanes normal sole source of attitude and heading information, although a standby attitude indicator and standby magnetic compass are still fitted.

The IRS outputs are independent of any external navigation aids, and in the normal NAV mode provides the following data to the various aeroplane systems:

- Attitude
- True and magnetic headings
- Acceleration
- Vertical speed
- Ground speed track
- Present position
- Wind data

A block schematic of the overall input/output functions for a typical IRS is shown on the next page.



The IRS also integrates with navigation aids and other equipment on the aeroplane. A multitude of data is additionally passed to the Horizontal Situation Indicator (HSI), which either may be an electro-mechanical instrument on its own, or alternatively may form part of an Electronic Flight Information System (EFIS) display.

The system normally consists of two independent IRSs, which can operate on either AC or DC power. If AC power is not normal, the systems automatically switch to backup DC power from the battery busbar. Backup power to the right IRS automatically terminates if AC power is not restored within 5 minutes.

IRS TRANSFER SWITCH

If either IRS fails, the IRS transfer switch is used to switch all associated systems to the functioning IRS.

IRS ALIGNMENT

Carry out the alignment of the IRS on the ground as follows:

NORMAL ALIGNMENT

Like the gyro stabilised INS, it is necessary to do the alignment of the IRS on the ground and initialise the system with the current aeroplane position, before it can enter the NAV mode. The position is normally entered on the POS INIT page of the FMC CDU during the alignment period, which like the INS must be carried out whilst the aeroplane is stationary. The IRS display unit may be also used to enter the necessary data.

Alignment between latitudes 70°12'N and 70°12'S is normally initiated by rotating the IRS mode selector switch from the 'OFF' position, directly to the 'NAV' position. The IRS performs a short DC power test, during which the 'ON DC' light illuminates, and when it extinguishes the 'ALIGN' light illuminates, indicating that the alignment process has begun. The aeroplane's present position should be entered at this time via the FMC CDU, and after approximately 10 minutes the IRS automatically enters the 'NAV' mode, at which time the 'ALIGN' light goes out.

At latitudes between 70°12' and 78°15' the mode selector switch must be left in the 'ALIGN' position for 12 minutes, and then manually rotated to the 'NAV' position, at which time the IRS immediately enters the 'NAV' mode.

FAST ALIGNMENT

During transit or through flight stop-overs, with only short ground hold over times, a 30 second realignment and zeroing of the ground speed error may be selected by selecting 'ALIGN' from 'NAV', whilst the aeroplane is parked. The present position is then simultaneously updated by manually entering the current latitude and longitude prior to reselecting the 'NAV' mode.

LOSS OF ALIGNMENT IN FLIGHT

If the alignment of the IRS is lost in flight due to the loss of DC or AC, or the mode selector switch is moved out of the NAV mode detent, the position and ground speed outputs are inoperative for the remainder of the flight. If the selector switch is rotated to the 'ATT' position, it will allow the attitude mode to be used to re-level the system, and provide attitude indications on the Attitude Director Indicator (ADI). In steady level flight (SLF), the levelling process takes approximately 30 seconds to complete. The attitude mode may also provide heading information, but in order to establish compass synchronisation the crew must first manually enter the initial magnetic heading. All heading information is invalid, and heading flags come into view until the actual magnetic heading is entered into the system. Thereafter the IRS is subject to drift (15.04° sin latitude), so when operating in this mode it is important to periodically cross check and update the magnetic heading in the IRS against the operating compass system as required.

Chapter 21
Radio Altimeter

INTRODUCTION

The Radio Altimeter (RA) is a device, which accurately measures the height above the surface immediately below an aeroplane up to 2500 ft, and is particularly suited to low-altitude terrain clearance measurement. It provides an instantaneous and continuous readout on the flight deck of the height above water, mountains, buildings, or other objects on the surface of the Earth, but gives no information regarding high ground immediately ahead of the aeroplane. This information is also supplied to the:

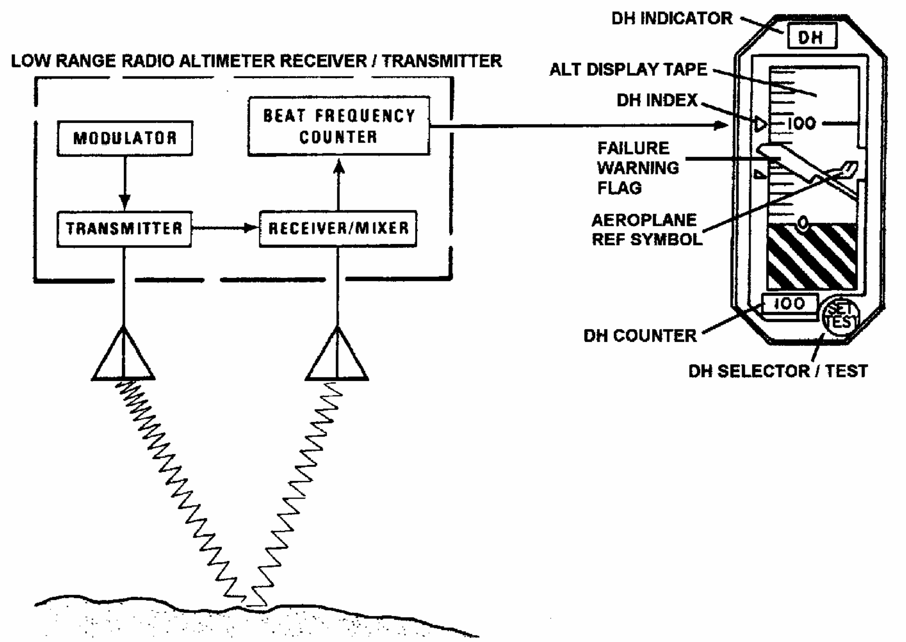
- Automatic Flight Control System (AFCS) to facilitate automatic landings using the Instrument Landing System (ILS)
- Ground Proximity Warning System (GPWS) to provide height, and rate of change of height information

The outputs from the Radio Altimeter can directly, or via a data bus, feed the Electronic Flight Instrument System (EFIS) and the Flight Management Computer (FMC).

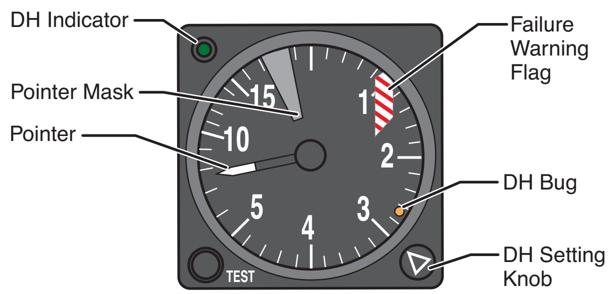
Importantly, the height measured by the Radio Altimeter is absolute, so flight over undulating terrain results in sympathetic variations in the indications of the height of the aeroplane on the display.

THE RADIO ALTIMETER SYSTEM

A Radio Altimeter determines the time taken for a radio wave to travel from the aeroplane to the ground directly beneath the aeroplane and back again. The system consists of a transmitter/receiver, a modulator, an integral timing or beat frequency counter, a transmitter aerial, a receiver aerial and a display as shown below.



Some Radio Altimeter systems alternatively use a mechanical circular display, as shown below, where the height displays linearly up to 500 ft and logarithmically from 500 - 2500 ft, making the lower range of altitudes easier to read more accurately.



In this type of display, the maximum height (2500 ft) is obvious, but it is not so apparent when using a moving vertical scale presentation, as shown on the previous page.

Notably, all radio altimeter displays have a method of setting the decision height, which is normally set at 100 ft, and a flashing DH indicator light is given when reaching this point. The required height is set using a decision height (DH) setting knob, and a bug or index marker indicates the set height. The setting control knob on some systems also normally doubles up as a press-to-test (PTT) facility, which when engaged, drives the display to a predetermined value, which is typically 100 ft.

With reference to the upper display, an ‘OFF’ or ‘FAIL’ flag is visible if:

- There is a power failure.
- The returning signal is too weak.
- Local reflections are received from the airframe itself.

A mask also covers the height pointer if:

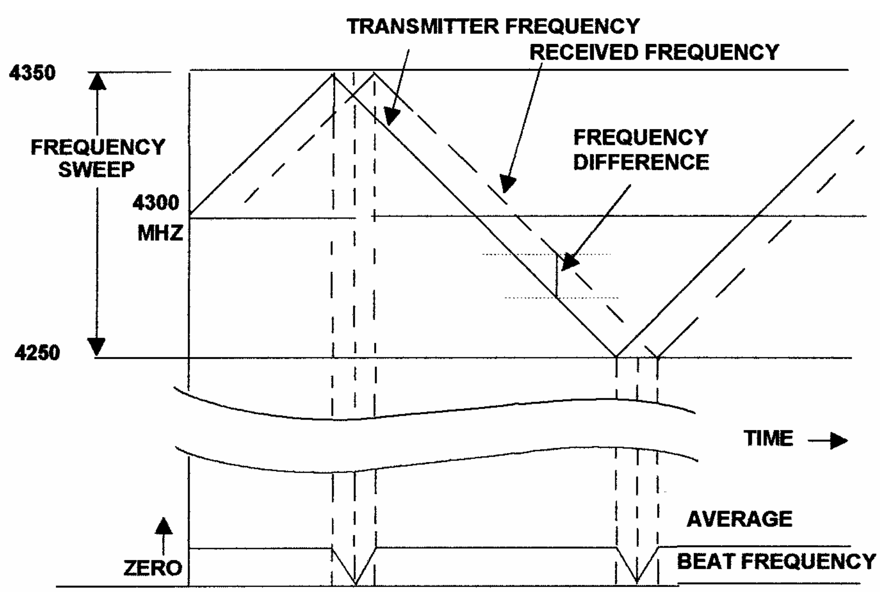
- The equipment is switched off.
- There is a fault in the transmitted signal.
- The altitude exceeds 2500 ft.

The Decision Height (DH) light flashes continuously if the aeroplane goes below the set height and remains so until the aeroplane climbs, or until setting the DH at a lower value. At approximately 50 ft above the set decision height, an audible alert sounds with increasing loudness until reaching the actual decision height.

PRINCIPLE OF OPERATION OF A RADIO ALTIMETER

A Radio Altimeter measures the time taken for a radio wave to travel from the aeroplane to the surface directly beneath and back again, and provided that the path followed by the wave is vertical, the total elapsed time is a function of the aeroplane's height. During this time, the transmitted frequency changes, and the equipment measures the difference between the transmitted and received signals. The frequency change is a measure of the time taken for the radio wave to travel to and from the surface and thus, the greater the frequency change the greater the height. To achieve this, the Radio Altimeter system makes use of primary radar principles and transmits a Frequency Modulated Continuous Wave (FMCW), at a frequency of 4250 MHz to 4350 MHz, which is in the Super High Frequency (SHF), or Centimetric wavelength band.

A complete modulation cycle or frequency sweep is illustrated on the next page. In this system, the total sweep of the modulated or carrier frequency is automatically varied by $\pm$ 50 MHz approximately 300 times per second, from an initial datum of 4300 MHz.



Throughout the cycle, there are two very short periods when the modulation changes from positive to negative and vice-versa. The frequency difference, which occurs when the transmitter alters the direction of its frequency sweep, is overcome by relating the aeroplane height directly to the average beat frequency (i.e. the difference between the transmitted and received frequency, observed over a short sampling period). The frequency changeover points can essentially be

ignored in the height calculation, so that the difference in frequency is directly proportional to the aeroplane's height.

At low altitudes, the reflected radio wave returns almost instantaneously, which gives an erroneous height, so a wider sweep is necessary to provide a measurable frequency difference. In order to overcome this ambiguity, the sweep rate is lowered (i.e. the time for a complete frequency sweep is made longer, so that all normal heights within the normal operating range of the radio altimeter are covered).

PERFORMANCE AND ACCURACY OF A RADIO ALTIMETER

The accuracy of the radio altimeter is normally:

- 0 - 500 ft: ±3 ft or 3% of the height, whichever is the greater.
- Above 500 ft: 5% of the height.

When the aeroplane is on the ground, the Radio Altimeter may show a small negative value, since the equipment is normally calibrated to indicate zero when the main wheels first contact the runway surface on landing. This effect is particularly noticeable on aeroplanes with multi-wheel undercarriage assemblies, which are inclined at an upward angle when deployed in flight.

ERRORS ASSOCIATED WITH A RADIO ALTIMETER

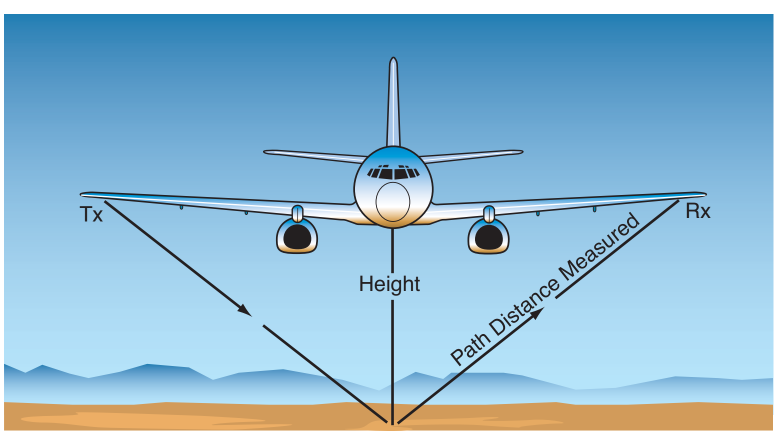
A Radio Altimeter may be susceptible to the following errors:

LEAKAGE ERROR

This may occur if the Transmitter (Tx) and Receiver (Rx) antennae on the underside of the aeroplane are fitted too close together (i.e. the spilling through of the side-lobes directly into the Rx antenna). Placing the antennae far enough apart to avoid any interference, also provides adequate screening.

MUSHING ERROR

This may occur if the antennae are placed too far apart. As the aeroplane comes close to the ground, the Tx antenna, reflection point, and Rx antenna form a triangle, so that the actual distance travelled by the wave can become greater than twice the vertical height between the surface and the aeroplane, thus giving a false height indication, as illustrated below.



THE ADVANTAGES OF A RADIO ALTIMETER

Radio altimeters have the following advantages:

- They indicate the actual (absolute) height of an aeroplane.
- They provide an easy crosscheck with the barometric altimeter for terrain clearance.
- They provide an aural warning signal prior to reaching the preset DH, and a visual warning when reaching the DH.

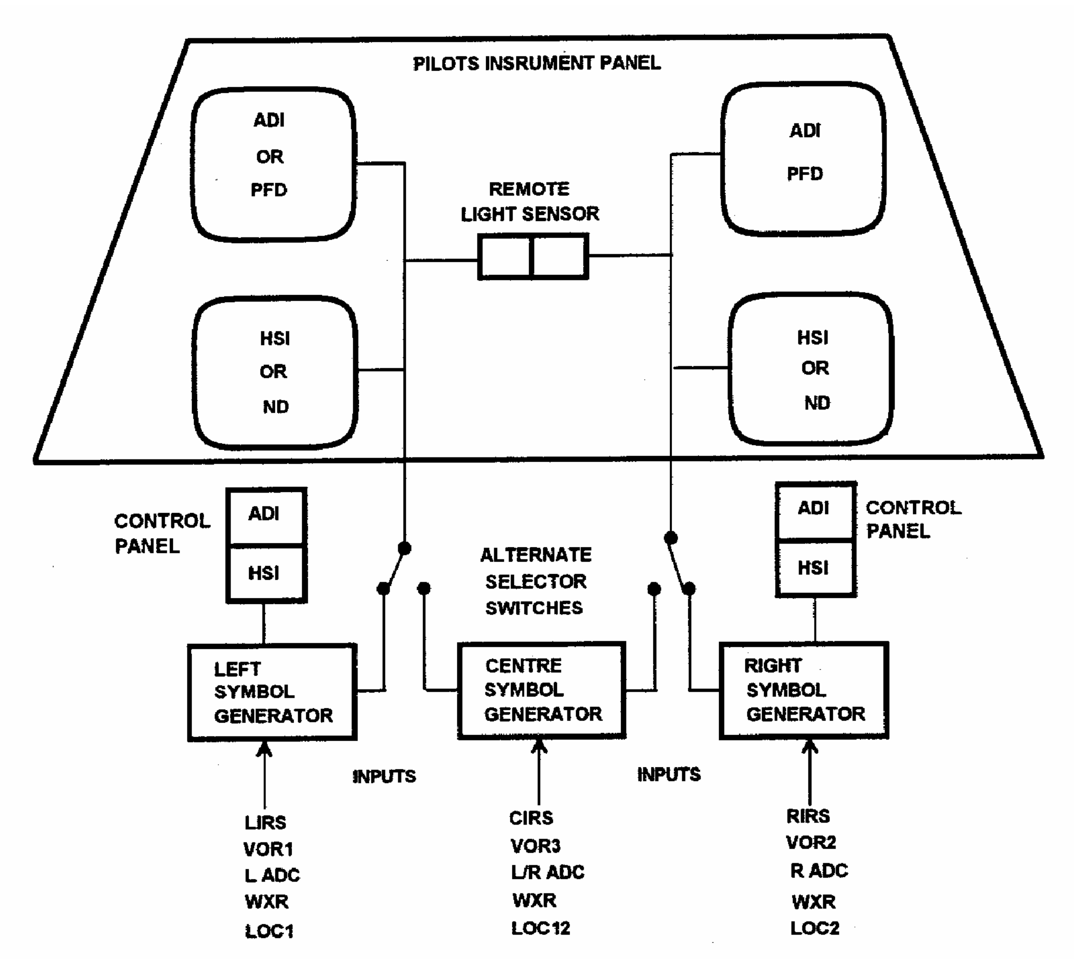
Chapter 22
Electronic Flight Instrument System

INTRODUCTION

The Electronic Flight Instrument System (EFIS) is a highly sophisticated Flight Director System (FDS), which uses cathode ray tube (CRT) or light emitting diode (LED) technology to provide attitude and navigation information. This system replaces the electro-mechanical type of instruments and provides the necessary high reliability for safe operations. In this system, all of the information is integrated into a single presentation and is placed in the flight crews preferred line of sight. The information, clearly presented using colour symbols, is easily understood. Selecting relevant information is easy without having to scan a large instrument panel.

EFIS ARCHITECTURE

Shown below is a basic EFIS layout:



The system comprises of:

2 x Attitude Director Indicators (ADI)
2 x Horizontal Situation Indicators (HSI)
3 x Symbol Generators (left, right and centre)
2 x Mode Control Panels
2 x Light Sensors

The Attitude Director Indicator, or the Primary Flight Display (PFD), and the Horizontal Situation Indicator as the Navigation Display (ND). The EFIS is capable of interfacing with multiple avionics, radio, and navigation (long and short-range) equipment, the Thrust Management Control System, and the Automatic Flight Control System (AFCS).

Inter-linked with this system is the Flight Management System (FMS), which provides flight progress, and MAP displays. It is additionally linked to the Inertial Reference System (IRS) to provide attitude and heading data, and the symbol generator to process, and display data.

SYMBOL GENERATOR (SG)

These components are a central part of the EFIS, and receive inputs from various aeroplane sensors and avionic equipments. The data is then processed and converted into suitable data for presentation on the ADI/ HSI. When powered, the symbol generators provide appropriate displays on these instruments, but the displays can be interrupted if the following faults occur:

- the screen goes blank if a power failure, over temperature, or failure of the relevant symbol generator occurs
- a partial loss of colour capability may cause an odd colour presentation, which may be due to an over-temperature
- when information is unreliable or the received radio signals are not received, the display will disappear
- if the aeroplane equipment fails, a failure flag is displayed

INSTRUMENT COMPARATOR UNIT (ICU)

This component detects data faults associated with the ADI/ HSI and constantly monitors the following cross-cockpit displays for disagreement:

- Attitude (pitch and roll)
- Heading
- Track

If conditions are detected outside set parameters, the appropriate Master Warning light illuminates, and an aural signal sounds. Instrument comparison monitoring is inhibited when both pilots are using the centre symbol generator.

COMPRESSION MODE

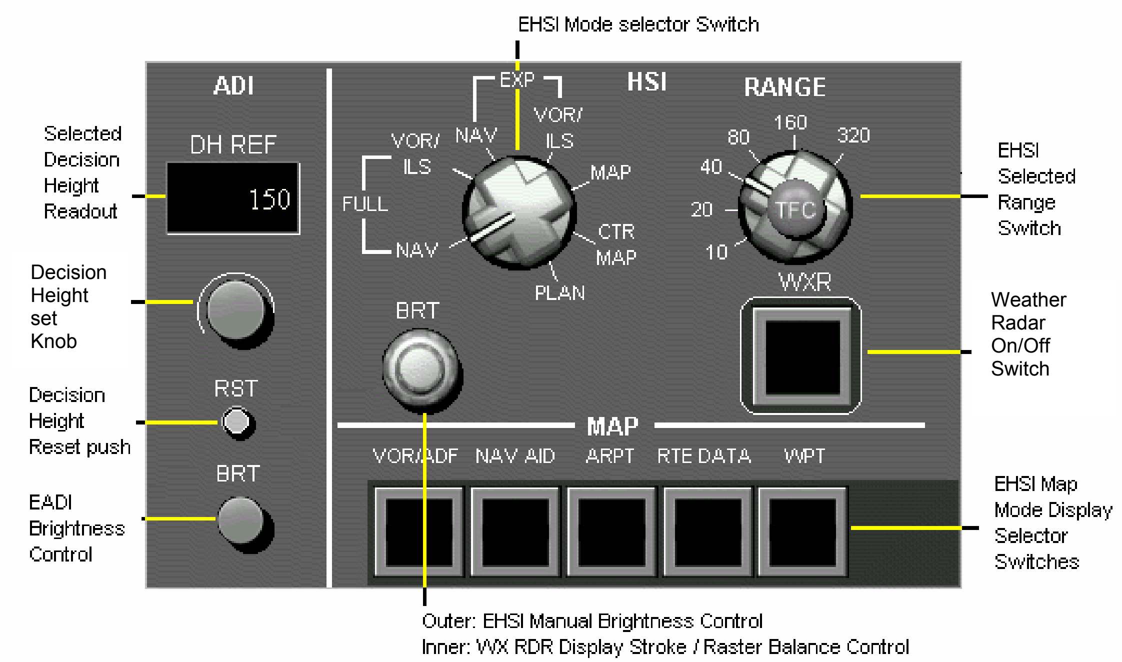
On most EFIS, if either ADI or HSI screen fails, modified information from the failed CRT will be automatically presented on the serviceable CRT in a compressed format.

TEMPERATURE SENSING UNITS

These units are fitted to each ADI and HSI, and are set to 'low' and 'high' values, which are spaced approximately 20°C apart. If the low value is exceeded on the ADI, the sky and ground shading switch off, thus alerting the pilot, but is automatically restored if the temperature returns to normal. If the temperature exceeds the high value, then the whole display switches off, but resets itself if it subsequently cools down.

MODE CONTROL PANELS (MCP)

The ADI and HSI mode control panels administer the symbology options, mode ranges, and brightness of the respective ADI and HSI displays. This panel also allows the radio altimeter decision height (DH) to be selected, as shown below:



LIGHT SENSORS

These are fitted in close proximity to the displays and ensure that the light intensity of the displays selected by the flight crew is compatible with the ambient light conditions on the flight deck.

ATTITUDE DIRECTOR INDICATOR (ADI)

The ADI is normally the upper of a pair of cathode ray tubes, sited in the instrument panel in front of both pilots. The information displays on approximately 16cm square colour screens, and provides indications on the current:

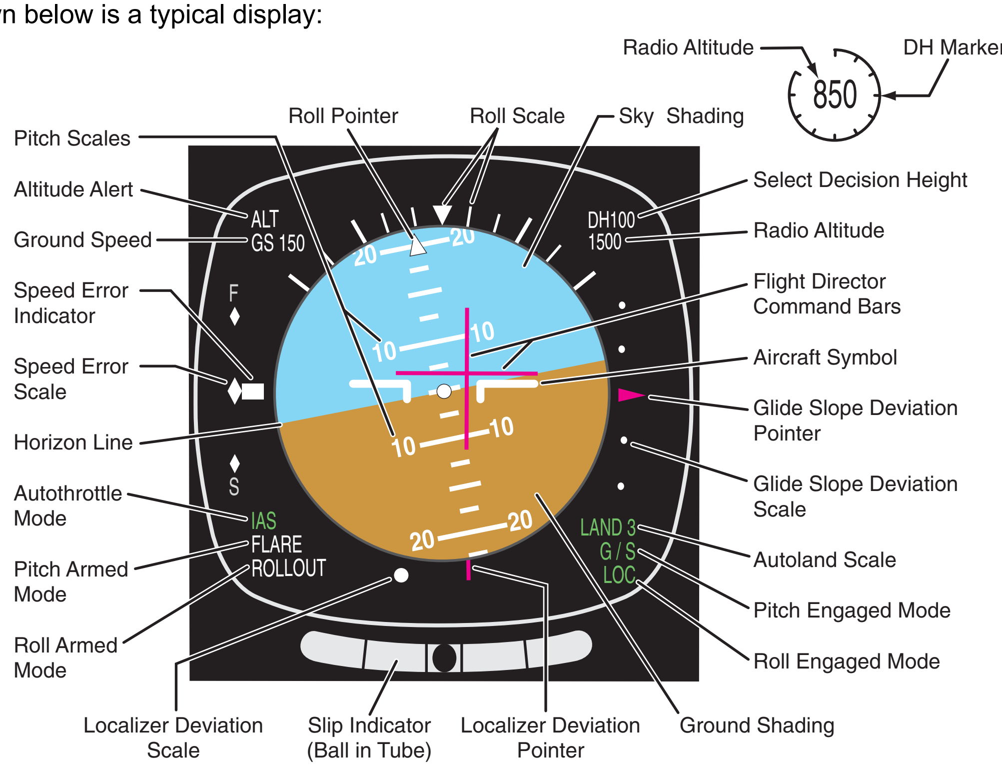
- Attitude (pitch and roll)
- Flight Director commands
- Localiser and Glide slope deviations
- Airspeed deviations

The ADI additionally displays:

- Auto-flight annunciations (e.g. auto-throttle)
- Ground speed
- Angle of attack
- Radio altitude
- Decision Height (DH)

The primary part of the ADI display is the aeroplane's attitude, which is supplied by the Inertial Reference System (IRS), when aligned in the navigation or attitude reference mode. Attitude data is however unavailable during the pre-flight checks, and whenever the 'ATT' flag appears.

Individual generators contained within the symbol generators produce the individual displays via a scanning process. The coloured background is produced by **raster scanning**, which is the method by which electron beams travel back and forth across the screen to form a solid image. The sky shading is cyan (light blue) in colour, whilst the ground shading is brown, and all other symbols/characters are produced by a digital stroking technique known as **stroke scanning**. Shown below is a typical display:



Flight Director (FD) commands are presented on the ADI by conventional command bars or in some systems by V-bars. A failure in either axis causes the respective command bar to disappear, but if both axes are unreliable, both bars disappear, and a FD flag appears. The FD bars do not display until the IRS properly aligns. The auto-flight modes are usually presented in the select mode in white, and in the acquisition mode in green. Additionally, if the normal data sources for the ADI are not available, access alternate data via an Instrument Source Selector Panel. Displayed on the ADI is the following data:

RADIO ALTITUDE

Above 2500 ft, the radio altitude display remains blank, but between 2500 ft and 1000 ft, the radio altitude digitally displays on the ADI. A white analogue ring scale and digital readout will replace the sole digital display when the aeroplane is below 1000 ft. Calibrated in 100 ft segments, this ring steadily disappears with reducing altitude.

DECISION HEIGHT (DH)

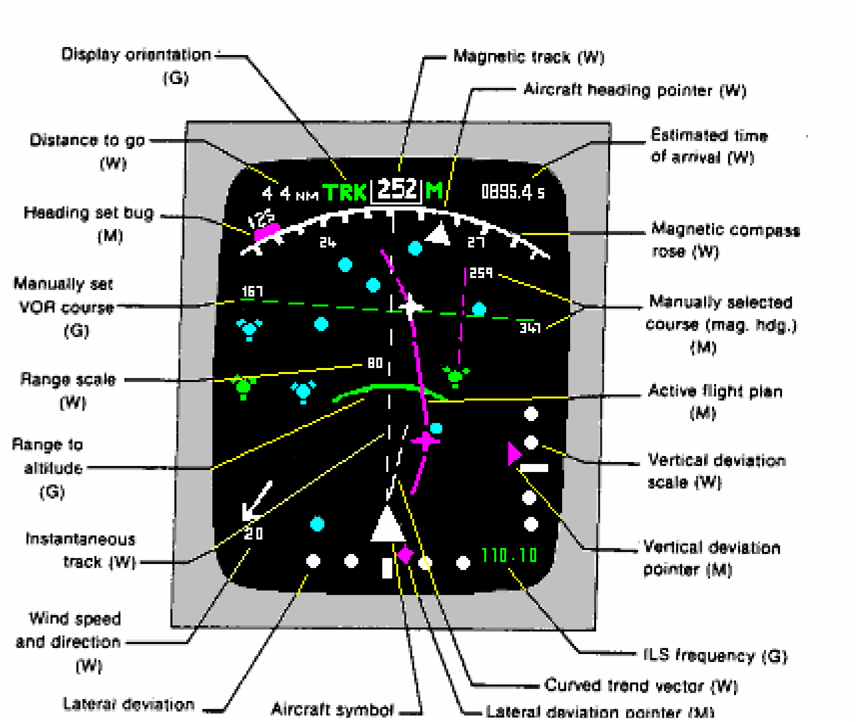
Normally displayed digitally, but from 1000 ft to touchdown the selected DH displays as a magenta triangle on the circular radio altitude display. During the descent, when the aeroplane is 50 ft above the selected DH, an aural alert chime sounds at an increasing rate until reaching the DH. At this point, the ring and scale change from white to amber, and the DH marker changes from magenta to amber. The ring, scale, and DH marker flash for a couple of seconds. To cancel the flashing, push the reset button on the mode control panel, causing the scale and marker to return to their original colours.

LOCALISER AND GLIDE SLOPE INDICATION

The ADI localiser deviation pointer sense reverses whenever the aeroplane's track is greater than 90° from the ILS forward course. This ensures that the ADI deviation pointer is usable on a back-course approach, and also retains the compatibility between the HSI and the ADI localiser deviation direction at all times. ILS Deviation Monitoring also alerts the flight crew of any ILS deviations during an autopilot or flight director approach, when flying below 500 ft AGL. With the APP (Approach) mode selected on the MCP, if the glide slope deviation is greater than one dot per second, or the localiser deviation is greater than one fifth of a dot per second, the respective localiser or glide slope scale automatically changes colour, from white to amber, and the pointer flashes. This alert condition ceases when the localiser and/or glide slope parameters return to within their normal limits.

THE HORIZONTAL SITUATION INDICATOR (HSI)

The HSI is a cathode ray tube (CRT), usually fitted beneath the ADI, and presents plan view orientation navigational information to the flight crew. It also presents a selectable, dynamic colour display of the flight's progress. The display modes include MAP, PLAN, ILS, and VOR. Heading data is forwarded to the HSIs by respective IRSs (CAPT HSI - L IRS, F/O HSI - R IRS), whilst the centre IRS is available as an alternate source. Shown below is a typical normal HSI display.



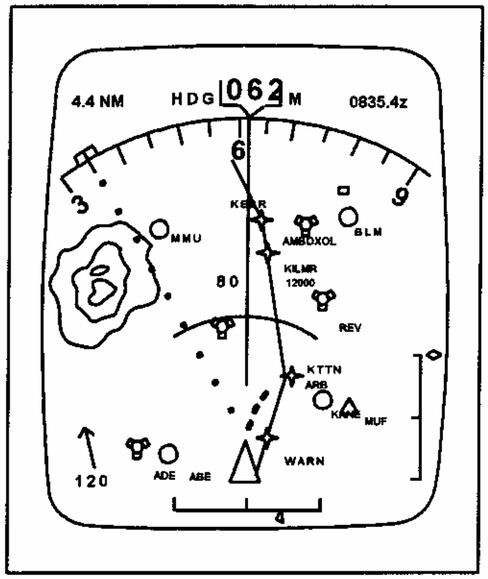
The HSI compass rose, automatically references to magnetic north when operating between 73°N and 60°S latitudes, with the NORM/TRUE switch selected to NORM, and to true north when operating outside these latitudes. The compass rose may be also be referenced to true north by manually selecting the NORM/TRUE switch to TRUE, regardless of the latitude.

TRU displays at the top of the HSI, and is enclosed by a white box if the HSI is referenced to true north. When the HSI is referenced to true north and the aeroplane descends 2000 ft at more than 800 fpm, the box changes from white to amber. The box then continues to flash for 10 seconds, and thereafter remains amber. The box returns to white if the aeroplane climbs 2000 ft at more than 500 fpm. A green box displays around the M for 10 seconds when the HSI is returned to magnetic referencing.

In the MAP mode, shown above, the HSI presents the following information against a moving map background:

- Heading
- Routes
- Curved trend vector
- Range to altitude
- Wind
- Distance
- Estimated time of arrival

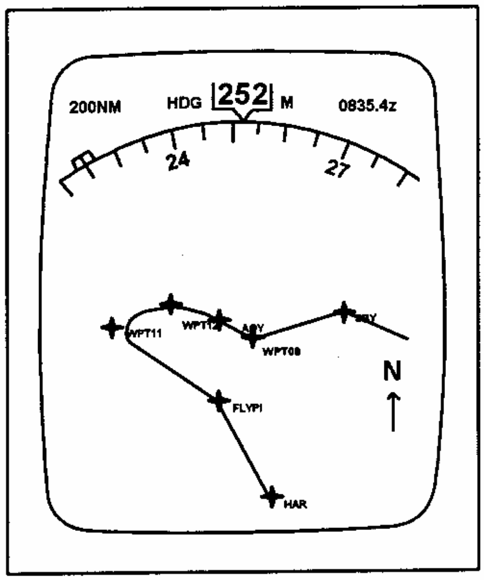
The respective Flight Management Computer (FMC) supplies track information, whilst the opposite FMC is available as an alternate source. If the information from the FMC is unreliable, an IRS automatically provides the necessary information. Selected navigation data points are also programmed into the FMC, so it is important to be familiar with the colours and Symbology used. Purely for reference, a comprehensive listing is provided at the end of this chapter, although these symbols may vary from manufacturer to manufacturer.



The recommended display mode for most phases of flight is the MAP mode, as shown above. Other available modes are shown on the following pages.

PLAN MODE

This display is presented on the bottom 2/3 of the HSI against a static map background, and shows the active route data oriented to true North.

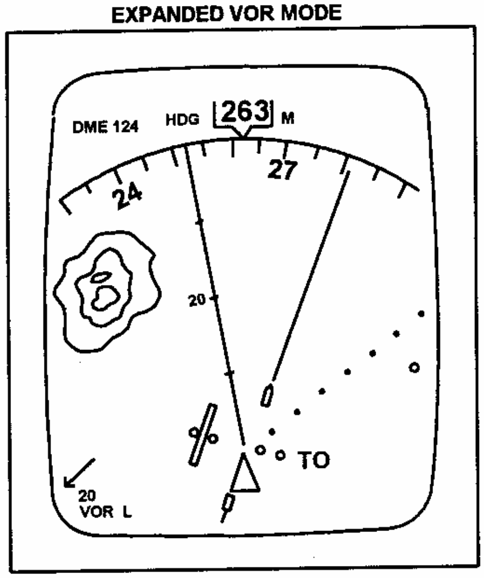


The top part of the HSI maintains a display of track and heading information as in the MAP mode. Sequencing through the Flight Management Computer Control Display Unit (FMC CDU) allows the pilot to view the active route.

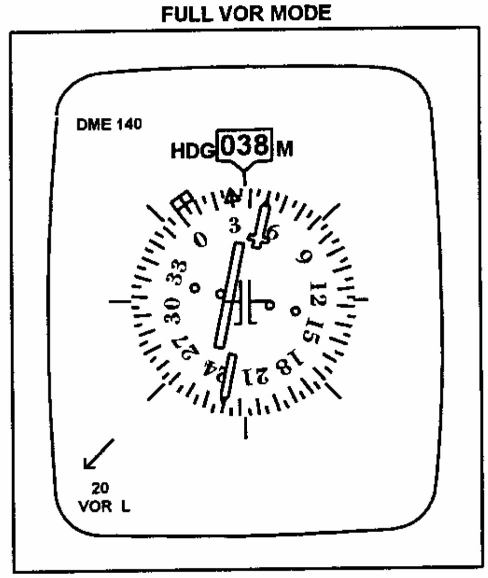
VOR AND ILS MODES

When these modes, as shown below, are selected, the HSI presents expanded track scale and heading orientation. Wind information and system source annunciation is provided with conventional VOR/ILS navigation information. Conventional full compass rose VOR and ILS modes are also available.

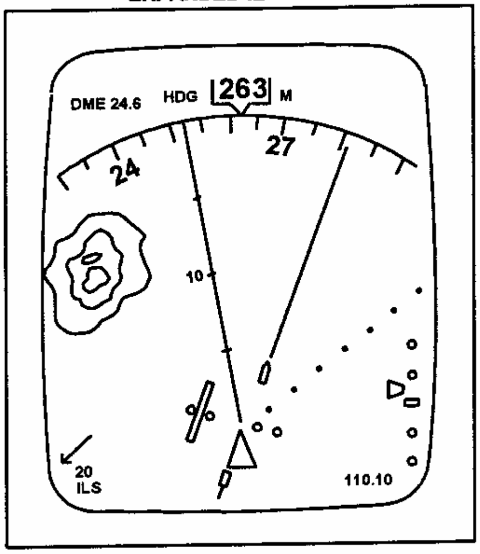
EXPANDED VOR MODE



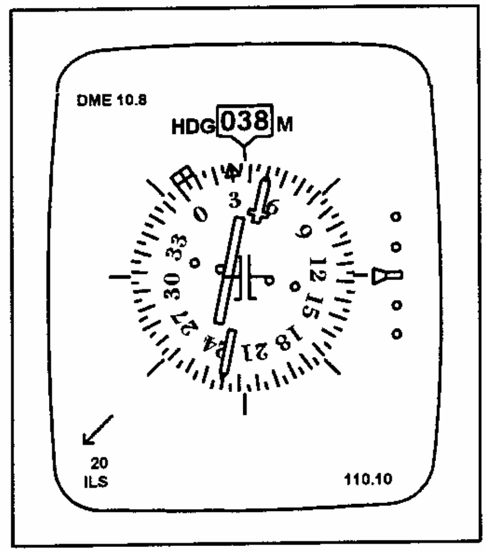
FULL VOR MODE



EXPANDED ILS MODE



FULL ILS MODE



The HSI displays weather radar data in cyan, green, amber, and red. Red depicts areas that have:

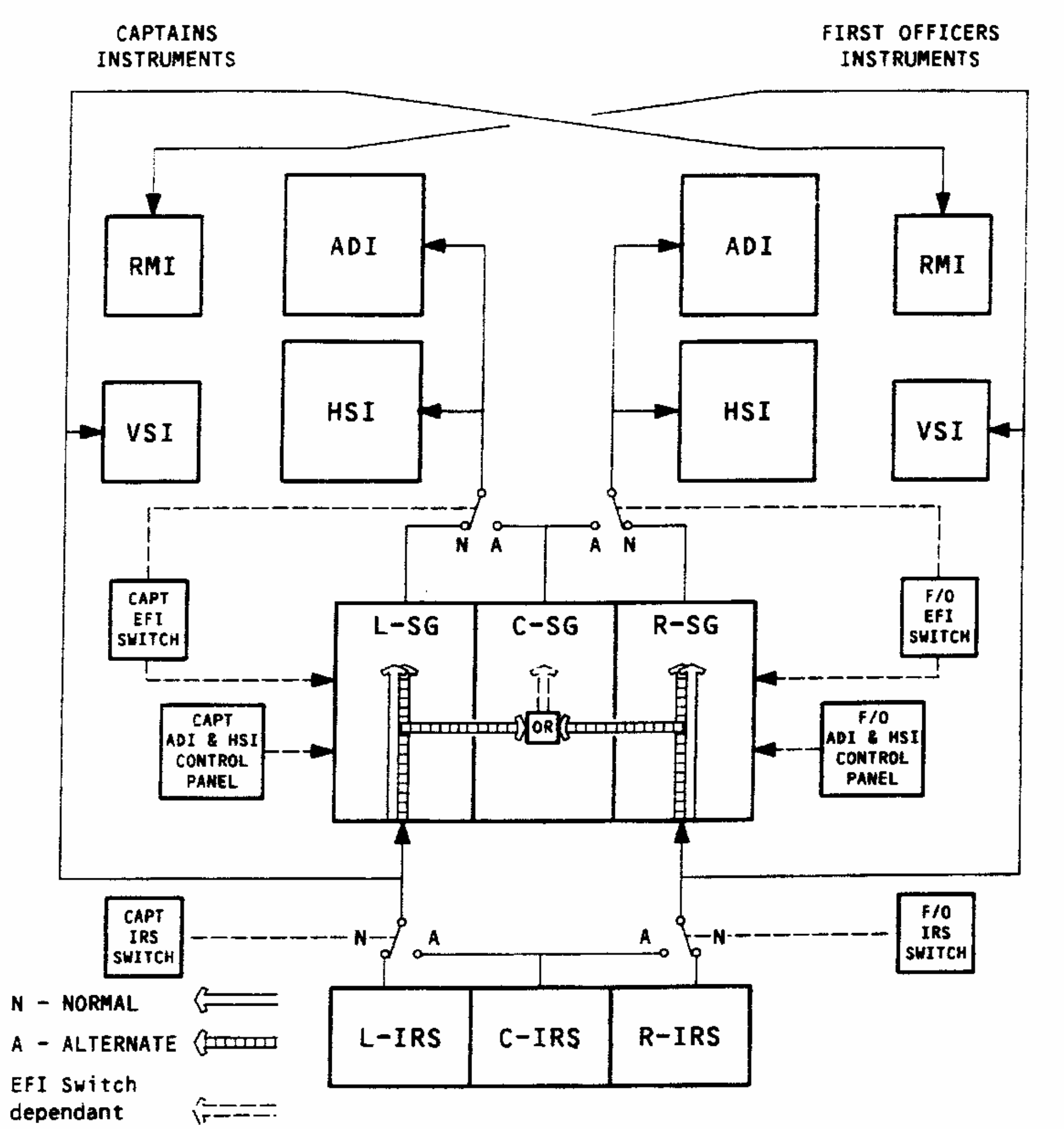
- Greatest return intensity, and thus the highest risk area for intense turbulence
- Reductions in the return intensity are indicated by a change of colour from red to amber, and then to green
- Cyan and amber are for use with message displays
- Weather radar data can only be displayed when the system is switched on and the respective HSI is in the expanded VOR, expanded ILS, or MAP mode

Note. When Weather radar (Wx) displays in the VOR or ILS mode, the scale shown applies only to the Wx display, and not to the deviation display.

Like attitude, heading/track data is unavailable until the associated IRS has completed its alignment and has entered the navigation mode. HDG (heading) flags do not appear in this case. In addition to previously mentioned EFIS failure indications, other discrepancy messages can be displayed on the HSI. For example WXR/MAP RANGE DISAGREE, indicates both Flight Management Computers, and the Weather Radar range, disagree with the current control panel range data.

EFIS/IRS INTERFACE

Shown below is a block schematic of a typical EFIS/IRS.



Electronic Flight Instrument (EFI) switching determines the Centre Symbol Generator (C-SG) inputs and outputs. If the left EFIS switch is selected to Alternate (ALTN), the left system instrument sources supply the C-SG. When the First Officer (F/O) EFI switch is in the 'ALTN' position, the right system instrument sources supply the C-SG. If both flight crew select 'ALTN' with their respective EFI switches, the left system instrument sources supply data to the C-SG. The C-SG always uses the centre ILS and centre radio altimeter. Each ADI and HSI Control Panel connects to the symbol generator via the EFI switch. Additionally each IRS switch permits flight crew selection of the alternate data source for heading/track, attitude, and vertical reference data.

HEADING REFERENCE SWITCH

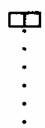
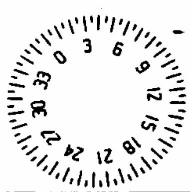
This switch is normally sited on the centre instrument panel and permits the selection of a magnetic or true heading reference for each Horizontal Situation Indicator (HSI), the Radio Magnetic Indicator (RMI), the Flight Management Computer (FMC), the Auto-pilot Flight Director System (AFDS) and the Flight Control Computer (FCC).

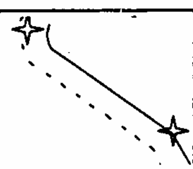
The AFDS uses true heading only when the switch is in TRUE, but if TRUE is selected when the AFDS is in the HDG SEL mode, it automatically changes to ‘HDG HOLD’. HDG SEL may then be re-selected.

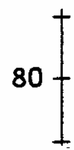
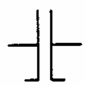
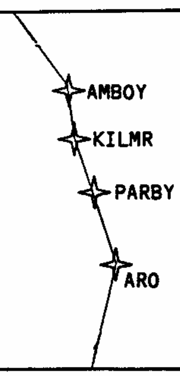
HSI SYMBOLOGY

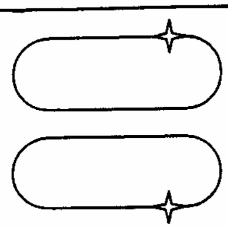
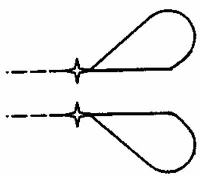
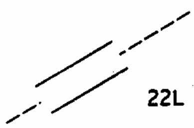
The following symbols can be displayed on each HSI/ ND depending on the switch selection on the EFIS control panel. Symbols can be displayed with different colours but the general colour presentations are as follows:

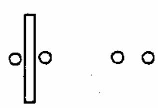
GREEN (G)	engaged flight mode displays, dynamic conditions.
WHITE (W)	present status situation, scales, armed flight mode displays
MAGENTA (M) (pink)	command information, pointers, symbols, ‘fly-to’ condition
CYAN (C) (blue)	non-active and background information
RED (R)	**WARNING**
AMBER (A) or YELLOW (Y)	cautionary information, faults, flags
BLACK (B)	blank areas, ‘off’ condition

SYMBOL	NAME	APPLICABLE MODE(S)	REMARKS
200NM / 4.4NM OR DME 124 / DME 24.6	DISTANCE DISPLAY (W)	PLAN, MAP OR VOR, ILS	Distance is displayed to next FMC Waypoint (nm) or tuned navaid (DME).
HDG 263 M	HEADING - orientation (G), indicator (W) and reference (G)	MAP, PLAN VOR, ILS	Indicates number under pointer is a heading. Box displays actual heading. Compass if referenced to magnetic north when between 73°N and 60°S latitude and true north when above those latitudes.
0835.4z	ETA DISPLAY (M,W)	MAP, PLAN	Indicates ETA at active waypoint based on present groundspeed.
	Selected Heading Marker (M)	MAP, ILS, VOR, PLAN	Indicates the heading set in the MCP. A dotted line (M) extends from the marker to the airplane symbol for ease in tracking the marker when it is out of view (except PLAN mode)
 24 27	Expanded Compass Rose (W)	MAP, ILS, VOR, PLAN	Compass data is provided by the selected IRS. 360° are available but approximately 70° are displayed.
	FULL COMPASS ROSE (W)	FULL VOR, FULL ILS	Compass data is provided by the selected IRS.

SYMBOL	NAME	APPLICABLE MODE(S)	REMARKS
	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 feet deviation.
	GLIDESLOPE POINTER (M) AND DEVIATION SCALE (W)	ILS	Displays glideslope position and deviation in ILS mode.
	DRIFT ANGLE POINTER (W)	FULL VOR, 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 oriented and referenced to true north.
	ALTITUDE PROFILE POINT AND IDENTIFIER (G)	MAP	Represents an FMC calculated point and is labeled on the flight plan path as, T/C (top-of-climb), T/D (top-of-desce-nt), and S/C (step climb).
	WEATHER RADAR RETURNS (G,A,R) MAPPING RADAR RETURNS (G,A,R)	MAP, VOR, ILS	Multicolored returns are presented when either WXR ON switch is pushed. Most intense areas are displayed in red. Lesser intensity amber, and lowest intensity green.

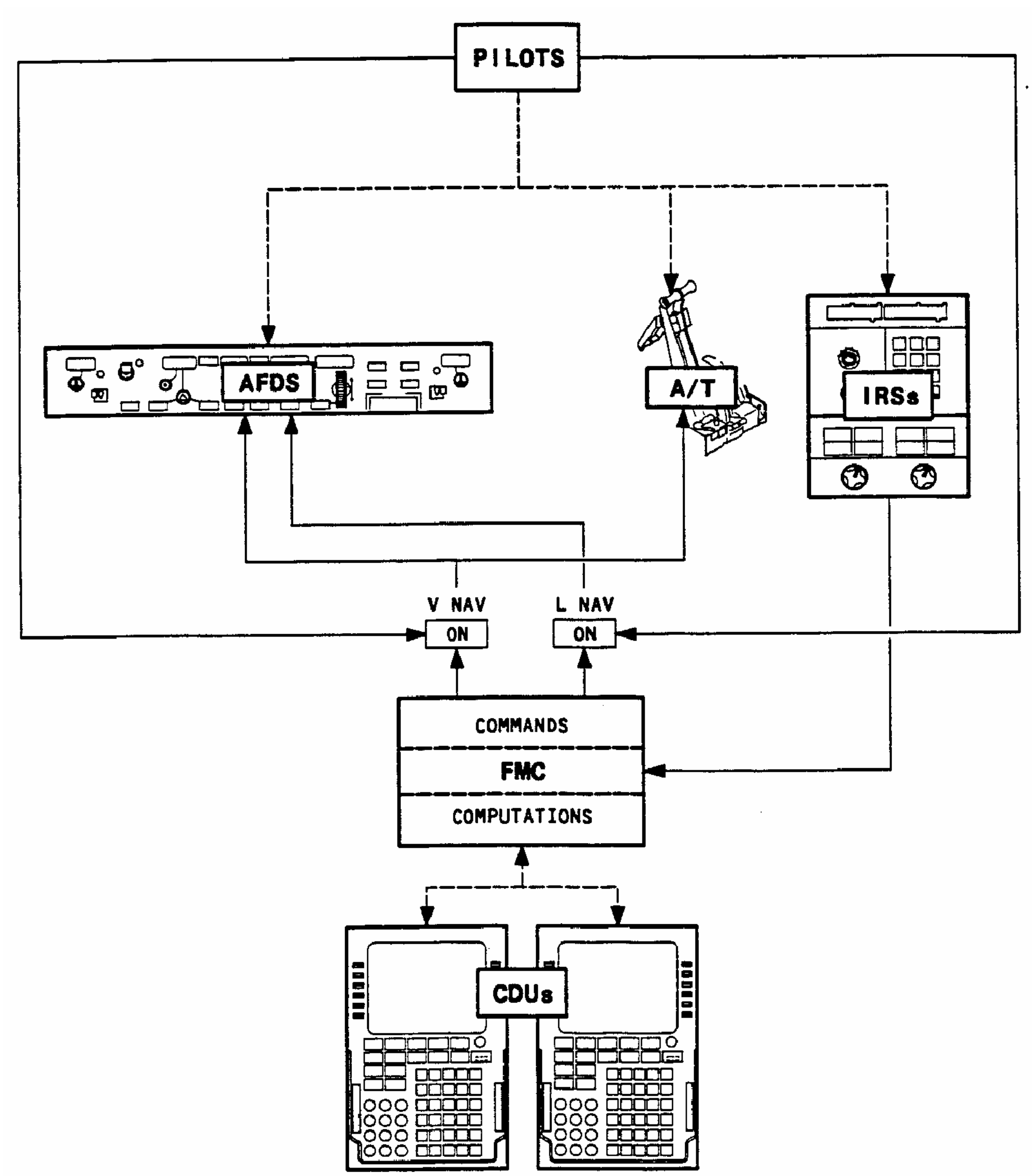
SYMBOL	NAME	APPLICABLE MODE(S)	REMARKS
	PRESENT TRACK LINE and (W) RANGE SCALE	MAP, ILS, VOR	Predicts ground track which will result with present heading and winds. Displayed range mark is one -half the actual selected range.
	AIRPLANE SYMBOL (W)	MAP, VOR, ILS	Represents the airplane and indicates its position at the apex of the triangle.
	AIRPLANE SYMBOL (W)	FULL VOR, FULL ILS	Represents the airplane and indicates its position at the center of the symbol.
	WAYPOINT: ACTIVE (M) INACTIVE (W)	MAP, PLAN	Active - Represents the waypoint the airplane 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 (W)	MAP	Predicts airplane directional trend at the end of 30, 60, and 90 second intervals. Based on bank angle and ground speed. 3 segments are displayed when selected range is greater than 20 nm. 2 segments are displayed on the 20 nm scale and one segment on 10nm scale.
	ACTIVE ROUTE (M) ACTIVE ROUTE MODIFICATIONS(W) INACTIVE ROUTES (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 with a continuous line. Inactive routes are displayed with long dashes (C) between waypoints.

SYMBOL	NAME	APPLICABLE MODE(S)	REMARKS
	HOLDING PATTERN (M)	MAP, PLAN	A fixed size holding pattern symbol appears when a hold entered in the RTE. The pattern increases to the correct size when the HSI range is 80 or less and the aircraft passes the way-point prior to the holding pattern.
	PROCEDURE TURN (M)	MAP, PLAN	A fixed size procedure turn symbol appears when entered in the RTE. The symbol increases to the correct size when HSI range is 80 or less and the air-plane passes the waypoint prior to the procedure turn.
	AIRPORT IDENT-IFIER AND RUNWAY (W)	MAP, PLAN	Appears when selected on FMC CDU. Available when HSI range is 80, 160 or 320 nm.
	AIRPORT AND RUNWAY (W)	MAP, PLAN	Appears when selected on FMC CDU. Available when HSI range is 10, 20 or 40 nm. Dashed line represents a 14.2 nm extended runway center line ending at threshold.
	SELECTED FIX CIRCLE (G) SYMBOL AND INDENTIFIER	MAP, PLAN	Presents a selected reference point (fix) via the FMC CDU FIX key. Can appear with any number of special map symbols (i.e. VOR, VORTAC, airport or waypoint, etc.) if contained in the existing data base.
	DOWN TRACK FIX (DNTKFX): conditional (W) active conditional (M)	MAP, PLAN	Defines the intersection of selected fix radials and present track. Appears with fix identifier and route data.

SYMBOL	NAME	APPLICABLE MODE(S)	REMARKS
	VOR (C) DME/TACAN (C) VORTAC (C)	MAP	When NAVAID switch is ON, all appropriate navaids in range appear in addition to those navaids which are standard or active waypoints and those which are tuned (G).
 KTEB	AIRPORT (C)	MAP	When the ARPT switch is ON, airports within the map area are displayed. Origin and destination airports are always displayed independant of ARPT switch.
 KILMR 12000 0835Z	ROUTE DATA (M,W)	MAP, PLAN	When the RTE DATA switch is ON, altitude and ETA for route waypoints can be displayed.
 MLF	OFF ROUTE WAYPOINT (C)	MAP	When WPT switch is ON, data base waypoints not on the selected route are displayed.
	COURSE INDICATOR (M) AND DEVIATION SCALE (W)	VOR, ILS FULL VOR, ILS	Displays ILS course when ILS mode is selected and valid signals are present. VOR course is displayed when VOR mode is selected and valid signals are present.
TO FROM	TO/FROM DISPLAY (W)	VOR	Display logic is the same as on non-EFI HSI's.
VOR L/R ILS	SOURCE NAV DATA (G)	VOR ILS	Displays source of nav radio data based on EFI control selection.
	SELECTED COURSE POINTER (W) AND LINE (M)	VOR, ILS	Displayed selected course as appropriately set by the VOR course selector or ILS Front Course Selector.
110.10	ILS FREQUENCY DISPLAY (G)	ILS	Appears when valid ILS frequency data is being received and the ILS Park signal is not being received.

INTRODUCTION

The Flight Management System (FMS), as shown below, is made up of the four primary systems; a Flight Management Computer System (FMCS), an Autopilot/Flight Director System (AFDS), an Autothrottle (A/T), and Inertial Reference Systems (IRS).



These components are all independent systems, and are operated either individually or in various combinations. Together these components provide continuous automatic navigation, flight guidance, and performance management. The FMS is capable of four-dimensional navigation (latitude, longitude, altitude and time), whilst optimising the aeroplane's performance, in order to achieve the most economical flight possible. It also provides centralised flight deck control of the aeroplanes flight path and performance parameters.

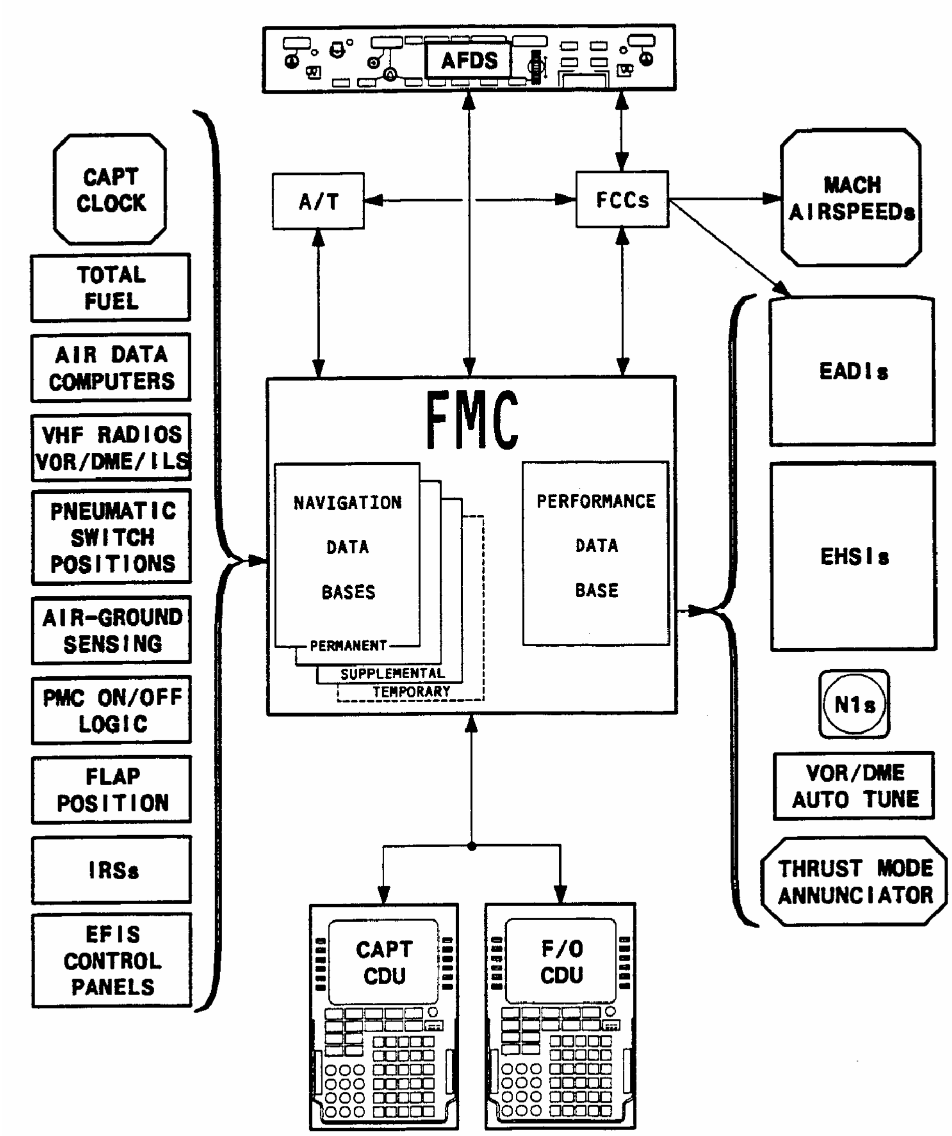
The primary function (flight management role) of the FMS is to provide performance management, navigation guidance, and automatic flight control. In this role, the FMS is additionally interfaced with the engine **Power Management Control (PMC)** and the **Automatic Flight Control System (AFCS)**. This isolates the flight crew from the control loop, and allows the FMS to act in a totally integrated fashion. The FMS thus provides optimum control over the aeroplane's engine power settings, and total control over its flight path.

The secondary function (advisory role) of the FMS is to provide inputs to the various flight deck displays in order to assist the flight crew in manually flying the aeroplane. For example, it provides the HSI map display (for orientation) and positions the bugs on the airspeed and EPR (N_1) indicators (to assist in manually flying precise flight profiles).

The FMS relieves the flight crew so that they can attend more closely to the tasks of monitoring and decision-making.

THE FLIGHT MANAGEMENT COMPUTER SYSTEM

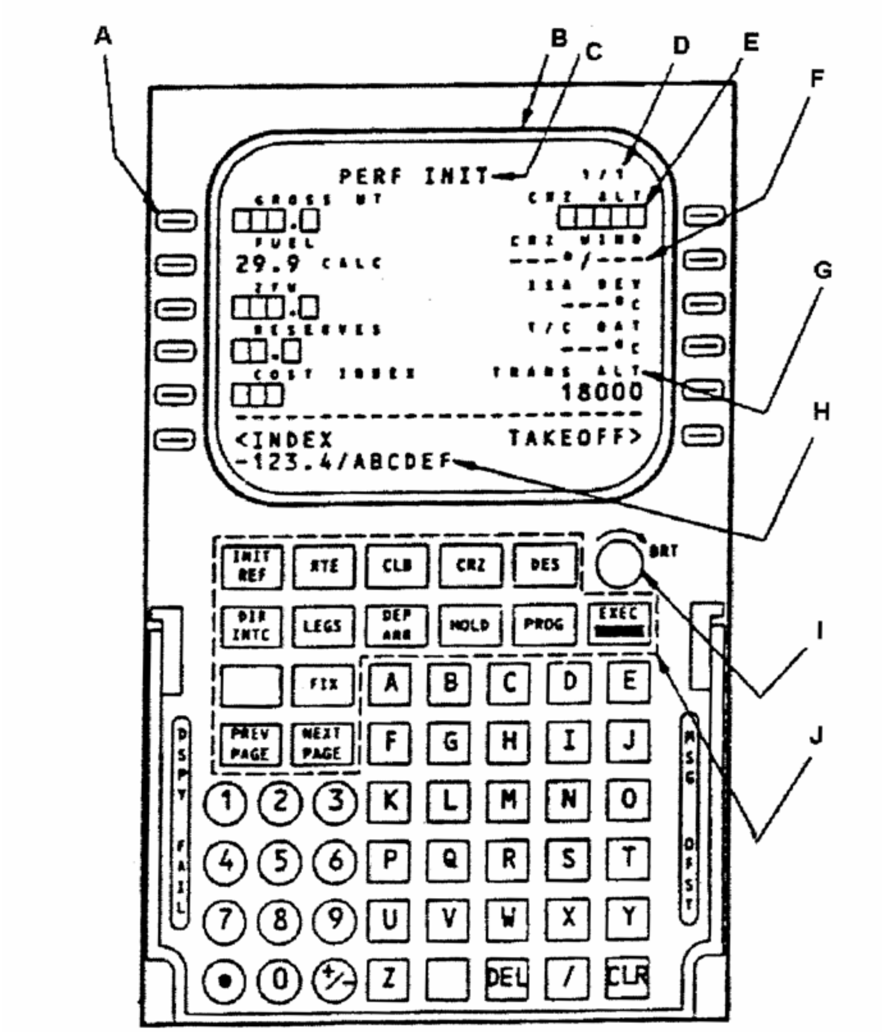
The Flight Management Computer System (FMCS) is the heart of a modern aeroplane's electronic systems, and gathers information from other subsystems.



The FMCS also allows the flight crew to manage the aeroplane's lateral (L-NAV) and vertical (V-NAV) flight path through a keyboard on the Command Display Unit (CDU). Computations relating to L-NAV include items such as courses flown, ETAs, and distances to go. Conversely, computations relating to V-NAV include items such as fuel-burn data, optimum speeds, and recommended altitudes. When operating in the Required Time of Arrival (RTA) mode, the computations include required speeds, take-off times, and enroute progress information. The FMC additionally integrates the information entered by the flight crew on the CDU and the information stored in its memory with information fed from external sources (e.g. navigation data, air data, and engine data). This information is used to calculate the present position of the aeroplane from which pitch, roll, and thrust commands can be derived, which in turn is used to optimise the aeroplane's flight profile. From this information, control and guidance commands are also fed to the AFCS and Autothrottle. This allows for an integrated FMS operation using both automatic lateral and vertical navigation, from the initial climb to the final approach. Advisory information is additionally forwarded to various flight deck displays (e.g. the EFIS and the Mach/Airspeed Indicators). The FMC thus reduces the need for the flight crew to make calculations, refer to maps, and read manuals. The flight crew must continue to monitor the FMC to ensure that it is following the planned route.

COMMAND DISPLAY UNIT

The Command Display Unit (CDU), as shown below, is the means by which the flight crew can communicate with the computer.



An alpha/numeric CRT screen dominates the upper part of the unit, while the lower part has a keyboard. Using the keyboard, the flight crew can enter the desired vertical and lateral flight plan data into the Flight Management Computer. The parts of the CDU and their individual functions are as follows:

A — Line Select Keys

(Six each side of screen) Push to select or enter data on the adjacent line.

B — CDU Display

Displays the page of data selected by function, mode, or Line Select key.

C — Page Title

Indicates the type of data displayed. When in lateral or vertical navigation modes, displays ACT or MOD, indicating the page status.

D — Page Number
The first digit represents the page number and the second digit indicates the total number of related pages.

E — Boxes
Indicates the data required by the FMC for full navigation capability. Data is entered from the scratch pad, using the Line Select key.

F — Dashes.
Indicate that a data entry is required by the system. If known, key in and transfer the data from the scratch pad using the appropriate Line Select key.

G — Line Title
Indicates the type of data on the line. This is blank if data is not recognised by the FMC.

H — Scratch Pad Line
This is the bottom line of the display and displays:

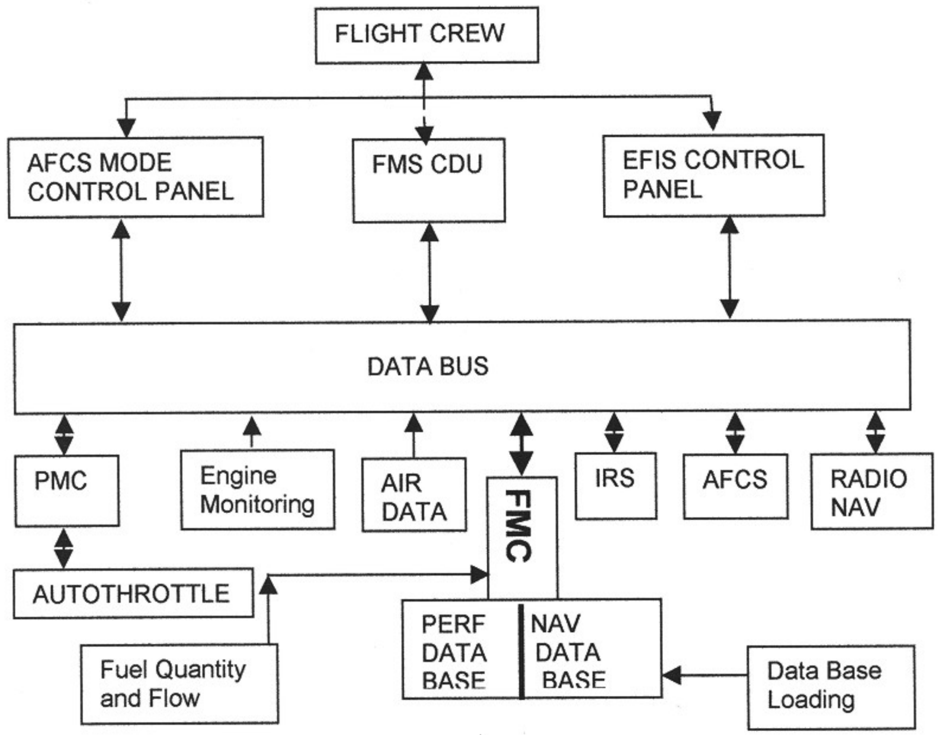
- System generated messages to the flight crew
- All keyboard entries (before transferring to the required line)
- Data moved from one line to the other

CONTROL PANELS

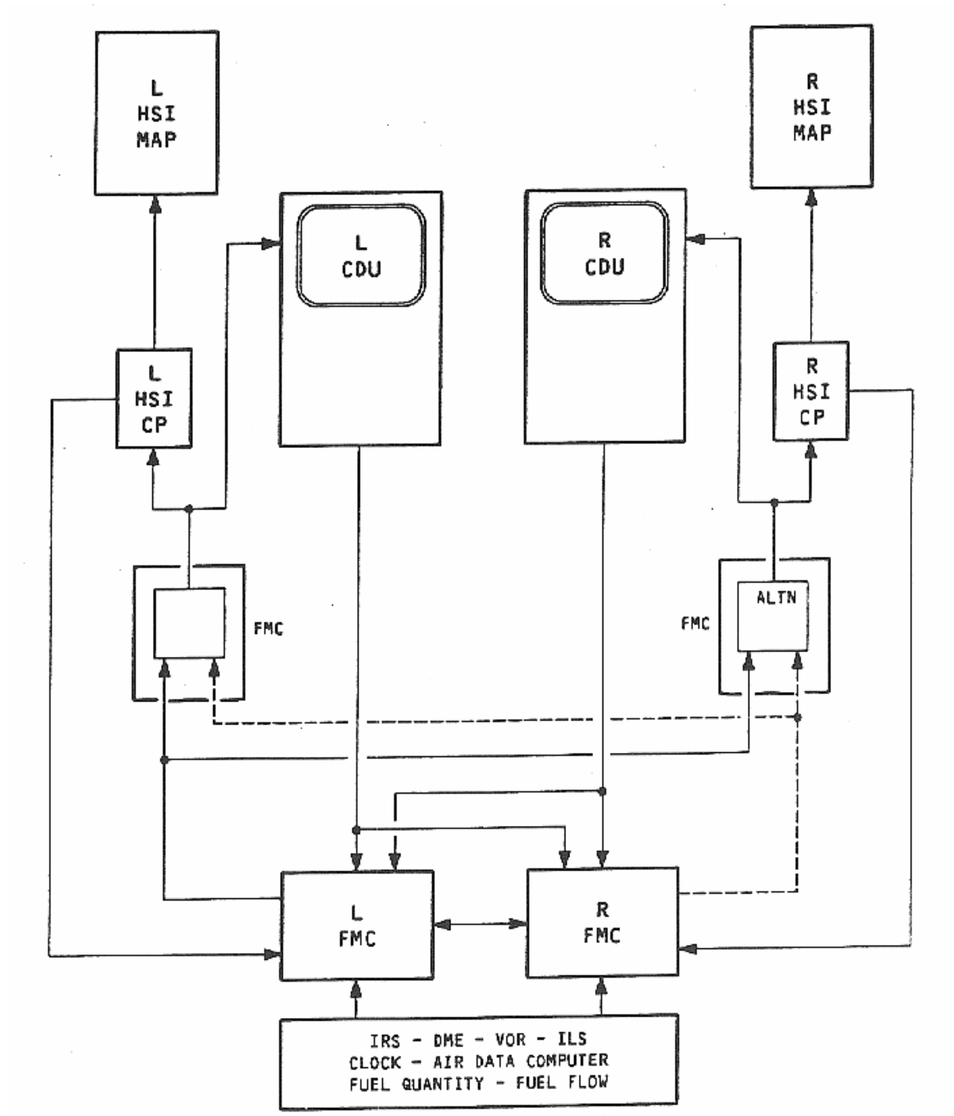
In a totally integrated system two further control panels exist, which are the:

- AFCS Mode Control Panel (MCP)
- EFIS Control Panel (ECP)

The diagram below illustrates how information flows to, within, and from the various FMS components.



Most modern aeroplanes are fitted with two FMS units, usually arranged as illustrated on the next page.



Both FMCs are cross-coupled, or matched as a dual system, so loading data into one CDU automatically feeds both FMCs. It is thus advisable to nominate one CDU for data loading while using the other to monitor or cross check the data loaded. In this arrangement, each CDU and EFIS map display will relate to its own computer (i.e. the left FMC feeds the left CDU and the left EFIS).

If the FMC switch is placed in the ALTN position, then the related CDU and EFIS displays connect to the other computer. This level of interconnection enables maintenance of all displays, provided at least one CDU and one FMC is operating.

CDU AND FMC TERMINOLOGY

Active

This refers to flight plan information, currently in use for calculating lateral (LNAV) or vertical (VNAV) guidance commands. For example, the active waypoint is the point the system is currently navigating toward; and the active performance (VNAV) mode is the climb, cruise, or descent speed schedule currently in use for pitch and thrust commands. In this mode, ACT displays in the associated page titles.

Activate

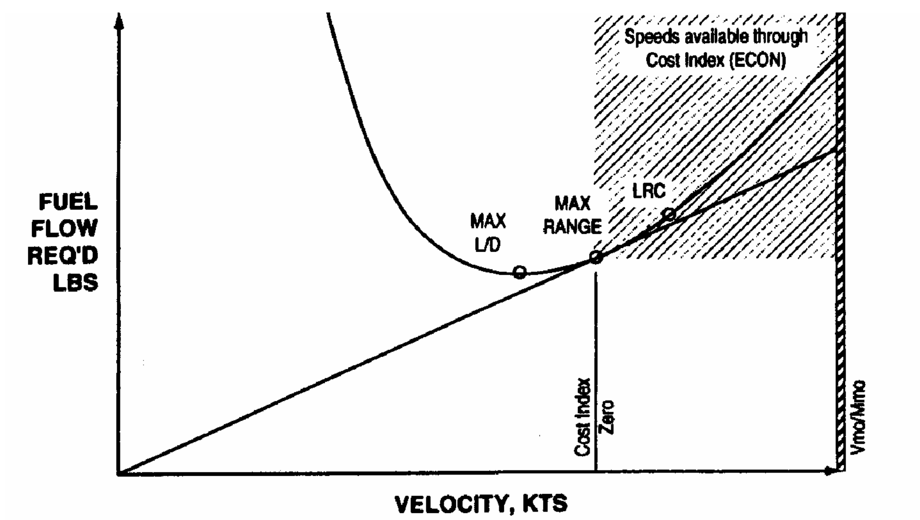
This refers to the process of designating one or two routes as active, and is a two-step process. First, push the ACTIVATE line select key for the desired routes, and second, push the illuminated EXEC key.

Altitude Constraint

This refers to a crossing restriction at a given waypoint.

Cost Index

This is a figure, determined by the operator, which is used to bias the computations for the speed schedule. It is based on a trade off to get the optimum balance between minimum fuel and least time. The index is determined by dividing the operating cost of the aeroplane by the cost of fuel (e.g. If the fuel costs are high, this number is low). A cost index of zero results in an economic speed (i.e. the aeroplane is flown at its maximum range speed), and the higher the Cost Index, the closer the aeroplane flies to V_{MO}/M_{MO}. Additionally, unless the Cost Index value has been arrived at in a scientific manner, it only provides a means of choosing appropriate climb, cruise and descent speeds.



Econ

This refers to a speed schedule, calculated to minimise the operating cost of the aeroplane, and is based on the ratio of time costs to fuel costs. The economy speed is based on a cost index that is entered into the FMC/CDU during pre-flight.

Enter

This refers to the process of typing or line selecting characters into the CDU scratch pad line, and then line selecting the desired location for the data.

Erase

This refers to removing modified data from the system by pushing the line select key adjacent to the word ERASE.

Execute

This refers to making entered data part of the active flight plan by pushing the illuminated EXEC key.

Inactive
This refers to route, climb cruise, or descent information not currently in use to calculate LNAV or VNAV commands.

Initialise
This refers to the process of entering data into the CDU that is required to make the system operative.

Message
This refers to information that the system automatically writes in the scratch pad to inform the flight crew of some condition.

Modified
This refers to active data that has been changed. When making a modification to the active route or performance mode, MOD displays in the associated page title. ERASE also appears next to one of the line select keys and the EXEC key illuminates. Pushing the ERASE line select key removes the modification, and pushing the EXEC key changes the modified data to the active status.

Prompt
This refers to something displayed on the CDU to aid the flight crew in accomplishing a task. Boxes or dashes remind the flight crew to enter information on the associated line, or alternatively a word indicates what action is required next.

Resynchronisation
This is the automatic process of one FMC loading data into the other, when detecting a significant difference between the two FMCs.

Select
This refers to pushing a key to obtain the desired data or action.

Speed Restriction
This refers to a flight crew entered airspeed limit below a specified altitude.

Speed Transition
This refers to an automatically entered airspeed limit below a specified altitude.

Waypoint
This refers to a point in the route. It may be a fixed point such as a latitude and longitude, VOR or NDB station, intersection on an airway, etc., or a conditional point. An example of a conditional point is when reaching 1000 ft.

THE FLIGHT MANAGEMENT COMPUTER MEMORY

The FMC storage is made up of three types of memory:

Erasable Programmable Read-Only Memory (EPROM) or Bubble Memory
This holds the bulk of the navigation and performance data bank.

Non-Volatile Random Access Memory (NVRAM)

This holds specific navigation and performance data, which has been downloaded from the EPROM. Power provided for this memory is from the computer power supply whenever applying power to the FMC. When removing power, the memory elements are automatically switched to a low-power standby state, which is specially designed for data retention.

High Speed Volatile Random Access Memory (HS RAM)

This holds the operating programme, which can be altered by CDU inputs.

The FMC contains storage space for two databases:

1. The **Performance Database** contains an average model of the aeroplane and its engines. This model provides the FMC with the data required to calculate pitch and thrust commands. Additionally, the relevant data can be displayed and this reduces the need for the flight crew to refer to a performance manual during flight. The data stored includes:
   - aeroplane drag characteristics
   - engine performance characteristics
   - maximum, optimum, minimum altitudes
   - speeds and speed limits
   - speed and altitude capability with one engine inoperative

   The maintenance group for individual aeroplanes can refine this data, by entering correction factors for drag and fuel flow, which are retained in the NV RAM for continued use.

2. The **Navigation Database** contains numerous elements of data, which relate to the normal operational area of the operator and type of aeroplane. Each data package originates in the Flight Operations department and is loaded onto a magnetic tape. The data is then loaded into the NV RAM, using a portable data transfer unit connected to the FMC.

   There are normally two Navigation Databases – one active and one inactive. The active part is effective until a specified expiry date, and the inactive part holds a set of data revisions for the next effective period.

   To cover changes in navigational data and procedures, each navigation database is renewed at intervals not exceeding 28 days, although the database holds up to 56 days of data.

The navigation databases contain items such as:

ITEM	DATA HELD
Radio Aids	Identifier, position, frequency, type of aid, DME elevation. VOR, magnetic variation, ILS category and centre line bearing. The maximum distance an aid can be tuned at normal cruise altitudes).
Waypoints	ICAO identifier; type (enroute/terminal). Position (latitude/longitude).
Enroute Airways	Designator; outboard magnetic course.
Airports	ICAO four letter identifier, position, elevation, alternates.
Runways	ICAO identifier, number, length, heading, threshold position, final approach fix (FAF) Ident. Any threshold displacement.
Airport Procedures	ICAO code; type (SID, STAR, ILS, RNAV etc) runway number/ transmission, path and termination code.
Company Route	Origin airport, destination airport, route number; details of SID, route, STAR, approach.

NB
It is the flight crew's responsibility, during the pre-flight checks, to ensure that the data package in use is valid. **CHECK THE DATE!**

At least one large aeroplane has crashed with a probable cause of an inaccurate data package loading.

GENERAL FMS OPERATION

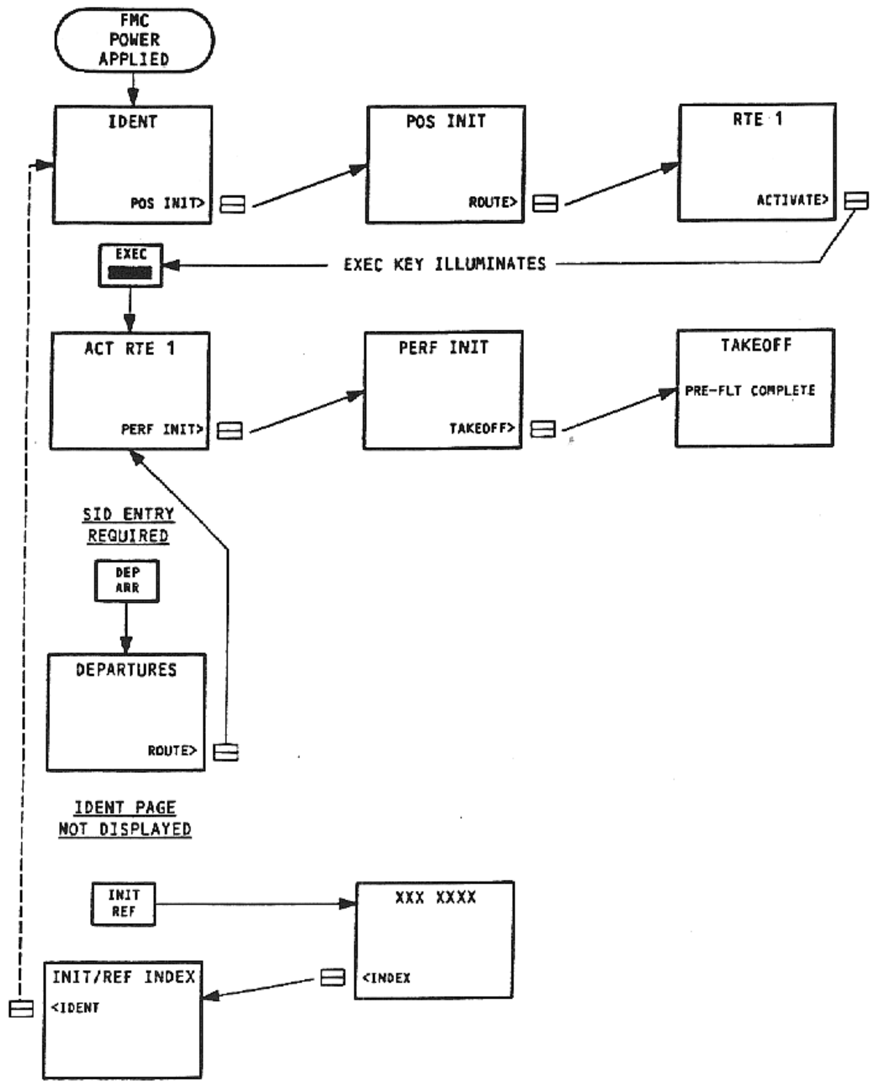
In order to make full and effective use of the FMS the following rules apply:

- To avoid errors, work in a slow deliberate manner while operating the CDU. Avoid pushing more than one key at a time. Avoid entering data into both CDUs at the same time. Do not push CDU keys when the system is going through a resynchronisation. Resynchronisations take about 30 seconds to complete. During this time, one map and CDU shows a failed condition, whilst the other CDU displays the **RESYNCING OTHER FMC** message.
- When selecting a CDU page, read the page title to ensure the correct page appears.
- Check that the scratch pad line is blank before trying to enter data on the line. To blank the scratch pad, use the CLR key as required.
- When entering data on the scratch pad line, ensure that it is correct before continuing with the procedure.
- Use care when pushing line select keys to ensure the correct key is pushed.
- Confirm that data displayed on the CDU is correct before pushing the EXEC key. If an error is made, correct the erroneous data or push the ERASE line select key, and then restart the procedure. Data cannot be entered on a blank line.
- Messages that commonly indicate an error has been made are: **NOT IN DATABASE**, **INVALID ENTRY,** and **INVALID DELETE**.

PRE-FLIGHT

During pre-flight, information from the flight plan and load sheet is entered into the CDU. This information defines the starting point of the flight for initialisation of the inertial reference systems, the desired route to the destination to initialise LNAV, and performance information to initialise VNAV. If necessary, the CDU may also be used to modify the flight plan while in flight.

When electrical power is initially applied to the aeroplane, the CDU displays the appropriate page for starting the pre-flight checks, as shown below.



After checking and entering the necessary data on each pre-flight page, push the lower right line select key to select the next page. After selecting ACTIVATE on the route page, the EXEC key illuminates. Push the EXEC key to complete the task of making the route active before continuing the sequence.

If entering a Standard Instrument Departure (SID) into the route, push the DEP/ARR key. Use the lower right line key to proceed with the pre-flight sequence, after selecting the desired SID.

When reaching the TAKEOFF REF page, it displays PRE-FLT COMPLETE, confirming that all of the required pre-flight entries are made.

If the IDENT page does not display at the beginning of the pre-flight, the IDENT page is alternatively selected by pressing the INIT REF (initialisation) key.

ENROUTE

In flight, the CDU is used to modify the flight plan and display navigation and performance information.

The first step is to select the appropriate page of data by pushing the Function/Mode key that specifies the desired command. For example, DIR = go direct; CLB = change climb conditions; HOLD = enter or exit holding pattern, etc. Then, if required, enter the desired modification.

For example, to fly from the aeroplane's present position direct to a waypoint, push the DIR/INTC key and enter the waypoint in the box prompts that appear. Alternatively, to answer a "what if" type question, select the page that displays the desired information and enter the modified conditions via the keypad. The CDU then displays predictions of what will happen if the modification is executed. The flight crew then has the option of either erasing or executing the modification.

LATERAL NAVIGATION (LNAV)

LNAV guidance outputs from the FMC are normally great circle courses between the waypoints that make up the active route. If a procedure stored in the FMC database is entered into the active route, the FMC is able to supply commands in order to maintain a constant heading / track, or follow a DME arc, as required to comply with the procedure.

The FMC determines the present position of the aeroplane using inputs from the IRS, DME, VOR, and localiser receivers. The FMS is certified to navigate accurately within a VOR/DME environment, although its tolerance is tighter if it uses GPS updating. The FMC System position may be solely based on IRS data, but if available, it normally uses DME or VOR/DME inputs to refine and update the FMC Position. The FMC then uses its calculated present position to generate lateral steering commands along the active leg to the active waypoint.

Whilst the aeroplane is on the ground, the FMC calculates its present position based only on data received from the IRS systems, for which the FMC requires a present position input from at least one IRS. Inertial systems accumulate position errors as a function of time, so the position information for use by the FMC, slowly accumulates errors. These position errors are detected by observing the position of the aeroplane on the HSI map. If an extended ground delay occurs and a significant map error is noticed, the IRS should be realigned, and the aeroplane's present position re-entered.

While the aeroplane is in flight, the FMC refines its position calculations based on inputs from GPS, IRS, DME, VOR, and ILS. The refinement of position calculations made by use of two DME stations, if available, or one DME and one co-located VOR. During an ILS approach, the position can also be refined by using the localiser signals.

The FMC additionally obtains basic position and velocity data from the IRS, which is then checked against data received from the ground and/or space based sensors to determine the IRS drift rates and offset corrections. Using all of this data the FMC is able to generate the Best position every 5 seconds.

The FMC normally automatically tunes the VOR and DME to provide the best available signals for updating the FMC calculated present position. The flight crew can alternatively select frequencies manually and the FMC can continue to use the signals for position updating.

VERTICAL NAVIGATION (VNAV)

CLIMB AND CRUISE

After takeoff, VNAV mode engages after a thrust reference other than take off is selected and the MCP altitude window is set to an altitude above the aeroplane. Once engaging VNAV, the MCP may be set to any altitude, even below the aeroplane, without causing a level off. The VNAV mode does disengage if there is interception of the MCP altitude before the aeroplane reaches its FMC cruise altitude.

The VNAV profile that the FMC commands, if not modified by the flight crew, is a climb with the climb thrust set to achieve the airspeed limit associated with the origin airport until above the limit altitude, and then a climb at the economy speed until the entered cruise altitude is reached. During the climb, the aeroplane remains within all altitude constraints that form part of a SID entered into the active route. The aeroplane then cruises at its economy speed until reaching the top of descent point, during which time the thrust is limited to maximum cruise thrust.

If flying the climb speed profile would cause a violation of any altitude constraint the UNABLE NEXT ALT message appears on the CDU. The flight crew must then manually select a different speed that provides a shallower climb angle.

DESCENT

When entering an (E/D) point, the FMC calculates a descent path. (An E/D is a waypoint altitude constraint that requires a descent from cruise altitude. The E/D is normally entered on the legs page because of selecting a STAR or APPROACH).

Target speeds are changed by entries on the legs or descent page. Wind and thrust assumptions are also changed on the descent forecast page.

If the set MCP is at an altitude below the aeroplane, when reaching the TOD point the FMC commands idle thrust and pitch to track the descent path. VNAV automatically disengages if the aeroplane reaches MCP altitude before the lowest altitude constraint. During the descent, the MCP may be set to an altitude above the aeroplane without VNAV disengaging or stopping the descent.

If an unexpected (not entered on descent forecast page) headwind is encountered that significantly decreases the airspeed, the engine thrust automatically increases to regain the target speed. If the autothrottle does not engage, a THRUST REQUIRED message displays on the CDU. Conversely, if encountering an unexpected tailwind that significantly increases the airspeed, a DRAG REQUIRED message alternatively displays. If the airspeed is limited, the aeroplane flies at the limit speed, even if it must leave the computed flight path.

For VFR and non-precision approaches, the FMC computed path, is designed so the aeroplane flies down to a point that is 50 ft over the approach end of the runway. It is then the flight crew’s responsibility not to descend below the Minimum Descent Altitude (MDA) until achieving adequate visual contact. At the missed approach point, the vertical profile initiates a climb to the missed approach altitude, using climb thrust.

OPERATIONAL NOTES

- When operating in LNAV and VNAV modes, continue to monitor the system operation for undesired pitch, roll, or thrust commands. If undesired operation occurs, switch over to the heading select and flight level change modes.
- The system should be carefully monitored for errors following the activation of a new database, resynchronisation, power interruption, or IRS failure.
- During twin IRS operation, each FMC uses a different IRS for position calculations. The IRS positions are not averaged as during normal operation, which can result in a difference between the two HSI maps and the descent paths when radio updating is not available.

When operating significantly off the active route, the active waypoint may not change when passed. When the LNAV mode is armed, it can only capture the active leg, and does not capture an inactive leg in the active route. The DIRECT TO or INTERCEPT LEG/COURSE TO procedures may be used to make the desired leg active.

- When the same waypoint is used more than once in the route, certain route modifications (such as DIRECT TO and HOLD) use the first waypoint.
- Some standard instrument departures contain a heading vector leg. These show on the CDU LEGS page as a VECTORS waypoint, and on the map as a magenta line leading away from the aeroplane symbol or waypoint. If the VNAV mode is engaged, the DIRECT TO or INTERCEPT LEG/COURSE TO procedure may be used to restore the waypoint sequencing.
- When entering airways into a route page, the beginning and ending waypoint must be in the database, otherwise the route segment must be entered as a DIRECT leg.
- Occasionally a procedure in the database contains a hidden discontinuity that appears on the LEGS page as **----** for the inbound course.
- If an ILS procedure is entered into the active route, and it contains a leg to intercept the inbound course, the LNAV mode does not sequence past the (INTERCEPT) waypoint until the LOC mode engages.
- If the engines are not shutdown on landing, a cruise altitude entry must be made prior to the next flight to ensure that the vertical profile is rebuilt. If in descent and a diversion to another airport is entered, a cruise altitude entry must be made to rebuild the vertical profile.
- When operating outside the FMC navigation database area the following operating characteristics will be noticed:

    1. Origin, destination, and runways cannot be entered into the route. However, any origin that is in the database may be entered. An origin entry is required for VNAV operation.
    2. All waypoints must be entered as latitudes and longitudes.
    3. The FMC will not use radio signals to update its calculated position and will not tune the VOR or DME.
    4. The HSI cannot display airports, navigation aids, or waypoints that are not in the route.

FUEL MONITORING

The FMC receives fuel data from the fuel quantity (totaliser) system and EICAS. The FMC additionally calculates a separate fuel quantity, which prior to engine start is set to agree and track the totaliser value unless the flight crew make a manual fuel quantity entry. When the FMC receives a positive fuel flow signal (engine start), the calculated value is disconnected from the fuel quantity system until the engines are shutdown after the flight. After start, the calculated value decreases at the rate the fuel flow signals indicate. The calculated value displays on Progress Page 2, and on the Performance Initialisation page, where it is labelled CALC, unless a manual entry of fuel quantity is made. In that case, it is labelled MANUAL.

If fuel is loaded after the FMC receives a positive fuel flow signal, the calculated value does not include the new fuel loaded. This could occur if the engines are shutdown at one location, then restarted to taxi to the fuelling location. Making a manual Fuel quantity entry on the Performance Initialisation Page, followed by deletion of the manual entry can restore normal operation.

The fuel flow signals are also used for calculating the fuel used by the engines. FUEL USED, displayed on Progress page 2, is reset to zero following a flight, before both engines are shut down.

If the FMC determines a significant difference between the totaliser and calculated values, the FUEL QTY ERROR-PROG 2/2 message displays on the CDU scratch pad. The flight crew may then select which value the FMC should use for fuel calculations for the remainder of the flight. The FMC also continually estimates the amount of fuel that remains on board the aeroplane when it reaches the destination airport if the active route is flown. If the estimate is less than the fuel reserve value entered on the Performance Initialisation Page, the INSUFFICIENT FUEL message displays.

FLIGHT CONTROL AND MANAGEMENT SUMMARY

The FMS is capable of commanding the aeroplane along a pre-selected lateral (navigation) and vertical path (performance). This takes place shortly after takeoff, and continues until the system captures the localiser and glide slope. A typical FMC receives twenty-four digital inputs and three discrete inputs, and outputs to nine different digital customers.

The FMC performs seven major functions (typical):

1. The input/output function of the FMC receives and transmits digital data to and from the various systems on board the aeroplane, and checks that all received data is valid.
2. The CDU function of the FMC, formats updates and sends data to the CDU fix display, and provides alerting and advisory messages to the CDU for display on the scratch pad.
3. The bit and monitoring function of the FMC performs a self-test of the FMC during power up and upon request. It continuously monitors the FMS during normal operation, and any failures are recorded (on the memory disk) for later retrieval.
4. The navigation function of the FMC houses the navigation data base, and is responsible for computing the aeroplane's current position, velocity and altitude. It also selects and automatically tunes the VOR receivers and DME interrogators. The navigation function computes the aeroplane's present position by determining the distance to two auto-tuned DME stations and GNSS, if installed. Positional information from the three IRUs is used to solve any ambiguity that may occur, or as a prime source when the aeroplane is on the ground. The aeroplane's velocity is computed using IRU inputs, and altitude is computed using both IRU and ADC inputs.
5. The performance function of the FMC computes performance parameters (limits) and predictions for the vertical path of the flight profile, by utilising the performance database, and the CDU input data.
6. The guidance function stores the active vertical and lateral flight plan input from the CDU. The current aeroplane velocity and position information is then calculated by the navigation function. The guidance function then compares the actual and desired position, and generates steering commands, which are forwarded to the appropriate flight control computer (FCC). Using the current computed vertical profile data from the performance function, the guidance function also compares the actual and desired altitude, and also the altitude rate. It then generates pitch and thrust commands, which are input to the appropriate FCC, and the thrust management computer (TMC).
7. The EFIS function of the FMC provides dynamic and background data to the EFIS symbol generator, and also provides the navigation function with a list of the closest NAV aid array for auto tuning.

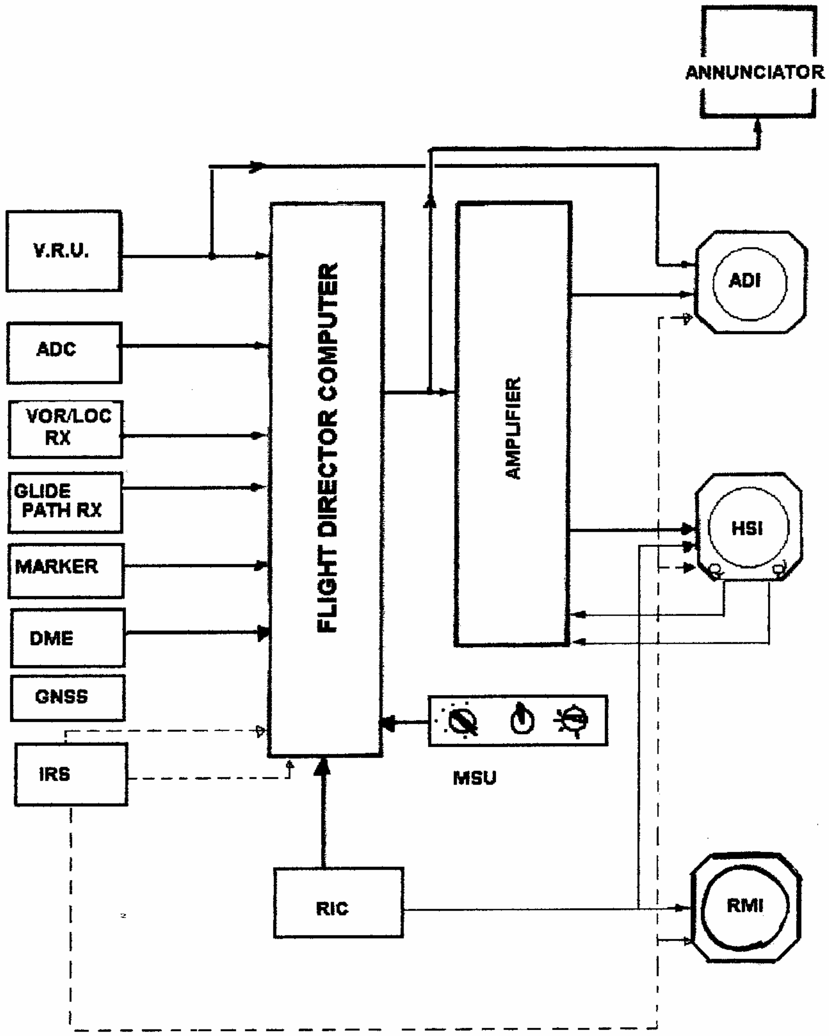
Chapter 24
Flight Director System

INTRODUCTION

In a Flight Director System (FDS), pitch and roll commands compute to achieve or maintain a particular condition of flight, which are then applied manually by the flight crew. The Flight Director is able to combine information from various sensor input sources, and display them as a series of pitch and roll commands. This reduces the scan, simplifies the interpretation, and removes the need for the flight crew to analyse the inputs before making the necessary control inputs. The commands from the Flight Director computer are also modelled so that any command given will not unduly over stress the airframe.

FLIGHT DIRECTOR ARCHITECTURE

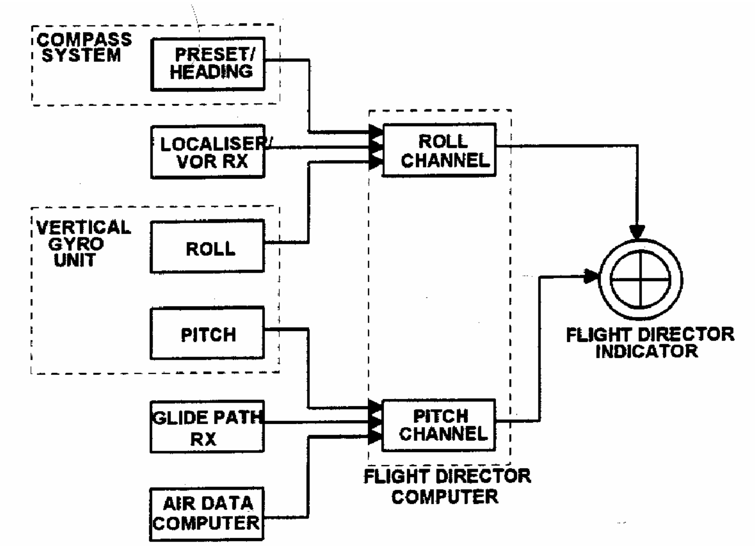
The Flight Director System is made up of the components and displays illustrated in the diagram on the next page. It also shows the range of input sources and flows within the system.



Key: Flight Director Computer (FDC) | Signal amplifier
Mode Selector Unit (MSU) | Annunciator Panel
Attitude Director Indicator (ADI) | Horizontal Situation Indicator (HSI)

FLIGHT DIRECTOR CONTROL INPUTS

In its operation, the Flight Director display receives pitch and roll commands from the Flight Director System Computer, which may form part of the Automatic Flight Control System, or may be completely independent. In the diagram below, the system typically receives inputs from the indicated sources.



These sources are:

Vertical Gyro

Pitch and roll attitude are prime to the system and essential to all computer modes. Any manoeuvre requires reference to a pitch and/or roll attitude. Any failure of the attitude reference results in a total failure of all Flight Director functions. The pitch and roll inputs may be derived from a Vertical Gyro unit, or more commonly from the gyro stabilised Inertial Navigation System (INS), or from the Inertial Reference System (IRS).

Compass

Magnetic Heading inputs are required for the heading hold and select functions of the system. Compass information is also an essential input during the radio modes of operation, where any heading deviation signals are obtained from the Remote Indicating (Gyro Magnetic) Compass (RIC) System.

Localiser/VOR

The VHF Navigation Unit provides inputs for the tracking of VOR radials, and for the capture and tracking of the localiser. The localiser inputs may be used as a pure lateral navigation function or may be displayed in combination with glideslope derived commands during the approach phase.

Glide Path

The Glideslope receiver provides inputs to the computer, equating to the direction and degree of displacement from the glide path during the approach phase. As a condition of the selected modes, when the pitch channels of the computer are processing the glideslope deviations, the roll channels will also be generating localiser commands.

Air Data Computer

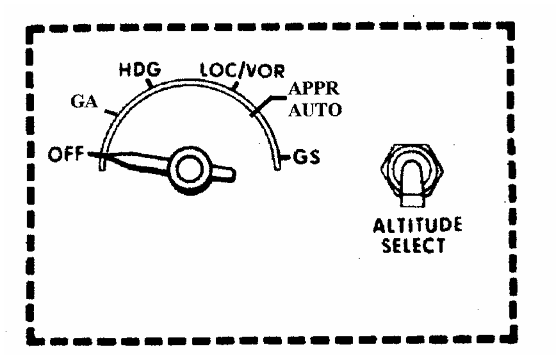
The ADC provides signal inputs related to IAS, Mach, Vertical Speed, and Altitude. A Flight Director Computer may also generate commands based on any, or all of these inputs.

THE FLIGHT DIRECTOR COMPUTER (FDC)

The information derived from the various sources is processed in a solid-state digital computer, and depending on the setting on the Mode Control Unit (MCU), provides the necessary attitude and steering commands. These commands are then fed to the displays, or alternatively to the Automatic Flight Control System (AFCS).

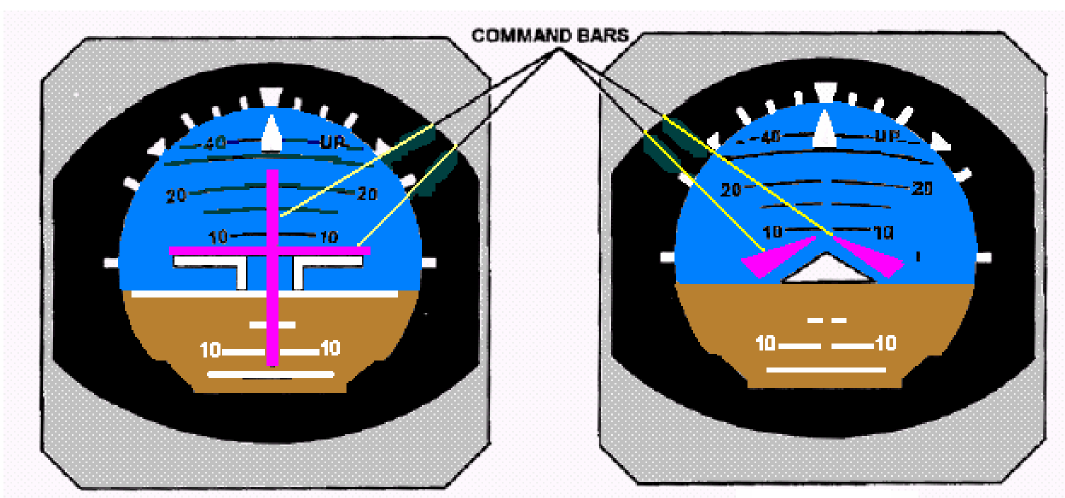
MODE CONTROL UNIT

This unit allows the flight crew to communicate with the FDC regarding the phase of flight, see the example below:

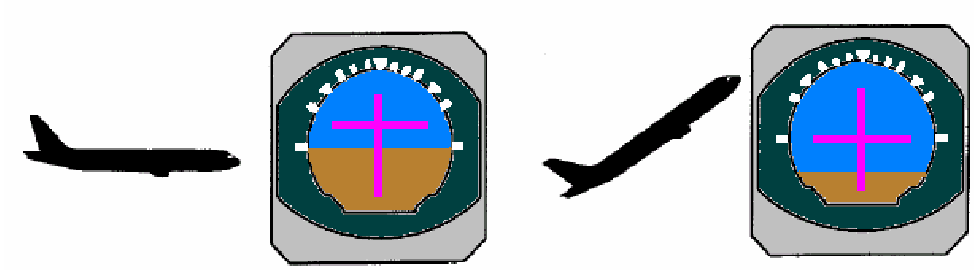


FLIGHT DIRECTOR DISPLAYS

In its simplest form, the display of the Flight Director consists of an indicator having pitch and roll command bars or 'V' bars, as shown below, which are often combined with those of the attitude indicator.

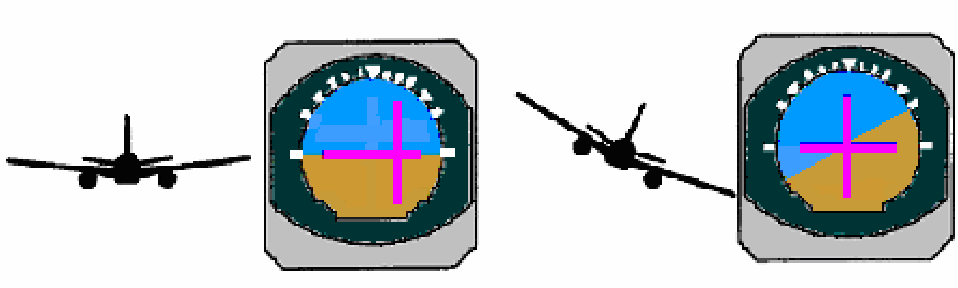


Movement of the Pitch Command bar up requires the positioning of the aeroplane by pulling the control column back until the command bar centres.



When centred, the command bar indicates the achievement of the required pitch attitude; similarly centred is the control column at this point.

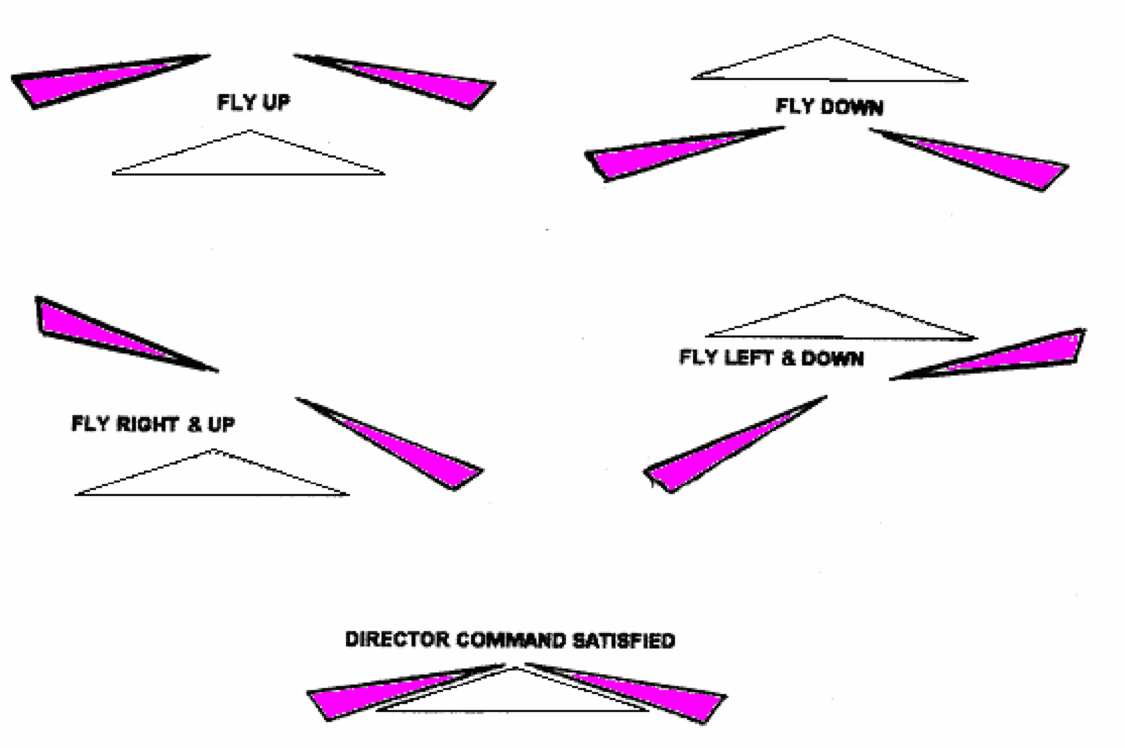
Conversely, movement of the roll command bar to the left or right requires rolling the aeroplane to the left or right, in order to centre the command bar, and to satisfy the computed command, as shown on the next page.



By virtue of the fact that the command bar deflections are the resultant of a combination of system inputs, do not regard them as positional to acquire a required state. They are simply a command for change in pitch and/or roll attitude, e.g.

- Pitch Command Bar UP - Pitch Up
- Roll Command Bar RIGHT - Roll Right

Alternatively if the delta and WING type of display is used the following commands will be given:

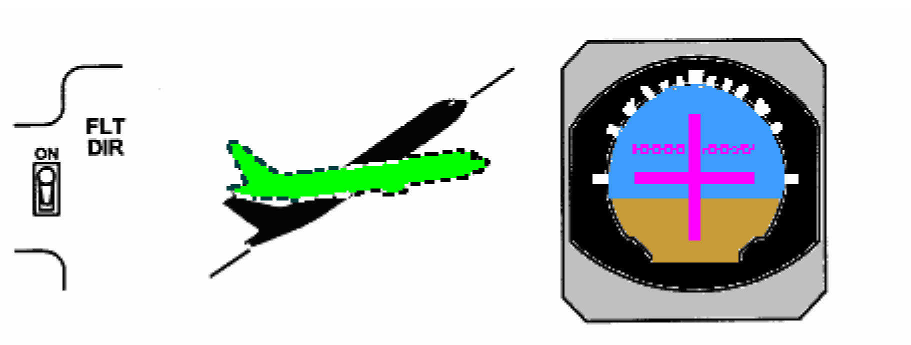


FLIGHT DIRECTOR MODES OF OPERATION

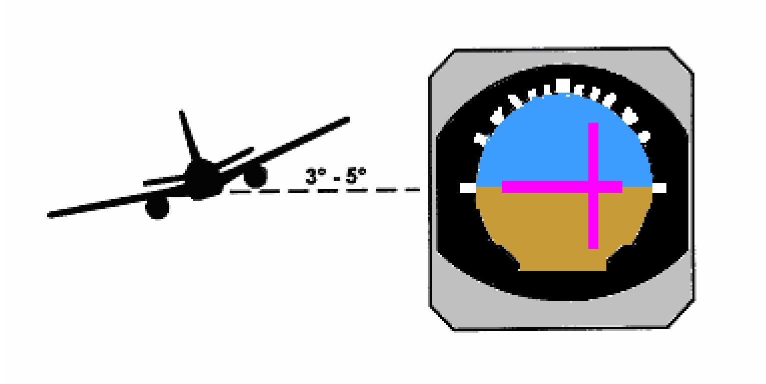
The following modes of operation are available:

ATTITUDE MODE

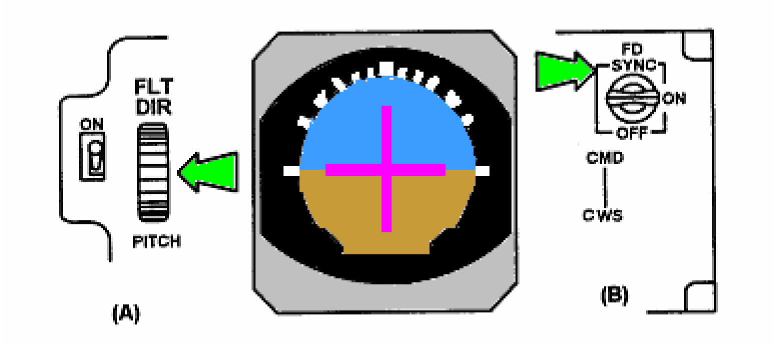
When no other mode is selected, this is the basic function of the system. The command bars are positioned to hold the pitch and roll attitude at the time the Flight Director is switched ON. The Attitude Reference, VG, resulting in the display of the corrective pitch and roll commands, senses any deviation from this datum.



In many systems if the aeroplane is in a near wings level state, typically 3° - 5°, the Flight Director Command bar calls for wings level and from this point, datum to wings level and Magnetic heading hold.



Some Flight Directors have the facility where the command bars can be re-datumed by the flight crew. This is achieved via a switch that enables the command bars to be re-centred at a particular pitch and roll, or by a thumbwheel that that enables the pitch command bar to be accordingly repositioned, as shown below.

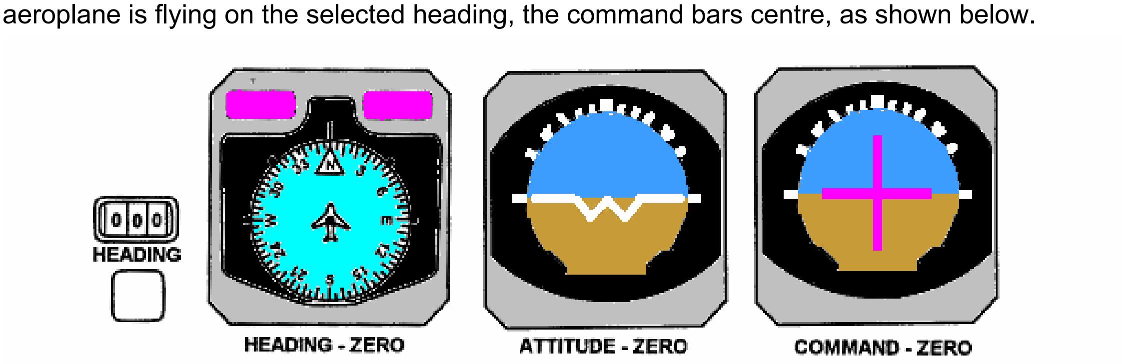


The following explanations assume that the Attitude and Flight Directors are separate instruments, but these are usually combined together on the same instrument, as an Attitude Director Indicator (ADI).

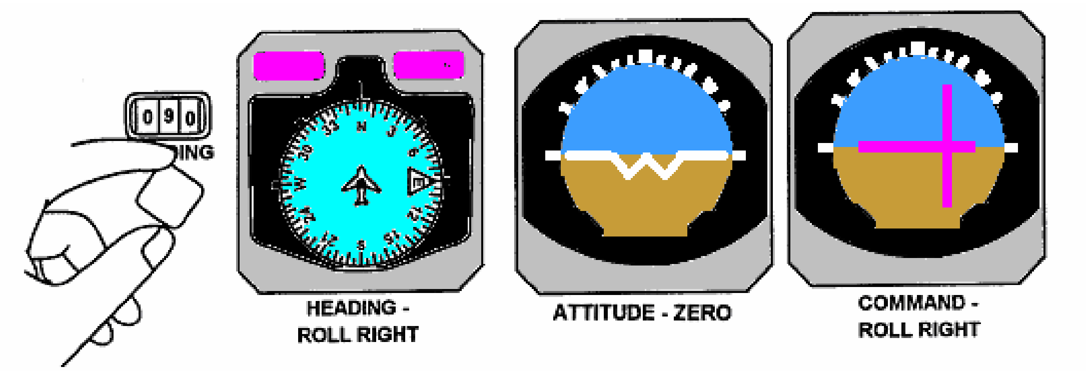
HEADING

With Heading selected the Flight Director roll bar commands the aeroplane to fly to the Pre-Selected Heading, and subsequently to hold that heading. To achieve this, the computer receives inputs from the attitude reference source, the VG, and the course from the compass system.

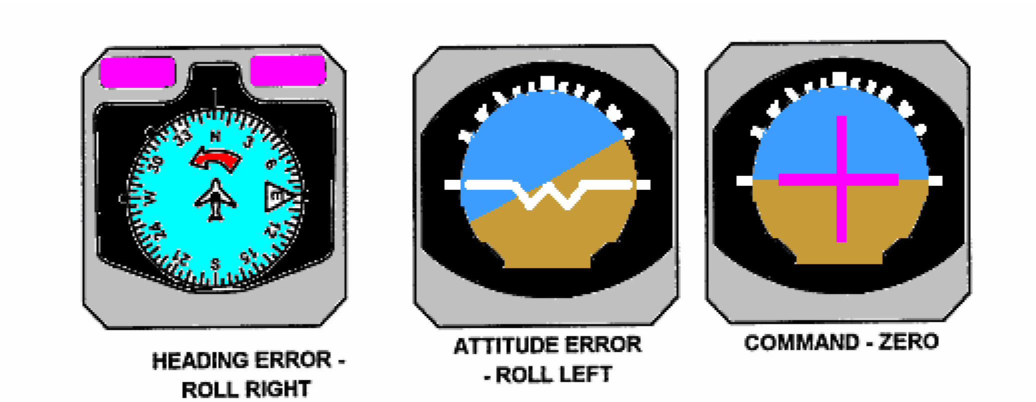
Either Pitch Attitude or Altitude Hold is compatible and available in the Heading mode, and when the aeroplane is flying on the selected heading, the command bars centre, as shown below.



If a new heading is subsequently selected, a heading error (i.e. the difference between the selected and actual heading) positions the roll command bar accordingly.



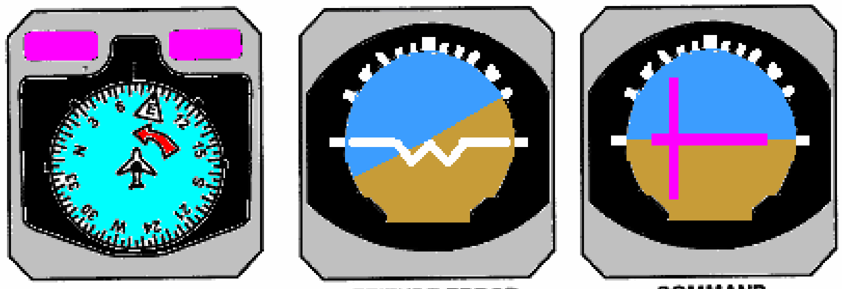
As the aeroplane banks to satisfy the command, the attitude reference inputs to the computer a roll attitude signal, which opposes the signal error. When the established bank angle is appropriate to the heading change required, the heading error and attitude signals cancel. The roll command bar then centres and the control wheel returns to its centre position.



With the aeroplane banked and turning, the heading error steadily reduces, resulting in the computer commanding opposite roll, in response to the attitude input, to roll the aeroplane out onto the new heading.

With the aeroplane on its new heading, the command bars and control wheel centre again, as shown on the next page.

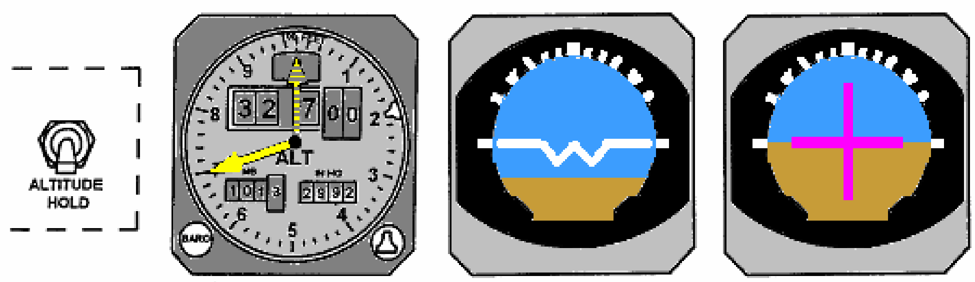
The heading/bank angle commands, which are determined by the computer are limited, primarily for passenger comfort, and typical maximum bank angle limits will be between 30° - 35°.



ALTITUDE HOLD

If this mode is selected, the attitude hold mode cancels automatically, and provides appropriate pitch commands to hold the existing altitude. This mode is compatible with all lateral command modes, and with the Glideslope mode, until capturing the glideslope.

Having selected this mode, any deviation from the datum altitude results in generated pitch commands. In response to the pitch command, a pitch attitude signal applies to the computer, opposing the altitude error signal.



Signal summation and the resultant command generated will be similar to that in the heading mode.

A sophisticated Flight Director may have a number of pitch modes (e.g. IAS Hold, Mach Hold, Vertical Speed) and some may have the facility to provide commands relative to Vertical Navigation profiles generated by the Flight Management Computers. In common however, they all allow only the engagement of **one** pitch mode at any one time.

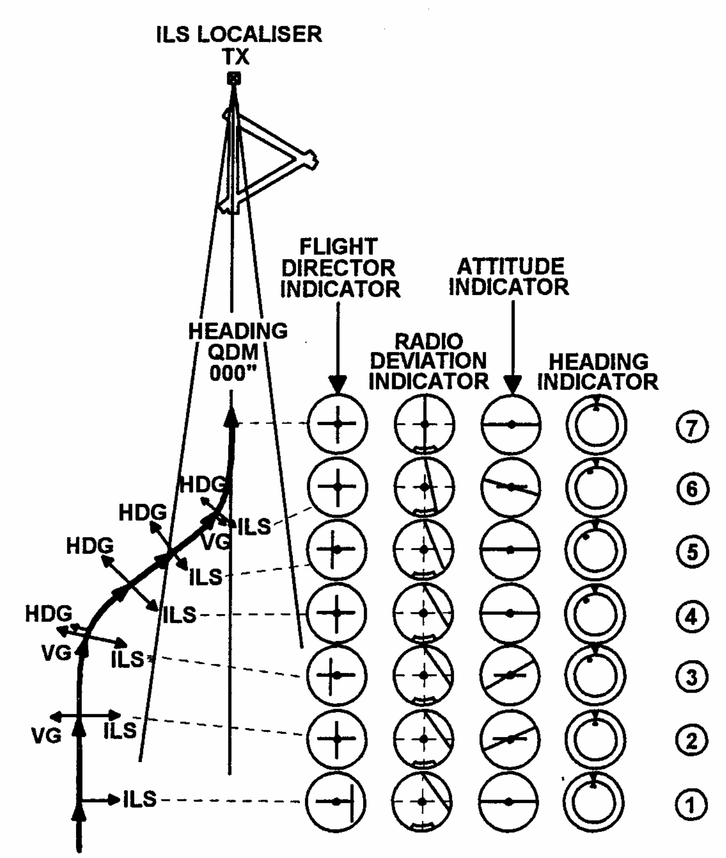
LOCALISER/VOR (LOC/VOR)

This selected mode is identified through the frequency of the particular navigation aid that has been tuned on the VHF Navigation Control Unit. VOR provides the facility for tracking a selected VOR radial, although the word Course is more appropriate since the Flight Director facilitates passage over the VOR station and continued guidance relative to what would now be an Outbound radial.

In the VOR mode, the aeroplane equipment automatically senses any over-station condition or **Cone of Confusion**, providing Flight Director commands in a memory capacity during the relatively short over-station condition.

The localiser mode provides for capture and tracking of the localiser signal. Apart from VOR over-station consideration, the principles of localiser mode operation apply equally to VOR. In this mode, the computer constructs its roll commands around the radio deviation signals generated because the aeroplane is to the left or right of the localiser. Importantly, the computer also sums the radio deviation signals with the heading inputs that correspond to the Runway Inbound Course or QDM. In the case of a VOR, this becomes the QDM, or the reciprocal dependent upon whether the aeroplane is flying TO or FROM the station. The Heading input is necessary since there is **NO** content in the radio deviation signal that tells the aeroplane how far it should fly to the left or right to reduce the error. In an extreme case, if an aeroplane is to the left of the localiser it flies to the right in circles unless the radio deviation signal was co-ordinated with the heading. Importantly, all manoeuvres require being co-ordinated against pitch and rolling attitude.

The following diagram shows how the Flight Director roll commands are computed relative to radio deviation, attitude, and heading. The arrows identify the sense of the input signals to the computer.



1. The aeroplane is wings level, and positioned to the left of the localiser, which results in the FLY RIGHT Localiser deviation signal giving a Flight Director ROLL RIGHT command.

2. Banking the aeroplane to the right, the ATTITUDE ERROR and RADIO DEVIATION signals oppose each other. The roll command bar centres and the aeroplane turns toward the runway centreline.

3. As the aeroplane turns away from the runway, commanding a MAGNETIC HEADING and a ROLL LEFT signal, the aeroplane rolls out until the wings are level.

4. The aeroplane is wings level and approaching the runway at an optimum angle. The HEADING ERROR and RADIO DEVIATION signals now act in opposition to each other, centering the roll command bar.

5. The RADIO DEVIATION signal is slowly diminishing, and the predominant HEADING ERROR results in the Flight director Computer commanding ROLL LEFT in order to reduce the relative angle to the runway centreline.

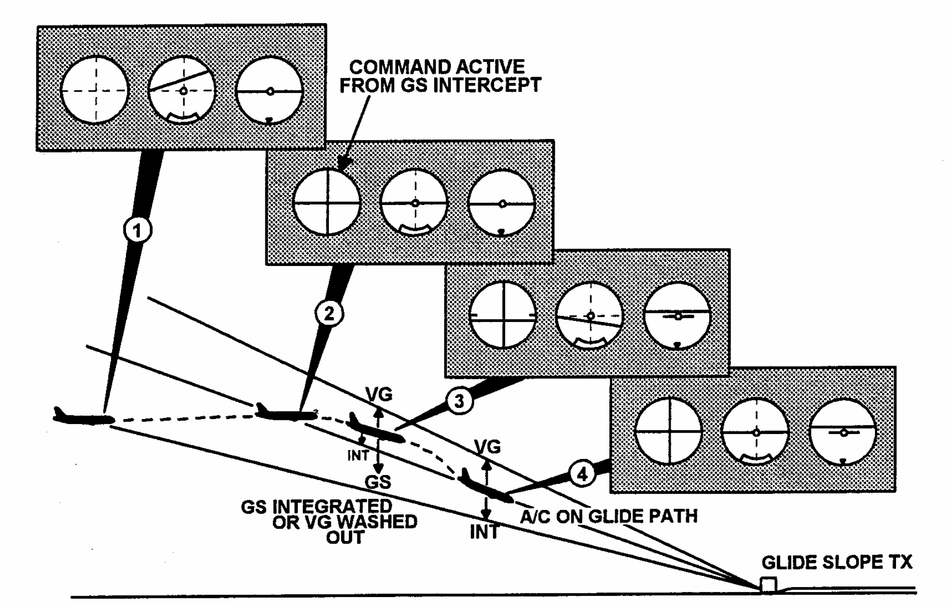
6. The aeroplane continues to roll out onto the centreline using co-ordinated radio deviation, roll attitude and heading signals.

7. The aeroplane is now on course and lined up with the runway centreline.

GLIDESLOPE (GS)

This mode provides for the capture and tracking of the Localiser/Glideslope, when the UHF glideslope frequency is paired with the localiser. With the mode selector switch in the Glideslope detent and the ILS frequency tuned, localiser and glideslope deviation signals will be directed to the computer.

The heading mode may be used to set up an initial approach to the localiser beam, or Altitude Hold may be used as a pre-capture mode for the glideslope. When the aeroplane is within the capture area for the localiser, the mode selector positions to Glideslope. The Flight Director now generates the necessary roll commands and, after capturing the glideslope, the Altitude Hold mode automatically disconnects. The diagram below shows the typical command indications when selecting the Glideslope mode.



1. The start of the capture phase. ALTITUDE HOLD disconnects and the computer disregards the initial FLY UP signal, thus preventing commands, which would call for the aeroplane to PITCH UP to acquire the glide path.

2. The aeroplane is on the glide path, with zero Glideslope deviation, and no pitch attitude error present.

3. The aeroplane passes through the glide path and the computer responds to the glideslope FLY DOWN by commanding PITCH DOWN. As the aeroplane assumes a nose down pitch attitude, it results in an increased PITCH ATTITUDE ERROR signal, in opposition to the glideslope signal. An integrator circuit within the computer then builds up a signal in opposition to the VG PITCH ATTITUDE signal, and pitch commands display in order to enable the positioning of the aeroplane on the glide path.

4. The aeroplane is now on the glide path with a slightly nose down pitch attitude.

GO-AROUND (GA)

This mode is selected for a GO AROUND after a missed approach, and the computer signals the command bars to instigate a wings level pitch up attitude. This mode also automatically selects if the GA button on the throttle activates.

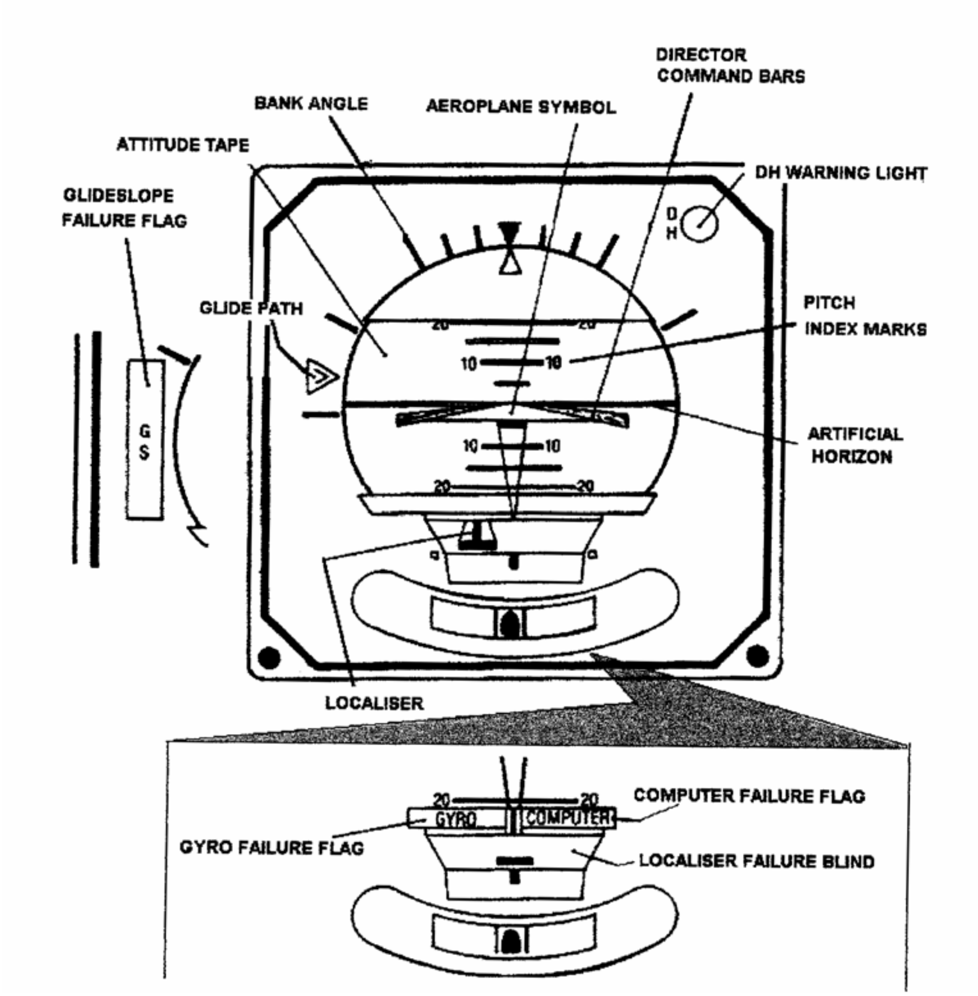
MODE ANNUNCIATOR

This unit provides a visual reminder of the mode selected by the flight crew, and is usually in the form of a panel showing a number of coloured lights, with one light for each mode. When selecting a mode, the appropriate light illuminates.

The Mode annunciators are sometimes solely dedicated to the Flight Director, whilst others combine both the FD and AFCS modes, with the FD annunciators normally sited on the left hand side, and the AFCS annunciators on the right hand side of the EADI.

OPERATION OF THE ATTITUDE DIRECTOR INDICATOR

For take off with the Flight Directors switched on before the start of the take off roll, the roll command bar provides wings level commands throughout the take off run, but the pitch command bar initially indicates 10° nose down. At 60 kt, a 15° nose up command is given and at the appropriate V_R speed the pilot flying pitches the nose up until the FD command is satisfied.



When changing flight levels, input the required level to the FMC via the MCP. If for example the aeroplane climbs to a specified cruise altitude, the new level inputs into the Flight Director Computer, which generates a FLY UP command. The flight crew then flies the attitude that places the aeroplane symbol into the command WINGS. As the aeroplane approaches the target altitude, the computer generates a FLY DOWN command. The flight crew then reduces the pitch attitude of the aeroplane progressively until reaching the attitude for a level cruise, as the designated altitude is reached.

If an essential input fails in flight it displays to the flight crew by red flags, which are easily identifiable by appropriate text as follows:

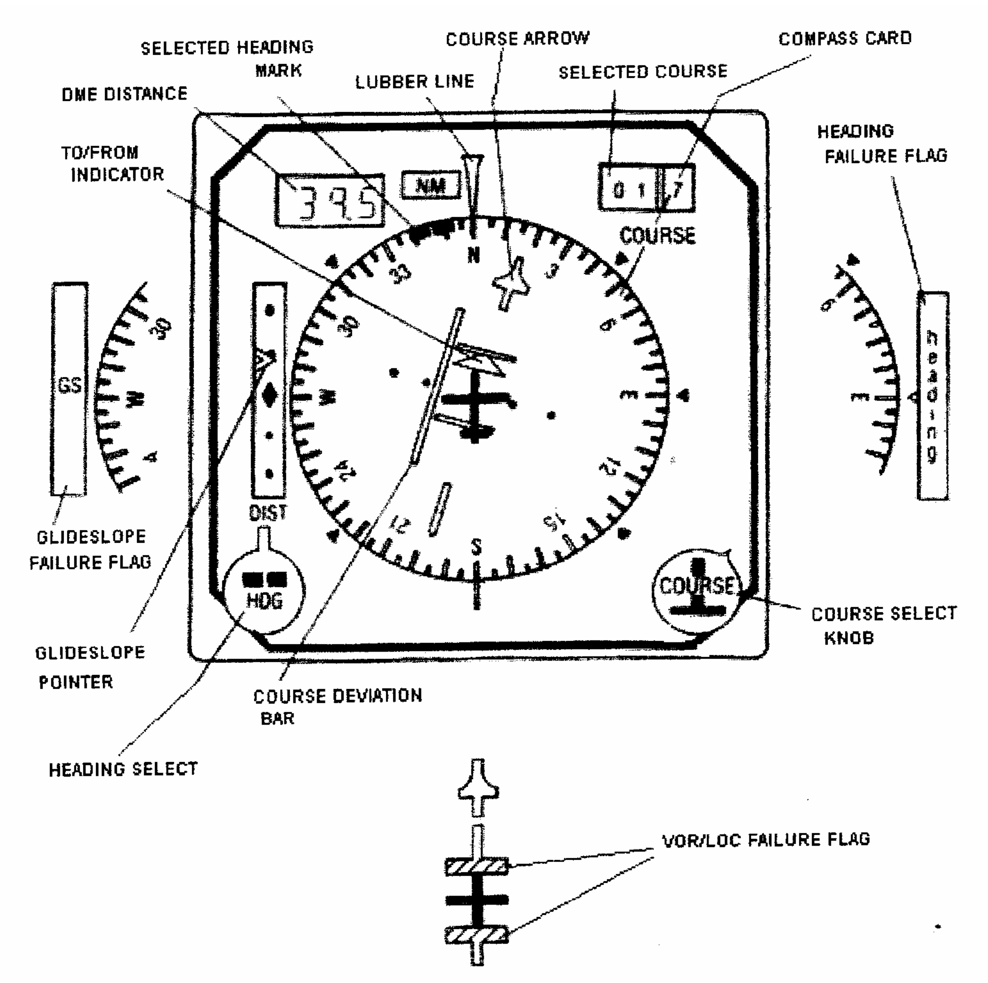
- **GS**
  This flag drops over the glideslope scale if the glideslope signal fails or is alternatively sufficiently weakened.

- **GYRO**
  This flag appears if the attitude information input fails.

- **COMPUTER**
  This flag appears if the system detects a failure in the computer output or command signals.

THE HORIZONTAL SITUATION INDICATOR (HSI) FLIGHT DIRECTOR COMMANDS

On a typical instrument a compass dominates the instrument display, as shown below, which is driven from the heading reference system (RIC or IRS), and rotates against a lubber line.



At the centre of the display is a fixed aeroplane symbol, and inside the compass ring is the symbology, which represents lateral guidance signals, similar to the ADI display. The symbology typically consists of:

- A course arrow adjusted/set by the course set knob.
- A course deviation bar driven by signals from the VOR/LOC receiver (as appropriate)
- A course deviation scale on which each dot represents a displacement of 1° from the selected course or localiser.
- A TO/FROM flag that indicates whether the selected course is extending TO or FROM the VOR, which disappears when a localiser frequency is selected.
- In the RNAV mode, each dot normally represents a 1NM displacement, but in the RNAV APR mode, the scale alters so that each dot represents ¼ NM.

To the left of the indicator is a glide path scale, which replicates the glide path indications on the ADI, and digital read outs are also provided. The range figure is determined from a DME while the flight crew selects the course. For intercept purposes, the flight crew may also manually select a heading, which will directly adjust the position of the heading bug on the compass card.

The warning flags on the HSI, like the ADI, are red in colour, and are identified by appropriate text, as follows:

- **GS**
  This flag appears if the glideslope signal fails, and will cover any Glideslope indications

- **COMPASS**
  This flag appears if the heading reference fails.

- **VOR/LOC**
  This flag appears if the VOR or localiser (as appropriate) signal fails.

INTRODUCTION

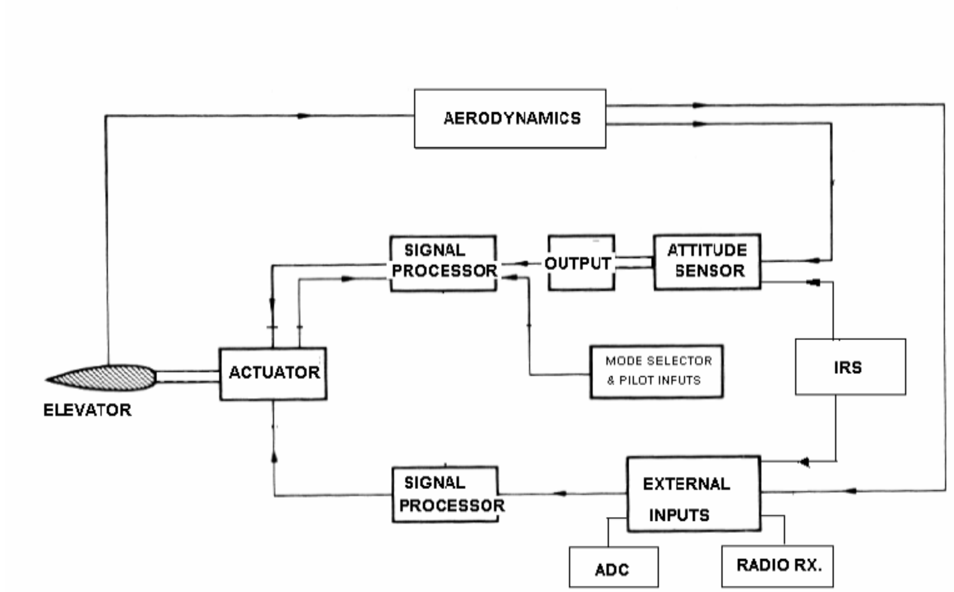
An Automatic Flight Control System (AFCS) is a complex system, designed to relieve the workload of the flight crew, allowing them to concentrate on the management/monitoring of the flight. An AFCS thus provides the following functions:

STABILITY AND CONTROL AUGMENTATION

These two functions are linked and apply equally to both manual and automatic flight control. Stability and control are closely allied to each other, and one cannot exist without the other (e.g. if the aeroplane stability is too low, its controllability is too high). The function of the AFCS is thus to ensure that the correct levels of each exist.

ATTITUDE HOLD

This function ensures the maintenance of the attitude of the aeroplane around selected axes.



FLIGHT CONTROL

In this function, the AFCS responds to externally sourced inputs such as altitude, airspeed, heading, and navigational information. The autopilot, when operating in this mode, maintains a set condition, but also responds to changes of flight profile (e.g. track guidance, automatic approach, flare, and landing, dictated by inputs from external sources). The external sources are not a part of the AFCS, but must fully integrate with it, if the system is to work effectively. The figure below shows the typical flow of information to the AFCS.

CLASSIFICATION OF AN AFCS

The degree of complexity of the AFCS fitted to an aeroplane is dictated by the:

- Size of the aeroplane
- Age of the aeroplane
- Length of the flights
- Intended route structure
- Complexity of the aeroplane
- Number of flight crew
- Cost benefits achievable
- Safety benefits

An AFCS is also classified according to the number of axes around which it exercises control, which are as follows:

Single Axis
This type of autopilot normally only provides control around the roll axis, using the ailerons for lateral control.

Two axes
This type of autopilot provides control around the roll and pitch axes, using the ailerons and elevators respectively. Stability about these axes is the most basic of functions for an autopilot. It can also be used to supply information directly to the Flight Director, to allow a pre-selected flight path to be followed manually, or automatically by the autopilot system.

Three Axes
This type of autopilot provides control around the pitch, roll and yaw axes, using the elevators, ailerons and rudder respectively. It is normally integrated with a Flight Director, and provides sufficient control to carry out automatic landings.

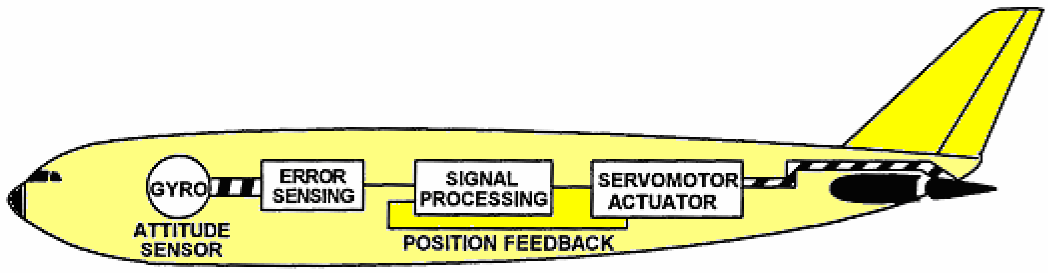
CONTROL CHANNELS

A control channel controls the movement about each axis, which are classified as the:

- Pitch channel
- Roll channel
- Yaw channel

INNER LOOP CONTROL (STABILISATION)

Each axis control channel works on the same basic principle, so it is appropriate to look in depth at one channel only (e.g. the pitch channel). The components of a basic single axis pitch control channel are shown below.



Together these components form a **Closed Loop Control System**, or an **Inner Loop Stabilisation System**. The functions of the individual elements are:

Attitude Sensing
Achieved by gyroscopes or accelerometers, which sense attitude changes about the relevant axis.

Error Sensing
Achieved by synchro-transmitters or E-I bars, which change the attitude signal into an electrical error signal.

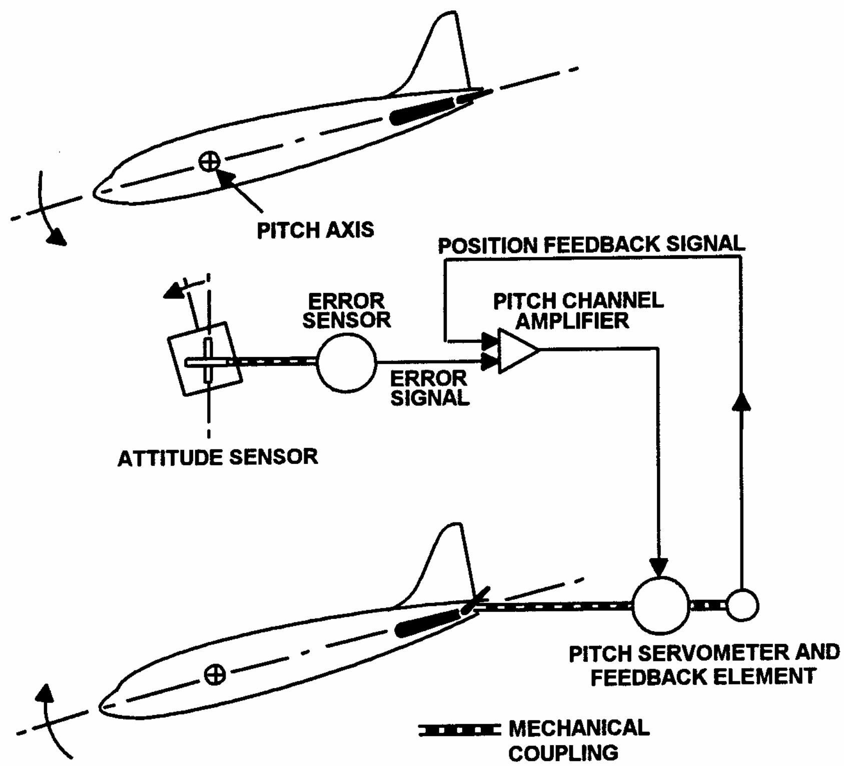
Signal processing
Achieved by a discriminator and amplifier circuit, they process the electrical error signals, providing an output to the servomotor actuator.

Servomotor Actuator
This device moves the control surface.

Position Feedback loop
This element advises the signal processor of any control surface movement.

OPERATION OF AN INNER LOOP PITCH STABILISATION SYSTEM

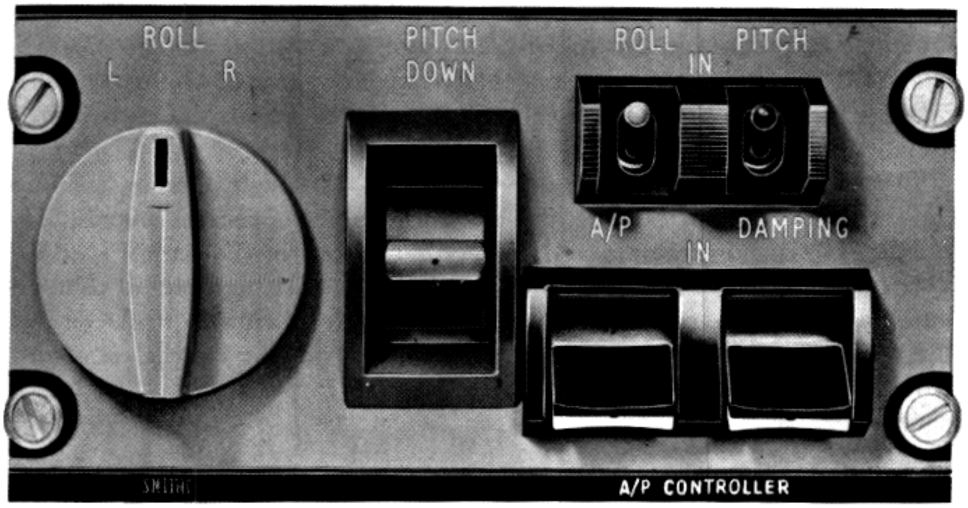
During normal operation, the autopilot system maintains the aeroplane pitch attitude prior to its engagement. If the aeroplane pitch however changes because of some aerodynamic disturbance or out of trim condition, an electrical pick off carries the displacement error signal to the signal processor. The error signal processes, and the demand determines the degree of corrective control input required to forward to the servomotor actuator. The servomotor then moves the control surface in the appropriate direction, and by the appropriate amount. The AFCS computer processes the signal, and is electrically advised of any control surface movement via a feedback loop.



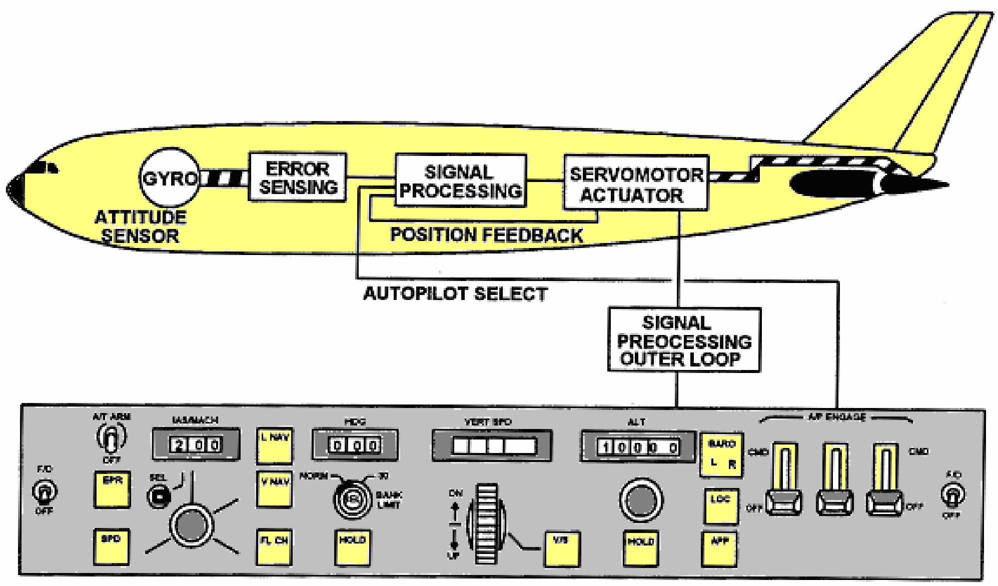
The position feedback signals amplify, and the resulting signal opposes the error signal. As the aeroplane is returned to its original attitude, an aerodynamic feedback signal steadily reduces the attitude error signal, and the position feedback signal begins to dominate, thus moving the elevator back into its former flared position.

OUTER LOOP CONTROL

The primary function of the autopilot is stabilisation, achieved by **inner loop control**, but by inputting signals from raw data {heading, airspeed, altitude, radio links, Lateral Navigation (LNAV), Vertical Navigation (VNAV)} into the inner loop, the system can perform other tasks. The flight crew, through an appropriate mode select panel, feeds these data inputs, known as 'outer loop control', into the system. For example, in a light twin-engine aeroplane, commands to roll or pitch the aeroplane insert manually via a Mode Control Unit (MCU), as shown on the next page. When these controls activate, a synchro transmitter rotor, or potentiometer applies the appropriate command to the channel servomotor.



In a modern transport category jet-engine aeroplane, the outer loop signals apply to the system via a more complex Mode Control Panel (MCP), as shown below.



These outer loop signals are categorised into either roll or pitch modes depending on their function.

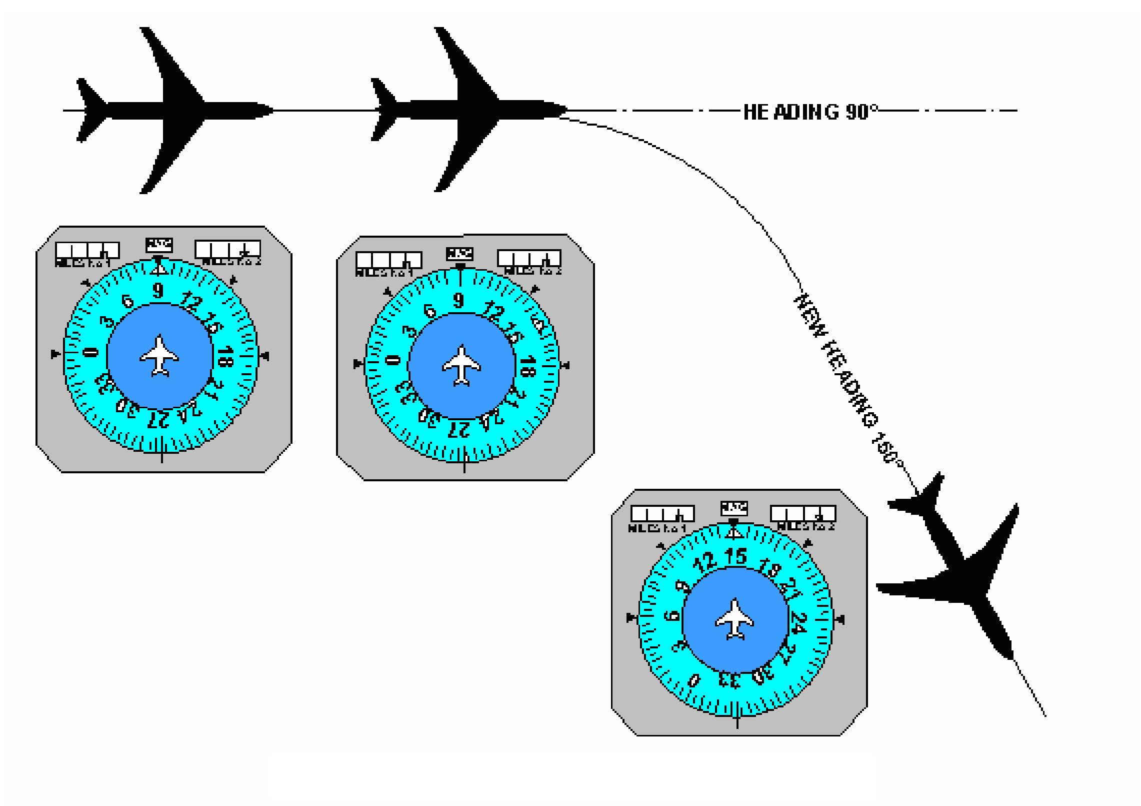
ROLL MODES

Heading Hold

When this mode engages, the autopilot, upon receiving a heading hold error signal from a Remote Indicating Compass (RIC) or Inertial Reference System (IRS), controls the bank angle to maintain the heading at the time of autopilot engagement.

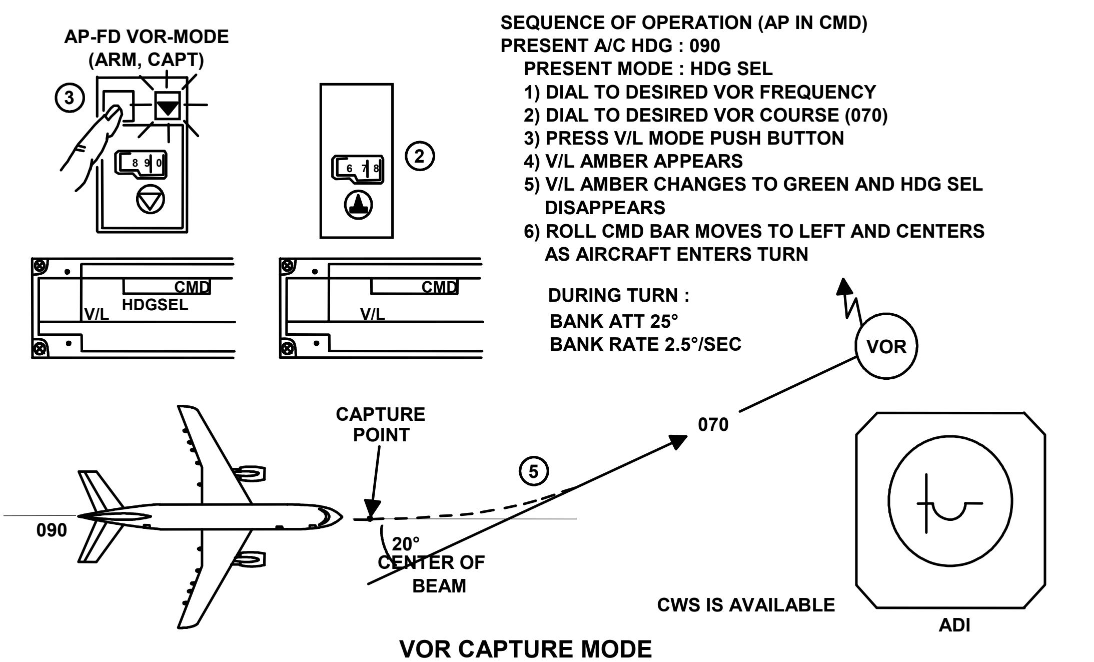
Heading Select

When this mode engages, the aeroplane turns onto a heading, digitally or manually input by the movement of a heading bug. The diagram on the next page illustrates how an aeroplane responds to a change of heading from 090° to a new heading of 150°.



VOR/LOC CAPTURE

The VHF Navigation Unit provides inputs for the capture and tracking of VOR Radials and a localiser. The aeroplane is controlled toward the radio beam in the Heading Select mode until a device called a **Lateral Beam Sensor** checks the strength of the signal and initiates the capture mode at a predetermined signal level.



VOR CAPTURE MODE

LATERAL NAVIGATION (LNAV)

In this mode, computed flight path information from the Flight Management system (FMS) feeds into the autopilot and steers the aeroplane along the designated route.

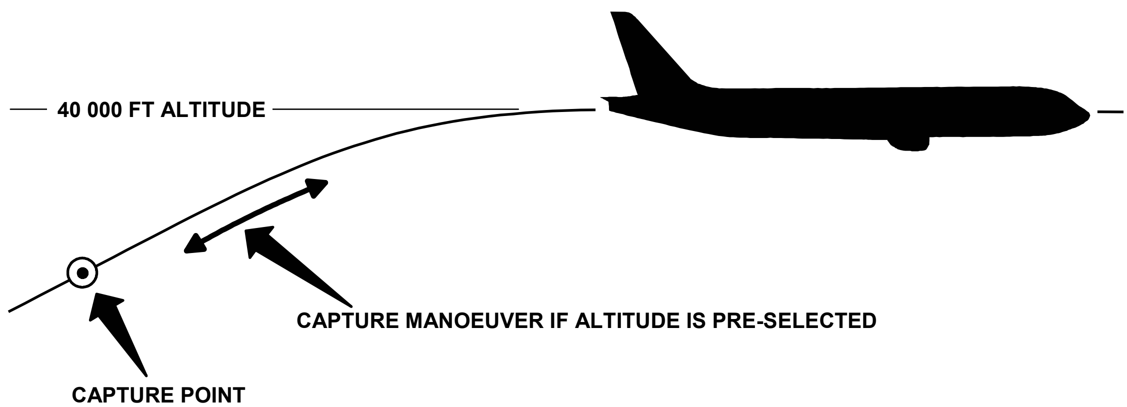
PITCH MODES

Altitude Hold

When engaging this mode, the aeroplane remains at the altitude at the time of selection. If deviation from this altitude is more than a preset amount (typically ± 200 to 300 ft), an altitude alert warning (aural and visual) sounds.

Altitude Capture (Acquire)

When activating this mode, the aeroplane captures a pre-selected altitude when climbing or descending, and automatically levels out and maintains that altitude. A warning of altitude capture is given approximately 1000 - 700 ft. prior to reaching the preset altitude, and stays illuminated until approximately 300 - 200 ft. from this altitude.



ALTITITUDE CAPTURE

Vertical Speed (VS)

When selecting this mode, the nose of the aeroplane pitches up or down in order to maintain a selected rate of climb or descent. Use this mode with caution. It is essential to ensure that the engine thrust setting is correct for the required rate of climb or descent, and that the aeroplane's speed is constantly monitored, normally done in conjunction with the Autothrottle system.

Glideslope (G/S)

When selecting the approach (APP) mode and the glide slope captures, the autopilot responds to the glide slope beam and automatically steers the aeroplane down to the runway threshold.

Vertical Navigation (VNAV)

When selecting this mode, like in LNAV, the aeroplane flies a computed flight plan in the vertical sense. The VNAV flight plan can connect to the autopilot and/or Autothrottle.

Flight Level Change (FLCH) (Speed Mode)

This mode is a **pitch speed** or **manometric lock**, which enables the autopilot to climb or descend the aeroplane at a selected speed, to a selected altitude, and may be either IAS or MACH defined. This mode requires pressure (manometric) data inputs, normally derived from the Air Data Computer (ADC).

COMBINED ROLL AND PITCH MODES

Go-Around (GA)

During an automatic approach, the flight crew can elect to abandon the approach and climb out with the wings level. Depending on the aeroplane type, the GA mode may be Flight Director and/or autopilot operated. This mode only initiates if the auto-approach mode is operative, by either operating the Take-Off Go-Around (TOGA) switch on the throttle lever, and/or by moving the throttle levers to the fully forward position. When the TOGA switch is pushed, the throttles automatically advance, and the aeroplane climbs out at maximum safe rate.

Control Wheel Steering (CWS)

If selecting this mode, the flight crew is able to control the aeroplane in pitch and roll using normal control forces on the control column, through the autopilot. When the control wheel moves, the aeroplane takes up the new attitude, and on releasing the wheel, maintains the new attitude. In this mode, force transducers detect the pitch and roll forces applied, and are built into the control column. The transducers may be in the form of piezo crystal elements, which vary their electrical resistance when they are under pressure. The generated signal outputs, which are proportional to the input forces, amplify and are fed as output signals to the appropriate control channel where the controls activate in proportion to the applied signal.

Touch Wheel Steering (TWS)

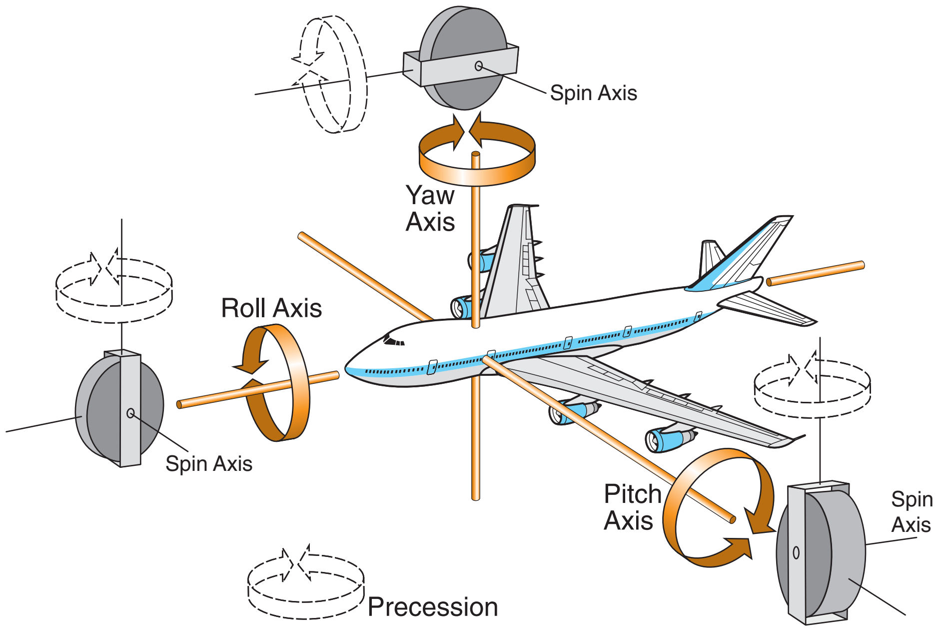
If selecting this mode via the TWS switch, temporary disengagement of the appropriate control channels and servos occurs. The flight crew is then able to manually manoeuvre the aeroplane to a new attitude and heading. Upon releasing the TWS switch, the autopilot automatically re-engages, maintaining the new attitude.

Turbulence Penetration

Flight in turbulent conditions can impose heavy structural loads on an aeroplane. In these conditions, it is thus normal to disengage the autopilot, and fly the aeroplane manually. In some aeroplanes turbulence penetration mode can alternatively be selected, which reduces the gain on the pitch and roll channels, and softens the AFCS response to turbulence.

ATTITUDE SENSING

This is achieved by using either a rate-sensing gyro, or a vertical axis (two degrees of freedom) displacement gyro, which are positioned as shown below, along each aeroplane axis.



Rate Sensing Gyro

When aligning a gyro with its sensitive axis parallel to an axis, it detects any rotation of the aeroplane around that axis, and is able to determine the rate of rotation experienced.

Displacement Gyro

In this system, the attitude of the aeroplane is monitored against a vertical gyro (VG) reference unit. When sensing a displacement, the signal processor determines the magnitude of the displacement, and the corrective control input required as compensation for the displacement. The vertical reference can be taken from an integral gyro source, or may be alternatively taken from an independent source, such as a remote Vertical Reference Unit (VRU), INS, or IRS.

In many systems, the detector signal is derived from a combination of both rate and displacement signals. This reduces the time delay incurred by deriving a rate from a displacement signal, and also has the advantage of **damping** the tendency to overshoot the correction, which is a common problem with the displacement type of system.

The pitch and roll channels are normally operated from the VRU, but the yaw channel requires an input from a horizontal axis tied gyro.

THE AFCS COMPUTER (SIGNAL PROCESSOR)

The function of this component is to process the displacement signal and determine the amount of control movement required to counteract the displacement. It must also monitor the feedback to ensure that the required control activates, and that the desired effect occurs.

The computations are very simple in an attitude hold system, but are extremely complex in a full multi-mode AFCS. The computations and processes carried out in a computer are:

Amplification

This boosts the power of the detected signal to a sufficiently high level to act as an output.

Integration/Differentiation

These are mathematical processes, which are used to derive information, such as attitude change from rate of rotation, or vice versa.

Limiting

This restricts the range of any parameter change (e.g. Pitch rate, to a specific limit).

Shaping

This modifies the computer output to achieve the required flight path or handling characteristics.

Programming

This is the individual process, designed to instruct the aeroplane to follow defined manoeuvres.

The degree of computer power varies according to the role and complexity of the AFCS. In all of these computations however, **Control Laws**, are used to determine exactly how the control demands are translated into control movements. The **C** control law, for example, is commonly for use in large transport aeroplanes in the pitch axis. This law provides stability of the aeroplane at a selected flight path angle, and compensation for problems associated with flight at low airspeeds.

These computations also consider the airframe load factors likely imposed on the aeroplane during a manoeuvre. These are a function of the airspeed (dynamic pressure) and thus require an input from the ADC, or other dynamic pressure source. On older or simple autopilots there is no allowance made for **gust loads**, which can exist during flight in turbulent conditions. If these conditions prevail with this type of autopilot the system should be immediately disconnected, and the aeroplane flown manually. In more modern and complex systems a function switch is available, which, when operated, **softens** the control demands, thus enabling the aeroplane to ride with the gusts. This is achieved by increasing the **limits** argument in the computations. The switch should, however, only be activated in turbulent conditions, as it desensitises the autopilot.

SERVOMOTOR ACTUATORS

The following types of actuators are available, but their design must consider the balance between the range of control surface movement, and the rate of movement (in the event of a failure occurring), the normal rate of movement, and the magnitude and accuracy of any movement for control and/or stability:

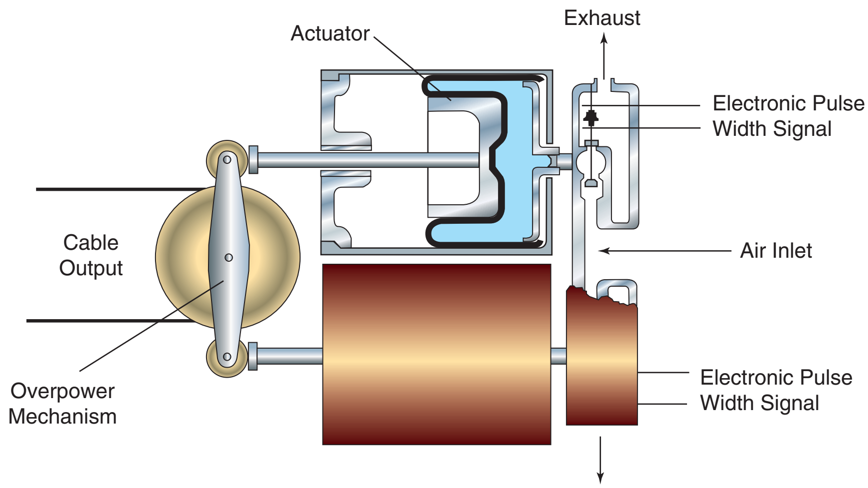
Electro Mechanical Actuator

These may be either DC or AC powered. In the DC system, a motor couples to the flight control via an electro magnetic clutch and a mechanical linkage. The provided feedback is from a potentiometer driven by the motor. In the AC system, the motors used may be either of the hysterisis type, or of the two-phase induction type. A synchro transmission system provides information on the position of the control surface whilst a tachogenerator provides the necessary feedback to the signal processor.

Electro Pneumatic Actuator

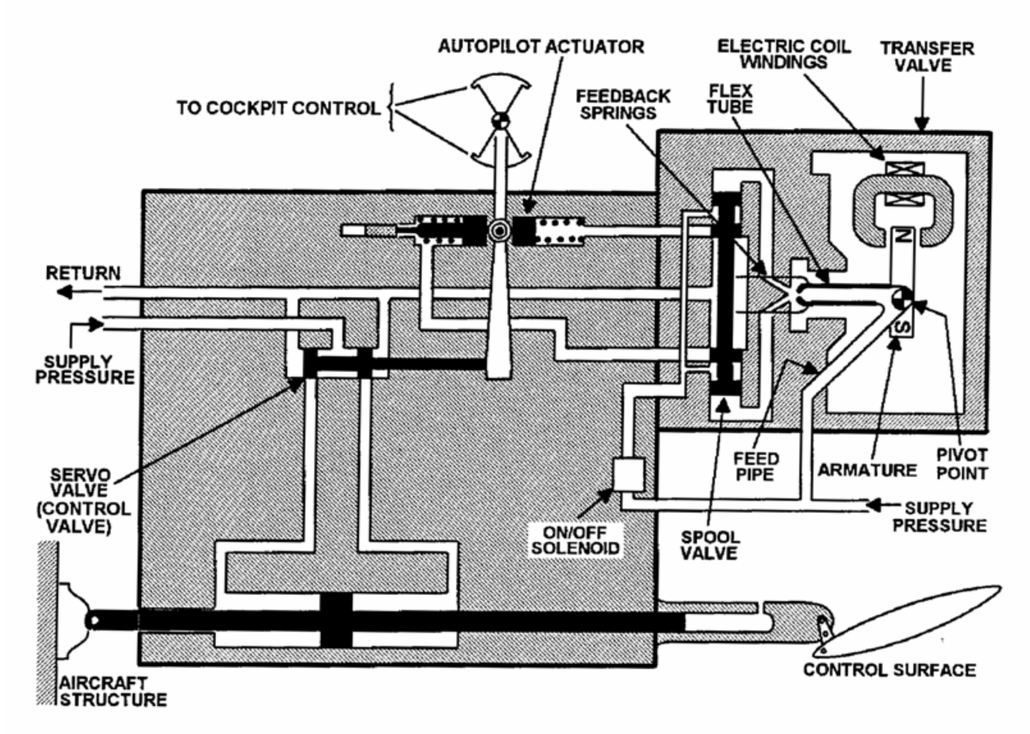
A typical actuator of this type is shown below, where the valve assembly is operated electro mechanically by signals from the AFCS computer.

The necessary power supply is from a pneumatic feed taken from a source such as an engine driven pump, or a compressor bleed from a jet engine. When receiving a command input, the opening of one valve increases, whilst reducing the other. This produces a differential pressure in the two cylinders and results in a differential motion of the two rods, which in turn causes the output linkage to rotate, and a resultant control movement to occur.



Electro Hydraulic Actuator

In most modern transport aeroplanes, the primary flight controls operate through Power Control Units (PCU), as shown on the next page, which utilise the muscle of hydraulic power to activate the control surfaces. These units respond directly to signals from the AFCS computer and do not require independent servo actuators. The signal from the AFCS computer is then fed to a solenoid, which operates a **Transfer Valve** within the hydraulic system. This activates the control surface and a position transducer provides a feedback to the computer.



AUTOPILOT TERMINOLOGY

The following terminology is associated with modern complex autopilot systems:

Rate Damping Systems

These systems have more to do with stability than control. They will not return an aeroplane to a specific attitude, but prevent unwanted divergence rates from developing, and help to smooth rate demands commanded by the flight crew.

System Protection

This system prevents control surface runaway, or any other undesirable malfunction, by limiting the actuator authority.

Comparators

These devices compare the outputs from both the sensors and the actuators. If the attitude change sensed is in the same direction as the actuator is moving the control surface, the comparator will automatically disconnect the circuit.

Rate Trigger Systems

The characteristics exhibited by an aeroplane during a system runaway are very marked, and are significantly different from those expected in normal flight. This system automatically disconnects the autopilot if the rate exceeds a set threshold.

Simplex System

This is a single automatic control system, allied to a number of sub-channels. Some of the components may be duplicated, although a single failure elsewhere renders the system unserviceable. This type of system is alternatively known as a **Single Non-Redundant System**.

Multiplex Systems

These are systems which comprise of two or more independent simplex systems, and sub-channels, such that in the event of a failure of a system or sub-channel, the remaining system is capable of performing the controlling function. The number of systems and sub-channels are qualified as, **Duplex, Triplex, and Quadruplex**.

Duplex Systems

These systems incorporate two complete control systems (**Lanes**) for each channel. They use independent sensors, computers, actuators, and actuator power supplies. The two control systems connect to the same control surface, and in the event of a runaway in one lane, the other lane senses a disturbance and applies a corrective actuator input. The speed of response of this system (to a runaway) can be improved by using a comparator system, which continually monitors the positions of the individual actuators, and if one is detected to be diverging at a significant (pre-set) rate, that channel automatically disengages. This system is therefore termed a **Single Failure Survivable system**.

Triplex Systems

These systems are an extension of the duplex system, and utilise three lanes on a control channel rather two. Monitoring of all three lanes occurs continuously and a comparator circuit is fitted to detect a divergence of any lane. Voting then takes place within the system to decide which lane is divergent, and that lane closes down, thus downgrading the operational channel to a duplex system.

Quadruplex Systems

This system uses four lanes per channel, and therefore provides additional redundancy.

Model Following

This system is for use in association with another system, such as a Duplex system. Programmed into the AFCS computer are the flight characteristics of the aeroplane using this software fix. The computer then uses this data to determine the anticipated response for a given control demand and compares this with the response detected by the sensors. This effectively gives a Duplex system the properties of a Triplex system.

Automatic Change of Gain

When shutting down a lane, the remaining lanes must carry the additional load. This is achieved by automatically adjusting the gain of the system so that a given disturbance demands a greater movement, or rate of movement from the remaining actuator.

Monitoring

This term applies to multiplex systems, where comparisons are made either between two or more outputs (or inputs) or between an output (or input) and a selected datum. If the values exceed a preset limit, the system automatically disconnects.

Duplicate-monitored

This term refers to a system comprising of two systems, which have separate power supplies, and operate in parallel. The systems are either self-monitoring or have their outputs checked by parallel comparator circuits. Only one system is active at any given time, whilst the other system is on, but acts only as an active standby system. If a fault is detected the standby system automatically switches over, and becomes the active system.

CROSS COUPLING

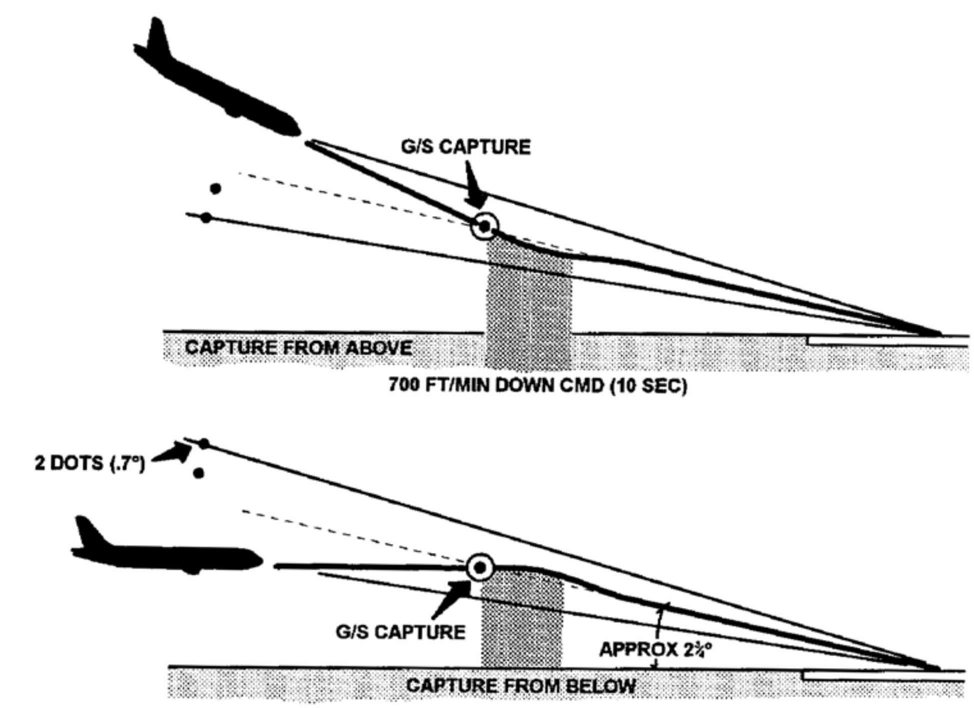
This is the process by which a yawing motion is set up when the ailerons are used to turn an aeroplane, which subsequently has to be countered by the application of some rudder. Additionally, the banked aeroplane also has its lift vector tilted toward the lower wing, so that the vertical component is reduced, which has to be recovered by increasing the aeroplanes angle of attack (pitch attitude). These factors are all taken into account and, in order to obtain a balanced turn, although the primary action comes from the roll channel, secondary inputs from both the pitch and yaw channels are added. In two axis systems the roll and pitch channels normally work together to achieve the necessary balance.

ILS COUPLING

Whenever an aeroplane navigation receiver is tuned to a specific Localiser, it also automatically tunes the glide slope. When the localiser and glide slope signals become strong enough they are captured and fed into the autopilot, thus enabling it to fly the aeroplane on an approach to the runway, which is normally done in two stages.

- First, the aeroplane intercepts and captures the localiser signal. A mix of the pre-selected heading and localiser signals enables the autopilot to align the aeroplane with the extended runway centre line.
- Second, the aeroplane captures the glide slope beam, and the autopilot modifies the pitch attitude so that the aeroplane flies down the beam.

During an automatic approach, create an allowance for the effects of crosswinds, otherwise the aeroplane would take up a position downwind of the localiser beam centre, causing the aeroplane to fly a stand off track parallel to the beam. An extra signal equivalent to the magnitude of the aeroplane drift must therefore be fed into the autopilot to correct for the crosswind, thus when the aeroplane correctly establishes on the localiser beam, the glide path will be intercepted. The mode prior to interception of glide slope is either ATTITUDE HOLD, ALTITUDE HOLD, INDICATED AIRSPEED HOLD, OR VERTICAL SPEED HOLD, but the selected mode automatically cancels when the aeroplane captures the glide slope.



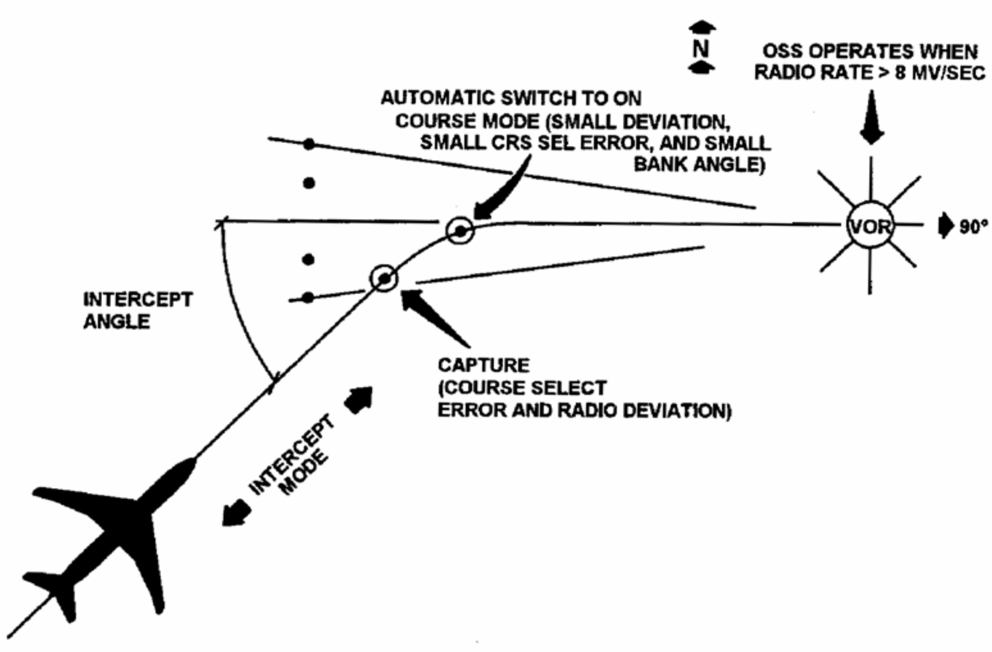
When the vertical beam sensor detects that the aeroplane is approaching the glide slope beam centre, the capture mode is automatically triggered, instructing the autopilot to achieve a vertical descent of approximately 700 ft. per minute. After a short period (approximately 10 seconds) during which the glide slope error signal reduces to approximately zero, the glide slope track mode engages. As the aeroplane approaches the runway, both the glide slope and localiser beams converges, and to reduce the possibility of the aeroplane entering an oscillatory motion, the error signals attenuate as an inverse function of radio altitude. During an automatic approach, some form of beam deviation warning is fitted, and this is indicated either via a **Beam Deviation Light** in association with a Land status light, or as part of the EADI. Deviation warning is normally given as a flashing localiser or glide slope indication.

VOR COUPLING

VOR coupling is similar to that in the Localiser mode, except in this instance the VOR captures when a lateral beam sensor selects it. As the aeroplane approaches the selected radial, the autopilot automatically captures it when the deviation is approximately one dot (5°).

Prior to capturing the beam, the aeroplane is normally on a pre-selected heading or alternatively on another intercept mode, and this mode automatically discontinues whenever the VOR capture mode initiates. Tracking of the VOR radial occurs when the deviation signal, heading error signal, and roll attitude signal approaches zero.

As the aeroplane nears the VOR transmitter, its error signals tend to become erratic due to **Beam Convergence**, and the **Cone of Confusion**. To prevent this it is usual to cut off the VOR signal as the aeroplane nears the ‘Cone of Confusion’, and this is achieved by an **Over Station Sensor (OSS)**, which automatically deselects the VOR signal for a preset time period, until the aeroplane has over flown the ‘Cone of Confusion’.



STABILITY PROBLEMS

The two most common stability problems encountered by an aeroplane are:

- **Dutch Roll**. This is an oscillation in yaw and roll, where roll predominates yaw. This motion is most common on aeroplanes with swept wings, and if not dampened out by the natural stability of the aeroplane, leads to a divergent Phugoid. To correct this, a **Yaw Damper System** is fitted, which is designed to pick up extremely small deviations in yaw away from the flight path, and automatically applies inputs to the rudder.

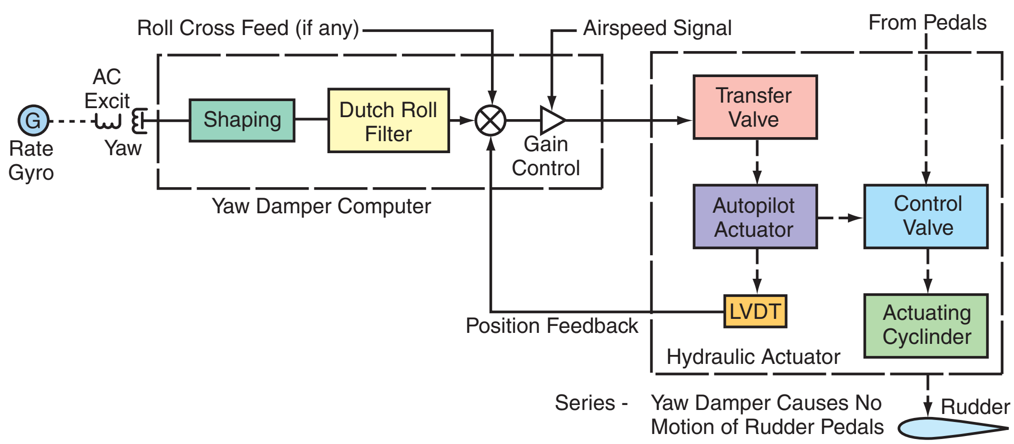
- **Tuck Under.** This is the phenomenon encountered in aeroplanes flying at high subsonic or transonic speeds. At these speeds, there may be a significant rearward movement of the centre of pressure, and a subsequent uncontrollable 'nose down' motion. To compensate for this a **Mach Trim System** is fitted.

YAW DAMPER

This unit is designed to counteract the yawing tendency at an early stage of onset, and before the roll motion associated with the onset of **Dutch Roll** can develop. This system is completely independent of the AFCS and is operational whenever ON. It is however an operational requirement that all 'Yaw Damper systems' are switched on, and are operational prior to take-off, although only one system is active at any one time. Shown below is a typical system.

The sensor unit is a rate gyro that has its sensitive axis vertical, and is contained within a unit known as the **Yaw Damper Coupler**. This unit also contains the yaw damper circuit computer, which provides the functions of:

- Filtering the detected error signals and comparing them to a reference signal before passing a command to the next stage of processing. This additionally removes any error that may be caused by fuselage flexing.
- Integration of the filtered signals to form the servo signal input
- Amplification of the servo signal



The amplified servo signal, fed via a **Transfer Valve**, is independent of the normal AFCS actuator, and directs hydraulic fluid under pressure to the yaw damper rudder actuator. The inputs from the yaw damper and from the rudder controls are normally arranged in **Series** or **Parallel**, which relates to the method by which the system affects the rudder control. In a Series system there is no resultant rudder pedal movement, but in a Parallel system operation of the yaw damper system results in rudder pedal movement. Some systems alternatively operate in Series or Parallel dependent on the phase of flight. During cruise the system operates in Series, when limited rudder is required, but during the **Autoland** stage, when increased rudder authority is required, the system switches over to a Parallel system. A feedback loop is also fitted so that when the yawing motion stops, the rudder automatically returns to its normal position.

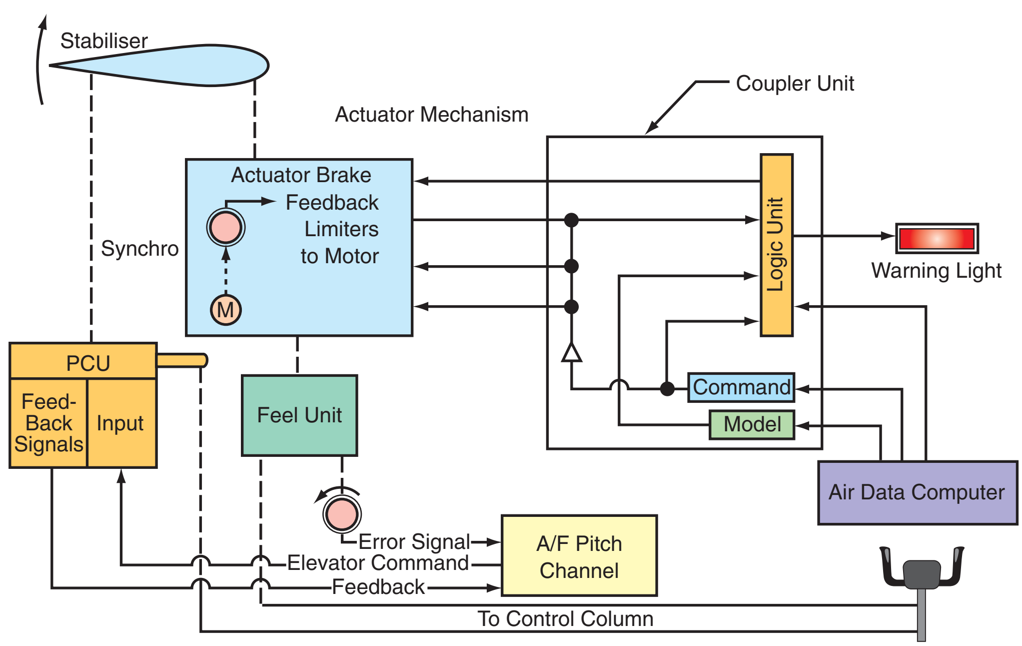
The yaw damper signal, for a given rate of oscillation, is inversely proportional to airspeed, so a signal from the Air Data Computer is required. The yaw damper signal may also require modification for different configurations. If this is necessary, a signal from the flap position indicator circuit applies to a gain circuit on the yaw damper output, increasing the rate of response when the flaps extend.

Operation of the Yaw Damper system is monitored on a flight deck indicator, which also shows the position of the rudder. A provided test circuit simulates a yaw oscillation by applying torques to the rate gyro sensor. The displaced gyro in turn generates an error signal, which displaces the rudder, and its respective movement relays to the flight deck indicator.

MACH TRIM SYSTEM

This system is attached to the pitch channel, and like the yaw damper does not depend on the AFCS for its operation. Shown on the next page is a typical system.

The heart of the system is the coupler unit, which receives signals, corresponding to Mach number, from the ADC. If these signals exceed a predetermined value (depending on the aeroplane type) the trim coupler unit releases the brake, and the speed signal from the ADC feeds to the motor. This causes the stabiliser to move in such a way to drive the elevators upward, and counteract any tendency for the aeroplane to **Tuck-Under**. This system is also internally monitored, and if the system fails, a fail indicator light illuminates on the flight deck.



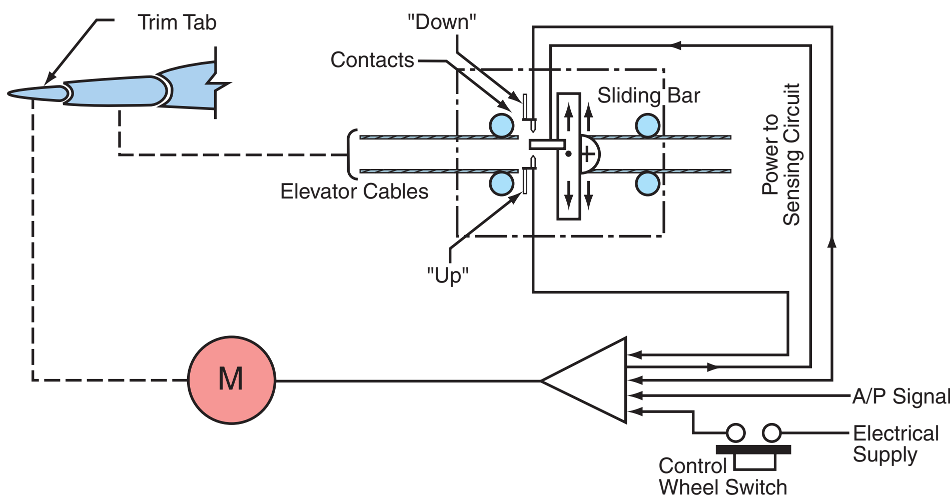
AUTOMATIC PITCH TRIM CONTROL

Automatic trim control in an AFCS is usually only provided about the pitch axis, and must be active whenever the AFCS is engaged.

The design of the trim control varies significantly from type to type, but for the purpose of this manual only two types are compared.

SMALL GENERAL AVIATION TWIN-ENGINE AEROPLANE

In this type of aeroplane, the elevators operate by a system of cables powered either manually or electrically. Shown below is a typical automatic trim system:

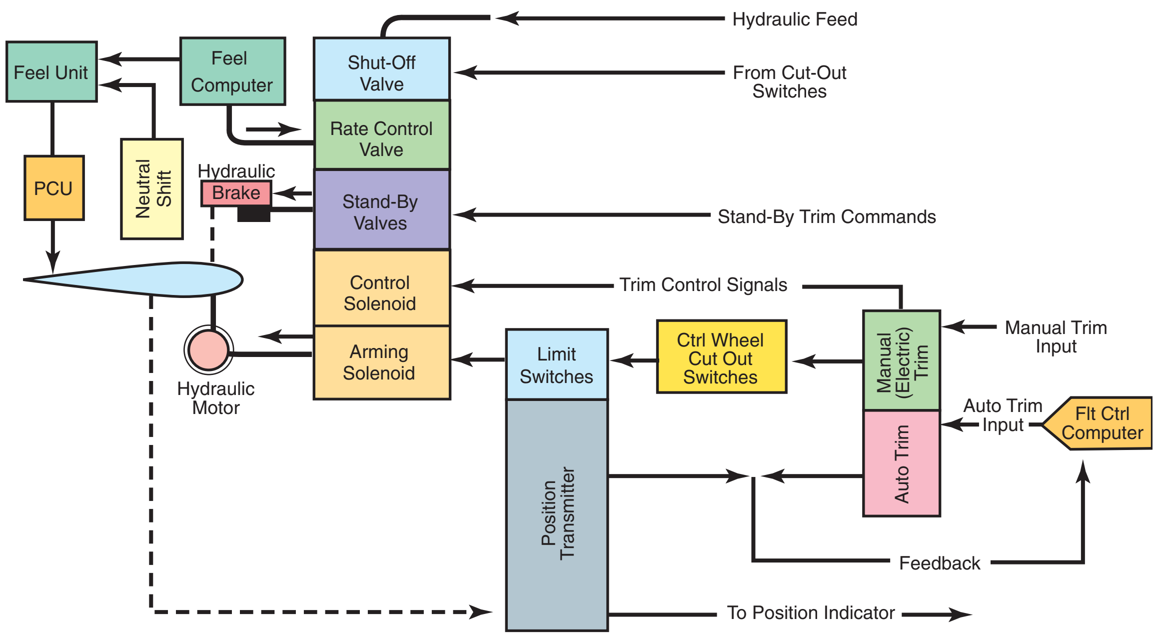


In this system, a sensor compares the tension of the **Up** elevator control cable with that of the **Down** control cable, and if the elevator is subject to an **Up** command, the sensor detects the imbalance in the cable tension. This in turn produces an error signal, which amplifies and forward to a trim motor. The trim tab drives downward until the control cable tensions reduce to a minimum. A **Flap Compensation** circuit may also be added to the command signal detection circuit to compensate for any pitch change, which may occur when the flaps are extended, and operates via a relay attached to the flap position indicator.

If the aeroplane is flown manually, trimming via a control switch controls the application of power to the servo system.

LARGE TRANSPORT CATEGORY JET ENGINE AEROPLANE

In this type of aeroplane, the normally hydraulically powered elevators and variable incidence tailplane, use an operating system that is much more complex. The control of the tailplane is affected by an automatic trim signal, which controls a hydraulic motor. The trim motor connects via a clutch assembly and screw jack to the tailplane. Shown below is a block schematic of such a system.



In this system, automatic trimming is achieved by moving the variable incidence tailplane only, whilst the elevators are left with full movement authority, to control any pitch changes commanded by the AFCS. If the detectors detect an elevator too far out of the faired position for several seconds, the tailplane moves up or down as required, reducing the need for the autopilot to hold the elevators out of the faired position. When the elevators get close to the faired position, the detectors stop the operation of the tailplane. For example, if a nose down demand from the AFCS is ordered, the pitch control channel activates the Power Control Unit (PCU), and this drives the elevator downward. At the same time, a signal is supplied to the trim circuit, and when the relative position between the stabiliser and elevator reaches a threshold value, the trim circuit is activated. A signal then passes to the solenoid, activating the hydraulic motor and repositioning the stabiliser. A new elevator **neutral** position establishes, such that a signal feeds back to the elevator positioning circuit, repositioning the elevator accordingly. The reason for elevator displacement or neutral shift is to augment the control authority of the tailplane, achieved via a neutral shift rod mechanism.

Limit switches prevent the tailplane from driving beyond pre-set limits, and a stabiliser trim indicator on the flight deck shows the position of the trimming surfaces. A fault detection circuit providing the following warnings monitors the system:

1. A warning light illuminates on the flight deck if the automatic trim system fails.
2. An aural alert sounds when detecting an excessive trim input.

INTERLOCKS

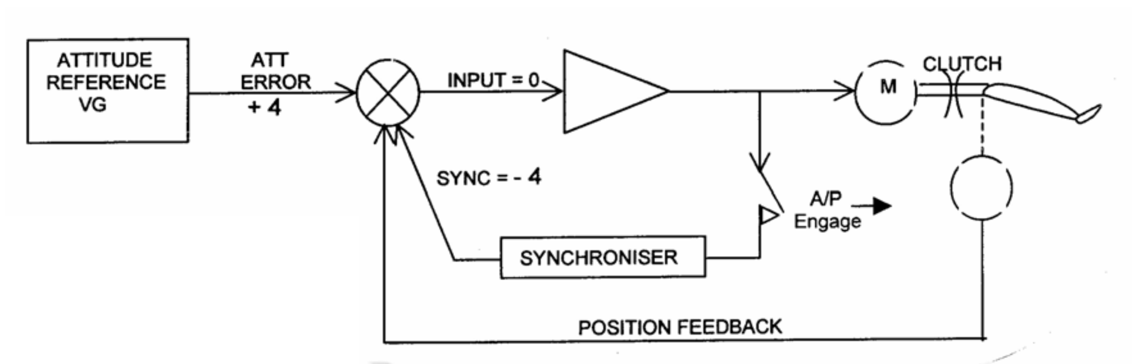
Before an autopilot can safely engage with the flight controls, certain conditions must be fulfilled. These conditions vary slightly between autopilots, but take the form of a number of switches called interlocks, connected in series. A list of some of the conditions that must be operative before the interlocks allow autopilot engagement is as follows.

- AC and DC power supplies in tolerance
- Trimmed and erected gyroscopes
- Yaw damper system engaged
- Automatic pitch trim system available
- Mach trim system available
- Do not press instinctive Cut-Out (Disengage buttons)
- Operative CADC
- Manual controls in their neutral position (i.e. Control wheel and column, turn knob, pitch switch).

Only after enabling all the interlocks, can the system engage.

SYNCHRONISATION

It is important that there is a smooth transition from manual to automatic flight (i.e. if an aeroplane is in a climb, it would not be desirable to have a zero pitch datum when engaging the autopilot). Prior to engagement, any signal present on a particular channel is passed to a **Synchroniser** circuit, where an opposing signal equivalent to the attitude error builds up, which continues right up until the autopilot is engaged.



For example, if the aeroplane pitches up prior to engagement, the resulting attitude error signal passes through the channel, and develops an opposing signal in the synchroniser circuit. When the autopilot engages the input to the synchroniser disconnects, and the synchroniser level freezes at its current value, providing a datum for the system.

INSTINCTIVE CUT OUT

It must be possible for the pilot to disengage the autopilot via **Instinctive Cut-Out Buttons**, which are located on the control column. These buttons are positioned on the outside of the control wheel, so the hand not controlling the throttles can operate them. It should also be possible to disengage the autopilot by manual operation of the control column. If disconnecting the autopilot via the ‘instinctive cut out’, the flight crew receives a short-term audio and visual warning.

INTRODUCTION

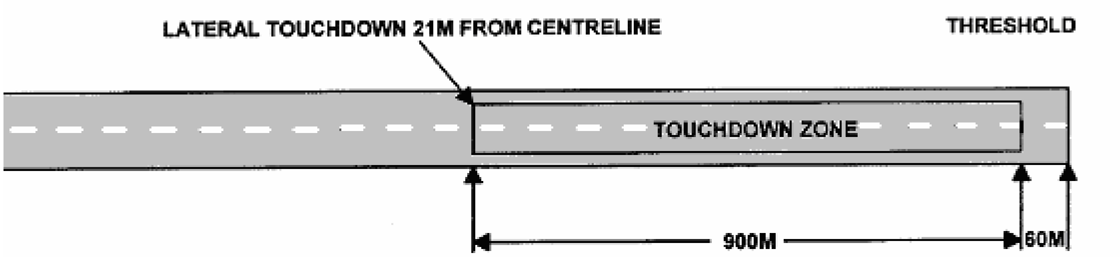
If the weather is poor at a particular destination it is still possible to land there and not divert to another destination, saving the expense of transporting passengers to their original destination by other means, by using an **Automatic Landing System**. Any diversion also means that the aeroplane is out of position, and must be flown to its original destination without fare paying passengers.

To certify an aeroplane with an automatic landing capability, it must comply with the minimum requirements laid down in **JAR-AWO (ALL WEATHER OPERATIONS)**.

BASIC REQUIREMENTS FOR AN AUTOMATIC LANDING SYSTEM

The basic requirements for an Automatic Landing System are:

- The safety achieved by an automatic landing must not be less than a manual landing, and the risk of a fatal accident should be better than 1×10^{-7}.
- The flight crew must be able to adequately monitor the landing phase, so that if a critical malfunction occurs, the autopilot can be manually disengaged, and manual control of the aeroplane taken at any time, with the minimum skill required, keeping the aeroplane under control.
- The aerodrome must have the required and suitably calibrated radio aids.
- It is improbable that the landing performance is outside the following limits, as shown in the following diagram.



a. Touchdown must occur between 60 m – 900 m after the threshold.
b. The outboard landing gear must be no more than 21 m from the runway centre line assuming a runway width of 45 m.

AUTOMATIC LANDING SYSTEM TERMINOLOGY

In order for an aeroplane to carry out a fully automatic approach and landing, it is necessary that the aeroplane be fitted with two or more autopilots, and an autothrottle system. The utilisation of multiple systems ensures that it is unlikely that in the event of a single major component failure in the autopilot system, it does not cause the aeroplane to deviate away from its approach path. The following terminology is associated with flight failures of an automatic landing system:

Fail-Operational

An automatic landing system is Fail-Operational if, in the event of a failure below the alert height during the approach, flare, and landing flight phase, the landing can still be completed by the remaining part of the automatic system. In the event of a failure, the automatic landing system downgrades to a **Fail-Passive** system. A Fail-Operational system is alternatively known as a **Fail-Active** system.

Fail-Passive

An automatic landing system is fail-passive if, in event of a failure, there is no significant out-of-trim condition or deviation of flight path or attitude, but the landing cannot be completed automatically. A Fail-Passive system is alternatively known as a **Fail Soft** system.

Dual-Dual

This term is used by some manufacturers to define a twin Fail Operational control system, having two passive monitoring systems. This type of system is not the same as a Duplex system, since the control systems may or may not be active simultaneously. In the event of a monitor detecting a failure in its associated system, the second system with its respective monitor automatically switches on.

Alert Height

The alert height is a specified radio height, based on the characteristics of the aeroplane and its Fail Operational Landing System. In operational use, if a failure occurs above the alert height in one of the required redundant operational systems in the aeroplane, (including, where appropriate, ground roll guidance and the reversionary mode in a Hybrid System) the approach discontinues, and a Go-Around (GA) 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 is ignored and the approach continues.

Decision Height

This is the wheel height above the runway elevation by which a GA must be initiated unless adequate visual reference has been established, and assessment of the aeroplane position and approach path is satisfactory to continue the approach and landing safely.

AUTOMATIC LANDING SYSTEM EQUIPMENT REQUIREMENTS

The following list of systems and equipment is required in aeroplanes with automatic landing systems, in order to achieve the following decision heights:

Decision Height 199-100 ft

1. Autopilot with an ILS coupling mode. (Note: A flight director system with an ILS coupling mode may be approved for use following failure or disconnect of the autopilot)
2. Autothrottle (unless it can be shown that speed control does not add excessively to the crew workload)
3. Radio altimeter
4. Excess ILS deviation warnings

Decision Height 99-50 ft

1. As for decision height 199-100 ft
2. Autopilot with autoland mode

Decision Height Below 50 ft

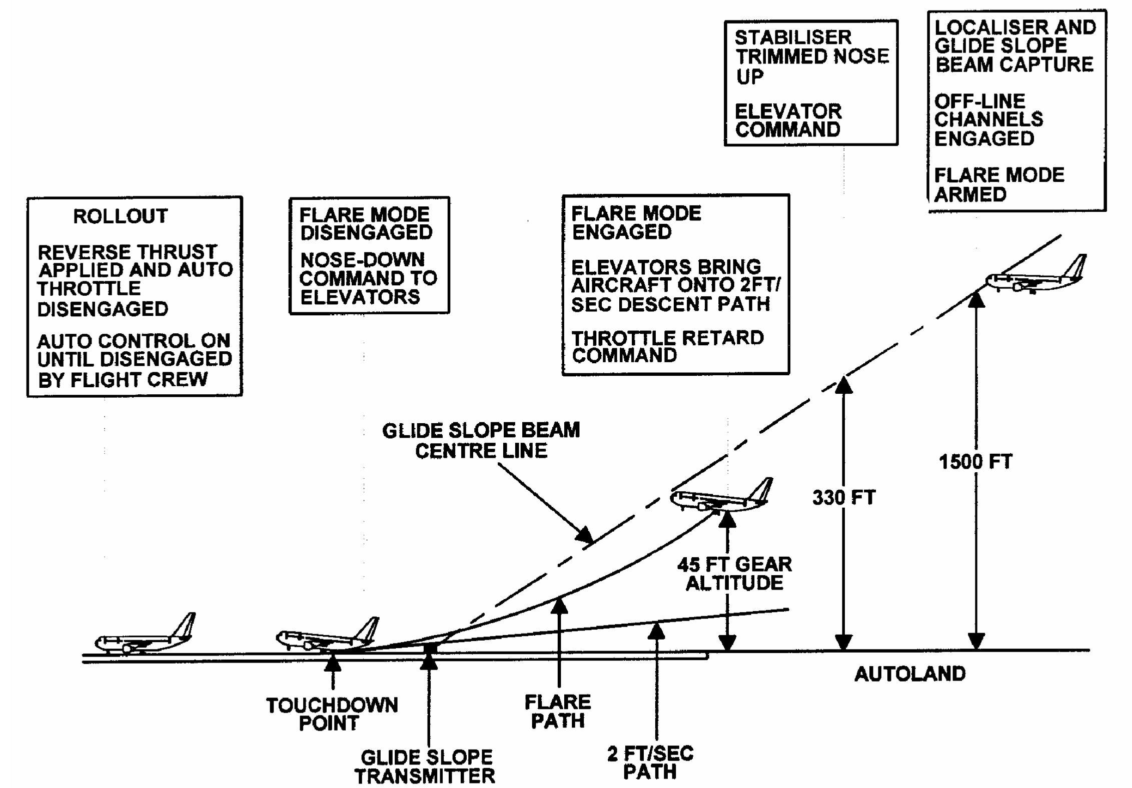
1. As for decision height 199-100 ft
2. (a) Autopilot with a fail-operational autoland mode and an automatic missed approach mode
   OR
   (b) Autopilot with automatic landing and missed approach modes and a landing guidance display

No Decision Height

1. As for decision height below 50 ft
2. (a) Autopilot with a fail-operational ground roll mode.
   OR
   (b) Fail operational head-up ground roll guidance display
   OR
   (c) Autopilot with ground roll mode and a head up ground roll guidance display
3. Anti-skid braking system

AUTOMATIC APPROACH, FLARE, AND LANDING SEQUENCE

The following diagram shows the stages involved during the automatic approach, flare, and landing phases of flight of an aeroplane fitted with a triple channel autopilot, such as that fitted to a Boeing 757.



Depending on the number of channels engaged, the system performs an automatic landing with a Land 2 or Land 3 status, which displays on the EADI.

> **Land 2** signifies two autopilot channels engaged giving the system a Fail Passive capability.
>
> **Land 3** signifies three autopilot channels engaged giving the system a Fail Operational capability.

During the cruise and approach to an airport, a single autopilot normally engages and controls the aeroplane along the designated route. As the aeroplane approaches the airport, the flight crew manually selects the other autopilots, and when selecting the APPROACH MODE (APP) on the AFCS control panel, the localiser/glide slope, together with the two remaining autopilots, arm. As the aeroplane passes through 1500 ft radio altitude, with the localiser and glide slope already captured, the two off line autopilots automatically engage. This indicates to the flight crew by a **Land 3** status, and the aeroplane continues to fly down the glide slope.

> **Note:** If a single failure occurs between 1500-200 ft radio altitude, the system downgrades itself to a Fail-Passive system, and a Land 2 status annunciates.

At a radio altitude of 330 ft, the aeroplane is trimmed nose up by an automatic adjustment of the variable incidence tailplane. As the aeroplane passes through the alert height (normally 200 ft radio altitude) the reversion to Land 2 due to a fault is inhibited until the aeroplane is below 40 kt during the roll out.

When the aeroplane is 45 ft above the ground the flare mode automatically engages, and the aeroplane rate of descent progressively reduces to achieve a rate of 2 ft. per second on touchdown. At the same time, the Autothrottle also reduces the amount of available engine thrust in order to maintain the flare path. At 5 ft gear altitude, the flare disengages and there is a transition to touchdown, and the subsequent roll out mode. The Land status remains engaged until de-selected by the flight crew.

These systems are designed to carry out automatic landings under all visibility conditions, and must have a reliability factor better than 1×10^{-7}.

WEATHER MINIMA

In low visibility operations, the given weather limits for landing are in terms of the following:

Runway visual range (RVR)
This instrumentally derived value represents the range at which high-intensity lights can be seen in the direction of landing along the runway. ATC transmits these readings the aeroplane, and provide the flight crew with the visibility conditions that currently exist at the airport.

Decision height
This is the wheel height above the runway threshold by which a go-around (GA) must be initiated, unless adequate visual reference has been established, and the position and approach path of the aeroplane have been visually assessed as satisfactory to safely continue the approach or landing.

The national licensing authorities specify minimum values for these two quantities, referred to as Weather Minima, for various types of aeroplanes, and for various airports.

ICAO CATEGORISATION FOR LOW VISIBILITY LANDING CAPABILITIES

This system is based on the principle that the probability of having an adequate short visual reference, for the range of permitted decision heights, should be as high as possible.

Category 1: Operation down to a minimum 200 ft decision height, and a RVR of 800 m, with a high probability of approach success.

Category 2: Operation down to minima less than Cat 1 but not less than 100 ft decision height and RVR of 300 m, with a high probability of approach success.

Category 3A: Operation down to, and along the surface of the runway, with external visual reference during the final phase of the landing down to an RVR minimum of 200 m.

Category 3B: Operation to, and along the surface of the runway and taxiways with visibility sufficient only for visual taxiing comparable to a RVR value in the order of 50 m.

Category 3C: Operation to, and along the surface of the runway and taxiways without external visual reference.

These categories also serve as an indication of the stages through which automatic approach and automatic landing development progresses, and thus designate the capabilities of an individual AFCS.

The designated capabilities of the ground guidance equipment available at a particular airport are as listed below.

- ILS localiser and glide path
- Approach lighting
- Runway and taxiway lighting

In connection with automatic landing systems, the term **All Weather Operations** is frequently used. This term usually means that there are no weather conditions preventing an aeroplane from successfully taking-off and landing. This is not strictly correct (e.g. an aeroplane cannot perform a landing task in wind conditions in excess of those for which it has been certificated, or on a runway which, because of contamination by water, slush, or ice, is not fit for such an operation).

THE FUNDAMENTAL LANDING REQUIREMENT

During an automatic landing, an aeroplane should be controlled in such a way that:

- Its wheels make contact with the ground comfortably within the paved surface of the runway and that the landing point is not too far down the runway.
- It lands at a very low vertical velocity in order to avoid collapse of the landing gear.
- The speed at touchdown should be sufficiently low to allow the aeroplane to be brought to a halt within the remaining length of the runway.

To facilitate the above, the aeroplane's:

- Final rate of descent should be no greater than 1 to 2 ft per second.
- Airspeed should be reduced from $1.3V_s$ during the approach to 1.15Vs, by a progressive reduction of engine thrust during the landing flare.
- Wings level prior to the actual landing, and any drift 'Kicked-Off' before touchdown.

To achieve all of the above requires that there is control of the aeroplane about all three axes simultaneously.

Any such system, as well as being capable of achieving at least the targets listed above, also requires the design for:

- Achieving the highest integrity and reliability of systems, bearing in mind that they need to be entrusted with very considerable authority over the controls of an aeroplane, including the throttles, and in the presence of the ground.
- The provision of adequate monitoring information on the progress of the approach and landing manoeuvre enables the flight crew to take over, under the most critical conditions of a system malfunction in the presence of the ground.
- The substitution of the flight crew's direct vision with an automatic externally referenced guidance system, having an integrity and reliability of the same high order as that demanded of the 'on board' system.

SYSTEM RELIABILITY AND INTEGRITY

Devices designed to limit the authority of automatic control systems in the event of runaway conditions that may result from malfunctions, may be incorporated in more conventional control systems. These are normally only effective for the intended purpose down to any specified **break-off** height (i.e. the approach height at which a control system is disengaged, but these systems do not satisfy the requirements for systems designed for autoland).

The setting of safety devices is dictated by the conflicting requirements listed below:

- They must limit the effect of a runaway such that a safe recovery can be effected by the flight crew.
- They must allow sufficient authority to the control system so that the required flight path is accurately in the presence of disturbances.

A further factor limiting the application of safety devices (in the manner of conventional control systems) is their inability to protect against passive failures. While not producing flight path changes directly, these failures would nevertheless mean that the predetermined and accurate flight manoeuvre of automatic landing, could not be maintained, and so could set up an equally dangerous situation.

It therefore follows that to achieve the objective of automatic landing, the operation of an AFCS must be of such a nature that it does:

- Not disturb the flight path because of an active malfunction.
- Have adequate authority for sufficiently accurate control along the required flight path.
- Warn of a passive failure.
- Not fail to complete the intended flight manoeuvre following an active or a passive failure.

In order to resolve these problems, the concept of **system redundancy** is applied (i.e. the use of multiple systems, operating in such a manner that a single failure within a system has an insignificant effect on the aeroplane's performance during the approach and landing operation).

- The autothrottle remains in control of the engines until reverse thrust is demanded.
- The AFCS remains in control of the aeroplane until the crew disengages it.

At any time from 2000 ft down to the decision height, the flight crew can elect to abort the approach, by pressing the TOGA switch on the throttle, causing the throttles to advance to a pre-set reduced thrust ‘Go-Around’ value. A second press of the TOGA switch commands the throttles to advance the engines to full power. The ‘Go-Around’ phase then interacts with the AFCS causing a GA annunciation on the ADI. The pitch channel generates a ‘pitch-up’ command, and places the aeroplane in the correct climb attitude.

Chapter 27
Thrust Management Systems

INTRODUCTION

Most modern aeroplanes are equipped with systems to control and compute engine thrust. Knowledge of how engine thrust is computed, how it is controlled, and how engine inputs are collected is therefore necessary.

DETERMINING THE THRUST REQUIRED

Thrust curves and charts published in the aeroplane manual and in the performance manuals are used to determine the required **Engine Pressure Ratio (EPR)** and/or **Fan Speed (N_1)** for any desired engine rating, which depends solely on the prevailing ambient temperature and barometric pressure. Time, engine speed, and the Exhaust Gas Temperature (EGT) limit the take-off rated thrust. Maximum continuous, maximum climb and maximum cruise thrust ratings are limited by EGT for a given length of time, or continuously, as defined below.

ENGINE PRESSURE RATIO (EPR)

This is the amount of useful thrust developed by an engine. It is the product of the mass of air passing through the engine and its velocity at the exhaust nozzle, minus the drag due to the air passing through the engine. Comparing the air pressure across the engine (i.e. the ratio of the exhaust pressure to the compressor inlet pressure) gives the Engine Pressure Ratio (EPR), which is an indication of the thrust output from the engine. EPR is usually a percent thrust value.

RPM, N1, N2, OR N3

These are normally a percentage of the maximum value.

EGT

The exhaust gas temperature must be monitored in order to prevent excessive heat from damaging the turbine.

RATED MAXIMUM CONTINUOUS THRUST

This is the amount of thrust that is approved for unrestricted periods of use, and according to JAR, is defined in the aeroplane flight manual.

MAXIMUM CONTINUOUS THRUST (MCT)

This is the amount of thrust that is authorised for emergency use at the discretion of the pilot. Aeroplane certification requirements and climb operations, as determined by the airframe manufacturer, use this amount.

MAXIMUM CLIMB THRUST

This is the maximum thrust approved for the climb phase, which on some engines is identical to the rated maximum continuous thrust level. To select this rating, position the throttle to give the required EPR or N_1 for the prevailing climb profile and engine inlet temperature. The climb thrust curves or charts are contained in the aeroplane performance manual.

MAXIMUM CRUISE THRUST

This is the maximum approved thrust for cruise operation. Closely adhere to these thrust limitations because they are allied to specific warranty limitations, and are established by the engine manufacturer to ensure optimum operation and engine life.

CALCULATION OF CLIMB AND CRUISE THRUST

To determine the necessary thrust to obtain the desired climb and cruise performance, the flight crew must consult the charts in the aeroplane performance manual. A simplified method to calculate the amount of reduced thrust is the assumed temperature method. This is based on using the take-off thrust and aeroplane speeds for an assumed temperature higher than the actual ambient temperature (OAT). The assumed temperature procedure enables the flight crew to make thrust reductions according to the prevailing conditions, and within practical limits.

CLIMB

This is the thrust established by adjusting the throttle to obtain the appropriate parameter indication (N_1 or EPR) in accordance with the published climb thrust setting charts or curves. On some engines, the maximum continuous and maximum climb thrust levels may be the same.

During the climb at a fixed throttle setting, as **Total Air Temperature (TAT)** falls with increasing altitude, the N_1 and EPR values progressively increase in accordance with the thrust curves. Normally only one or two throttle adjustments should be necessary throughout the climb, depending on whether a high-speed or long-range climb is performed. Significant temperature and speed deviations may however require slight throttle adjustments in order to avoid exceeding the thrust curve values. By comparison, typical changes in the ambient temperature and Mach number will not require any such adjustments.

Continuous monitoring of the EGT is necessary throughout the climb so that the established maximum climb EGT is not exceeded.

CRUISE

Upon reaching the selected flight level, maintain the climb thrust only long enough to allow the aeroplane to accelerate to its designated cruising speed. To maintain the desired speed, the required N_1 or EPR is set in accordance with the cruise charts, or curves, which are applicable to the prevailing cruise conditions.

Cruise thrust values are based on cruising speeds, in KIAS and Mach number, aeroplane gross weight, TAT, and pressure altitude. The charts and curves cover all realistic combinations of these factors within the operational scope of the engine and aeroplane up to, and including the maximum cruise thrust rating.

The Maximum Cruise Thrust is an engine warranty limitation, and should thus not be exceeded whilst attempting to maintain a given altitude and speed.

CRUISING METHODS

Procedures used for cruising depend primarily on the length of time spent in the cruise during a flight. For short flights, within given periods, depending on the aeroplane-engine type and cruising environments, the fixed throttle cruise provides a favourable or acceptable balance of fuel consumed versus time saved. Once the thrust level has been set to obtain the desired cruising speed, the throttle position may remain fixed throughout the cruising portion of the flight.

ELECTRONIC ENGINE CONTROL (EEC)

The first Electronic Engine Control System (EEC) was a supervisory control, utilising proven hydro-mechanical controls. The major components in this system included the control itself, the fuel control of the engine, and the bleed air and variable stator vane control.

In this system, the flight crew simply moved the thrust lever to a desired thrust or maximum climb position, and the control system automatically adjusted the EPR to maintain the thrust rating irrespective of changes in flight, and ambient conditions. The control also limited the engine speed and temperature, which ensured that the engine/engines were operated safely throughout the entire flight envelope.

If a fault were to occur in this system, the control automatically reverts to a hydro-mechanical system, whilst maintaining the required thrust level. Full reversion to the hydro-mechanical system can also be instigated at any time.

Some electronic control systems alternatively function as a limiter only. For example, if the engine shaft speed or EGT approaches the limits of safe operation, then an input is automatically made to a fuel flow regulator to reduce the fuel flow, and thus maintain the desired shaft speed or EGT at a safe level.

Modern Full Authority Electronic Engine Control (EEC) are fully redundant systems, which control all engine functions, and also eliminate the need for a back-up hydro-mechanical control system. This control system is more commonly referred to as a Full Authority Digital Engine Control (FADEC) system.

FULL AUTHORITY DIGITAL ENGINE CONTROL (FADEC)

FADEC is a digital electronic fuel control system, specifically for use on gas turbine engines, and functions during all engine operations. It includes total electronic engine control, and operates with the Flight Management Computer to schedule the fuel to the engine. Shown on the next page is a typical FADEC system. One of the basic purposes of FADEC is thus to reduce the flight crew workload, particularly during the critical phases of flight. Achieved by the FADEC's control logic, it simplifies the power settings for all engine-operating conditions. The thrust levers thus achieve engine thrust values for a set lever position, regardless of the flight or ambient conditions. For example, assume a given EPR at a particular OAT; if the OAT consequently changes, the system is designed to automatically adjust the amount of fuel supplied to the engine, in order to maintain a set EPR.

The FADEC system establishes the amount of engine power through direct closed-loop control of the EPR, which is the thrust rating parameter. The selection of EPR normally calculates as a function of thrust lever angle, altitude, Mach number, and TAT.

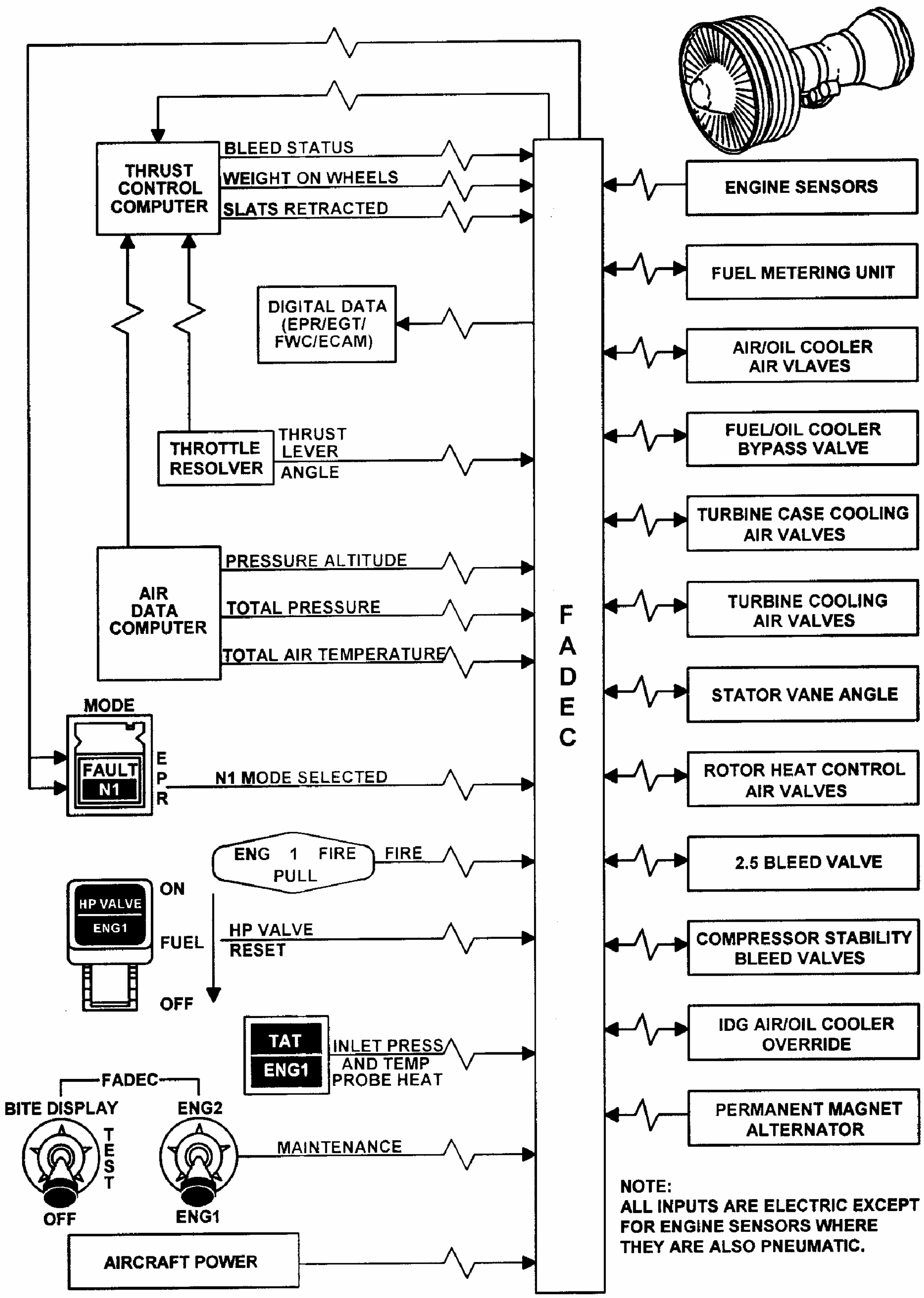
The Air Data Computer supplies altitude, Mach number and TAT to the control system, whilst sensors provide measurements of engine temperatures, pressures, and speeds. This data is used to provide automatic thrust rating control, engine limit protection, transient control, and engine starting. The control system also implements EPR schedules to obtain the EPR rating at various throttle lever angle positions, and provides the correct rating at a constant throttle lever angle during changing flight or ambient conditions.

The FADEC has the following advantages over a mechanical system:

- The system requires no engine adjustment, and therefore no engine running, which saves fuel.
- The system reduces fuel consumption through improved engine bleed air control.
- The system fully modulates the active clearance control systems, producing a substantial benefit in performance by reducing the engine blade tip clearances.
- The higher precision of the digital computer ensures more repeatable engine transients (i.e. acceleration/deceleration) than that possible with a hydro-mechanical system. The latter is subject to manufacturing tolerances, deterioration, and wear, which affects its ability to consistently provide the same acceleration and deceleration times.
- The system ensures improved engine starts by means of digital schedules and logic that adjusts for measured conditions.
- The system provides engine limit protection by automatic limiting of the critical engine pressures and speeds. Direct control of the rating parameter also prevents inadvertent overboost of the selected rating when the power set.
- The engine idle speed remains constant regardless of changes in ambient conditions and bleed requirements, whereas with the mechanical system, the engine speed changes with ambient conditions.

FADEC takes over virtually all of the steady state and transient control intelligence and replaces most of the hydro mechanical and pneumatic elements of the fuel system. The fuel system solely reduced to a fuel pump and control valve, an independent shut-off cock and a minimum of other additional features, which are necessary to keep the engine safe in the event of extensive electronic failure. FADEC also furnishes information to the engine instrument and crew alerting system.

The FADEC, mounted on the engine compressor casing on anti-vibration mounts, is air-cooled. The figure below indicates the signals transmitted between the engine, mounted components, and the engine/aeroplane interface.


THRUST CONTROL COMPUTER
BLEED STATUS
WEIGHT ON WHEELS
SLATS RETRACTED
DIGITAL DATA (EPR/EGT/ FWC/ECAM)
THROTTLE RESOLVER
THRUST LEVER ANGLE
AIR DATA COMPUTER
PRESSURE ALTITUDE
TOTAL PRESSURE
TOTAL AIR TEMPERATURE
MODE
FAULT
N1
E P R
N1 MODE SELECTED
ENG 1 FIRE PULL
FIRE
HP VALVE
ENG1
ON
FUEL
OFF
HP VALVE RESET
TAT
ENG1
INLET PRESS AND TEMP PROBE HEAT
FADEC
BITE DISPLAY
ENG2
TEST
OFF
ENG1
MAINTENANCE
AIRCRAFT POWER
FADEC
ENGINE SENSORS
FUEL METERING UNIT
AIR/OIL COOLER AIR VLAVES
FUEL/OIL COOLER BYPASS VALVE
TURBINE CASE COOLING AIR VALVES
TURBINE COOLING AIR VALVES
STATOR VANE ANGLE
ROTOR HEAT CONTROL AIR VALVES
2.5 BLEED VALVE
COMPRESSOR STABILITY BLEED VALVES
IDG AIR/OIL COOLER OVERRIDE
PERMANENT MAGNET ALTERNATOR
NOTE:
ALL INPUTS ARE ELECTRIC EXCEPT FOR ENGINE SENSORS WHERE THEY ARE ALSO PNEUMATIC.

The control has dual electronic channels, each having its own processor, power supply, programme memory, selected input sensors, and output actuators. A dedicated engine gearbox driven alternator also provides power to each electronic control channel. If computational capability is lost in the primary channel, the FADEC automatically switches to the secondary channel. If a sensor is lost in the primary channel, cross talk with the secondary channel automatically supplies the necessary information.

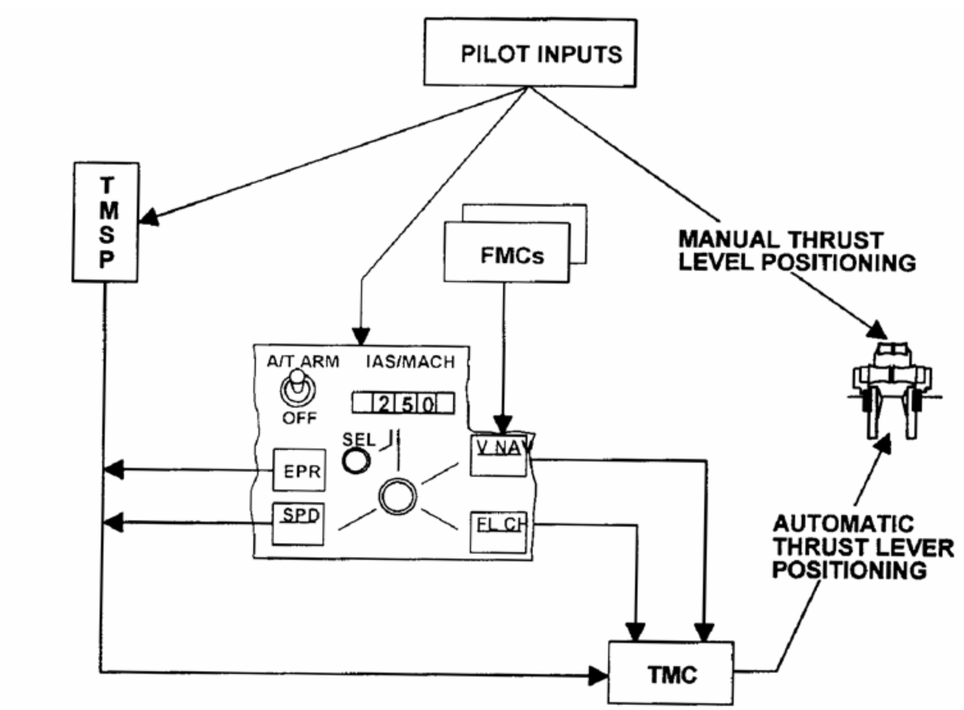
In the unlikely event of the loss of both channels of the electronic control, the torque motors are spring loaded to failsafe positions. The fuel flow goes to minimum flow, the stator vanes are set to fully open (to protect take-off), the air/oil cooler goes to wide-open, and the active clearance control shuts off.

AUTOTHROTTLE (A/T)

The Autothrottle (A/T) system is a computer controlled electromechanical system that controls engine thrust within engine design parameters. The system:

- Computes and displays EPR and/or N1 and speed (IAS or MACH) information.
- Provides automatic control from start of the take off, through the climb, cruise, descent, approach, and go-around or landing.

The throttle position for each engine is controlled in order to maintain a specific engine thrust (N_1 or EPR) or target airspeed, for all flight regimes as directed by the Thrust Management Computer. Fitted in modern jet engine aeroplanes are autothrottles, primarily to conserve fuel. Shown below is a typical system:



It has inputs from a **Thrust Management Computer (TMC)**, which integrates signals from the engines, a Thrust Mode Select Panel (TMSP), and a Flight Management Computer (FMC), and receives signals from the Air Data Computer (ADC). With full forward throttle, the TMC provides maximum engine power without exceeding the specified operating limits during the take-off and go-around phases of flight. The system can however be manually over ridden at any time.

The A/T and AFCS operate together to maintain a specific airspeed and vertical path profile. With the A/T ‘ON’ and either the autopilot or the flight director ‘ON’, one system or the other maintains a set airspeed. When the AFCS mode is controlling the airspeed, the A/T automatically controls a specific engine thrust value, but when the AFCS mode is controlling the aeroplanes vertical flight path, the A/T automatically maintains the desired airspeed through thrust control.

The engines on Airbus aeroplanes are driven by FADEC, but alternatively use electrical signals for thrust control in order to eliminate the weak points of the conventional Autothrottle system, which is mechanically operated. On these aeroplanes, the actual throttle (thrust lever) position does not move automatically, unlike the early auto-throttles, thus making them much more reliable.

Operation of the Airbus system is in either manual or auto-thrust modes. In the manual thrust mode, the flight crew moves the thrust levers between idle and full thrust as usual, but in auto-thrust, the thrust levers are set in a fixed position, which is defined by the maximum amount of available thrust. Whether in manual or auto-thrust, speed and power changes are monitored via N_1, the IAS, and speed trends, as on any aeroplane.

THRUST LEVER OPERATION

The Autothrottle system when engaged moves the thrust levers together, in response to error signals generated, and compares the aeroplane’s actual flight conditions, against selected datums. In the majority of aeroplanes, the throttles can be manually repositioned at any time without disengagement of the autothrottle.

> **Note:** In the Airbus fly-by-wire type of aeroplane the throttle levers are in a fixed position on the throttle box, and the throttle levers do not move.

THRUST MANAGEMENT VIA THE AUTOTHROTTLE

The Autothrottle operates in response to flight crew mode control panel inputs from the AFCS, or alternatively via automatic FMC commands. The Autothrottle system:

- Uses reference thrust limits calculated by the FMC
- Commands the thrust levers
- Commands thrust equalisation through the electronic engine controls

The FMC calculates a reference thrust for the following modes (typical):

- Takeoff
- Reduced takeoff (also called de-rated takeoff)
- Assumed temperature takeoff
- Climb
- Reduced climb
- Cruise
- Go-around

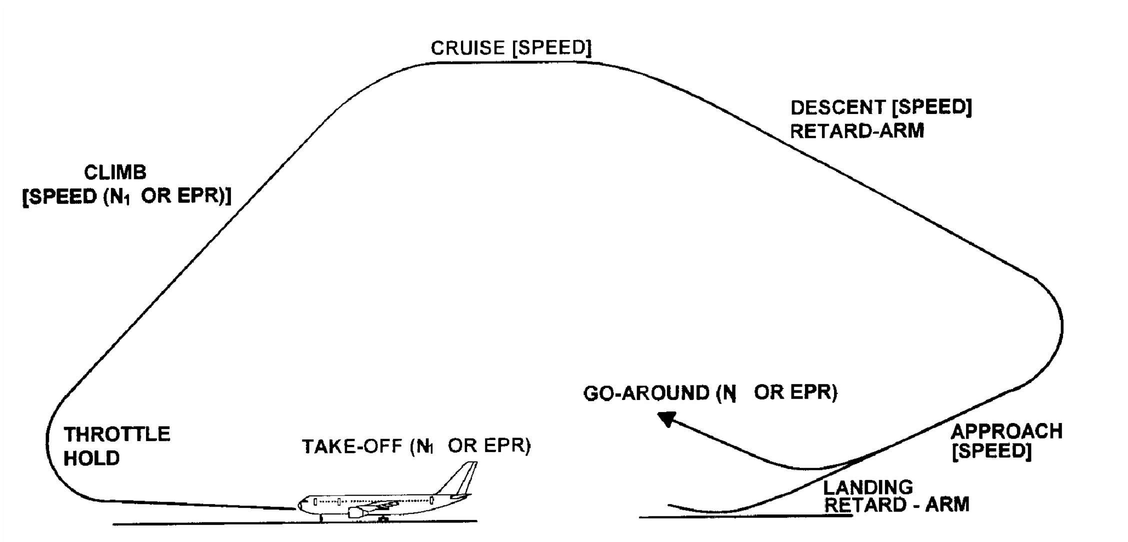
When used for take off, the autothrottle sets the required power, and indicates TO on the gauges and annunciates TOGA in the FMA (Flight Mode Annunciator) of the EADI until take off speed has increased beyond 80 kt. Beyond 80 kt, the autothrust mode changes to Thrust Hold mode (THR HLD) and replaces TO with THR HLD on the EADI. The thrust hold mode maintains the take off thrust but exerts no control over throttle position, allowing the crew to exercise full control over the throttles if required.

The mode used for thrust reference automatically changes during the respective flight phase, as selected by the FMC, and the selected thrust reference mode displays on the thrust mode display.

THRUST MANAGEMENT COMPUTER (TMC)

The TMC operates the throttles in response to manual inputs from the Thrust Mode Selector Panel (TMSP), and automatic commands from the Flight Management Computer (FMC) when operating in the VNAV mode. The basic functions of the TMC are to:

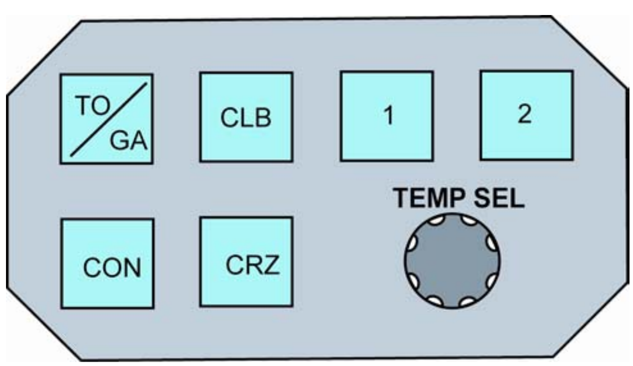
- Calculate thrust limits (using outside pressure and temperature) and thrust settings, or follow any FMC thrust commands
- Provide automatic control of the three primary modes (EPR, MACH HOLD, and Speed) for various flight phases, as shown on the next page
- Detect and transmit A/T failures



THRUST MODE SELECT PANEL (TMSP)

The TMSP, as shown below allows:

- The TO/GA, CLB, CON, CRZ reference thrust modes to be selected
- Temperature de-rating to be selected, in order to prolong the engine life



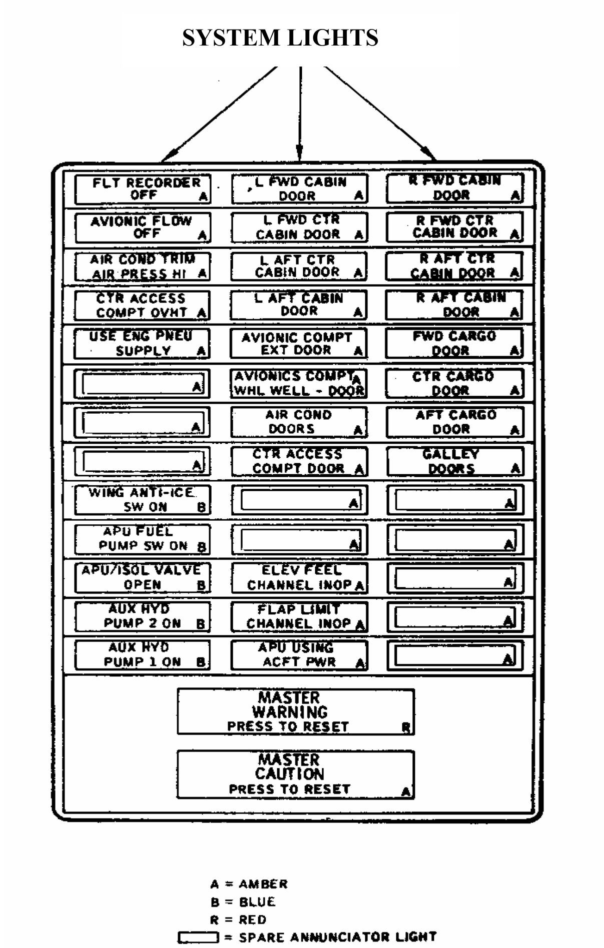
Chapter 28
Central Warning System

INTRODUCTION

In modern aeroplanes, many systems require constant monitoring, which in turn require a corresponding multitude of warning devices; both visual and aural. These warnings consist of flashing lights, horns, and bells which, if operating in various parts of the flight deck, could pose an unnecessary distraction. In order to reduce this probability, aeroplanes are equipped with a Central Warning System (CWS).

CENTRAL WARNING SYSTEM ANNUNCIATOR PANEL

In its basic form, the system comprises of a group of warning and indicator lights connected to signal circuits allied to the various aeroplane systems. Each light displays a legend, denoting the system, and a malfunction or advisory message. The lights are blue, red, or amber, and are on an annunciator panel, normally installed on the centre control panel, as shown below.



Also installed in aeroplanes carrying a flight engineer is a functionally integrated duplicate warning panel. The character of the signals also varies according to the degree of urgency or hazards involved. Aural, visual, and tactile signals, given singularly or in combinations, provide both warning and information simultaneously, regarding the nature of the problem. If the condition requires immediate corrective action by the flight crew, a red warning light indicates. These lights indicate engine, wheel well, or Auxiliary Power Unit (APU) fires, autopilot disconnect, and landing gear unsafe conditions. If the condition requires timely corrective action by the flight crew, a cautionary amber warning light indicates it. A series of blue lights alternatively indicate whether any associated system valves are in transit, or merely in disagreement with the appropriate control switch.

If a fault occurs in a system, a fault-sensing device additionally transmits a signal to an electronic device known as a **Logic Controller**, which determines whether the fault is of a hazardous nature, or is one that simply requires caution. If the fault is considered hazardous, the controller output signal illuminates the red **master warning** light. When caution is required, the signal illuminates the amber **master caution** light. Each master warning light incorporates a switch unit, so that if the caps are pressed in, the active signal circuits disconnect, and the lights extinguish. This action will also reset the master warning system, so that it can accept signals from other faults, which might subsequently occur in other aeroplane systems. The system lights are not of the resetting type and remain illuminated until the system fault corrects. Dimming of the lights and testing of the bulb filaments is normally carried out by means of switches mounted adjacent to the annunciator panel.

AURAL WARNINGS

Various aural signals on an aeroplane alert the flight crew if warnings or cautions exist, and some typical ones are below:

- A clacker sounds if the airspeed limits are exceeded.
- A warning tone sounds if the autopilot is disconnected.
- An intermittent horn sounds if the cabin altitude limits are exceeded.
- A steady horn sounds if a landing gear disagreement exists.
- An intermittent horn sounds if the take-off configuration is incorrect.
- A fire-warning bell sounds if a fire exists.
- Ground Proximity warnings and alerts indicate by voice warnings.

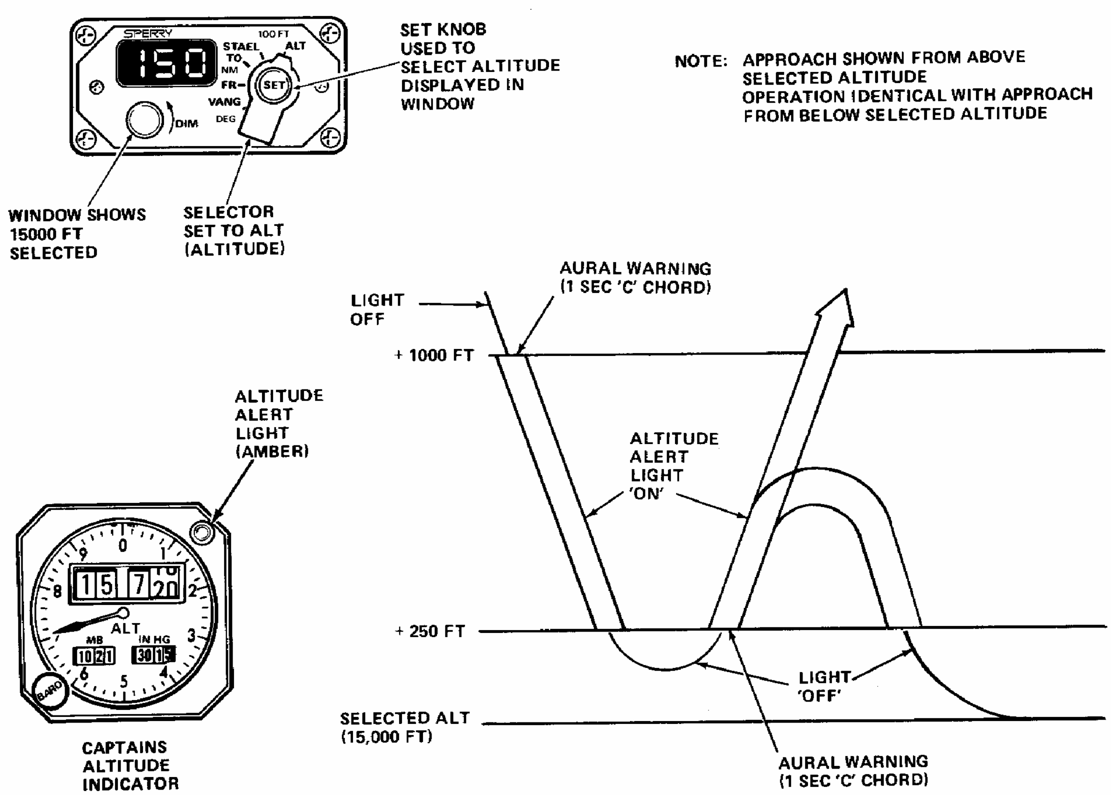
Chapter 29
Altitude Alerting System

INTRODUCTION

The Altitude Alerting System (AAS) provides both aural and visual warnings of an aeroplane reaching, or deviating from a pre-selected altitude, by utilising an output from a pressure altimeter (or ADC).

ALTITUDE ALERTING SYSTEM OPERATION

If selecting a target altitude on the mode control panel (MCP), a signal from an altitude sensor mixes with the signal created by the control panel pre-set altitude, and as long as the actual altitude and pre-set altitude are different, a signal difference exists.



For example, the sequence of events that takes place during a descent from 31 000 ft to 15 000 ft is as follows:

- Prior to leaving 31 000 ft, the control unit is set to 15 000 ft, with no warning given at this stage, as the signals are too different.
- At approximately 1000 ft above the target altitude, an aural warning sounds (C-chord) for approximately one second, and an altitude alert light illuminates adjacent to each primary altimeter.
- The light remains illuminated until approximately 250 ft above the target.
- If the aeroplane moves greater than 250 ft above the target altitude or continues through the target altitude to 250 ft below it, an aural warning sounds and the light illuminates again.

The same sequence of events also occurs if the target altitude is approached from below. Altitude alerting is inhibited when the flaps are in the landing configuration, or when the glide slope (G/S) is captured.

INTRODUCTION

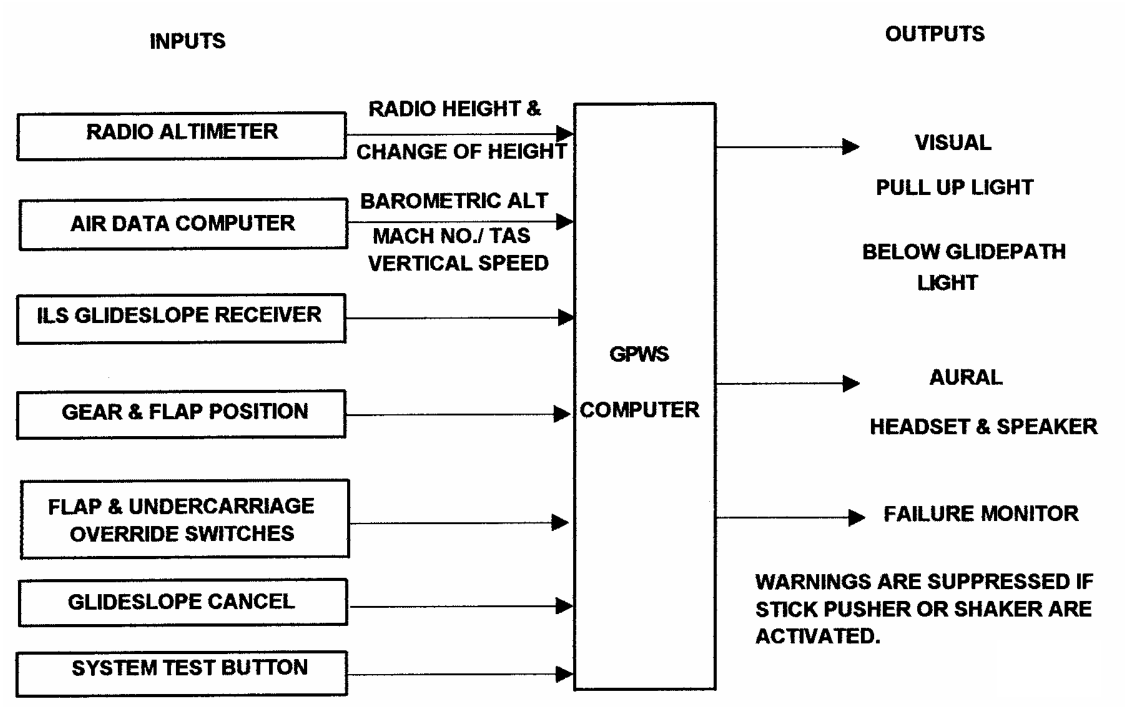
Most transport aeroplane crashes occur due to **controlled** flights into the ground for reasons such as inattention, confusion, vertigo, distraction, instrument reading error, poor visibility, and navigation error. The Ground Proximity Warning System (GPWS) is thus designed to prevent this sort of accident, by giving the flight deck crew some advanced warning, both aurally and visually, if an unsafe flight condition close to the ground exists.

The radio altimeter and the Air Data Computer (ADC) are continuously monitored for the aeroplane height above the ground, and the barometric rate of change of height; which enables continuous assessment of the rate of closure with the terrain immediately beneath the aeroplane. In a typical GPWS, a red PULL-UP light, together with a “WHOOP-WHOOP PULL-UP” audible command, gives a warning of an unsafe proximity to the ground. After correcting the dangerous condition, the warnings cease and the system automatically resets itself.

Importantly, no warning is provided of steeply rising ground directly ahead of the aeroplane. Additionally, the system does not prevent a properly configured aeroplane from landing short of the runway in the absence of an ILS glide path. The GPWS normally activates between 50 ft and 2450 ft above the surface, determined by the radio altimeter. Never deactivate the GPWS (i.e. by pulling the circuit breakers) except when using approved procedures at airports where GPWS inhibition is specifically required.

GPWS SYSTEM ARCHITECTURE

The GPWS comprises of a Central Processing Unit (CPU), which accepts inputs from various sources. The CPU continually examines these inputs, and if a collision risk with the terrain exists, appropriate visual and aural warnings generate. Computer failure and any failure in the input signals display on a warning panel



The GPWS system additionally incorporates a fully integrated self-test function, capable of checking the signal path from all of the inputs described above. If the system checks carry satisfactorily when the test switch is depressed, the normal indication to the flight crew is the simultaneous activation of both visual and aural warnings. Notably, system testing by this means is normally prohibited when the aeroplane is airborne.

GPWS MODES

Four main conditions give rise to audible and visual alerts. These four conditions are described as MODES, and are numbered from 1 to 4. The condition associated with each mode is as follows:

Mode 1 Excessive descent rate below 2450 ft AGL

Mode 2 Excessive terrain closure:

- In the cruise configuration
- In the landing configuration

Mode 3 Height loss after T/0, or during a missed approach

Mode 4 Unsafe terrain clearance when not in the landing configuration below:

- 500 ft AGL with the landing gear not locked down
- 200 ft AGL with the flaps not set for landing

Other modes which exist are:

Mode 5 Excessive downward departure from the ILS glide path

Mode 6 Aural altitude call-outs on approach, including decision height (DH)

Mode 7 Windshear detection, warning, and guidance

In addition to the inputs from the ADC and the radio altimeter, the GPWS CPU also obtains information from the following sources:

Main Landing Gear Selector Assembly
The position of the landing gear governs whether or not MODE 3 activates, and determines the height/barometric rate of descent conditions that activate a MODE 4 warning.

Flap Selector Assembly
The position of the flap governs whether or not MODE 3 activates, and determines the height/terrain closure rate which activates a MODE 2 warning, and the height/barometric rate conditions which activate a MODE 4 warning.

ILS Receiver
MODE 5 uses the degrees of deviation from the glide path together with glide path validity signals.

Stall Prevention
The stall prevention devices fitted to the aeroplane (e.g. stick shakers and stick pushers) also feed a signal to the GPWS computer to inhibit the GPWS warnings during the incipient stall and/or stalled condition).

WARNING SYSTEM

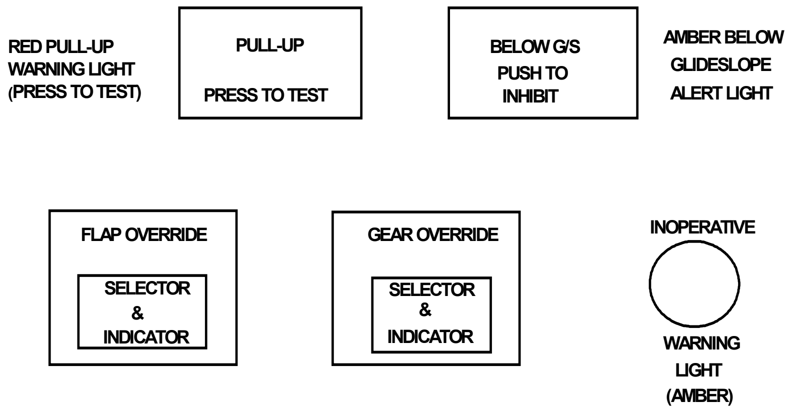
To avoid violent manoeuvres, current equipment generates alerts and warnings. The alert is regarded as preliminary to a warning, and the flight crew must respond immediately by correction of the flight path or configuration, so that the alert ceases. This is a cautionary indication, and if ignored, an imperative command or warning follows. If a warning is initiated, the flight crew must immediately level the wings and initiate a maximum angle of climb until a minimum safe altitude is reached. Where an advanced GPWS is fitted, verify the cause of the warning after initiating the climb. The aural alerts and warnings for both the basic and advanced systems are summarised in tabular format on the next page.

GPWS Mode		Vertical Limits (ft)	Basic Equipment		Advanced Equipment	
			Alert	Warning	Alert	Warning
1	Excessive descent rate	50 - 2450	—	Whoop Whoop Pull-up	Sink Rate Sink Rate	Whoop Whoop Pull-up
2	Excessive terrain closure 2A Cruise	50 - 2450	—	Whoop Whoop Pull-up	Terrain Terrain	Whoop Whoop Pull-up
	Excessive terrain closure 2B Configured for landing	200 - 790		Whoop Whoop Pull-up	Terrain Terrain	Whoop Whoop Pull-up
3	Altitude loss after take-off or go-around	50 - 700	—	Whoop Whoop Pull-up	Don't Sink	
4	4A Proximity to terrain gear not down & locked	50 - 500	—	Whoop Whoop Pull-up	Too Low Gear	Too Low Terrain
	4B Proximity to terrain & flaps not set for landing	50 - 200	—	Whoop Whoop Pull-up	Too Low Flaps	Too Low Terrain
5	Below glide slope	50 - 1000	Glide slope	—	Glide Slope	—
6	Below minimums	50 - 1000	—	—	Minimums	—
7	Windshear encounter	—	—	—	Windshear	—

The alert for Mode 5 consists of a steady amber glide slope light, and an aural glide slope warning that becomes more frequent the greater the deviation below the glide slope, and also becomes louder if the deviation exceeds 2 dots below 300 ft.

GPWS CONTROL PANEL

The GPWS warning and alert lights are on a control panel, as shown below.



Pressing the PULL-UP light tests the system integrity. This causes the PULL-UP, BELOW GLIDE SLOPE and INOP lights to illuminate, and activates aural warnings. The INOP light also illuminates if the system or power supply fails, or any input to the computer is lost.

A flap-override switch is also available that should be operated whenever a non-standard flap setting is used during the approach to land. With the switch in this position, the associated warning suppresses, and the GPWS flap-override indicator illuminates. Similarly, a gear-override selection disables the warning if the aeroplane has to land with one or more of its wheel assemblies retracted.

DISCRETIONARY RESPONSE

Regardless of the type of GPWS, basic or advanced, on receipt of an alert or a warning, make a response. In order to avoid an excessive manoeuvre, a warning may sometimes be considered as if it were an alert, but only when the following conditions exist:

- The aeroplane is operated during the day in meteorological conditions, which enables it to remain 1 nm horizontally and 1000 ft vertically away from clouds, with a visibility of at least 5 nm.

  and

- If it is obvious to the Commander that the aeroplane is NOT in a dangerous situation with regard to the terrain, configuration, or present manoeuvre.

Note: Although some manufacturers of GPWS equipment may show in their literature TOO LOW TERRAIN to be an alert, the view of the JAA is that the response should be as it is for a warning.

WARNING INHIBITION

The Mode 5 indication alone may be inhibited so that localiser-only or back-beam approaches may be flown without provoking an alert. Inhibition is achieved by pressing the lamp housing of the amber indicator light, which is only effective in the soft alerting height band above 300 ft. In all other cases, climbing out of the envelope that initially triggered the event only silences the warning or alert. There is intentionally no on/off switch. Do not, under any circumstances, trip the power supply circuit breaker to silence a warning.

THE REPORTING OF GPWS EVENTS

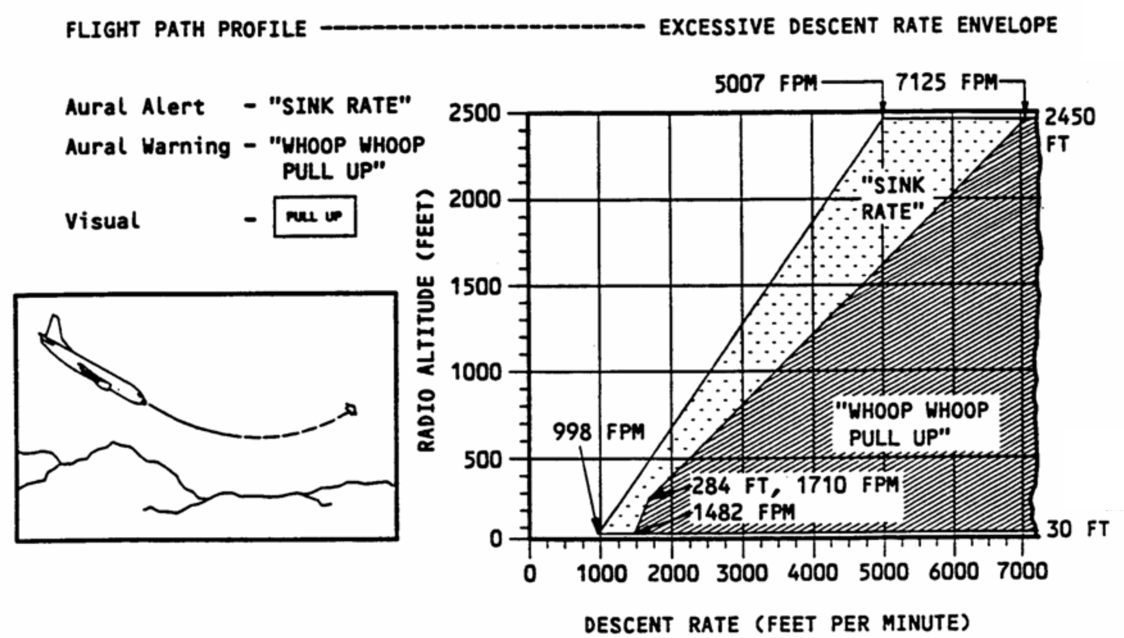
Report all GPWS events to the regulatory authority on purpose-designed forms. Also, submit reports when not receiving a warning, but when normally expected.

OPERATION OF THE GPWS

A typical GPWS operates as follows, although some differences do exist between manufacturers (e.g. for the Boeing 737-400 the lower limit is 30 ft, but for JAA purposes the range is always between 50 and 2450 ft):

MODE 1

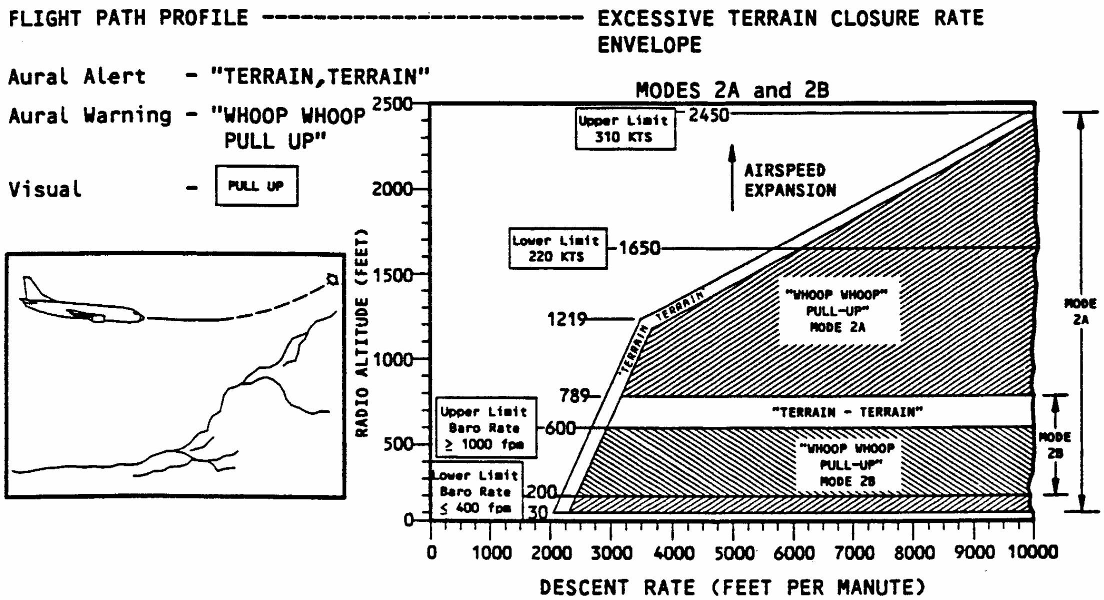
This mode is activated whenever the barometric descent rate is excessive with respect to the aeroplane's height above the terrain, as determined by the radio altimeter, and the barometric rate signal, obtained from the ADC. The mode 1 envelope is divided into two areas: the **initial penetration area**, or **sink rate area**, and the **inner warning area**, or **pull-up area**. This mode is also completely independent of the landing gear and flap positions. The warning envelope for MODE 1 has an upper limit of 2450 ft above the ground, and at this height a warning is given if the barometric rate of descent exceeds 7125 ft/min. At the lower limit of the envelope, which is 50 ft above the ground, a barometric descent rate of 1500 ft/min or more causes MODE 1 activation. The full operating parameters for MODE 1 are shown on the next page.



Penetration of the first boundary activates the PULL-UP light and a repetitive aural alert of "SINK RATE". Penetration of the second boundary results in a repetitive aural warning of "WHOOP WHOOP PULL-UP".

MODE 2

This mode activates whenever the aeroplane has an excessive closure rate with respect to rising terrain, achieved by measuring the terrain closure rate as determined by the radio altimeter. This mode also consists of two sub modes; Mode 2A if the flaps are NOT in the landing configuration, and Mode 2B if the flaps are in the landing configuration. The full operating parameters for Mode 2 are shown below:



Like Mode 1, this mode also possesses two boundaries; the initial penetration area and the inner warning area. Penetrating the first boundary results in an aural alert of “TERRAIN” repeated twice and followed by a repetitive aural alert of “WHOOP WHOOP PULL-UP”

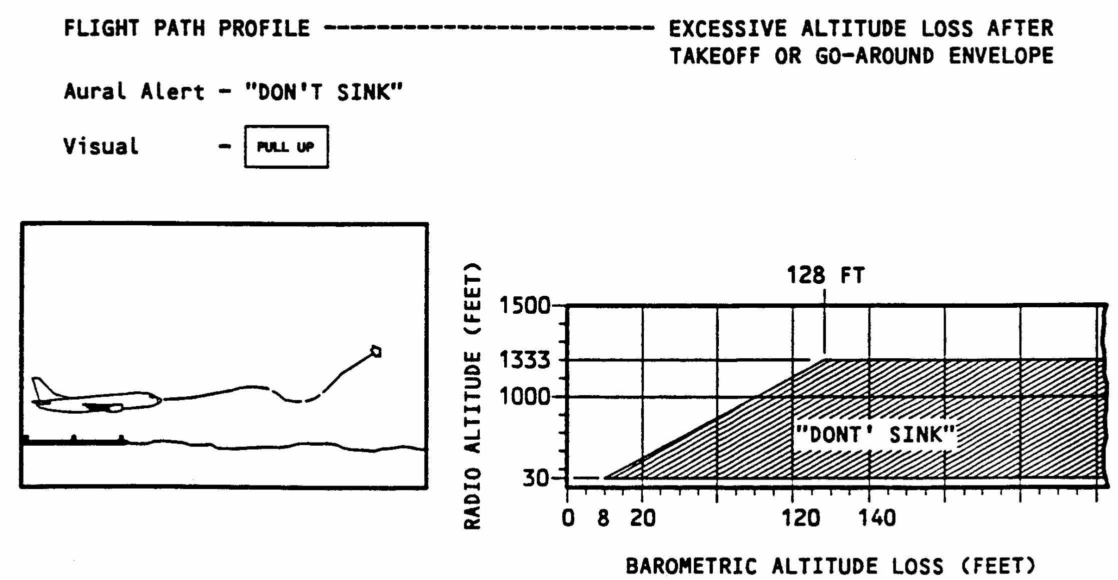
A **Mode 2A** “TERRAIN” alert, is also given if the landing gear and flaps are NOT in the landing configuration when the aeroplane is exiting the ‘PULL-UP’ area, and continues until gaining at least 300 ft of barometric altitude, or lowering the landing gear. As the airspeed increases from 220 kt to 310 kt with the landing gear retracted, the radio altitude at which Mode 2A alerting or warning varies according to an airspeed expansion function up to a maximum of 2450 ft.

At airspeeds less than 220 kt, the upper boundary is 1650 ft radio altitude and a Mode 2A warning activates if the terrain closure rate is equal to, or in excess of 6000 ft per minute. At the lower limit of MODE 2A operation, a warning is given if the terrain closure rate exceeds approximately 2000 ft per minute, and the flaps are NOT in the landing configuration.

A **Mode 2B** warning activates if an excessive terrain closure rate occurs with the flaps in the landing configuration. It activates at the upper parameter of 789 ft radio altitude if the terrain closure rate is equal to, or exceeds, 3000 ft/min, or at the lower parameter of 200 ft radio altitude if the terrain closure rate is equal to, or exceeds, 2500 ft/min. The lower boundary cut-off varies between 200 ft and 600 ft and is dependent solely on the barometric rate. In this mode, altitude gain is also required to silence the “TERRAIN” alert after the “PULL-UP” warning activates. Additionally, during the approach, with the landing gear and flaps fully extended, the altitude gain function is inhibited, and the “TERRAIN -TERRAIN” annunciation replaces the “WHOOP WHOOP PULL-UP” annunciation.

MODE 3

This mode activates if an excessive height loss occurs during the initial take-off climb or during a go-around procedure. The full operating parameters for Mode 3 are shown below.

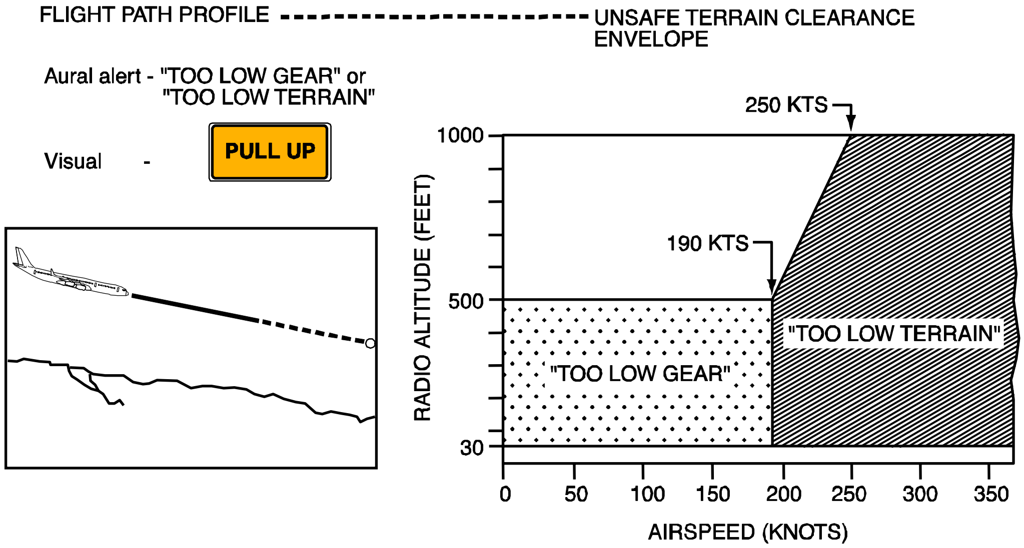


If the aeroplane enters the envelope, a repetitive aural alert of "DON'T SINK" sounds until a positive rate of climb is established between 30 ft and 700 ft radio altitude. The alert repeats based on the original descent altitude if the aeroplane descends again before climbing. This mode is only active during take-off or when either the flaps or undercarriage retracts during a missed approach. When the aeroplane passes through 700 ft radio altitude with the undercarriage retracted, Mode 4 normally arms if the ADC senses an accumulated barometric height loss of 70 ft or more. An expanded upper altitude limit of 1333 ft radio altitude is used at airspeeds in excess of 190 kt in order to prevent premature switching from Mode 3 to Mode 4 warning during the climb-out, when the accumulated barometric height loss is in excess of 128 ft. Mode 3 is also inactive when the landing gear and flaps are both in the landing configuration.

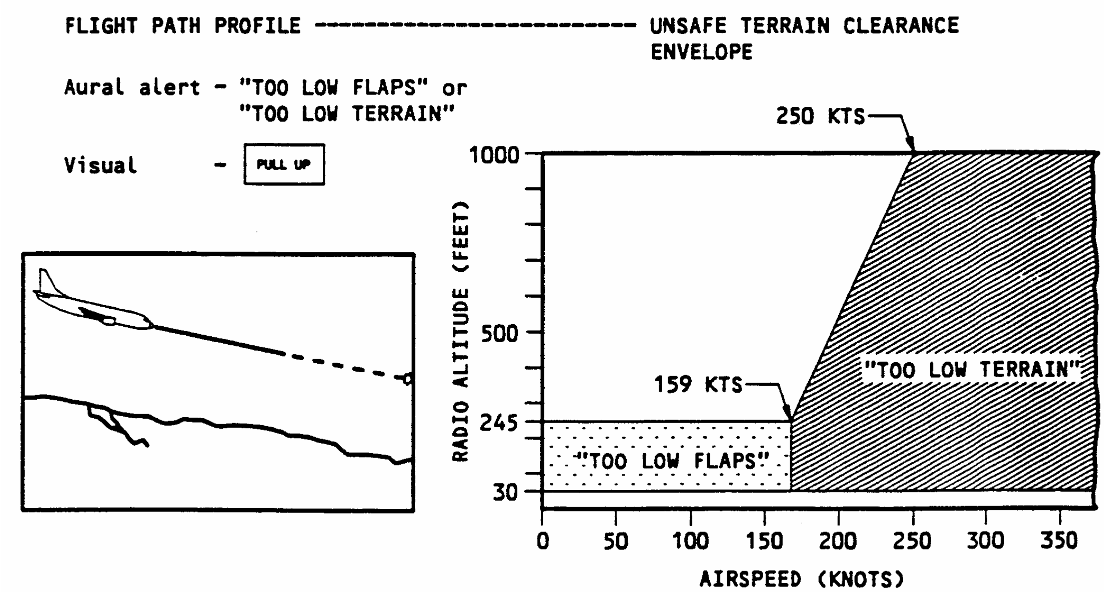
MODE 4

This mode activates whenever an unsafe terrain clearance situation is experienced, and the aeroplane is NOT in the landing configuration. Similar to Mode 2 this mode also has two sub modes; Mode 4A with the undercarriage retracted, and Mode 4B with the undercarriage extended and the flaps not in the landing position.

MODE 4A activates whenever the terrain clearance reduces to 500 ft radio altitude, regardless of the barometric rate, UNLESS the landing gear is fully extended. The standard upper boundary for this sub mode is 500 ft radio altitude, and at airspeeds less than 190 kt a repetitive aural alert of "TOO LOW GEAR" sounds. At airspeeds between 190 kt and 250 kt, a repetitive aural alert of "TOO LOW TERRAIN" and the associated warning sounds, and the upper boundary of the envelope extends to 1000 ft. The full operating parameters associated with Mode 4A are shown below.



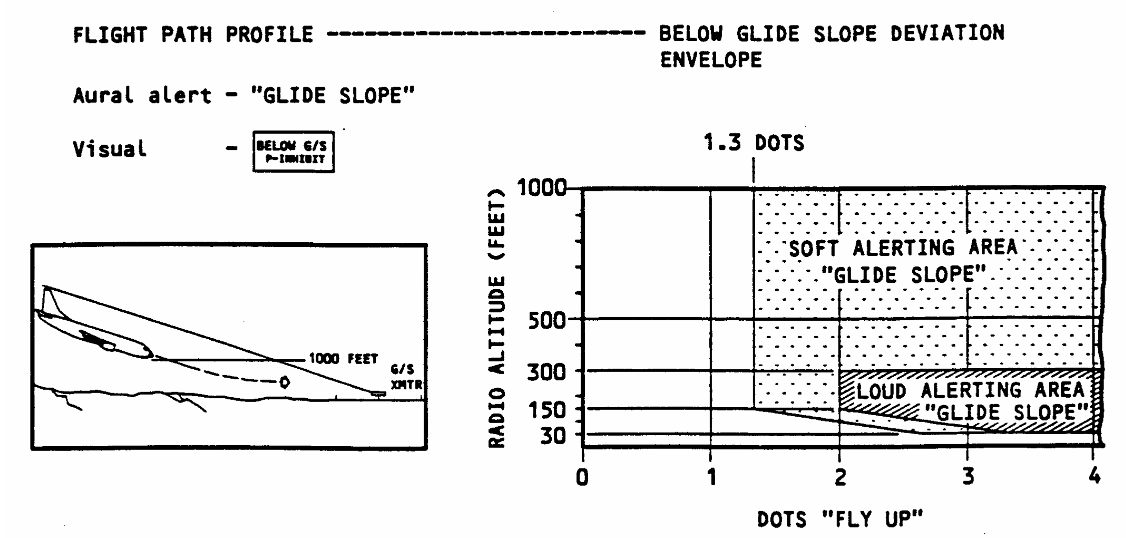
MODE 4B activates UNLESS the flaps are also in the landing position. The standard upper boundary for this sub mode is 245 ft radio altitude and at airspeeds less than 159 kt, a repetitive aural alert of "TOO LOW FLAPS" sounds. Shown below are the full operating parameters associated with Mode 4B.



At airspeeds between 159 kt and 250 kt, the upper boundary of the envelope similarly increases to 1000 ft radio altitude and the aural alert "TOO LOW TERRAIN" repeats.

MODE 5

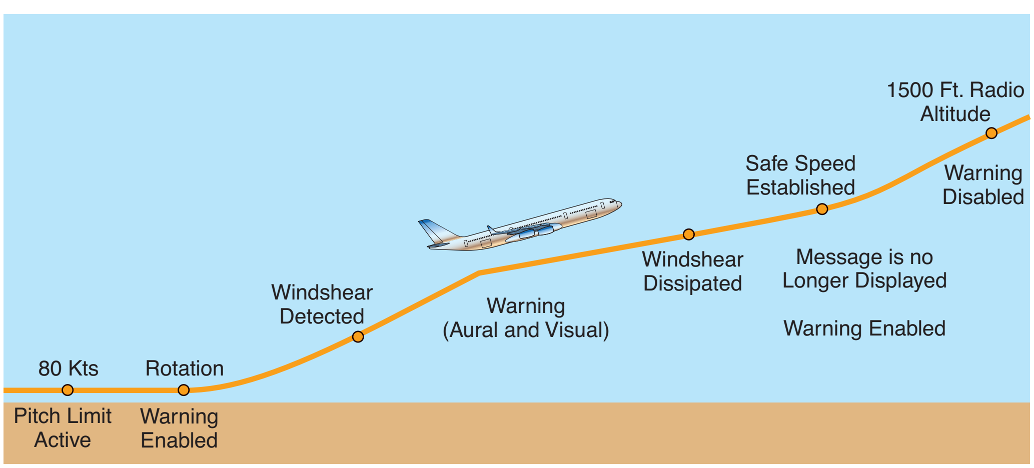
This mode activates whenever the aeroplane falls significantly below the ILS glide path by more than 1.3 dots. This mode has two specific alerting areas: soft and loud, although the repetitive aural alert of "GLIDE SLOPE" sounds if entering either area. The 'BELOW G/S' lights also illuminate at the same time. The amplitude of the repetitive aural alert additionally increases in volume on entering the 'loud area' and the repetition rate increases as the glide slope deviation increases, whilst the radio altitude decreases. The full operating parameters associated with Mode 5 are shown below.



This mode arms whenever the Commander's glide slope receiver receives a valid signal and whenever the radio altitude is 1000 ft or below. Inhibiting or canceling this mode is accomplished when flying below 1000 ft radio altitude by pressing either flight crew BELOW G/S light and rearms whenever the aeroplane climbs above 1000 ft radio altitude or below the lower limit. Additionally, the glide slope warnings may occur at the same time as pull-up warnings when the 'PULL-UP' alert is due to an active MODE 1, 2, or 4, but NOT Mode 3. If Mode 3 activates, Mode 5 automatically inhibits.

MODE 7

This mode provides both aural and visual warnings if a windshear condition exists during the take-off and approach phases of flight, below 1500 ft radio altitude. The following diagram shows the typical sequence of events during a windshear encounter during the take-off phase of flight.



The aural warning consists of a two-tone siren, which is followed by an aural warning of "WINDSHEAR, WINDSHEAR, WINDSHEAR". A visual warning message also displays on each Electronic Attitude Director Indicator (EADI), and remains there until the windshear condition subsides. The windshear warnings also take priority over all other GPWS modes, and only become active on take-off after rotation.

JOINT AVIATION REQUIREMENTS

The requirements state that:

- All turbine-engine aeroplanes of a maximum certified take-off mass in excess of 15 000 kg or authorised to carry more than 30 passengers, for whom the individual Certificate of Airworthiness is first issued on or after 1 July 1979, shall be equipped with a GPWS.
- All turbine-engine aeroplanes of maximum certified take-off mass in excess of 5700 kg or authorised to carry more than 9 passengers shall be equipped with a GPWS from 1 January 1999.
- A GPWS shall provide automatically a timely and distinctive warning to the flight crew when the aeroplane is in a potentially hazardous proximity to the Earth's surface.
- From 1 January 1999 a GPWS shall provide, as a minimum, warnings of the following circumstances:

    - Excessive descent rate
    - Excessive terrain closure rate
    - Excessive altitude loss after take-off or go-around
    - Unsafe terrain clearance while in the landing configuration
    - Gear not locked down and the flaps not in the landing position
    - Excessive descent below the instrument glide path

Chapter 31
Traffic Collision Avoidance System

INTRODUCTION

With the ever-increasing traffic flow, the risk of an airborne collision has dramatically increased and, in order to preserve the safety element, modern aeroplanes are fitted with equipment that provides collision avoidance assistance. **TCAS**, which is an abbreviation for **Traffic Alert and Collision Avoidance System**, is the current equipment used to achieve this function, and provides traffic information within approximately 30 miles of the aeroplane. The purpose and function of the TCAS is to alert the flight crew to the presence of other aeroplanes in their vicinity, and where possible provide an escape manoeuvre, should a collision risk arise. This is achieved by equipment on board the aeroplane only, and without any reference to the ground installations used by air traffic control. It is designed to complement, and not replace conventional air traffic management methods.

The TCAS has the following levels of capability:

TCAS I

This system provides a Traffic Advisory (TA) only (i.e. information that would advise the flight crew of a potential traffic hazard as an aid to visually acquiring the target and avoiding it). The TA display shows the range and bearing of the aeroplane posing a potential threat. This capability serves only as a warning, and simply provides an aid to visual acquisition and avoidance, but does not recommend an escape manoeuvre.

TCAS II

This system has the same capability as TCAS I, but is additionally capable of providing an escape manoeuvre recommendation, called a Resolution Advisory (RA), in the vertical plane only. Bearing information also displays, but only as an aid to visual acquisition. The recommended escape manoeuvre is based on Mode C reports from any conflicting traffic.

TCAS III

This system has the same capability as TCAS II, but is also able to provide RAs and manoeuvre guidance in the horizontal plane.

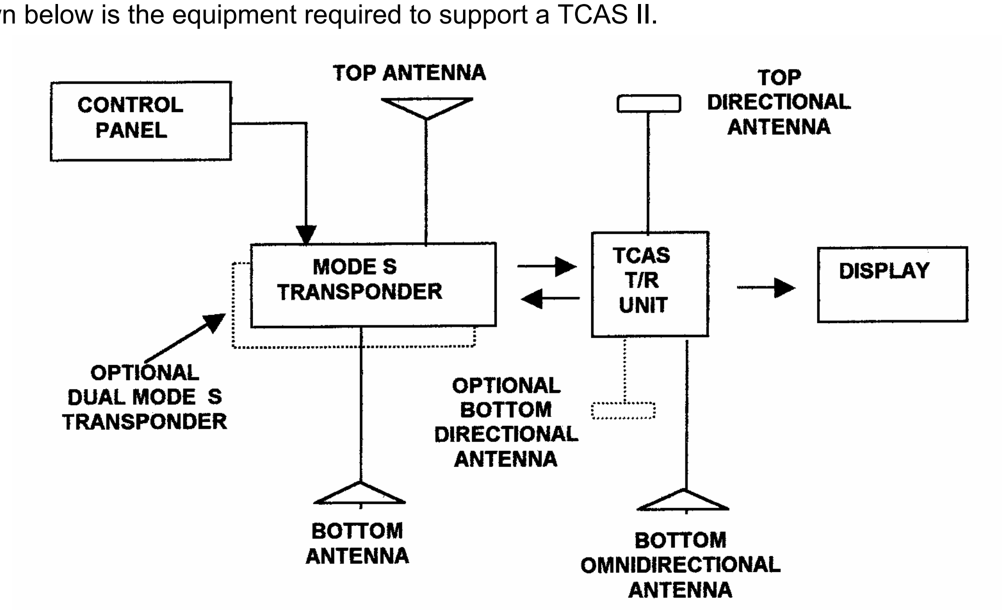
TCAS II is the most common system and provides the necessary vertical manoeuvre advice. In accordance with the Joint Airworthiness Requirements, as of the year 2000, all commercial aeroplanes over 15 000 kg, and with a seating capacity of 30 or more passengers, when operating in European airspace, must carry TCAS II. This system provides:

- The generation of Traffic Advisories (TA)
- Threat detection
- The generation of Resolution Advisories (RA)
- Co-ordination
- Surveillance
- Communication with ground stations

The first four of these items process during each complete cycle of operation, and take approximately 1.2 seconds to complete.

AEROPLANE INSTALLATION

Shown below is the equipment required to support a TCAS II.



The above equipment includes:

- A TCAS Computer (transmit/receive unit)
- Two TCAS antennas; an upper directional antenna, and a lower antenna, which may be directional or omni directional.
- A combined ATC Transponder/TCAS control panel
- A flight-deck display.
- A Mode S transponder(s) with top and bottom omni-directional antennas.

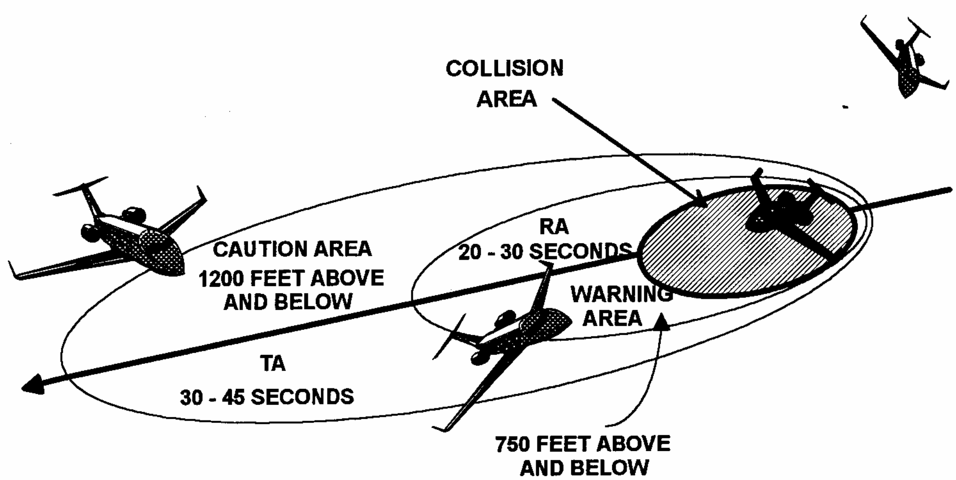
TCAS II receives inputs from:

- The altimeter and /or the Air Data Computer (ADC) with regard to pressure altitude
- The Radio Altimeter in order to establish heights for the various operating restrictions described later
- The gear and flap circuits which provide aircraft configuration status
- The FMC which provides operational performance data such as operational ceiling

Note: TCAS II is not connected to the autopilot, nor the FMS (this includes INS/IRS), and remains independent of them, therefore it continues to function in the event of the failure of either of these systems.

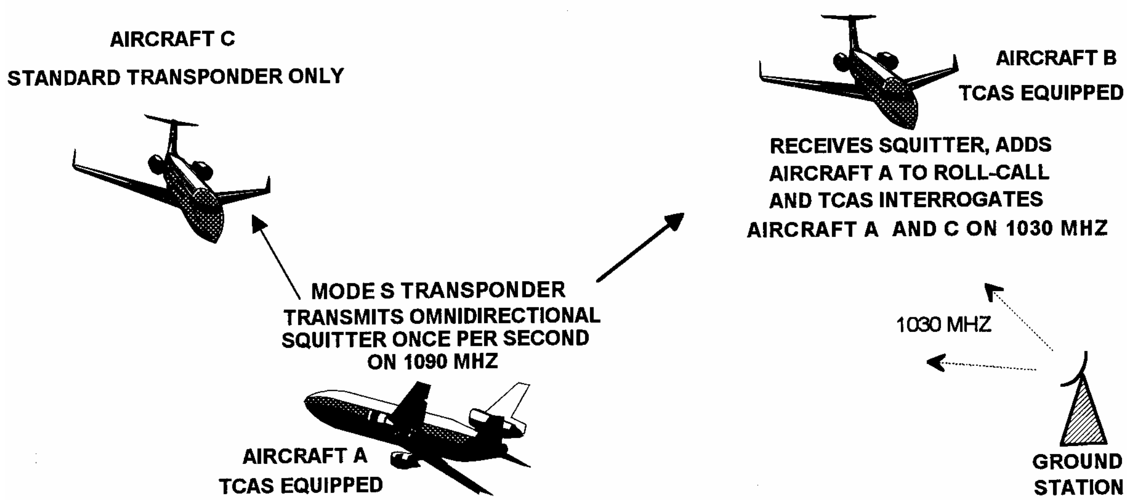
TCAS relies on other aeroplane transponders to indicate their presence, so aeroplanes not equipped with this system will appear transparent. If targets respond with Mode A only, it assumes that they are at the same flight level as the interrogator, and only a traffic advisory generates. TCAS-protected airspace is defined horizontally by time-to-convergence, and by ATC vertical separation minima (400 to 740 ft depending upon the altitude).

Traffic Advisories (TA) appear on targets detected 30-45 seconds away from convergence within the protected vertical envelope, and **Resolution Advisories (RA)** appear on targets 20-30 seconds from convergence.



OPERATION OF TCAS II

With reference to the diagram below the TCAS II system operates as follows:



- Aeroplane A is TCAS equipped, and uses a Mode S transponder to transmit its unique aeroplane identifier once per second in the form of an omni-directional broadcast known as **Squitter**.
- Other TCAS equipped aeroplanes in the area monitor the 1090 MHz transmissions, and when a valid 'Squitter' signal is detected, the identity of the transmitting aeroplane adds to a **List** or **Roll Call** for subsequent interrogation.
- Aeroplane A also compiles a Roll Call of its own, derived from Squitter from other TCAS equipped aeroplanes.
- The TCAS II transmitters of aeroplanes A and B additionally transmit interrogating signals to detect other aeroplanes in the area that are equipped only with standard transponders. In this mode, the TCAS transmitter radiates a general interrogation signal to all standard transponders. To separate replies when a number of aeroplanes are in the vicinity, the transmitter has the following features:
  - The antenna design permits individual bearing transmissions to each of the four quadrants in sequence.
  - The transmissions consist of a series of pulses, starting at low power and progressively increasing to a higher power. Known as **Whisper/Shout**, this causes close-by aeroplanes to respond before ones that are more distant. TCAS is therefore able to track individual replies, and also limit unnecessary overall energy radiation.
- Aeroplane C replies to such interrogations, but the Mode S equipment on aeroplane B only responds to Mode S interrogations.
- Having compiled the roll-call, aeroplane A interrogates each aeroplane on the list. The TCAS antennas then receive the replies, and this enables the relative bearings of intruders to be determined. Should the intruder be able to respond with altitude, the TCAS also uses this information. The range of intruders is determined from the time interval between the interrogation signal and the reply.

- The altitude, altitude rate, range, and range rate are also determined by tracking the replies from each interrogation. By computer analysis of the replies, the TCAS determines which aeroplane represents a potential collision threat, and then provides the appropriate advisory to the flight crew if any aeroplane might pass within 1200 ft. Where multiple threats exist, each threat processes individually to produce the optimum avoidance solution.
- When receiving and processing a Mode S signal, if a collision risk establishes, the computer establishes an air-to-air Mode S data link with the TCAS II computer on the other aeroplane. The computers then ‘agree’ and co-ordinate **Resolution Advisories** as necessary.

TCAS AURAL WARNINGS

If the computer determines the likelihood of a conflict, an aural announcement “TRAFFIC, TRAFFIC” generates. Should evasive action later be required, a Resolution Advisory (RA) follows a Traffic Advisory (TA), and provides a verbal statement of the action needed (e.g. “CLIMB, CLIMB”). The RAs may be:

Corrective
This instructs the flight crew to, “CLIMB” or “DESCEND” or “DON’T CLIMB/DESCEND”.

Preventative
This advises the flight crew to avoid certain manoeuvres in order to maintain the separation between other aeroplanes. A selection of possible aural resolution advisories are in the following table.

Aural Resolution Advisories	
No.	**Advisory**
1	Climb, Climb, Climb
2	Descend, Descend, Descend
3	Reduce Descent, Reduce Descent
4	Reduce Climb, Reduce Climb
5	Monitor Vertical Speed, Monitor Vertical Speed
6	Clear of Conflict
7	Climb, Crossing Climb, Climb, Crossing Climb
8	Descend, Crossing Descend, Descend, Crossing Descend
9	Increase Climb, Increase Climb
10	Increase Descent, Increase Descent
11	Climb-Climb Now, Climb-Climb Now
12	Descend-Descend Now, Descend-Descend Now

When TCAS determines that a threat has passed, a statement “CLEAR OF CONFLICT” generates.

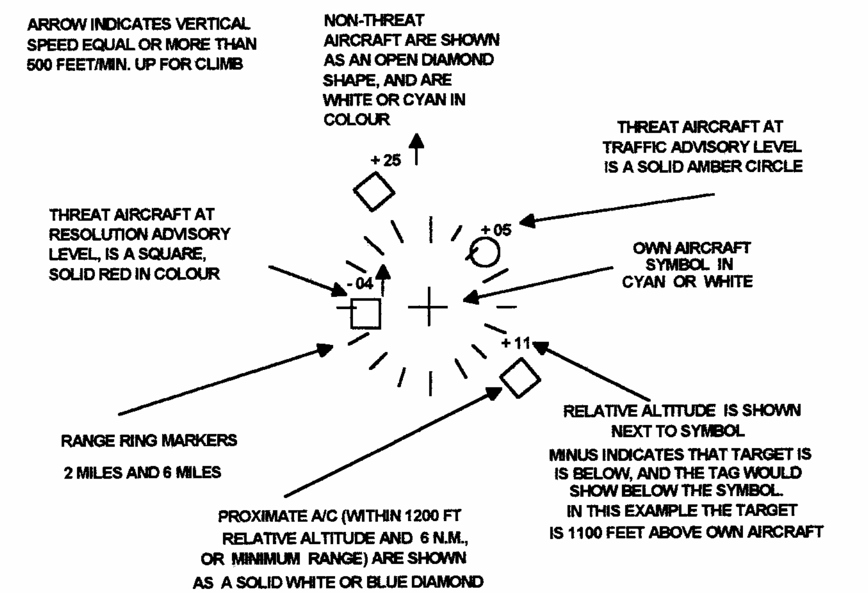
INFORMATION DISPLAY

The action required to comply with the Resolution Advisory displays on a specialised Resolution Advisory/Vertical Speed Indicator (RA /VSI) instrument using a colour liquid crystal screen, which shows the rate of climb, and intruder information as shown on the next page. On other aeroplanes, the intruder information displays on the EFIS Navigation Display screen.

In either case, the symbology used is the same:

Hollow White or blue diamond.	Other traffic not offering a threat
Solid White of blue diamond.	Traffic within 6 nm and 1200 ft vertically.
Solid Yellow Circle.	Traffic advisory.
Solid Red Square,	Resolution advisory.

Each symbol has a data tag attached to it, which appears in the same colour as the associated symbols.



This data provides the altitude of the target in reference to your own aeroplane's altitude (e.g. + 06 indicates 600 ft above). An arrow also indicates whether the 'target' is climbing or descending. There is also often provision to display momentarily absolute pressure altitude in hundreds of feet of pressure altitude.

Where the target is a mode A transponder with no height transmissions, no data tag appears and no RA is given. If the target does not have a functioning transponder, the TCAS system is unable to detect its presence.

The nominal maximum tracking range of TCAS is 14 nm. However, in areas of high-density traffic, the system range can be reduced to 5 nm.

An RA generates between 15 to 35 seconds before the point of closest approach of the intruder, and a TA generates 5 to 20 seconds in advance of the RA.

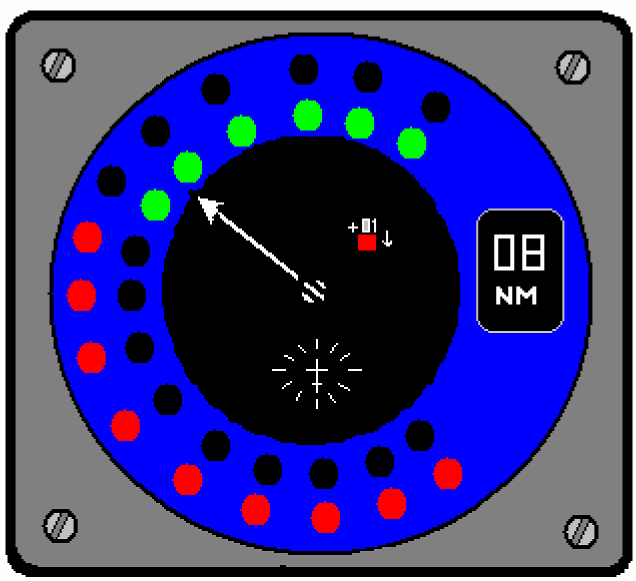
The TCAS equipment is capable of handling a maximum surveillance capacity of 30 aeroplanes, but is nominally capable of surveillance of approximately 27 'high closing speed' targets, within 14 nm of the aeroplane.

RESOLUTION ADVISORY / VERTICAL SPEED INDICATOR (RA / VSI)

The required action to comply with the resolution advisory usually commands by means of a specialised vertical speed indicator, as showing on the next page. The TCAS VSI uses a colour liquid crystal display to show conventional rates of climb, plus intruder information, which displays as symbols in the centre. The two lower buttons change the displayed range between 4, 8, and 16 nautical miles. The aeroplane symbol also appears 2/3 from the top. A range ring is always present at 2 miles, and a second ring appears at 6 miles when selecting the 16-mile scale.

A Resolution Advisory is commanded by part of the VSI scale appearing in red, and the target climb or descent rate as a green sector to show the preferred rate.

A panel dimmer also controls the display brightness, and an automatic sensor adjusts the display for differing flight deck conditions.



The audio annunciation would be:

CLIMB;CLIMB

The indication would show a danger area defined by red LED lights, and a safe area in green LED lights.

The object is to climb at a rate higher than that indicated by the red LED lights

TCAS CONTROL PANEL

A Mode S Control panel as shown below controls the TCAS.

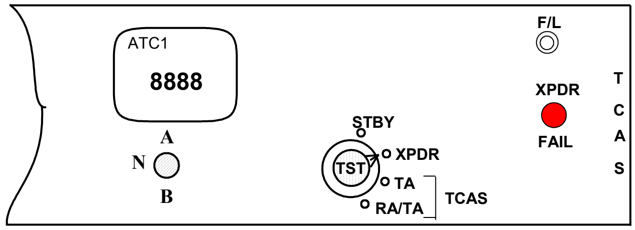


DIAGRAM 26.6 TCAS CONTROL

The control panel switch labelled A/N/B permits the selection of various altitude bands in which intruders display as follows:

A 7000 ft above to 2700 ft below
N 2700 ft above to 2700 ft below
B 2700 ft above to 7000 ft below

Note: This switch also has no effect on the generation of advisories.

OPERATING RESTRICTIONS

TCAS may be inhibited either totally or partially, to avoid conflicts with other handling requirements, as follows:

- When either the ground proximity warning or windshear system is active to ensure that these alerts take priority.
- Below certain altitudes:
    - TCAS aurals are suppressed below 400 ft AGL
    - All RAs are inhibited below 500 ft AGL
    - Descend RAs are inhibited below 1000 ft AGL and sometimes 700 ft.
    - Increase Descent RAs are suppressed below 1800 ft AGL.
- Climb or Increase Climb RAs may be inhibited above a defined barometric altitude.
- Climb and/or Increase Climb RAs are inhibited in those circumstances where the manoeuvre cannot be safely executed due to performance limitations.
- The flight crew also has the capability through the control panel to select TA only in order to eliminate unnecessary RAs during some operations (e.g. during parallel approaches).

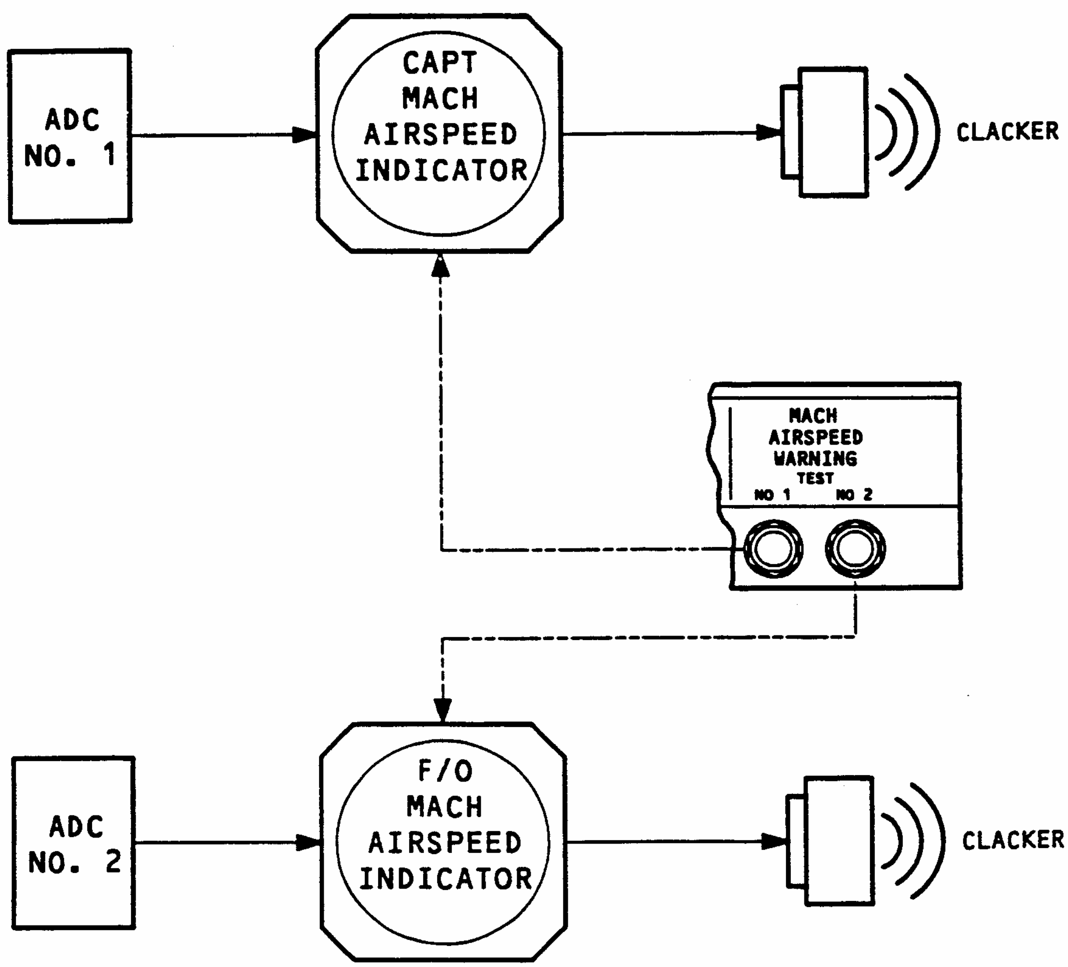
Chapter 32
Mach/Airspeed Warning System

INTRODUCTION

The Mach/Airspeed Warning System provides a distinct aural warning when exceeding the maximum operating speeds of V_{MO} or $M_{MO.}$ These airspeeds exist due to structural limitations at low altitudes, and the handling characteristics of the aeroplane at higher altitudes.

SYSTEM ARCHITECTURE AND OPERATION

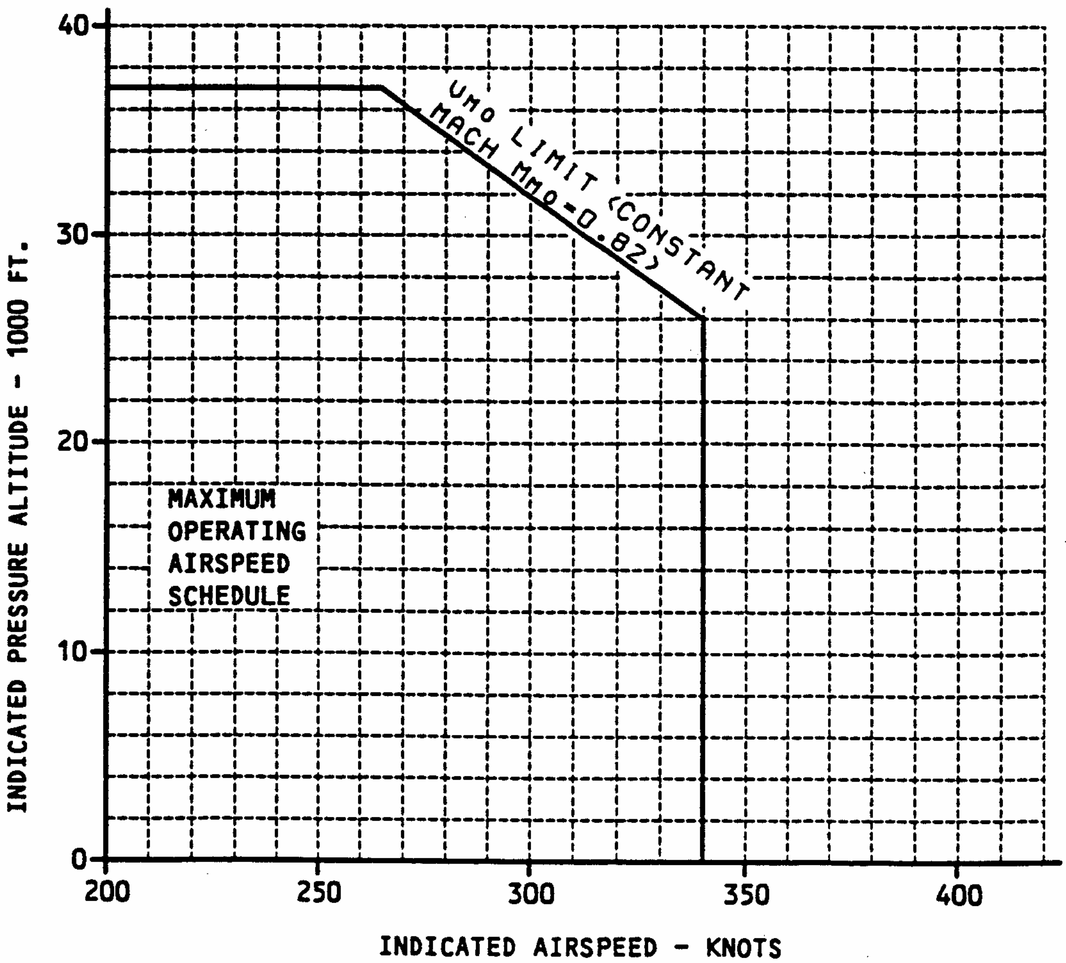
The system consists of two independent Mach/airspeed units, normally driven by outputs from the **Central Air Data Computer (CADC)**, via an internal mechanism comprising of interconnected altitude and airspeed capsules. Shown below is a typical system layout:



If the limiting speed, as indicated by the bugs on the Mach/airspeed indicator is exceeded, an audio warning (clacker) will be activated, which can only be silenced if the airspeed is reduced below V_{MO}/M_{MO}. To test this system at any time, push the TEST function switch, which in turn activates the 'clacker'. Notably, this system provides the only audible warning of **over-speed**.

MAXIMUM OPERATING AIRSPEED SCHEDULE

The diagram on the next page shows a typical airspeed envelope, where V_{MO} at sea level is 340 kt (IAS). This airspeed remains constant until the aeroplane reaches approximately 26 000 ft, when the M_{MO} limit of 0.82 Mach is reached.



V_{MO} then continues to decrease with increasing altitude, if the selected Mach number is maintained.

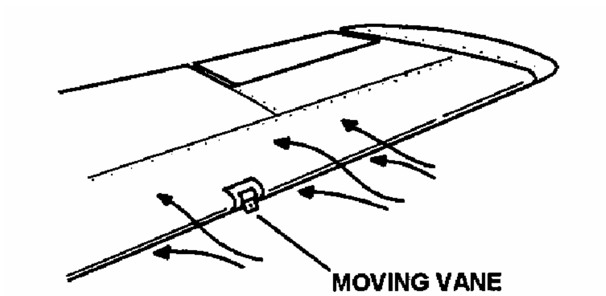
Chapter 33
Stall Warning

INTRODUCTION

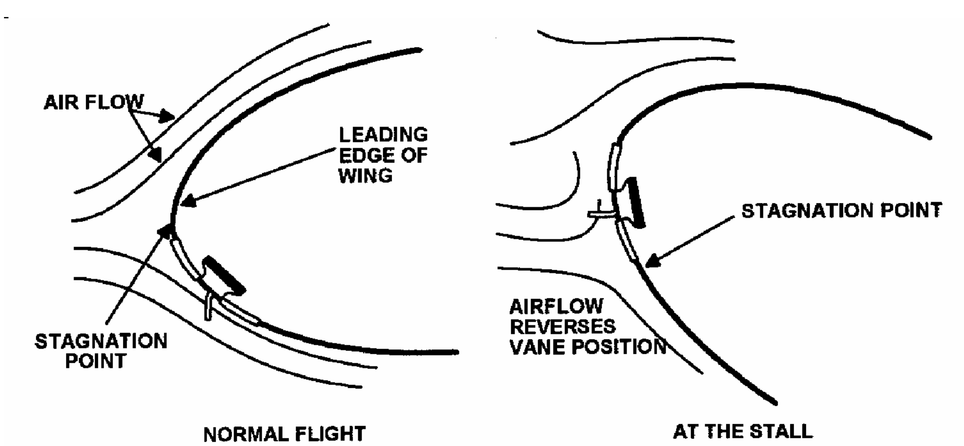
In order to warn the flight crew that the aeroplane is approaching a stall condition, a warning device is fitted, which varies depending on the type of aeroplane.

LIGHT AEROPLANE STALL WARNING DEVICE

In addition to pre-stall buffet, light aeroplanes are normally fitted with an audible stall-warning device, which operates just before the stall. This device activates by way of a stall warning sensor in the form of a moving vane, positioned approximately midway along the wing, just below the leading edge, as shown below:



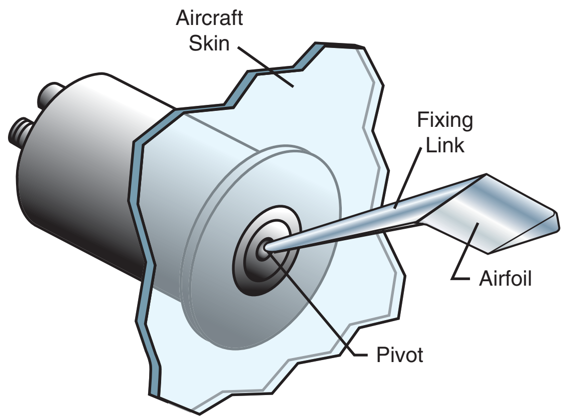
They are held in their non-active position by static pressure, due to the forward motion of the aeroplane, and the pressure distribution around the wing, with the stagnation point positioned above the device. As the airspeed reduces/angle of attack increases, the stagnation point moves below the device, and the spring-loaded flap moves upward, as shown below.



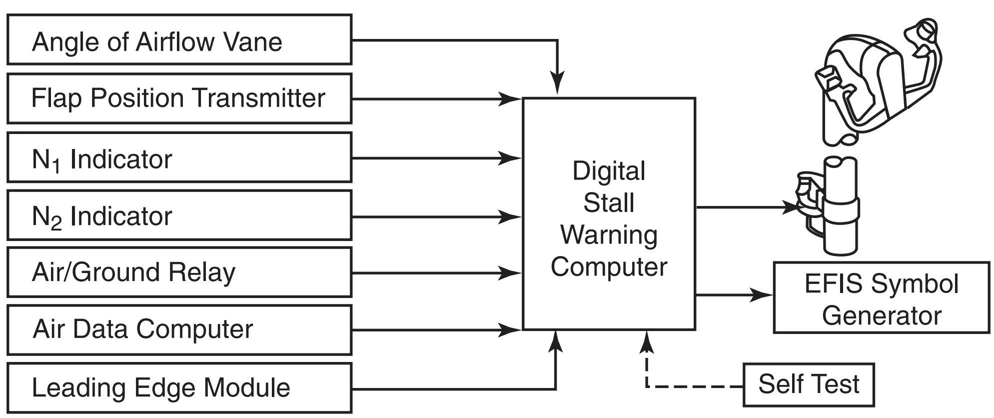
When reaching a pre-set deflection, an electrical circuit completes and a warning horn activates. The warning is usually a continuous sound that stops only when the angle of attack reduces (i.e. when the airspeed increases).

TRANSPORT CATEGORY AEROPLANE STALL WARNING DEVICE

On most modern transport aeroplanes the stall-warning device activates via an **angle of attack** or **Alpha sensor**. One of these sensors is located on either side of the fuselage near the nose, and detects the change/rate of change of angle of attack of the aeroplane as the airspeed varies. The sensor, shaped like an aerofoil surface, varies its position relative to the airflow. An electrical signal generates, and when reaching a predetermined limit, nominally 12 – 14°, a stall warning activates. These devices are also heated so that they remain operational throughout the flight.



The sensor, connected via a pivot to a digital stall warning system as illustrated below, deactivates on the ground via a Squat switch on the nose wheel gear (weight on undercarriage or air/ground safety sensor).



Two independent digital computers are normally installed which compute the proper stall warning based on the angle of attack, flap configuration, and thrust. The computers receive inputs from the angle of attack sensor (Alpha probe), the flap position transmitter, the N_1 and N_2 indicators, the air/ground relay, the air data computers, and the leading edge module.

At a pre-set angle of attack, the circuit completes and an eccentric weight motor attached to the base of either or both control columns operates. The motor in turn vibrates the control column, thus imitating the effect of aerodynamic buffet, and alerts the flight crew before a stall develops. This also displays visually on the Primary Flight Display (PFD).

The stall warning system additionally incorporates a ‘test’ function facility, which when operated activates the stick shaker by completing the circuit from the sensor to the motor.

INTRODUCTION

Recording devices installed in aeroplanes are a part of the never-ending quest to prevent accidents, since no transport system has a 100% safety record. The two root causes of all incidents and accidents are:

- Mechanical failure
- Human error

If an aeroplane incident or accident occurs, it is important to establish whether the cause was the result of mechanical or human failure, or even a combination of both.

To make the investigation of a mechanical failure easier, it is better if a complete record of the behaviour of every mechanical/structural component is available right up to the instant of the occurrence. To facilitate this, extensive pre-flight and maintenance documentation for the aeroplane is retained on the ground. A complete record of the flight crew actions prior to flight, is maintained together with any relevant certification and training certificates, which may provide invaluable information into the cause of the event.

To enhance the recording system certain aeroplanes are required to carry a **Flight Data Recorder (FDR)** and a **Cockpit Voice Recorder (CVR)**. The FDR is designed to record mechanical features, whilst the CVR is designed to record all voice communications with and on the flight deck. It is also a requirement for the flight crew and ATC to keep in-flight records although, in the event of an accident, the on-board documentary records could be destroyed.

FLIGHT DATA RECORDER (FDR) REQUIREMENTS

In accordance with JAR-OPS commercial transport category aeroplanes with a Certificate of Airworthiness (C of A) first issued on or after 1st April 1998 with more than 9 passenger seats,\ and a maximum take-off mass over 5700 kg should not be flown in a JAA Member State or elsewhere unless it is equipped with a FDR. This device must also use a digital method of recording and storing data, and be equipped with a method of readily retrieving that data from the storage medium.

The parameters recorded may also vary according to the maximum certificated take-off mass and age of the aeroplane, as specified in JAR-OPS 1 Section K.

For aeroplanes required to carry an FDR the following data requires recording against a common time scale:

- Altitude
- Airspeed
- Heading
- Acceleration
- Pitch and roll attitude
- Radio transmission keying, unless an alternative means is provided to enable the FDR and CVR recordings to be synchronised
- Thrust or power on each engine
- Configuration of lift and drag devices
- Air temperature
- Use of Automatic Flight Control Systems (AFCS)
- Angle of attack

For those aeroplanes with a maximum certified take-off mass over 27 000 kg the following additional parameters must also be recorded:

- Positions of the primary flight controls and pitch trim
- Radio altitude
- Primary navigation information being displayed to the flight crew
- Cockpit warnings
- Landing gear position

The FDR should furthermore:

- Be capable of retaining the data recorded during at least the last 25 hours (10 hours for aeroplanes of 5700 kg or less) of its operation
- Be able to obtain accurate data correlation (matching) with information displayed, or presented to the flight crew from aeroplane sources
- Must automatically start to record the data prior to the aeroplane being capable of moving under its own power, and must also automatically stop after the aeroplane is incapable of moving under its own power
- Be fitted with a device to assist in locating the recorder in water

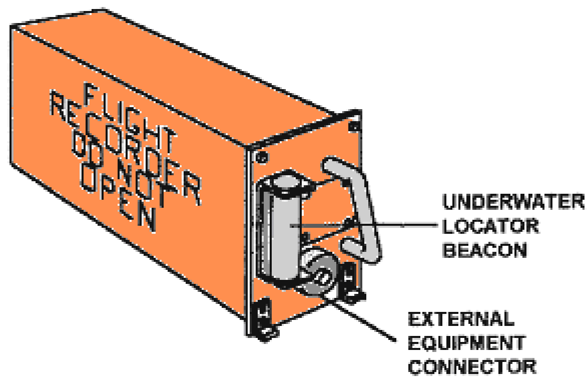
On some modern aeroplanes, a large number of the parameters come from the aeroplane's integrated data source, and on some FDR models certain parameters can also be transmitted at regular intervals through a data link with a ground station.

If the FDR is unserviceable, the flight may still be conducted, as listed in JAR-OPS 1, if:

- It is not reasonably practicable to repair or replace the unit before the commencement of the flight.
- The aeroplane does not exceed eight further consecutive flights with the FDR unserviceable.
- Not more than 72 hours have elapsed since the FDR was first reported unserviceable.
- Any CVR required to be carried is operative, unless it is combined with the FDR.

FDR DESIGN

The FDR, powered from the aeroplane's 24V DC or vital DC busbar, is contained in a shockproof yellow or orange box, as shown below.



The box, fitted at the rear of the aeroplane, normally behind the rear pressure bulkhead, must be capable of sustaining extremely high impact forces. The box must also be fireproof and waterproof. The box is additionally fitted with an underwater locator (pinger) or small beacon transmitter, which enables it to be located in deep water, and it is designed to survive crash conditions. The transmitter has a self-contained power supply and commences operation as soon as it enters water. The transmitter can operate continuously for 30 days, and has a range of 2–3 miles.

COCKPIT VOICE RECORDER (CVR) REQUIREMENTS

In accordance with JAR-OPS, commercial transport category aeroplanes with a Certificate of Airworthiness (C of A) first issued on or after 1 April 1998, with more than 9 passenger seats, and a maximum take-off mass over 5700 kg should not be flown in a JAA Member State or elsewhere unless it is equipped with a CVR.

The CVR must be capable of recording, with reference to a common time scale:

- Voice communications transmitted from or received on the flight deck by radio
- The aural environment of the flight deck, including without interruption, the audio signals received from each boom or mask microphone in use
- Voice communications of flight crewmembers on the flight deck using the aeroplane's interphone system and public address system
- Voice or audio signals identifying navigation or approach aids introduced into a headset or speaker

The CVR should furthermore:

- Be capable of retaining the data recorded during at least the last 2 hours (30 minutes for aeroplanes of 5700 kg or less) of its operation.
- Must automatically start to record the data prior to the aeroplane being capable of moving under its own power, and must also automatically stop after the aeroplane is incapable of moving under its own power. In addition, depending on the availability of electrical power the recorder must start to record as early as possible during the cockpit checks prior to engine start at the beginning of the flight, until cockpit checks immediately following engine shutdown at the end of the flight.
- Be fitted with a device to assist in locating the recorder in deep water.

If the CVR is unserviceable, the flight may still be conducted, as listed in JAR-OPS 1, if:

- It is not reasonably practicable to repair or replace the unit before the commencement of the flight.
- The aeroplane does not exceed eight further consecutive flights with the CVR unserviceable.
- Not more than 72 hours have elapsed since the CVR was first reported unserviceable.
- Any FDR required to be carried is operative, unless it is combined with a CVR.

CVR DESIGN

The CVR is powered from the 24V DC or vital DC busbar, and like the FDR is located in a box at the rear of the aeroplane. The box is orange or yellow in colour, and must be impact resistant, shockproof, fireproof, and waterproof. Additionally in aeroplanes over 5700 kg, the CVR must be a separate unit from the FDR.

Chapter 35
General Engine Instrumentation

INTRODUCTION

Instrumentation is vital to maintain the safe and efficient operation of the aeroplane engines and their associated systems. It varies according to the type of engines fitted (e.g. piston, turboprop, or gas turbine), and in many cases the individual instruments are identical.

PISTON ENGINES

The level of instrumentation required for an aeroplane fitted with a piston engine is largely governed by the complexity of the engine, for example, the pilot of an aeroplane with a supercharged piston engine needs to monitor the:

Engine Speed
This is a measure of how much work is being done by the engine, and is measured in revolutions per minute (rpm).

Induction Manifold Pressure, or Boost Pressure (MAP)
This is a measure of the engine power of a supercharged engine (absolute pressure). MAP is usually given in inches of mercury (in Hg), and boost is normally measured in pounds per square inch (psi).

Torque or Turning Moment
Acts on the output shaft of the engine and is proportional to the horsepower developed. It is occasionally used to provide information for power control.

Cylinder Head Temperature
This temperature is important because excessive temperatures can cause engine damage. The temperature is measured in degrees Celsius (°C).

Lubricating Oil Pressure and Temperature
Ensures adequate lubrication of the engine. In pressure terms it may be HIGH or LOW, whereas the temperature is usually in degrees Celsius (°C).

Fuel Flow
Provides a measure of the economy of the engine, and is measured in pounds, kilograms or gallons/hour.

Fuel Quantity
Ensures that there is sufficient fuel to complete the flight, and is measured as Fuel Mass or Volume.

Fuel Pressure
Measured using a pressure gauge, and any drop in fuel pressure may indicate a partially blocked fuel filter.

TURBO PROPELLER ENGINES

For an aeroplane with a turbo propeller engine, the parameters that require monitoring are the:

- RPM
- Torque
- Engine Exhaust Gas Temperatures (EGT), measured in °C.
- Lubricating oil pressure
- Lubricating oil temperature
- Fuel flow
- Fuel quantity
- Fuel Pressure

GAS TURBINE ENGINES

For an aeroplane with gas turbine engines, the pilot must monitor:

Engine Pressure Ratio (EPR)
This is the amount of useful thrust being developed by the engine. It is the product of the mass of air passing through the engine and its velocity at the exhaust nozzle minus the drag due to the air passing through the engine. Comparing the air pressure across the engine (i.e. the exhaust pressure to the compressor inlet pressure [EPR]) provides an indication of the thrust output from the engine. EPR is usually given as a percent thrust value.

RPM, N1, N2 or N3
This is normally a percentage of its maximum value.

EGT
The exhaust gas temperature requires monitoring in order to prevent excessive heat damaging the turbine.

Oil Temperature and Pressure
Monitoring ensures the safe operation of the engine.

Fuel Pressure and Temperature
Monitoring ensures that a supply of non-cavitated fuel is at an acceptable pressure and temperature. A low fuel pressure warning light may back this up.

INTRODUCTION

Devices are fitted to aeroplanes that sense or measure pressure and temperature, and then create a signal, proportional to that measurement.

PRESSURE MEASUREMENT

In aviation, pressure is measured in pounds per square inch (psi), inches of mercury (in Hg), millibars (mbar), or kilopascals (kPa). Pressure is also compared to some reference value, and the three most common types are:

Absolute Pressure
This is the pressure compared to a perfect vacuum, which is either equal to, or greater than this value. It is given as PSIA or in Hg absolute.

Gauge Pressure
This pressure compares to ambient pressure, and is given as PSIG.

Differential Pressure
This is the difference between two different pressures in an aeroplane, and is given as PSID.

Aeroplane instruments used to register these pressures are typically the:

Manifold pressure gauge
This gauge measures absolute pressure.

Oil pressure gauge
This gauge measures gauge pressure

Cabin differential pressure gauge
This gauge measures the difference in pressure between the inside and the outside of the aeroplane, and is calibrated in PSID

Pressure measurements are required for various applications such as:

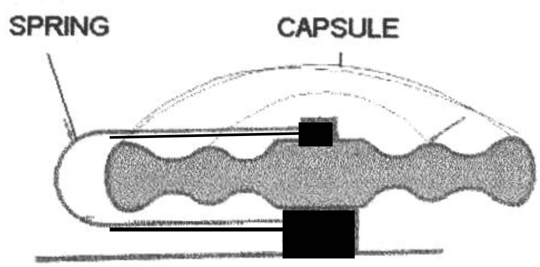
- Static air pressure
- Fluid pressure
- Manifold pressure
- Differential pressure
- Pressure ratios

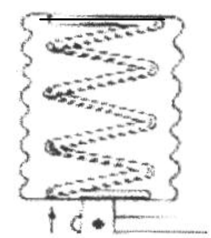
Pressures are usually measured by using a flexible metal chamber (aneroid capsule or bellows), which is spring loaded against the effect of changes in pressure, or a Bourdon tube.

Aneroid Capsule

To measure static pressure, the capsule is partially evacuated and sealed, and is prevented from collapsing by the action of a spring. The spring may be fitted externally or, for some applications, may be fitted internally, as shown on the next page.

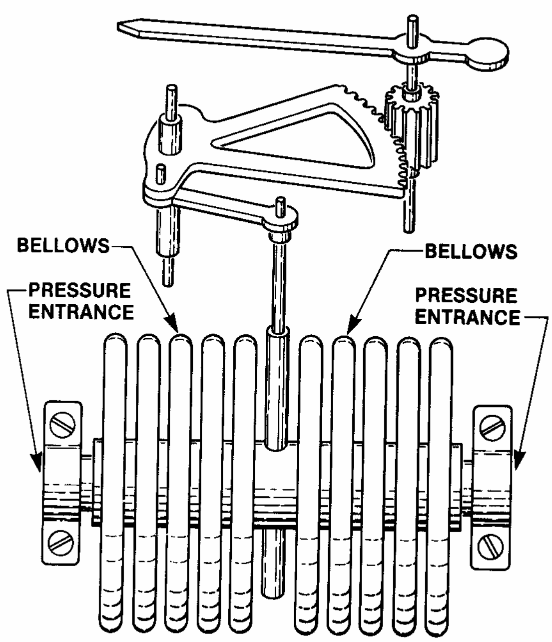
If the pressure acting on the external face of the capsule reduces, the spring causes the capsule to open, but if the external pressure is increased, the effectiveness of the spring reduces, and the capsule collapses. This type of device is used to measure relatively low pressures.





Bellows

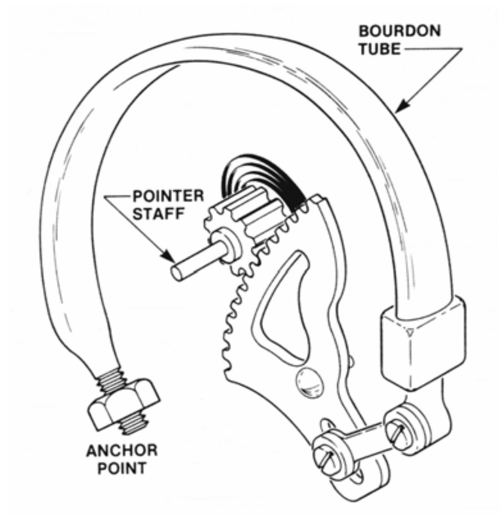
In some cases, it is necessary to measure the difference between two sensed pressures. One common use is to measure the differential pressure, where the bellows divide into two separate chambers, and a different pressure source connects to each side.



Expansion and contraction of the bellows is dependent on the algebraic sum of these pressures. This type of device is used to measure medium pressures.

Bourdon Tube

This device is manufactured from a metal such as phosphor bronze or beryllium-copper. It is in the form of a coil, as shown on the next page, and when affected by a change in pressure, extends or contracts. The Bourdon tube may be used to measure oxygen pressure, hydraulic pressure, and engine oil pressure. This type of device is used to measure relatively high pressures.



In all of these types of pressure sensors, the change in pressure acting on them converts into a mechanical motion by the change in shape of the sensor. The sensors have an initial resistance to any change in shape that results in a time lag between the time, pressure changes, and resultant change in shape.

TEMPERATURE MEASUREMENT

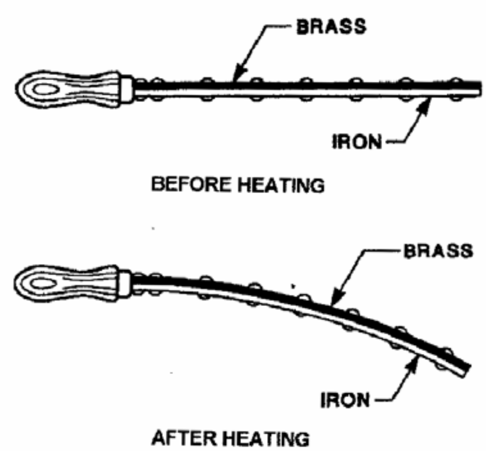
The temperatures requiring measurement on an aeroplane are:

- Air temperatures
- Gas temperatures
- Component temperatures
- Fluid temperatures

The variation in the physical properties of a substance is used in measuring temperature, and any devices used on aeroplanes to measure temperature are called **Temperature-Measuring Systems**. Aeroplane temperature indicators give readings in degrees Celsius (°C) or in degrees Fahrenheit (°F). To make a comparison the temperatures in the following paragraphs are in °F.

Bi-metallic Temperature System

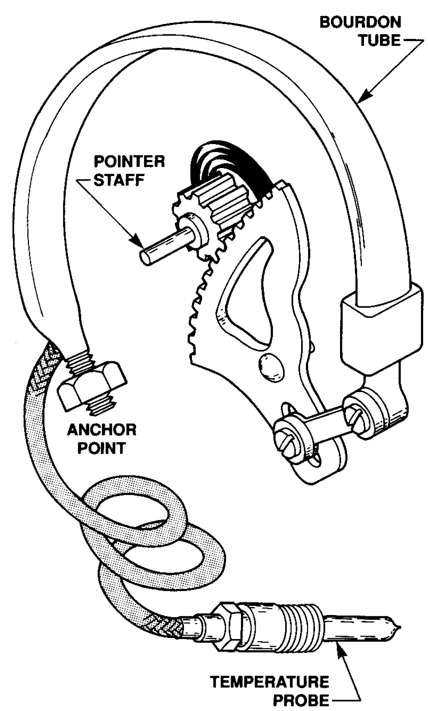
This system, used to measure temperatures up to 140°F, uses the property of expansion. Different materials expand and contract at different rates when subjected to the same change in temperature. If two thermally dissimilar metals (e.g. iron and brass), are strapped together and heat is applied, one expands more than the other, and the bi-metallic strip distorts.



The bi-metallic strip can be formed into a coil so that any change in temperature causes the strip to wind or unwind and produce a rotational movement. This motion then rotates a needle around a scale, and displays the temperature on a temperature gauge. This system is commonly used on small aeroplanes, normally installed through the side window.

Mechanical Bulbs Temperature System

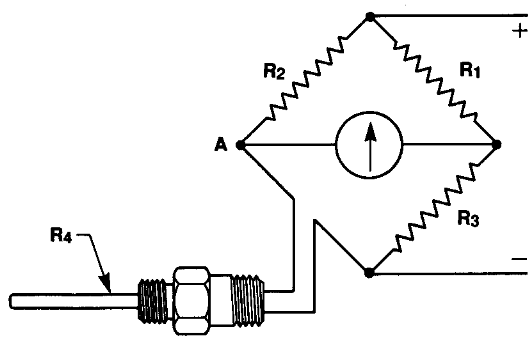
This system consists of a Bourdon tube gauge, which measures pressure, a thin walled bulb, which is at the point of measurement, and a thin capillary tube to connect them together, as shown below.



It uses the principle of the increase in vapour pressure within a confined space to measure temperature. The system is filled with a chemical (e.g. Methyl Chloride), which in its natural state is part liquid and part gas. The system is sealed, and as the temperature increases, the pressure changes within the tube, giving an accurate reading of temperature on the Bourdon tube gauge. This method is used on small aeroplanes to measure engine oil pressure, and on some jet aeroplanes to measure the compressor inlet temperature of the engine.

Wheatstone Bridge System

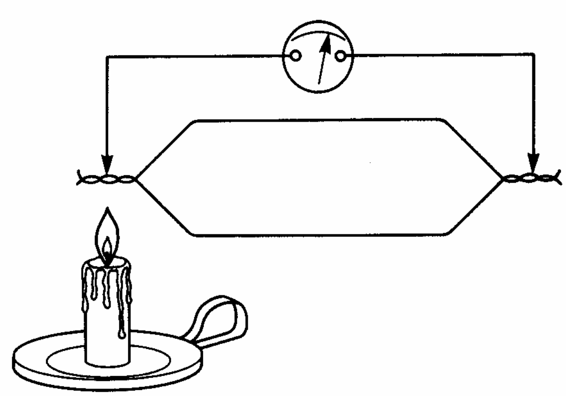
This method of measuring temperature requires electrical power, and is useful for measuring temperatures up to 300°F. The bridge circuit consists of three fixed resistors and one variable resistor.



The variable resistor is the temperature probe, and contains a coil of fine nickel wire. As the temperature of the coil increases, its resistance also increases, and current flows in the bridge. This in turn moves the needle on the gauge, giving a temperature reading. A disadvantage of the Wheatstone bridge is that any bad connections can drastically affect the resistance value, and produce an inaccurate temperature reading.

Thermocouple Temperature System

This system is used to measure temperatures up to about 500°F or more, and is the most commonly used device. Unlike the Wheatstone bridge, this system does not require an electrical power source.



When one junction of two 'dissimilar' metals is heated a voltage proportional to the temperature between the 'hot' and 'cold' junctions occurs, and current flows in the circuit, with a reading taken at the cold junction. Two commonly used metals are Copper-Constantan and Iron-Constantan, which are both able to withstand high temperatures and produce a useable voltage. The actual voltage produced is very low, so this device is not usually used to measure temperatures below 400°F. To measure high temperatures up to 1000°F and above a combination of Chromel-Alumel is used.

Thermocouples are used on piston engines to measure the cylinder head temperature. If only one probe is used it is fitted in the hottest running cylinder (e.g. the rear cylinder on a horizontally opposed engine). The probe is often in the form of a spark plug gasket, which fits under the spark plug, or alternatively a bayonet type probe, which fits into a special recess in the cylinder head. Thermocouples are also used to operate exhaust gas temperature (EGT) gauges on jet engines, because high temperatures can severely damage the turbine sections. They are fitted around the engine jet pipe and connected in parallel, so that the failure of one thermocouple does not adversely affect the overall reading.

INTRODUCTION

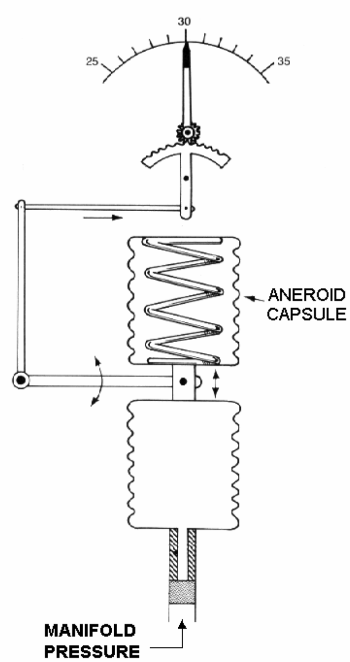
Various pressure and temperature indicators are used on board an aeroplane to control engine parameters.

PRESSURE INDICATORS

A number of pressures on an engine are sensed. These include gas pressures and fluid pressures, although all indicators use the same basic principles.

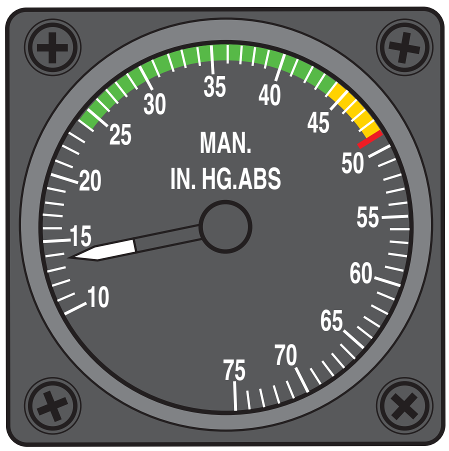
Manifold Pressure (MAP)

These indicators, colloquially termed **boost gauges**, are of the direct-reading type, and are calibrated to measure absolute pressure in inches of mercury. This pressure is representative of that produced at the induction manifold of a supercharged piston engine. In order to measure the pressure delivered by the supercharger and obtain an indication of engine power, it is necessary to have an instrument that indicates absolute pressure. This measures between the throttle valve and inlet valve. Illustrated below is the mechanism of a typical indicator.



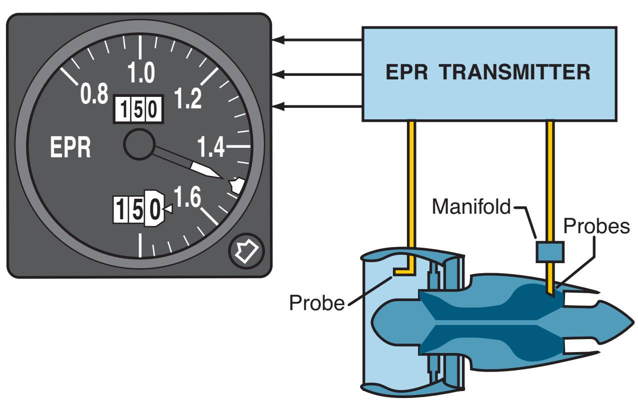
In this system, two bellows make up the measuring element, one open to the induction manifold and the other evacuated and sealed. A controlling spring fits inside the sealed bellows and the distension of both bellows, transmits to the pointer via a lever, quadrant, and pinion mechanism. A filter is located at the inlet to open the bellows, where there is also a restriction to smooth out any pressure surges.

When pressure applies to the open bellows, the latter expands causing the pointer to move over the scale (calibrated in inches of mercury) and indicate a change in pressure from the standard sea level value of 29.92 in Hg (zero boost). With increasing altitude, there is a tendency for the bellows to expand a little too far because the decrease in atmospheric pressure acting on the outside of the bellows offers less opposition. This tendency is counteracted by the sealed bellows, which also sense the change in atmospheric pressure, but expands in the opposite direction. Thus, a condition is reached at which the forces acting on each bellow are equal. This cancels out the effects of atmospheric pressure, and the manifold pressure measures directly against the spring. Shown below is a typical manifold pressure gauge.



Engine Pressure Ratio (EPR)

This measuring system consists of an engine inlet pressure probe, a number of pressure-sensing probes projected into the exhaust unit of the engine, a pressure ratio transmitter, and an indicator. Schematically shown below is the interconnection of these components based on a typical system.



The inlet pressure (P1) sensing probe is similar to a pitot probe, and is mounts so that it faces into the airstream in the engine intake or, as in some power plant installations, on the pylon, and near the air intake. The probe is also protected against icing by a supply of warm air from the engine anti-ice system.

Pipelines terminating at a manifold interconnect the exhaust pressure-sensing probes (PEX) in order to average out the pressures. In some engine systems, pressure sensing is done from chambers contained within the EGT sensing probes. A pipeline from the manifold, and another from the inlet pressure probe, is each connected to the pressure ratio transmitter. The indicators may be of the servo-operated type, but in electronic display systems, the transmitter output signals are supplied directly to the appropriate system computer. If a circuit malfunction occurs, an integrity monitoring circuit within the indicator activates a warning flag circuit, which obscures the digital counter display.

Provided in some aeroplanes is a maximum allowable EPR limit indicator. Integrated with a TAT indicator and with the CADC, its purpose is to indicate limits related to air density and altitude values from which thrust settings have been predetermined for specific operating conditions. These conditions are climb, cruise, continuous, and go-around. They are selected as appropriate by means of a mode selector switch, which is connected to a computing and switching circuit, and generates a datum signal corresponding to each selected condition. The signal, supplied to a comparator, also receives temperature signals from the TAT sensor and altitude signals from the CADC. These signals are compared with the datum signal and the lower value of the two is automatically selected as the signal representing the maximum EPR limit for the selected operating condition. The Comparator transmits this signal to an amplifier, and then to a servomotor, which drives a digital counter to display the limiting values.

Fuel and Oil Pressures

The oil pressure gauge is the most important gauge for satisfactory engine operation. If the oil pressure fails, bearing failure occurs quickly. The face of the gauge has a green arc showing the normal pressure range, a yellow arc for the caution range and a red line for maximum oil pressure. It is important that the oil pressure register on the gauge within 30 seconds of the engine starting. Metal capsules (or diaphragms) are normally used as sensors for measuring oil and fuel pressures. These connect by an electrical transmission system to the indicators on the flight deck.

TEMPERATURE INDICATORS

The types of sensors used in indicating engine-associated temperatures are:

Cylinder Head Temperatures

High temperatures are sensed in this area of the engine so a probe capable of withstanding these values is used. This would typically be a thermocouple, which can be in the form of a gasket beneath the spark plug, or a bayonet type probe placed in the hottest part of the engine within a recess. The gauge is positioned at the **cold** junction.

Exhaust Gas Temperatures

In a reciprocating engine aeroplane, the EGT gauge is used to manually lean the fuel-air mixture for better economy. Rich mixtures reduce the engine exhaust gas temperature and weak mixtures increase it. Any adjustment of the mixture using the EGT occurs in the cruise and below 75% power. A thermo-couple probe, installed in the exhaust pipe and connected to a simple gauge on the flight deck, facilitates this. If the EGT gauge is used for leaning the mixture there is no redline on the gauge, but in a turbo-supercharged reciprocating engine aeroplane the gauge has a redline because the turbo-supercharger can be damaged by high temperatures. In gas turbine engines, a thermo-couple is also for use in measuring the **Turbine Gas Temperature (TGT)**, which is sometimes referred to as **exhaust gas temperature**, or **Jet Pipe Temperature (JPT)** This is because this type of engine can be severely damaged by high temperatures in the turbine sections, which in some regions can reach up to 1100°C. Thermocouples are used in parallel with multiple pick-up points, so that the failure of one does not affect the reading.

Fuel and Oil Temperature

These are of the variable resistance type with the sensitive element contained in a bulb and immersed in the fluid. This data transmits by a Wheatstone bridge arrangement to a gauge on the flight deck.

Chapter 38
RPM Indicators and Propeller Synchroniser Systems

INTRODUCTION

The measurement of engine speed is an extremely important parameter. Together with manifold pressure, torque, and exhaust gas temperature, it allows the accurate control of the engine's performance.

The measurement of the speed of a reciprocating engine is at the crankshaft, whilst the measurement with turboprop and turbojet engines is by the rotational speed of the compressor shaft, which gives a useful indication of the amount of thrust being produced. These instruments are normally referred to as **Tachometers**, and operate either mechanically or electrically.

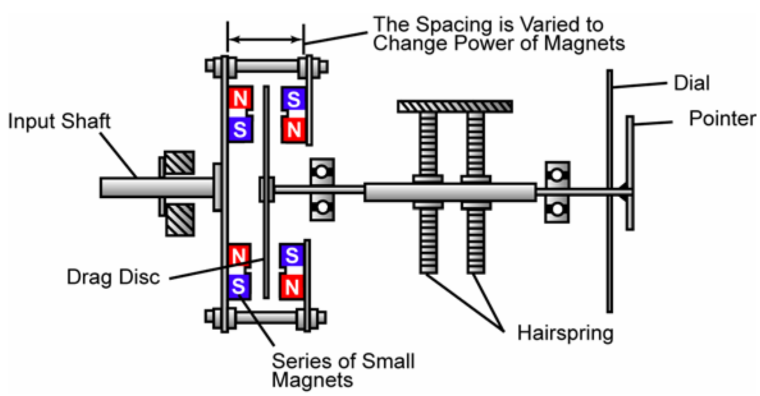
In the case of aeroplanes fitted with multi-propeller installations, are carefully matched to reduce flight crew workload by automatically reducing the noise and vibration during the cruise.

TACHOMETERS

The main types of tachometer are:

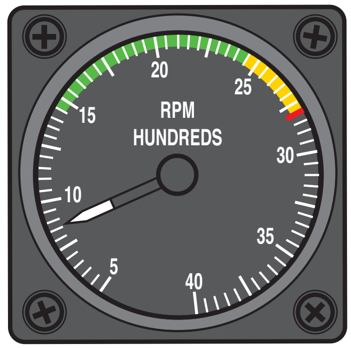
MAGNETIC DRAG TACHOMETER

This type of tachometer is like a car speedometer and for use on a light aeroplane. It uses a series of small permanent magnets, which are rotated via a flexible shaft at half the engine speed, from a spur gear on the engine.



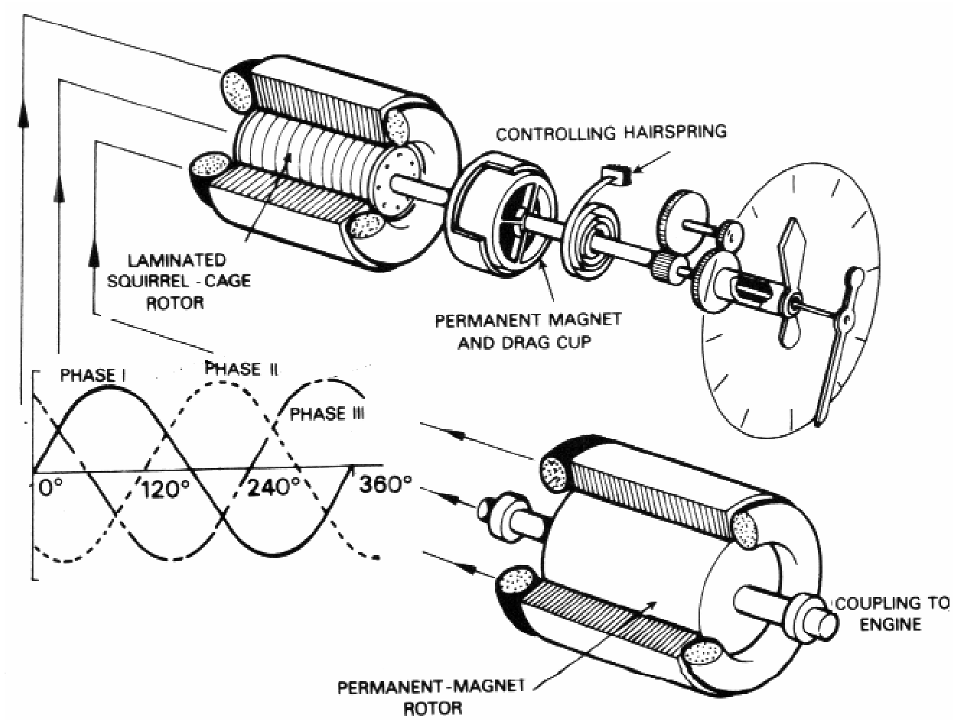
A highly conductive metal cup or disc (copper or aluminium alloy) mounts on a shaft, which is free to rotate in very low friction bearings within the rotating magnetic field. The shaft also carries a pointer, positioned by a calibrated hairspring so that it registers zero when the magnets are at rest.

As the magnets rotate the resultant magnetic field induces eddy currents in the disc, which then interact with the magnetic field, and drag it along with it, hence the name **Drag Cup**. The eddy currents are such that the amount of drag increases proportionally with speed, whilst hairsprings apply torque to the system. The torque produced is proportional to the rotation of the drag cup shaft, and the pointer correspondingly rotates over a linearly spaced dial, as shown on the next page. The cautionary operating range is clearly marked with a yellow arc, whilst a green arc indicates the normal operating range.



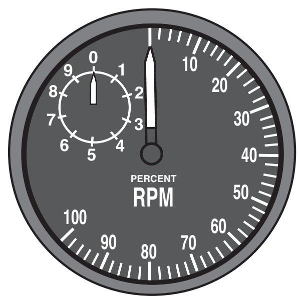
TACHO-GENERATOR AND INDICATOR SYSTEM

This system uses a remotely driven tachometer. The detector (or pick-up) is an AC type generator, which consists of a permanent magnet rotor that rotates within a slotted stator. The AC generator bolts directly to a mounting pad at the appropriate accessories drive gear outlet from an engine, and a splined shaft coupling drives the rotor. In order to limit the mechanical loads on generator, ratio gears are used in the engine drive system to reduce the operating speed of the rotor.



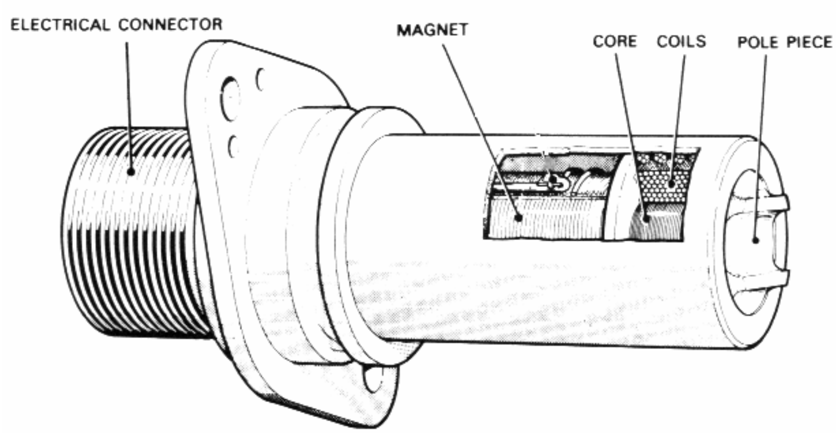
The signal from the detector unit passes through a synchro system to the indicator unit. A typical indicator consists of two interconnected elements, a **driving element,** and a **speed-indicating element**.

Three wires connect the AC generator to the indicator unit, and as the permanent magnet rotor rotates within the stator, a three-phase AC supply, whose frequency and voltage is proportional to the engine speed, generates. The output from the generator feeds directly to stator of an AC three-phase Squirrel Cage Induction motor, which in turn drives a cylindrical permanent magnet on a shaft. As the magnet rotates, it induces eddy currents into a drag cup, whose control of rotation is by a calibrated hairspring, which attaches to one end of the shaft. At the front end of the shaft, a gear train is coupled to two concentrically mounted pointers; a large one indicating hundreds of rpm, and a small one indicating thousands of rpm, as shown below. The indicator reads percentage of rpm.

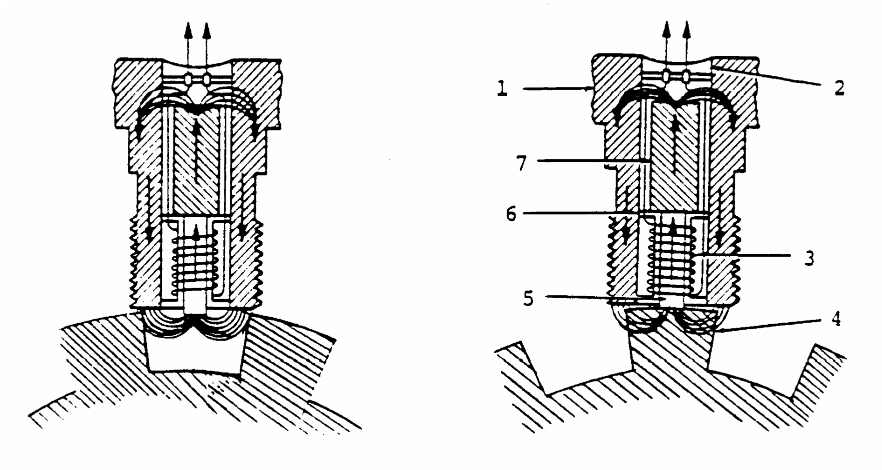


TACHOMETER PROBE AND INDICATOR SYSTEM

This system has the advantage of providing a number of separate electrical outputs in addition to those required for speed indication (e.g. automatic engine control and flight data recording). It has the advantage of no moving parts, and thus is not subject to high rotational loads.

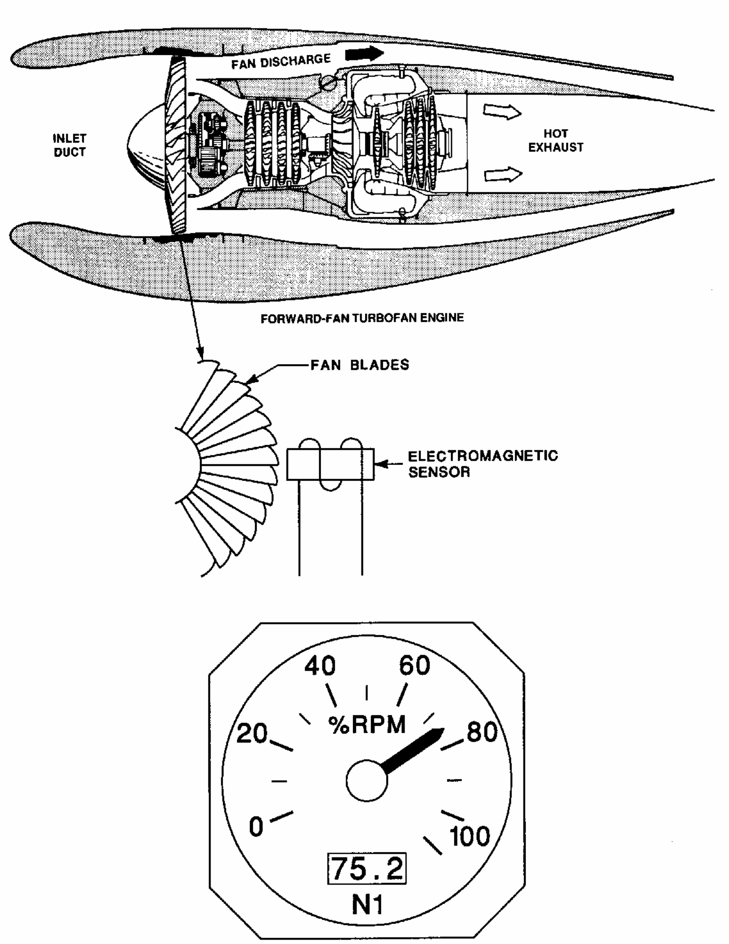


The stainless steel, hermetically sealed probe comprises of a permanent magnet, a pole piece, and a number of ‘Cupro-Nickel’ or ‘Nickel-Chromium’ coils that wind around a ferromagnetic core, as shown below. Separate windings (from five to seven depending on the type and application of a probe) provide outputs to the indicator and other processing units requiring engine speed data.



1. Body
2. Potting
3. Coil
4. Gear
5. Soft Iron Core
6. Spool (insulation)
7. Permanent magnet

This type of probe mounts on the engine at a station in the high-pressure compressor section so that it extends into this section. In some turbofan engines, a probe may mount at the fan section for measuring fan speed. The fan speed indicating system is, in effect, a fan blade counting device. The mounted sensor heads are flush in the fan shroud panel, and contain permanent magnets.

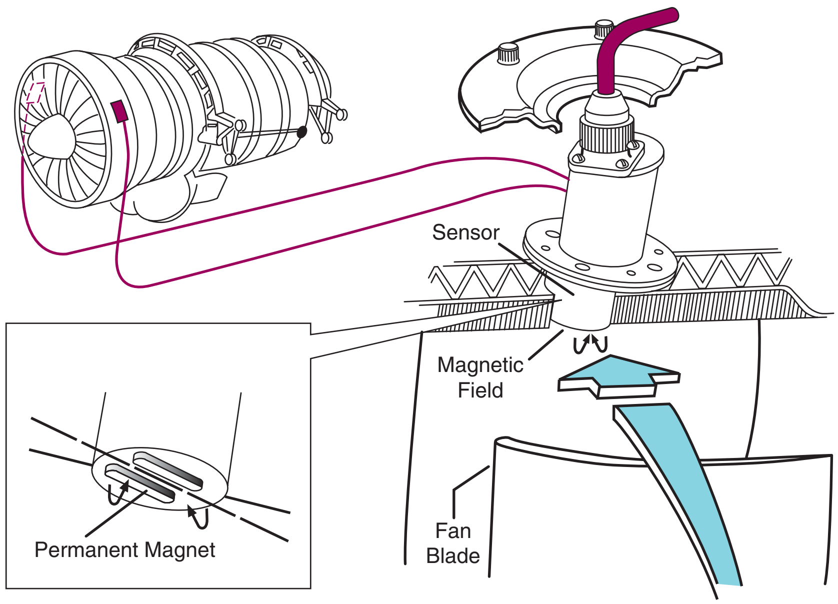


The passage of each fan blade disrupts the magnetic field set up by the sensor magnets, and causes an electrical signal pulse. The frequency of the pulses is equal to the number of blades times the rpm, thus giving a signal frequency, which is proportional to the fan speed. The signal is then amplified, conditioned and transmitted to the cockpit indicator in order to provide a N_1 readout in percentage rpm.

To ensure correct orientation of the probe, provided in the mounting flange is a locating plug, as shown on the next page. The permanent magnet produces a magnetic field around the sensing coils, and as the fan blades pass the pole pieces, the intensity of flux through each pole varies inversely with the width of the air gap between the poles and the blades. As the blades move, the air gap varies, and an induction of EMF occurs in the sensing coils, the amplitude of which varies with the rate of flux density change

The output signals (from the probe) then pass through a signal-processing module, and then through a servo amplifier to a torque motor. This in turn rotates a pointer and indicates the change in probe signals, in terms of speed. Supplied to the servo potentiometer is a reference voltage that controls the summation of signals to the servo amplifier and ensures that signal balancing occurs at the various constant speed conditions. In the event of a power supply or signal failure, the design of the pointer of the indicator is to return to an ‘off-scale’ position under the action of a pre-loaded helical spring.

In this type of indicator, the indication of a power failure differs in that a flag also energises to obscure the counter display.

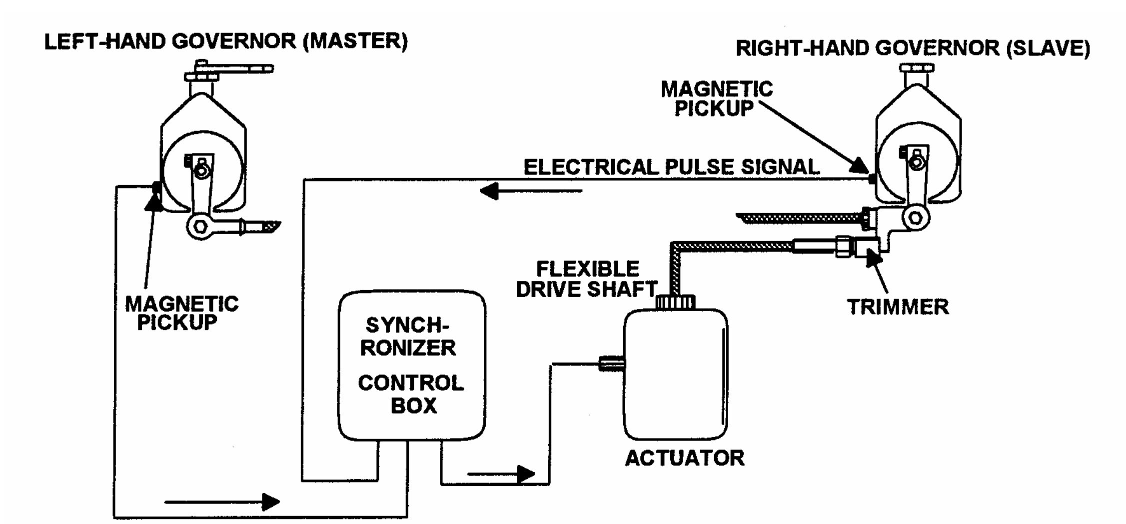


PROPELLER AUXILIARY SYSTEMS

Propeller auxiliary systems include systems, which increase the efficiency of the propeller operation, and provide automatic operation of the Constant Speed Unit (CSU) or Propeller Control Unit (PCU), and feathering mechanisms. This increases safety, and reduces the workload of the flight crew. The following systems are found on either light twin-engine aeroplanes, or large turbo-propeller aeroplanes.

SYNCHRONISATION SYSTEM

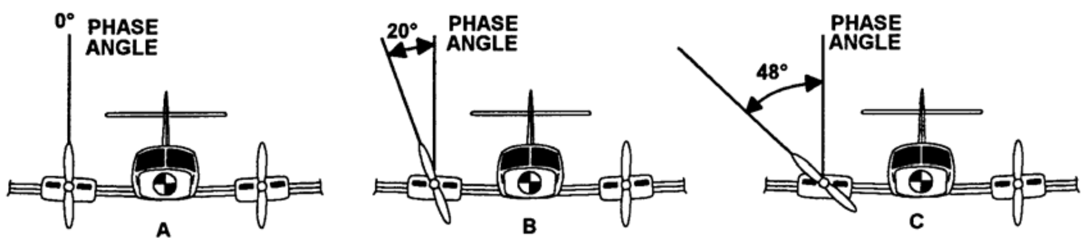
The synchronisation system is used to set all of the engine's CSUs/PCUs at exactly the same rpm, thus eliminating excessive noise and vibration. It also avoids the need for the flight crew to continually adjust the engine controls. Fitted to each engine is a **Tacho-generator** or a **Frequency Generator**, and these generate a signal proportional to engine speed. One of the engines acts as the **Master** engine, whilst the other engine(s) act as the **Slave** engine(s). The slave engines always maintain the same rpm as the master. On four-engine aeroplanes any engine can be selected to act as the master, but on a light twin-engine aeroplane it is always the left engine. The diagram below shows a typical two-engine synchronisation system.



A synchroniser control box compares the rpm signal of the slave engine(s) to the rpm signal of the master engine. The engine that generates the higher voltage or frequency, determines the direction in which the actuator rotates, and adjusts the CSU/PCU spring setting, which in turn adjusts the rpm. Generally, the rpm of the slave engines must be within approximately 100 rpm of the master engine for synchronisation to occur. This system is used during all phases of flight, except for take-off and landing, when failure of the master engine would result in all the engines attempting to follow the master engine.

SYNCHROPHASING SYSTEM

Synchrophasing is a refinement of Synchronisation, and allows the pilot to set the blades of the slave engines a number of degrees in rotation behind the blades of the master engine.

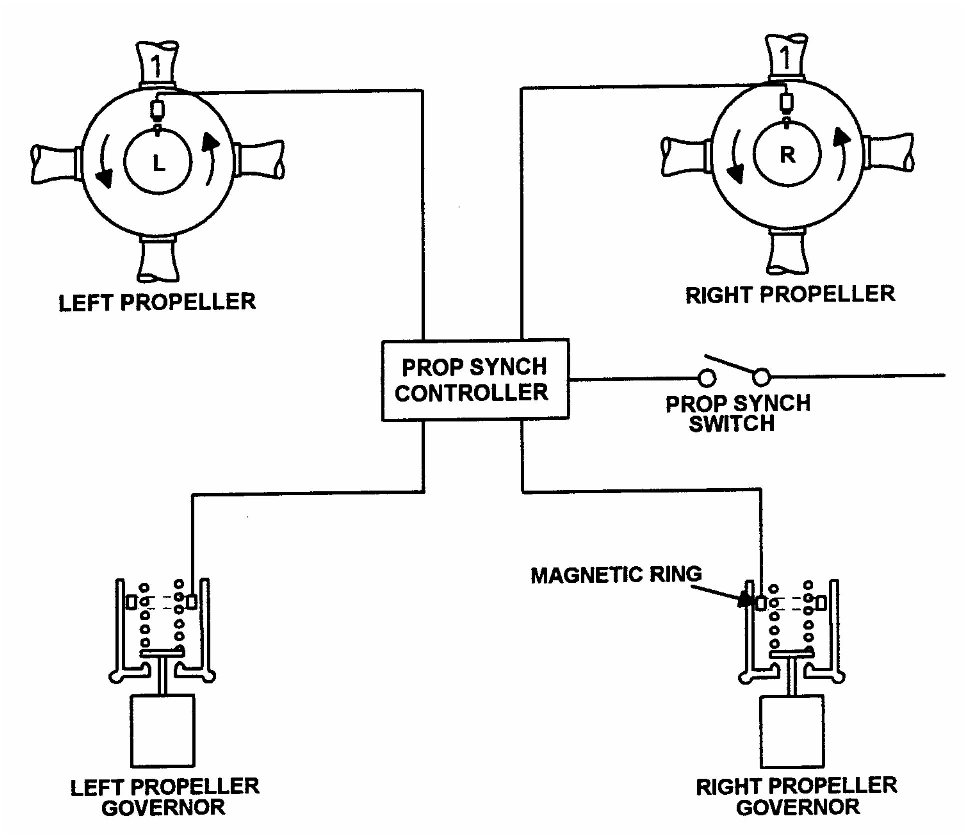


SYNCHROPHASING SETS THE BLADES OF THE PROPELLERS AT DIFFERENT ANGLES AND MAINTAINS THE SAME RPM

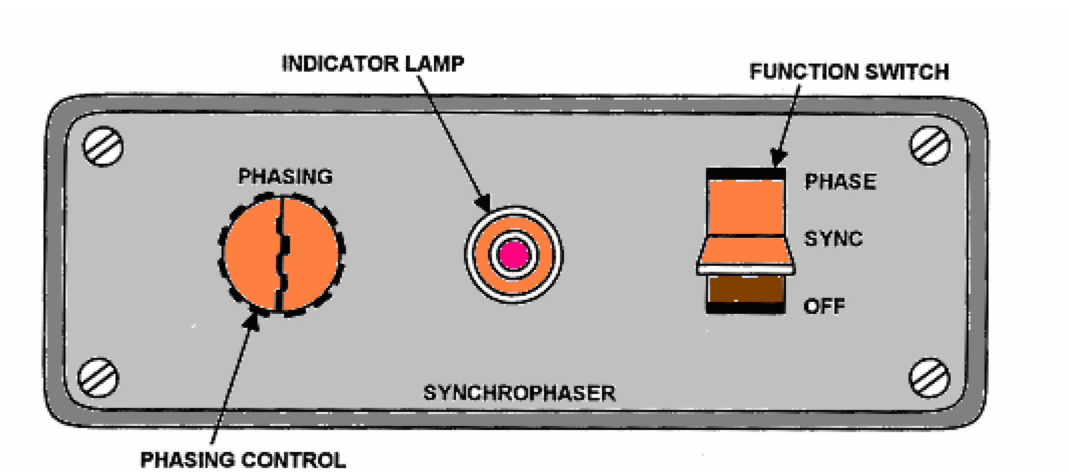
This system further reduces noise and vibration. The Synchrophasing angle varies by the pilots' adjustment for different flight conditions to achieve a minimum noise level.

OPERATION OF A SYNCHROPHASING SYSTEM

Fitted to the same blade root of each propeller is a pulse generator (e.g. No. 1 blade), and the signals generated are ensure that all of the No. 1 blades are in the same relative position at the same instant.



The pulse generator serves the same function as the tacho-generator does in the synchronisation system. By comparison, when the signals from the slave pulse generators occur in relation to the master engine pulse, the mechanism synchronises the phase relationship of the slaves to the master engine. A propeller phase control in the cockpit then allows the flight crew to select the phase angle, as shown below, which gives the minimum amount of vibration.



Chapter 39
Engine Torque Measurement

INTRODUCTION

The power produced by a propeller is proportional to the torque, described best as the turning moment produced by the propeller about the axis of the output shaft.

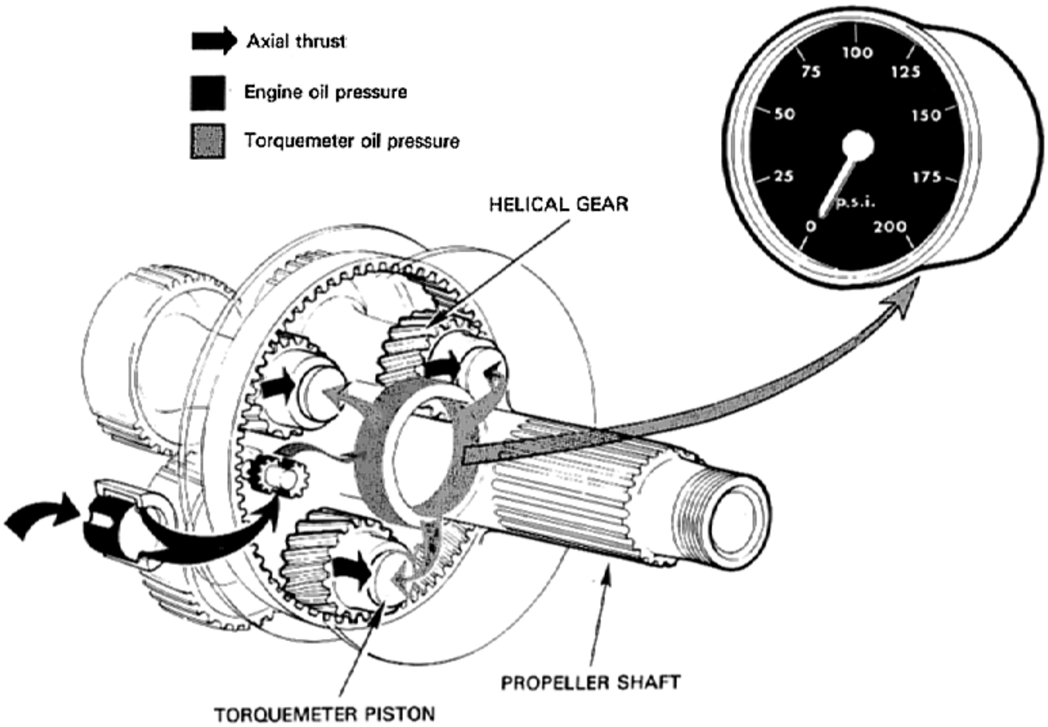
A torque meter on the flight deck indicates the power produced by a turbo-propeller engine. The torque meter system forms part of the engine itself, and is usually built in with the reduction gear assembly between the output shaft and the propeller shaft.

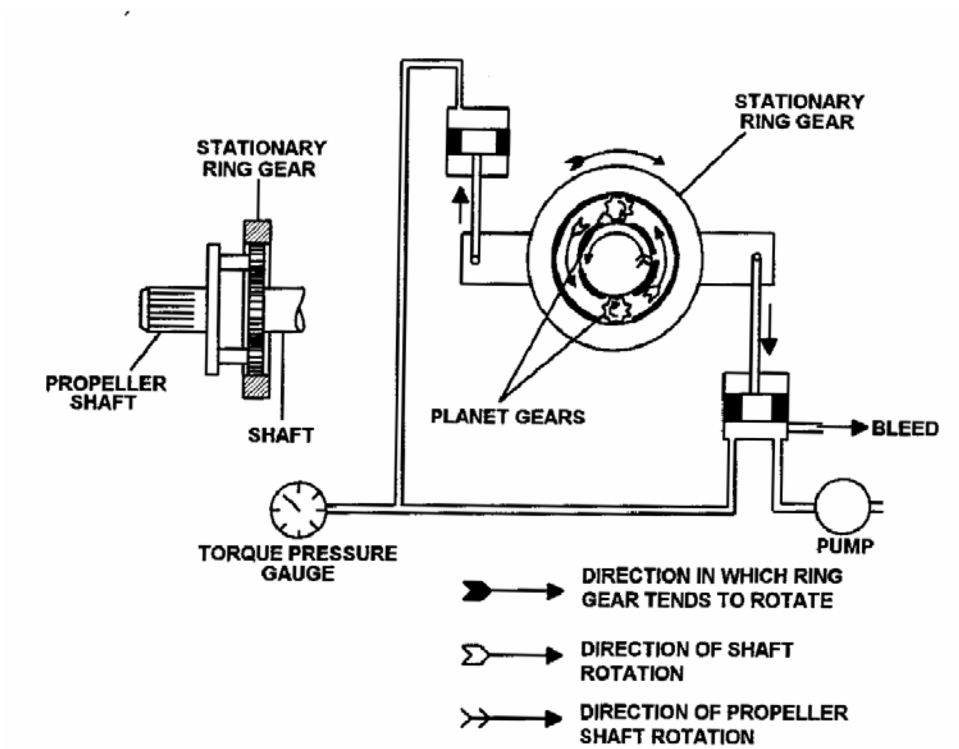
TORQUE METER

Illustrated on the next page is a typical torque meter system.

In this system, the helical gears, driven by the engine shaft, create an axial thrust, and oil pressure, acting on a number of pistons, resists this axial thrust, which is in turn directly proportional to the torque. This value then transmits to a suitably calibrated indicator dial. One advantage of this type of torque measuring device, as fitted on a turboprop aeroplane, is that the system is also for use in operating the propeller-feathering device if the torque meter oil pressure suddenly reduces due to a power failure. On some aeroplanes, it is additionally used to automatically operate the water injection system that boosts the take-off power when operating at high altitude/ high temperature aerodromes.

The system on the next page is alternatively based on the principle of the tendency for some part of the reduction gear to rotate, and is resisted by pistons working in hydraulic cylinders, which are secured to the gear casing. The pressure created by the pistons then transmits to a pressure gauge on the flight deck.





Torque may also be sensed by an electrical strain gauge system. A strain gauge is a fine insulated conductor wire bonded to a component. When applying strain (i.e. deflection under load), the resistance of the wire changes, and this is measured in a ‘Wheatstone Bridge’ circuit.

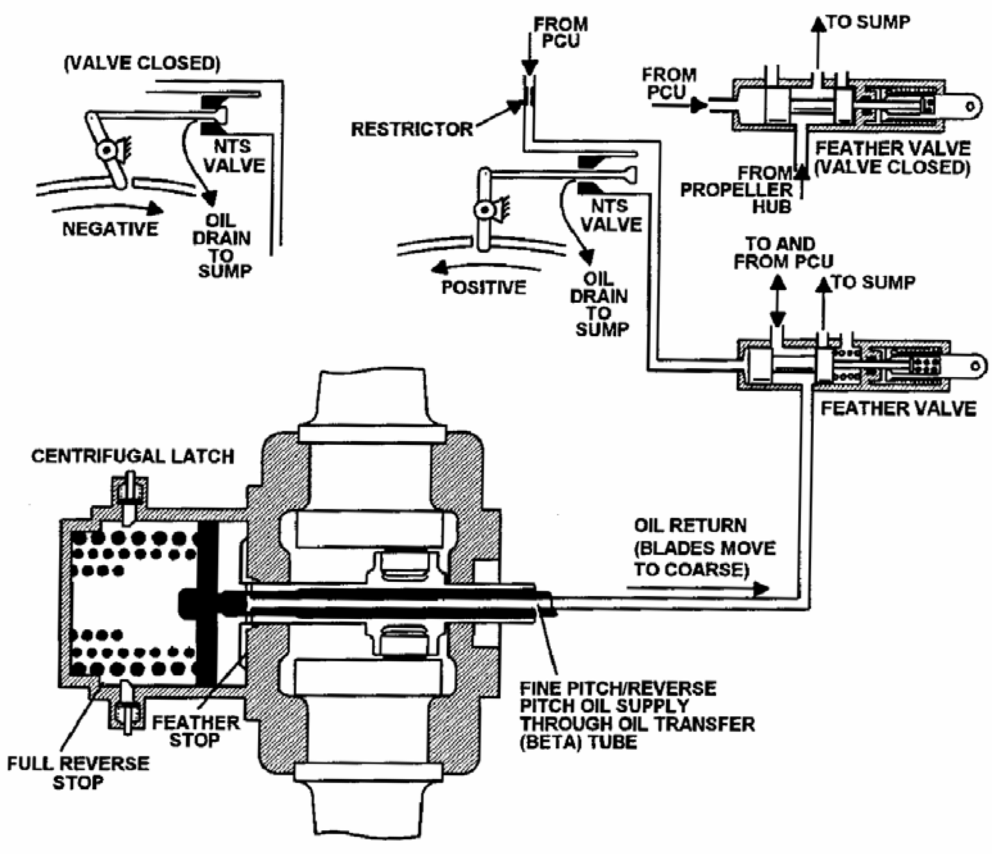
NEGATIVE TORQUE SENSING

The Negative Torque Sensing (NTS) system, as shown on the next page, is designed to prevent the propeller driving the engine (e.g. in the in-flight idle power setting) during a lean fuel schedule, or temporary fuel interruption, or due to air gusts momentarily acting on the propeller.

The NTS device takes its signal from the torque indicating system, and at a specified torque, the NTS activates the propeller control system, causing the propeller blades to coarsen until the torque value rises above a specified value. This system is therefore a blade coarsening system, and not necessarily a feathering system.

During negative torque, the gear ring closes the NTS valve, and oil pressure feeds to the feather valve, forcing it to the right. Oil releases from the propeller hub, allowing the feathering spring to coarsen the blades. When reapplying positive torque to the gear wheel, the NTS valve opens, thereby releasing the oil, allowing the feather valve to move to the left, and allowing propeller operation to resume normally.

When activating the NTS, the torque meter fluctuates, tending to indicate engine failure, and manual feathering should be carried out.



Chapter 40
Vibration Monitoring

INTRODUCTION

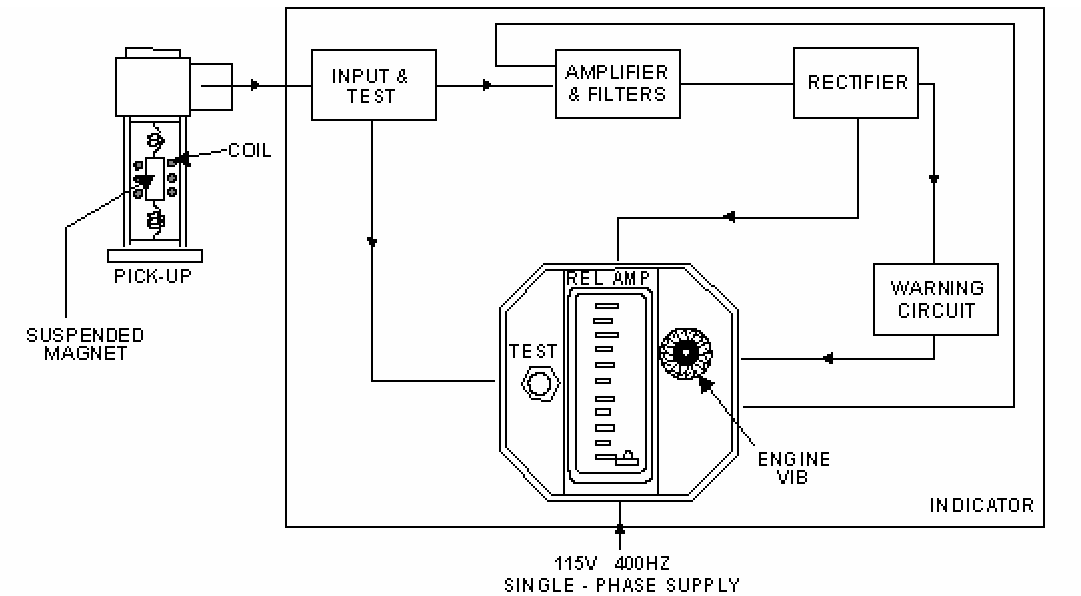
A turbo jet engine has a very low vibration level, and any change in vibration could be the first indication of an impending problem, which could easily go unnoticed. Any problems may be caused by:

- A damaged compressor blade
- A turbine blade that has a crack or is subject to creep
- An uneven temperature distribution around the turbine blades and rotor discs may be set up.

Any of these problems could result in an unbalanced condition of the main rotating assemblies, and could lead to possible disintegration. In order to provide the flight crew with a timely warning of increased vibration, jet engines are fitted with vibration monitors, which continually monitor their vibration levels. On a multi-jet aeroplane, a monitor is fitted to each engine, and these are designed to indicate if the maximum amplitude of vibration of the engine exceeds a pre-set level. These indicators are located within the control group of instrumentation and are usually milli-ammeters that receive signals through an amplifier from engine-mounted transmitters, and display as relative amplitude.

VIBRATION MONITORING SYSTEM

The monitor is mounted on the engine casing and consists of a vibration pick-off (sensor), which is mounted at right angles to the engine axis, an amplifier-monitoring unit, and an indicator calibrated to show vibration amplitude in thousandths of an inch (mils) as shown below.



The sensor (transducer) is a spring-supported permanent magnet, suspended in a coil attached to the interior of the case. As the engine vibrates, the sensor unit and core move with it, but the magnet tends to remain fixed in space because of its inertia. The motion of the coil causes the turns to cut the field of the magnet, thus inducing a voltage in the coil and providing a signal to the amplifier unit. The signal, after amplification and integration by an electrical transmission system, feeds to the indicator via a rectifying section.

An amber indicator light also forms part of the system, together with a test switch. The light is supplied with dc from the amplifier rectifying section and comes on if the maximum amplitude of vibration exceeds a pre-set value. A test switch also permits functional checking of the system's electrical circuit. In some engine installations, fitted to an engine are two sensors: for example, in a typical turbofan engine, one monitors vibration levels around the fan section, and the other around the engine core section. A system of filters in the electrical circuit to the gauge makes it possible to compare the vibration recorded against a known frequency range, and so enables the source of vibration to be located. A multi-selector switch enables the flight crew to select a specific area, and to obtain a reading of the level of vibration.

In systems developed for use in conjunction with LCD and CRT display indicators, the vibration sensors are of the type whereby vibration causes induced signals in a piezoelectric sensor stack.

INTRODUCTION

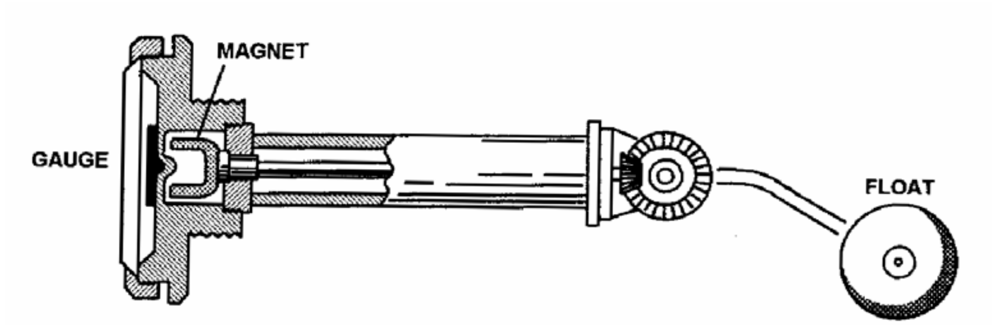
Fuel quantities and flows are essential measurements for the safe and efficient conduct of a flight. Fuel quantity is of great interest to the flight crew, since there must be sufficient fuel on board to complete the flight, and maintain adequate reserves. A check on the amount of fuel remaining against the planned quantity at significant route points reveals any excessive fuel burn. Fuel quantity may also be given in volume or mass.

MEASUREMENT OF FUEL QUANTITY

The following are types of fuel contents gauges.

FLOAT TYPE

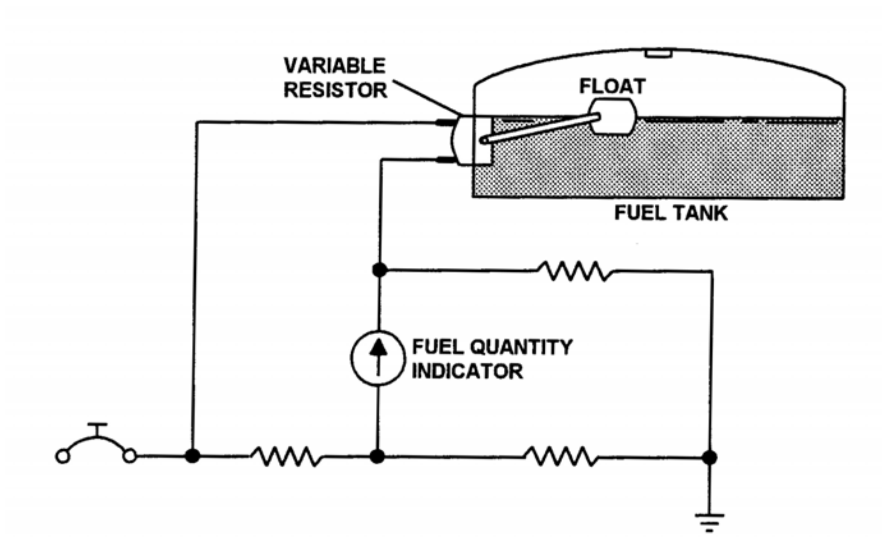
This type of gauge may be either electrical or mechanical. In a direct reading mechanical fuel gauge, a float drives a pointer directly, via gearing and a magnet. The pointer rotates over a dial, calibrated in units of volume (e.g. gallons), and separated from the fuel.



The electrical gauge is similar to the mechanical type, except that in this system the float is arranged so that it drives an electrical design type transmitter, enabling the remote positioning of the gauge from the fuel tanks.

RATIO METRE TYPE FUEL GAUGE

Most commonly used on light aeroplanes, this type of system indicates the quantity of fuel by volume in a fuel tank.

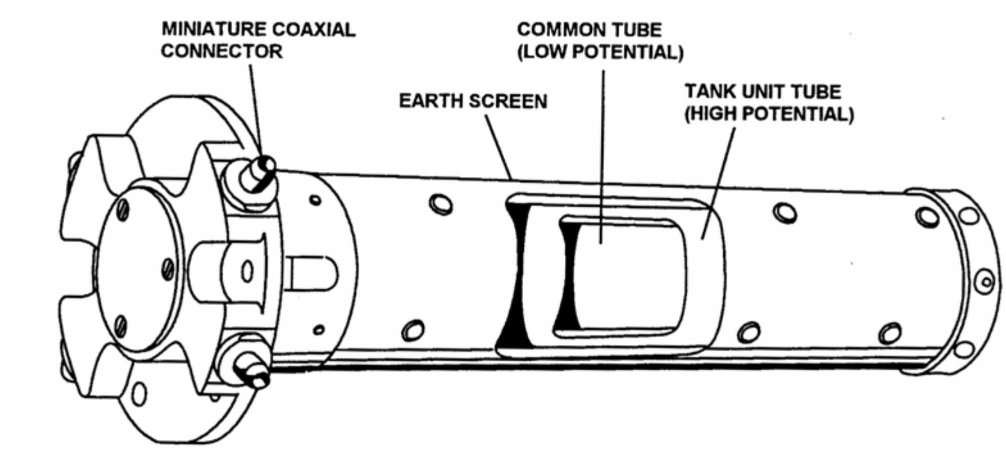


As the level of the float changes, it moves a wiper arm, which alters the resistance of a variable resistor, and changes the overall resistance in the circuit. This in turn alters the current flow in a DC circuit, giving an indication on a gauge similar to a voltmeter or ammeter. The gauge uses two opposing magnetic fields and a pointer, which react to the ratio of the current flow in the two sections.

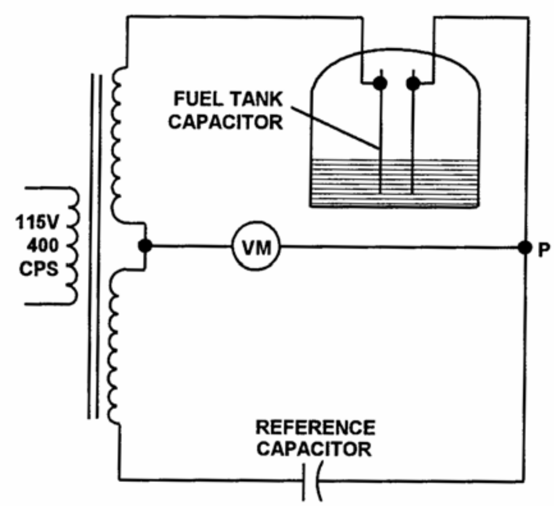
The float and resistance systems are only accurate when the aeroplane is in steady straight and level flight. Additionally, if the aeroplane accelerates or the temperature changes, the volume of fuel indicated on the gauge also varies. This type of gauge is also subject to inaccuracies if the voltage fluctuates, which may be caused by voltage regulator settings or a weak battery.

CAPACITANCE TYPE OF FUEL GAUGE

This system is the most common type used on modern jet engine aeroplanes, and uses an electronic fuel-measuring device, which indicates fuel quantity. The quantity is measured in mass or weight, in pounds or kilograms, but not volume. The principle used in this type of gauge is that the capacitance of a capacitor of fixed dimensions is dependent on the dielectric between the plates. The tank units that form the plates of a capacitor consist of two concentric tubes, as shown below.

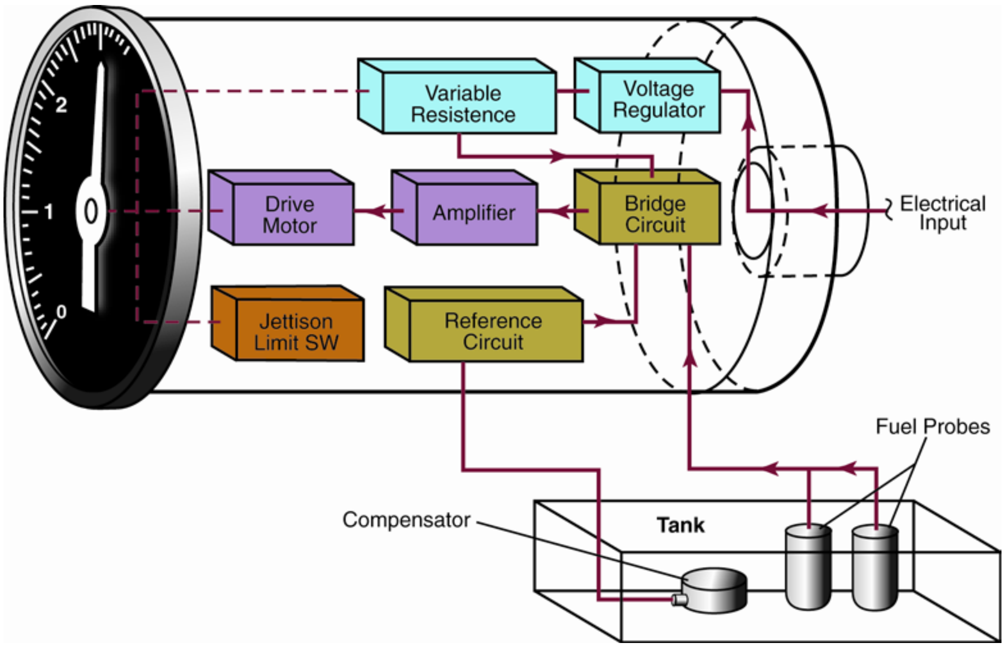


The space between the plates is filled with fuel, air, or a mixture of both, forming the dielectric of the capacitor; thus, the capacitance of the tank units is directly proportional to the amount of fuel in the tanks. The dielectric constant of jet engine fuel compared to air is approximately 2:1. Shown below is a simplified capacitance bridge circuit.



If the two capacitances are equal, the voltage drop across them is also equal, and the voltage between the centre tap and point P is zero. If the fuel quantity increases, the tank unit's capacitance similarly increases, and the bridge circuit becomes unbalanced. A voltmeter therefore indicates a voltage proportional to the change in capacitance, hence fuel mass. This is useful with jet engines because the amount of thrust produced by them is more a factor of the amount of mass consumed, rather than the volume of fuel consumed. This system also enables accurate readings to be produced in large or irregular shaped fuel tanks. A number of probes are normally fitted in each tank, depending on its size, and are connected in parallel, ensuring that the indications remain the same regardless of the attitude or wing flex of the aeroplane. The capacitors must also be matched or characterised to their specific locations, and the sum of their capacitance gives a measure of the actual quantity of fuel in the tanks. The tank units connect to an amplifier in place of a voltmeter, and the output drives a pointer, showing the total mass of fuel in the tank.

Any changes in temperature also affects the density of fuel, so the volume occupied by a given quantity of fuel increases if the temperature increases, and its density falls. To compensate for this error, a balancing (short) capacitance unit is installed at the lowest part of the tank where it will be completely immersed whenever there is a useable quantity of fuel in the tank, as shown below.



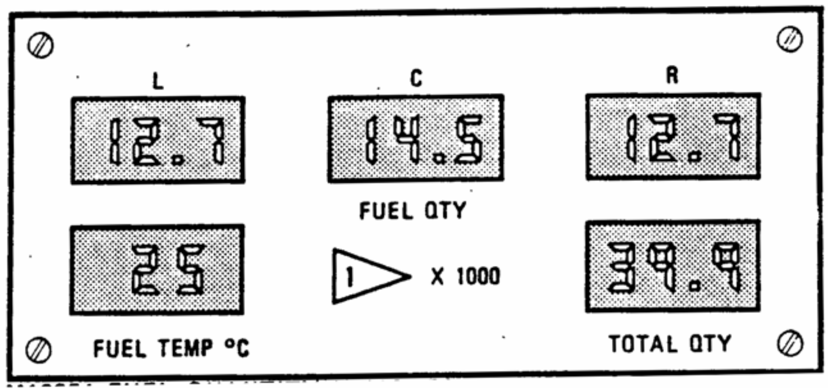
By totally immersing this unit, the only variation in the capacitance it measures is due to changes in the fluid density. This variation is thus used to adjust the calibration of the fuel quantity probes.

If there is a significant amount of water in the tank, it lies at the bottom of the tank, affecting the compensating or balancing probe, which results in an incorrect adjustment of the calibration. For example, if the tanks are drained and then refilled to the same level with water, the pointer will tend to move up the scale to a new point of balance. This is because water has a greater density and dielectric constant than aeroplane fuel.

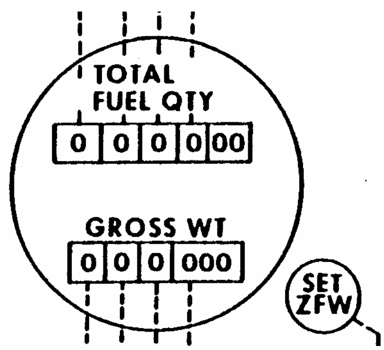
This method of quantity measurement may also be used for other fluids (e.g. Hydraulics).

FUEL TOTALISER

This device digitally displays the amount of fuel in each tank on a flight deck indicator and provides an indication of the total amount of fuel on board the aeroplane, as shown below.



On some aeroplanes, the Totaliser is combined with a gross weight (mass) indicator and uses two digital scale windows as indicated.



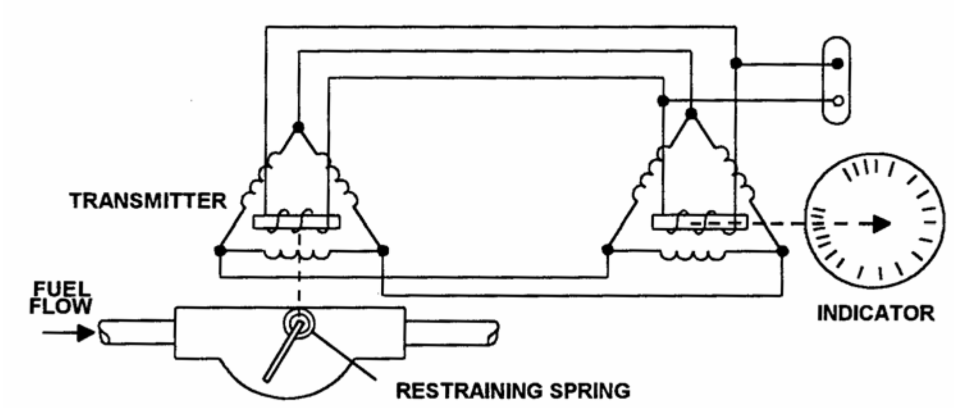
The upper scale shows the sum of the individual tank readings and the lower scale shows the aeroplane's gross weight, which is normally manually set after it has been loaded. As the fuel is used, the fuel remaining, and the weight scales reduce.

On some indicators, the flight crew may also be able to enter the zero fuel weight (ZFW) at the commencement of the flight, to display the gross weight throughout the flight.

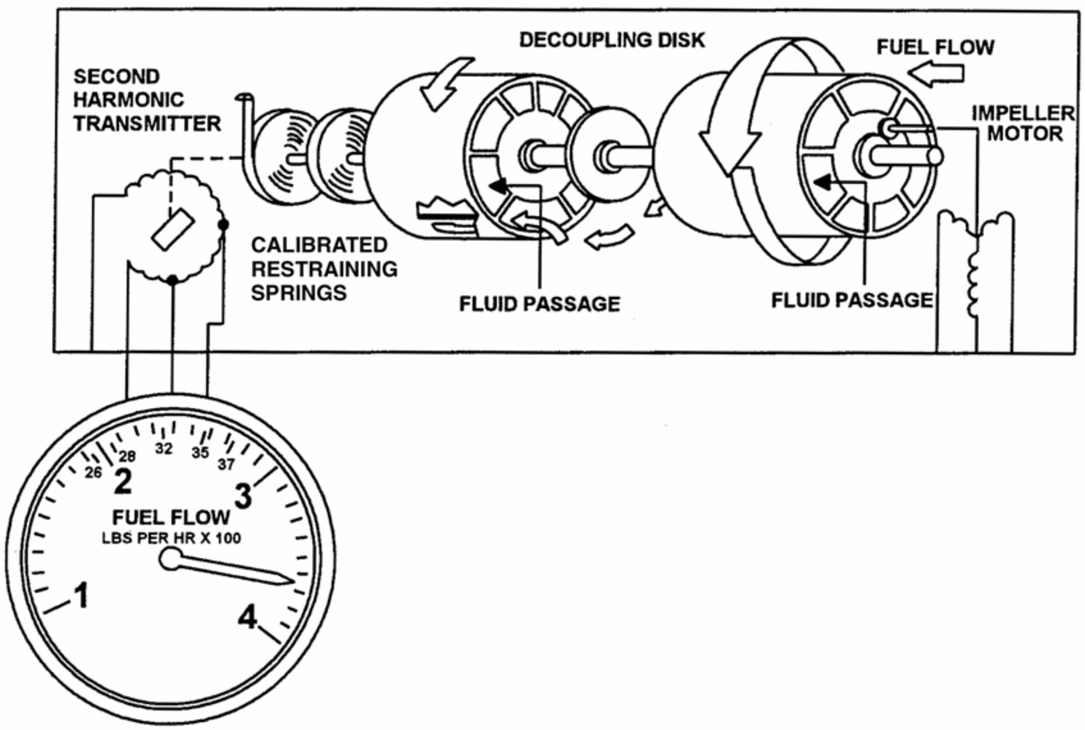
FUEL FLOW

Fuel flow metering systems are designed to provide the crew with a continuous indication of the instantaneous rate of fuel flow to each engine and in some instances, the amount of fuel consumed. Measurement of fuel flow is at the fuel intake of each engine (LP fuel supply line on a gas turbine). If the fuel flow increases for a particular engine power, it indicates a reduction in efficiency and probably an impending mechanical problem.

The basic method of measuring fuel flow is by a rotating vane flow meter, as shown on the next page. In this system, the shaft carrying the vane forms the transmitter of a synchro transmission system. The receiver rotor attaches to, and drives a needle moving against a scale calibrated in fuel used per unit time, which can be either volume or mass related.



A more complex system, which is in common use in large aeroplanes, is the integrated flow meter system, as shown below.



In this system the fuel flow to the engine is measured, and is not only presented on a suitable scale, but also passed to an integrator where it is processed (integrated) with respect to time to obtain the amount of fuel used. The transmitter/sensor unit consists of a tube, narrowed at the ends and fitted into the appropriate engine fuel-supply line. Within the tube, there is a motor driven impeller through which the fuel passes, which makes the fuel swirl at a rotation rate that varies with the flow rate. On leaving the impeller, the swirling fuel impacts a receiver turbine, inducing a rate of rotation of that turbine that is directly proportional to the swirl rate. This rotation is electronically detected and transmitted to the fuel flow indicator via a synchro system. Signals are also sent via the integrator, to the fuel-consumed indicator.

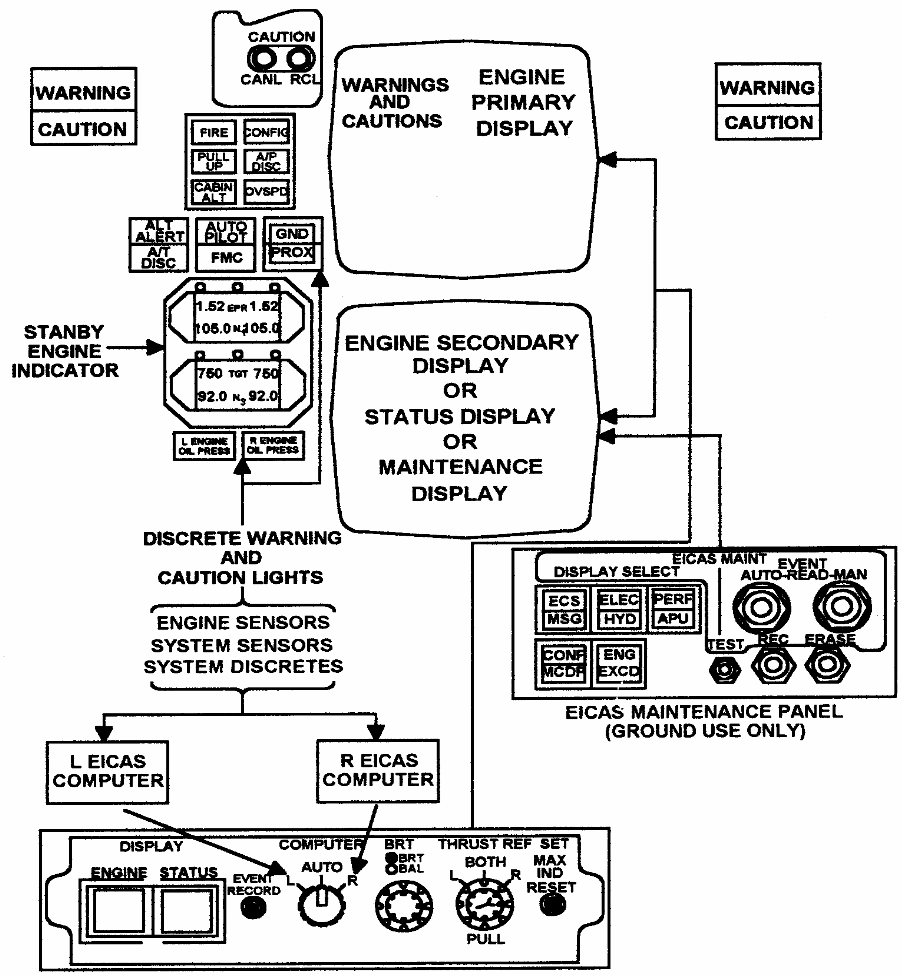
Chapter 42
EICAS

INTRODUCTION

The **Engine Indication and Crew Alerting System (EICAS)** displays primary engine indications and provides a centrally located crew alerting system for non-normal situations. The system also shows the status of systems not otherwise displayed on the flight deck. On the ground, EICAS additionally provides maintenance personnel with a variety of system data.

EICAS ARCHITECTURE

Two EICAS computers receive inputs from engine and system sensors. The information from the sensors displays on two Cathode Ray Tubes (CRTs) as dials and digital readouts of warnings, cautions, and advisory messages.



The parameters that require setting and monitoring by the flight crew permanently display on the screen, but the system also monitors the remaining parameters and displays them only if one or more exceed safe limitations.

A Computer Selector panel determines which computer controls the EICAS. When the selector is in the AUTO position the left computer is used, and if the left computer fails, control automatically switches to the right computer. When the L position is selected only the left computer can control the EICAS and when the selector is in the R position only the right computer can control the system. The EICAS computers monitor over 400 inputs through a comprehensive warning and caution system, to provide a quick and unambiguous identification of problems as they arise.

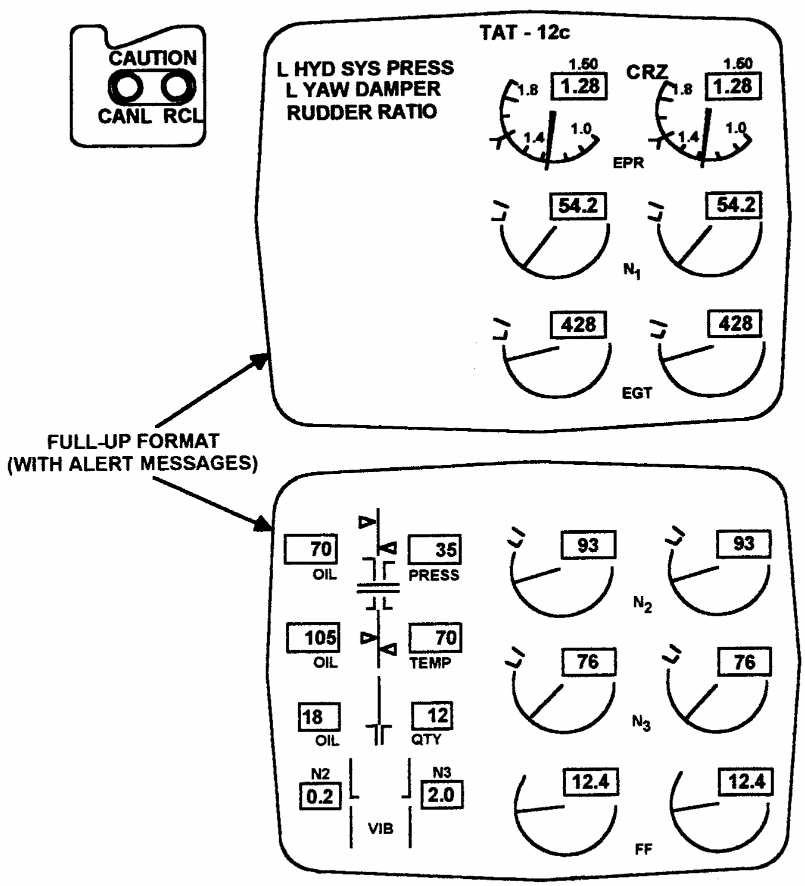
The Brightness and Balance Controls are used to adjust the brightness level of both CRTs.

The Event Record Switch is for use in storing systems data in an EICAS memory for later use by maintenance personnel. To record current data from the engine and system sensors, and erase any previously recorded data from the memory, push the switch.

System lights and a Standby Engine Indicator (SEI) provide backup indications for the CRT displays.

ENGINE DISPLAYS

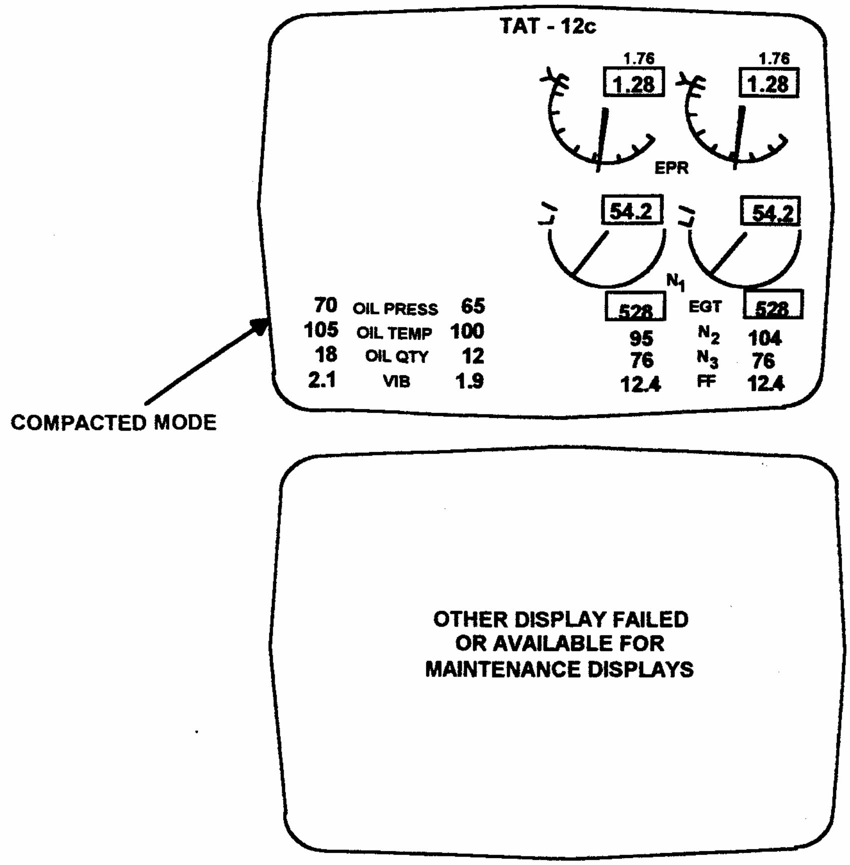
The CRT screens, located in the centre of the instrument panel above the throttle pedestal, typically display as follows:



Primary engine indications appear on the upper screen.

Secondary Engine Indications appear on the lower screen.

In the event of an upper or lower CRT failure, all engine indications transfer automatically to the other CRT, and the display appears in a compacted format.



This allows the flight crew to retain full capability in the event of a CRT failure, or if the lower CRT is being used for status displays (or maintenance displays on the ground). In the compacted mode, the primary engine indications show in their normal format, whilst the secondary information and oil system indications are shown in a digital format.

CREW ALERTING

The crew-alerting portion of EICAS continually monitors all of the aeroplane systems, and if a fault occurs, or any system fault light illuminates in the cockpit, the EICAS displays a crew-alerting message on the upper CRT. In addition to the display messages, aural tones and Master Warning/Caution lights also indicate crew alerts.

All crew alert messages divide into one of three categories:

WARNINGS (LEVEL A)

These are red, and reflect an operational or aeroplane system condition that requires immediate crew awareness and prompt corrective action. These are the most urgent types of crew alert, of which an engine fire is a typical warning.

CAUTIONS (LEVEL B)

These are amber, and reflect an operational or aeroplane system condition that requires immediate crew awareness and future compensatory action. These are less urgent than warnings; of which an engine overheat is a typical caution.

ADVISORIES (LEVEL C)

These are also amber, and reflect an operational or aeroplane system condition only for crew awareness that requires corrective action on a time available basis. Advisories are the least urgent type of crew alert. A yaw damper fault is a typical advisory.

If a parameter goes out of tolerance, an alert, status, or maintenance message generates automatically, depending on the urgency of the malfunction.

MASTER WARNING/CAUTION LIGHT

Two Master Warning Lights illuminate when a warning occurs, and they remain on as long as the warning exists, or until pressing either Master Warning/Caution reset switch.

Pushing the reset switch silences the fire bell and cabin altitude siren, and may silence the landing configuration siren, depending on the reason for its activation.

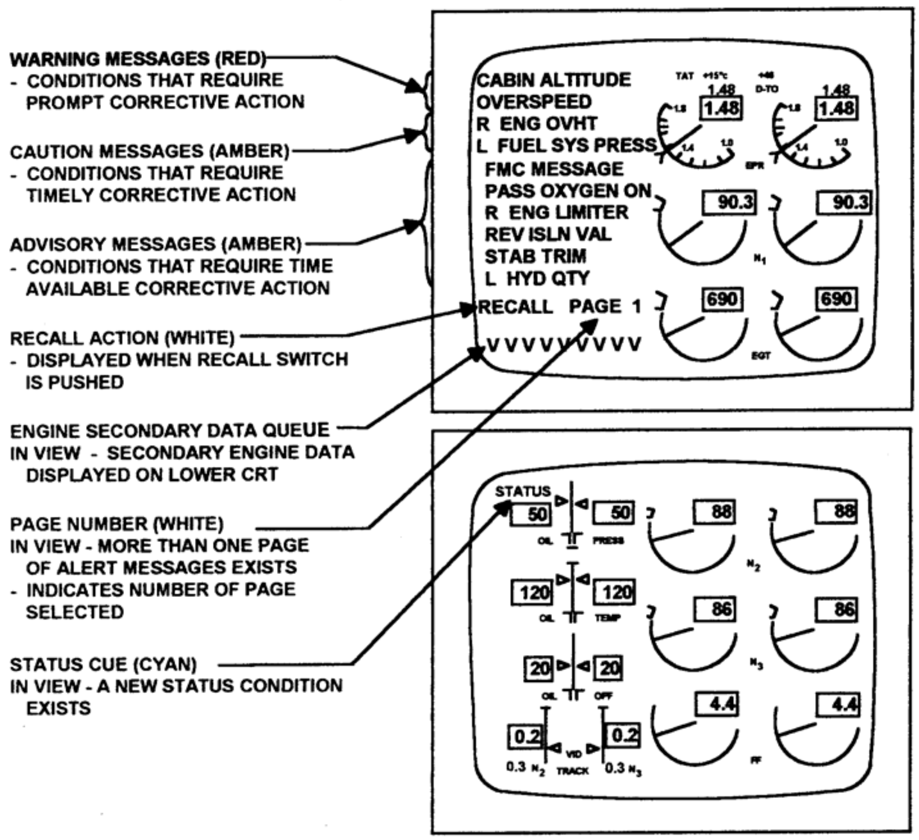
INHIBITS

Parts of the crew alerting system are inhibited or deactivated during certain phases of the flight to prevent distractions (e.g. the gear light illuminates as soon as the landing gear begins to retract, but the GEAR DISAGREE message is inhibited for 25 seconds to allow their normal stowage).

DISPLAY MESSAGES

Crew alerting messages appear on the upper CRT to indicate all non-normal conditions detected by EICAS, where up to 11 messages display. If more than 11 messages generate, the last message is removed and replaced by a page number. Page 2 can display by pressing the CANL (CANCEL) switch, and recall page 1 by pressing the RCL (RECALL) switch.

Shown below are some typical examples.



Warnings, indicated by red messages at the top of the message list, arrange according to their urgency and order of occurrence.

Cautions appear as amber messages below the lowest warning.

Indicated by amber messages, advisories appear below the lowest caution. They are indented by one space to distinguish them from cautions.

The most recent warning, caution, and advisory messages appear at the top of their respective group of messages.

When the associated condition no longer exists, a message from the display is automatically removed, and then all messages that appeared below the deleted message move up one line.

If a new fault occurs, its associated message is inserted on the appropriate line of the display, which may cause older messages to move down one line. For example, a new caution message causes all existing caution and advisory messages to move down a line.

If there are more messages than can display at one time, the lowest message is removed, and a white page number appears on the lower right side of the message list. Messages 'bumped' from the bottom of one page appear automatically on the next page.

The Cancel and Recall switches are used to manipulate the message lists. Pushing the Cancel Switch will remove the caution and advisory messages from the display.

Warning messages cannot be cancelled.

If there is an additional page of messages, pushing the Cancel Switch displays the next page. Warning messages carry over from the pervious page. When the last page of messages displays, pushing the switch once more removes the last caution and advisory messages, and the page number.

Pushing the Recall Switch displays the caution and advisory messages that were removed with the Cancel Switch if the associated faults still exist. If there is more than one page of messages, page one displays. After pushing the Recall Switch, a white RECALL message appears for about one second on the lower left side of the message list to indicate that the Recall Switch has been pushed.

New display messages appear on the page being viewed. For example, if selecting page three and a new caution occurs, the caution message appears on page three below any warning messages. If the Recall Switch is subsequently pushed, the new caution message appears as the top caution message on page one.

A single, more general display message can sometimes replace multiple display messages of a similar nature. For example, if only the forward or aft entry door is open on the left side, an L FWD ENT DOOR or L AFT ENT DOOR message appears. If both doors are open, only an L ENTRY DOORS message appears.

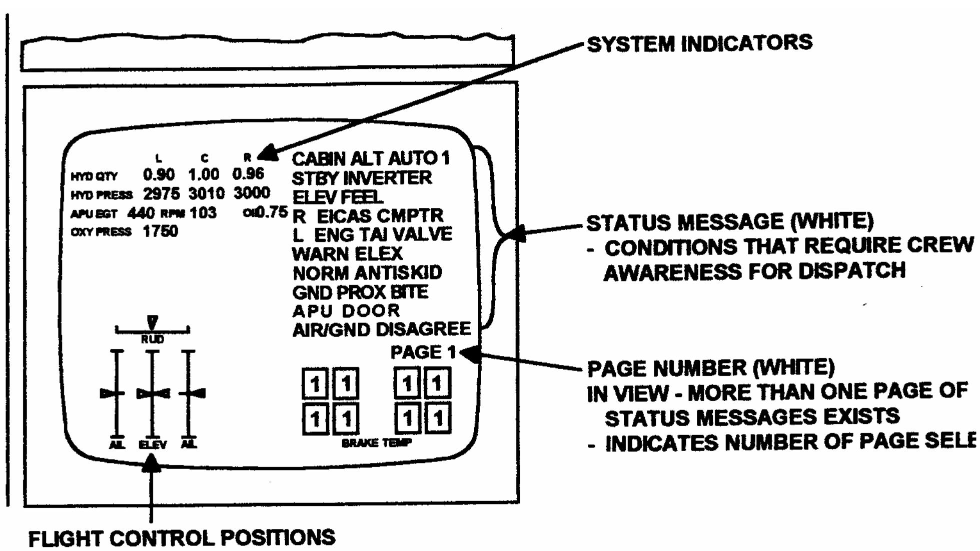
STATUS

The status portion of EICAS is used to determine the aeroplane's readiness for dispatch, and if the STS (Status) Switch is pushed, the status display appears on the lower CRT, as shown on the next page.

The status display includes system indicators, flight control position indicators, status messages and brake temperature indicators.

System indicators will appear in the top left corner of the display, and shows hydraulic quantity and pressure, APU EGT, rpm, oil quantity, and oxygen pressure.

The flight control positions of the rudder, ailerons, and elevators also appear in the bottom left corner of the status display.



White status messages appear on the right side of the status display. These messages indicate any equipment faults that require awareness at dispatch, particularly those not otherwise shown on the flight deck.

Status messages are arranged by order of occurrence, and the most recent status message appears at the top of the list.

A message is automatically removed from the display when the associated condition no longer exists, and messages that appear below the deleted message each move up one line.

If a new status fault occurs, inserted at the top of the list is its associated message, and all other messages moves down one line.

If there are more messages than can display at one time, the lowest message is removed and a white page number appears on the lower right side of the message list. Messages bumped from the bottom on one page automatically appear on the next page.

If there is an additional page of status messages, pushing the STS (Status) Switch displays the next page. When the last page of messages is displayed, pushing the switch removes the status display from the lower CRT, and will blank the screen.

New status messages appear at the top of the page viewed. If the status display is deselected and subsequently reselected, the message list is reordered, with the newest status message now appearing at the top of the first page.

A Status Cue appears in the left upper corner of the lower CRT if a new status message occurs whilst the status display is not currently selected, and the aeroplane is in the air. The cue disappears if the status page displays. Status messages do not need to be checked in flight; however, they can be useful in anticipating possible ground maintenance actions.

Brake temperature indications additionally appear on the lower right side of the status display.

MAINTENANCE

The maintenance portion of the EICAS provides a flight deck display of system data for use by maintenance personnel. Maintenance displays can only be used on the ground and are designed to provide flight deck display of maintenance information for the use of flight deck crew post flight logbook entries and ground crew. For convenience, all status messages are repeated on the Maintenance page, and any significant information not covered by the alert messages. Maintenance messages display on the right hand side of the lower CRT by pressing the ECS/MSG button on the EICAS maintenance panel when the aeroplane is on the ground.

EICAS FAILURE MODES

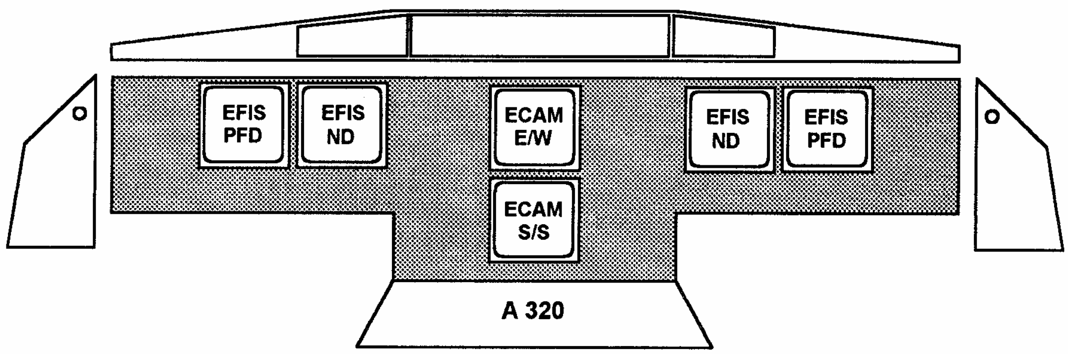
If a CRT fails, status can only display on the ground. If the EICAS Control Panel fails, an EICAS CONT PNL advisory message displays, and the EICAS full up engine mode displays automatically. The cancel and Recall Switches do not operate when the EICAS Control Panel fails. In the event of the failure of both EICAS computers, or both CRTs, a Standby Engine Indicator (SEI) automatically activates. The SEI system lights and system indicators are used to monitor the engines and system operation if a total EICAS failure occurs.

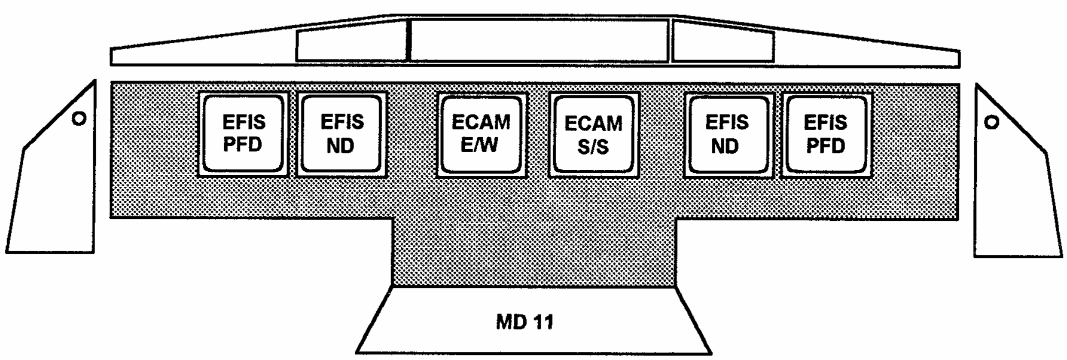
INTRODUCTION

The Electronic Centralised Aircraft Monitoring (ECAM) System is part of the electronic instrument system (EIS) consisting of six screens, providing the flight crew with assistance in system management in both normal / abnormal conditions. This operational assistance is given by the EFIS system and two centre mounted CRTs identified as the:

- Engine / Warning Display (E/W)
- System / Status Display (S/S)

The physical layout of both CRT displays is dependent upon the flight deck layout. Shown below are the layouts on the A320 and MD11.

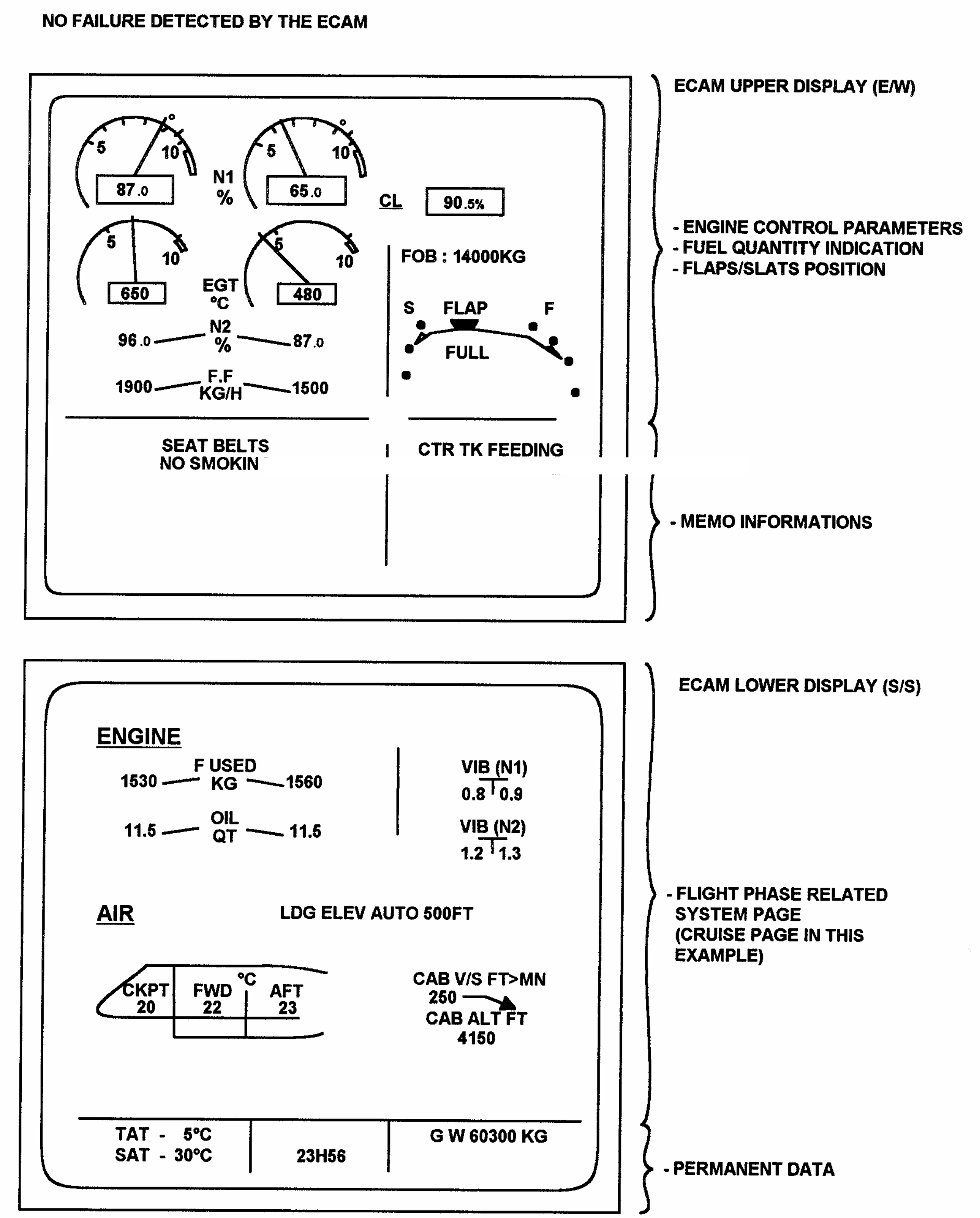




ENGINE / WARNING (E/W) CRT DISPLAY

The E/W Display shows the:

- Engine parameters
- Fuel on board
- Slats and flap position
- Warning and caution messages
- Memos when no failures exist



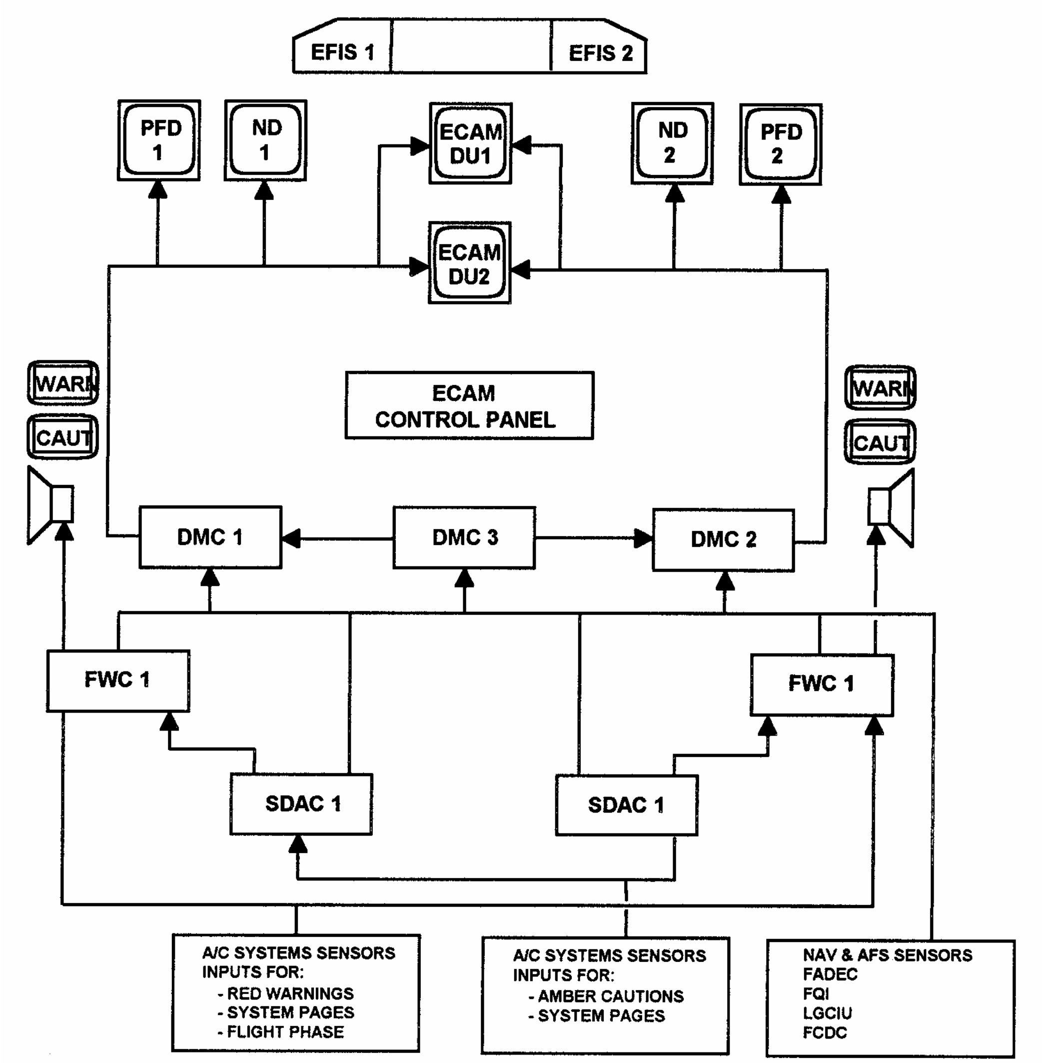
THE SYSTEM / STATUS (S/S) CRT DISPLAY

The S/S Display above shows the:

- System synoptic diagrams
- Status messages

ECAM SYSTEM ARCHITECTURE

Shown below is a typical ECAM system.



In this system, information is derived from the various aeroplane systems. After processing in the following units, it displays on the ECAM Display Units.

FLIGHT WARNING COMPUTERS (FWCS)

These units generate all of the alert messages, aural alerts, and vocal messages (Radio Height), derived:

- Directly from aircraft sensors or systems to generate **red warnings**
- Through the system data acquisition concentrators (SDACs) to generate **amber cautions**

DISPLAY MANAGEMENT COMPUTERS (DMC)

These units are common to EFIS/ECAM and generate the images displayed on the PFD, ND, E/W and S/S display units. They thus provide a similar function to that of the symbol generators in an EFIS system.

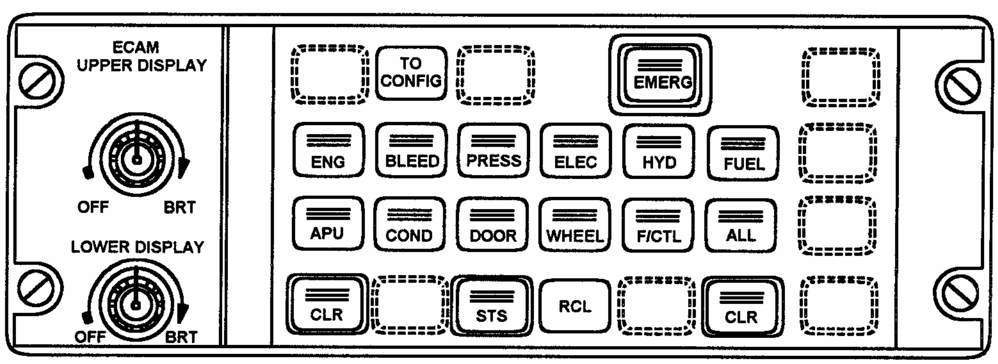
SYSTEM DATA ACQUISITION CONCENTRATORS (SDAC)

These units acquire data, process and then distribute it to the:

- DMCs, for the display of engine parameters and system page data
- FWCs, for the generation of alert and procedure messages

ECAM CONTROL PANEL (ECP)

The ECP shown below allows selection of the system pages or the status (STS) page prior to being displayed on the S/S Display.



The ECP has the following functions:

System Page Push Switches

Depressing these switches causes the S/S display to change to the corresponding system page, illuminating following a manual selection, or when detecting an advisory.

STS Push Switch

Depressing this switch causes the status page to be displayed on the S/S display, but if there is no status message present, NORMAL displays on the CRT for 5 seconds.

CLR Push Switch

This switch illuminates if a warning/caution message on the ECAM display unit requires flight crew action or acknowledgement. Whilst the CLR switch is illuminated, pressing it changes the ECAM display until all actions required by the ECAM system have been carried out.

ALL Push Switch

Depressing this switch causes all of the system pages to successively display at one second intervals. This switch also allows the system pages to present successively in the case of ECAM control failure, and allows the flight crew to stop at the desired page.

EMERG Cancel Push Switch

If this switch is depressed any present:

Aural Warning (including GPWS) cancels as long as the failure condition is present.

Caution (CRT message + master caution + single chime) cancels for the remainder of the flight. The status page then automatically displays and the CANCELLED CAUTION message displays, together with the title of the failure, which was inhibited.

TO CONFIG Switch

Depressing this switch causes a simulation of take-off power application, and a warning triggers if the aeroplane is not correctly configured for take-off.

OFF / BRT Knobs

These knobs are used to control the on/off and brightness of each ECAM display unit.

ATTENTION GETTERS

In abnormal operation, the ECAM system directs the flight crew's attention by visual and audio attention getters. The visual attention getters are the:

Master Warning Light (red)

This gives an indication of any system faults, which require immediate crew awareness and immediate corrective action.

Master Caution Light (amber)

This gives an indication of any system faults, which require immediate crew awareness, but not immediate corrective action.

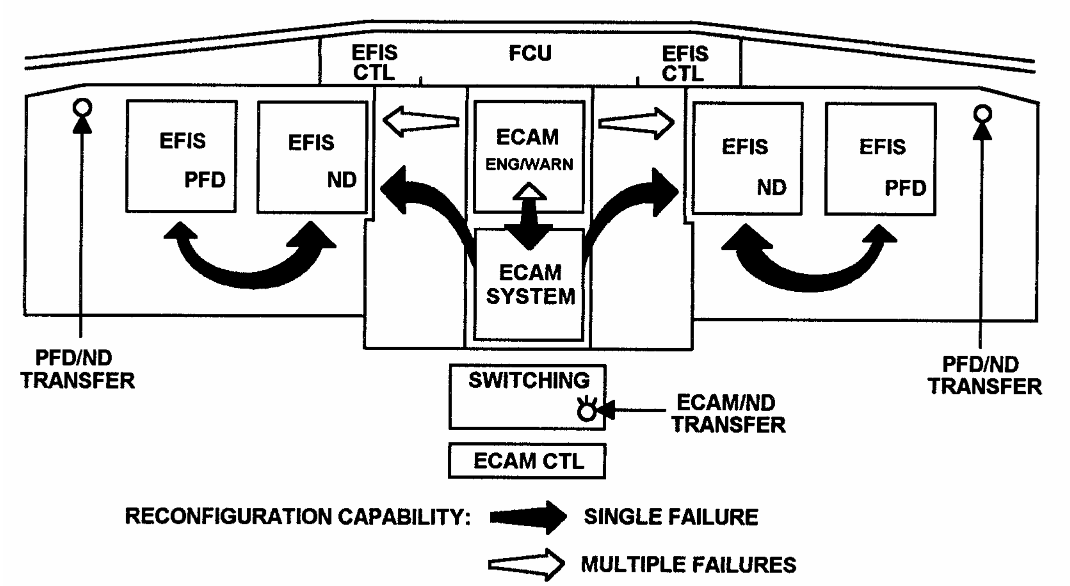
DISPLAY COLOUR CODING

The displays are colour coded as follows:

Red	Warnings
Amber	Cautions
Green	Normal long-term operation
White	For functions not in normal operation (switch in off position)
Blue	To carry out an action

ECAM SYSTEM FAILURE

If a system failure occurs, the flight crew can manually switch the display functions to other units within the electronic instrument system, and still maintain full operational capability as follows.



FAILURE CATEGORISATION

The following categories of failure can occur on an aeroplane:

Independent Failure

This type of failure only affects an isolated item of equipment or a system and does not affect any other systems in the aeroplane.

Primary Failure

This type of failure affects an item of equipment or a system, resulting in the loss of other systems in the aeroplane.

Secondary Failure

This type of failure results due to the primary failure of an item, or system.

SYSTEM OPERATION

The ECAM system provides warnings and cautions whenever the aeroplane flies out of its flight envelope, or the failures affect the aeroplanes integrity. These warnings and caution messages are in plain English, and:

- Provide improved failure analysis.
- Provide guidance for any corrective action required.
- Minimise the need for the flight crew to refer to paper checklists.
- Improve the understanding of the aeroplane and system configuration after failure

In normal operation, the ECAM system eases the flight crew workload by eliminating the need for frequent scanning of the various aeroplane system panels. For example, if the SEAT BELT or NO SMOKING signs are on, or the APU is running, this displays on the E/W display unit as a memo message. Checklists display on the memo page during the take-off and approach phases of flight. Synoptic diagrams are routinely presented on the S/S display unit, and are automatically adapt to the current flight phase, or can be manually called up by the flight crew, using the ECP push buttons. The synoptic pages are used to amplify system failures and switch selections made in conjunction with warnings displayed on the E/W display. Continual monitoring of some system parameters occurs throughout the flight, and automatically display on the relevant system page when their values drift out of the normal range, but well before reaching the warning level.

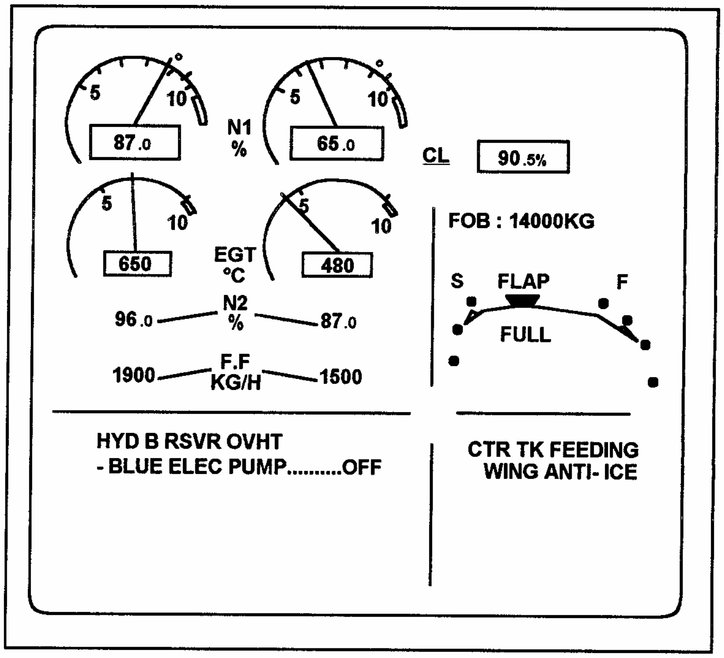
In the case of a failure detected by the ECAM system, the following actions occur:

- The E/W display presents the warning and caution messages. The warning message includes a failure title and any associated procedures, as well as the title of the systems affected by the failure.
- The master warning or master caution lights illuminate.
- An audio alert triggers.
- The S/S display presents the affected system page.
- The CLR push switch illuminates on the ECP.
- When all of the messages and synoptic diagrams associated with the failure have been reviewed by the flight crew, the CLR push button must be depressed until the normal configuration returns, and the CLR push button is extinguished.

The sequence of events, which occur during a typical failure are detailed on pages 8.9.7– 8.9.13 inclusive.

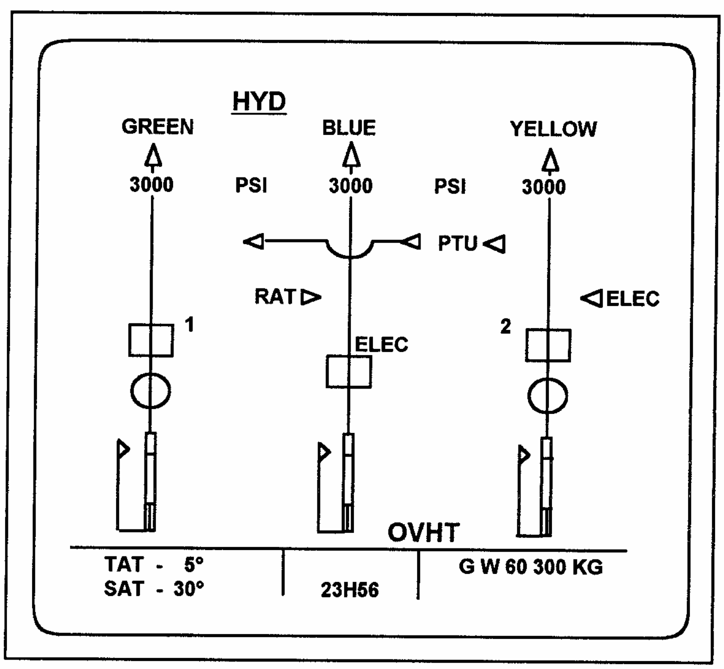
FAILURE IS DETECTED - EXAMPLE : HYDRAULIC RESERVOIR OVERHEAT -

COCKPIT INDICATIONS
- SINGLE CHIME SOUNDS
- MASTER CAUTION LIGHTS ILLUMINATE STEADY
- FAULT LIGHT ILLUMINATES ON OVERHEAD PANEL
- CLR PB ILLUMINATES ON ECP



ECAM UPPER DISPLAY (E/W)

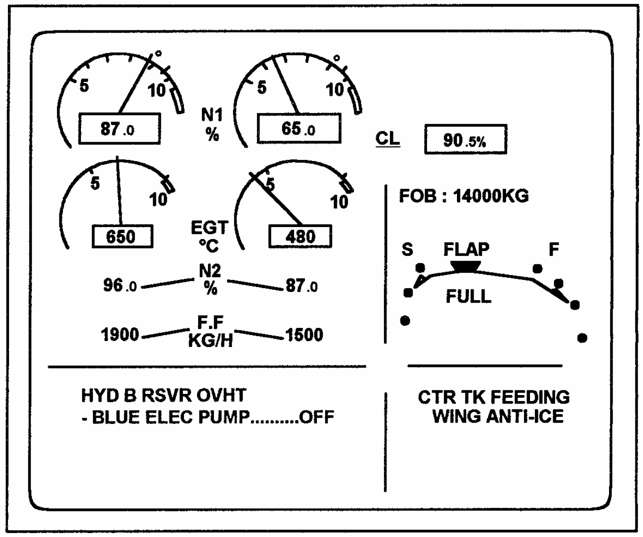
- LEFT PART
  . INDEPENDENT FAILURE
    - TITLE OF THE FAILURE
    - ACTIONS TO BE PERFORMED
- RIGHT PART
  . MEMO INFORMATION

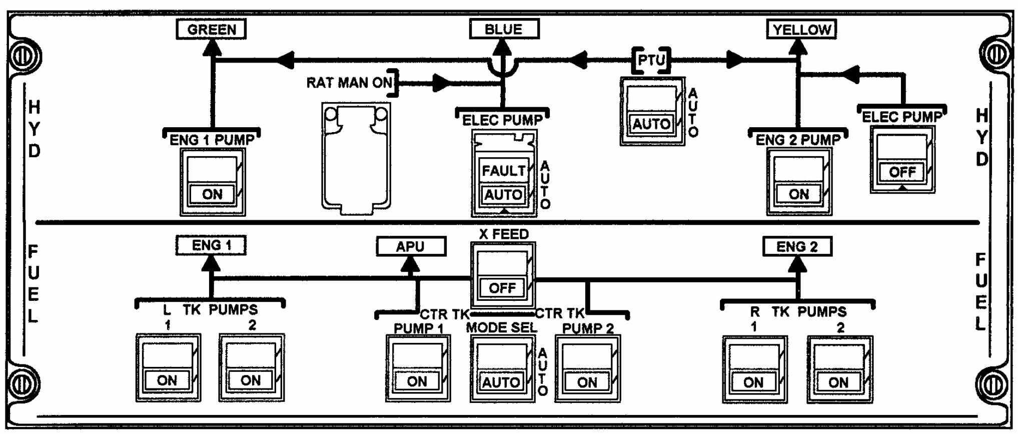


ECAM LOWER DISPLAY (S/S)

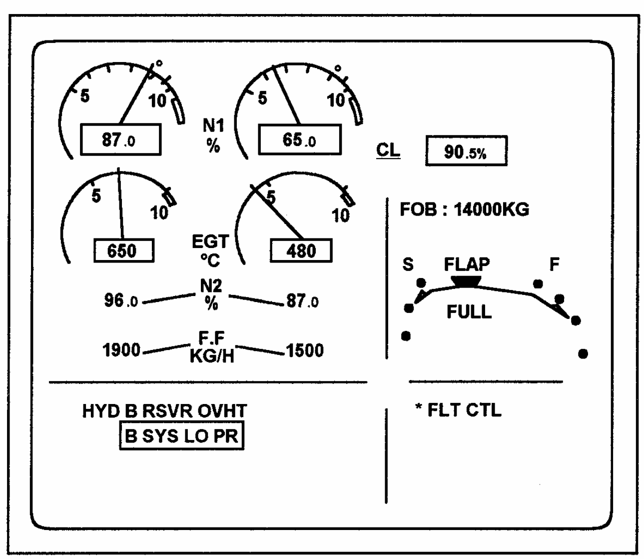
- SYNOPTIC OF THE AFFECTED SYSTEM AUTOMATICALLY CALLED OVHT ID DISPLAYED IN AMBER

Step 1





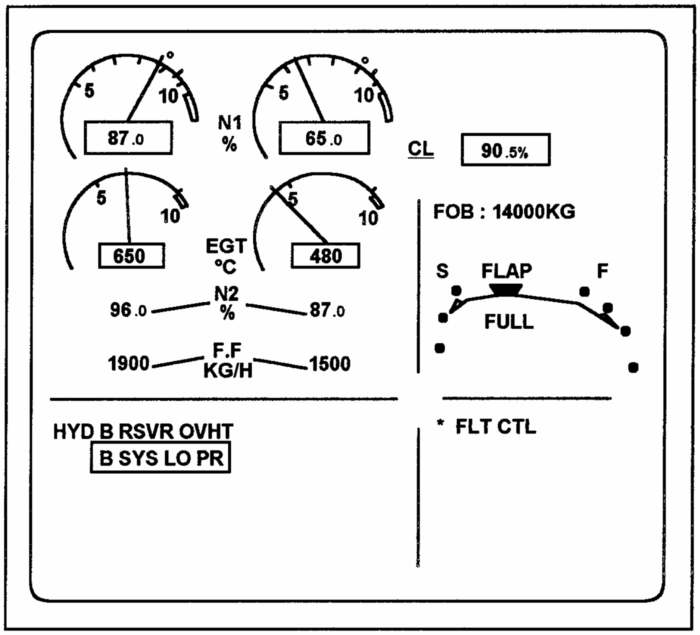
THE CREW APPLIES THE ECAM REQUIRED ACTIONS BY SWITCHING OFF THE BLUE ELEC PUMP,
THE BLUE HYDRAULIC CIRCUIT IS DEPRESSURIZED



Step 2

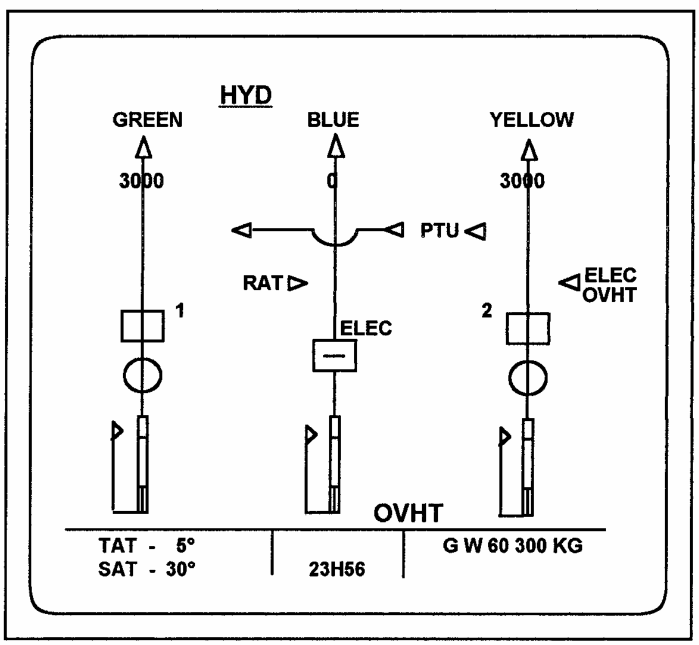
THE CREW APPLIES THE ECAM REQUIRED ACTIONS
BY SWITCHING OFF THE BLUE ENG 2 PUMP , THE BLUE HYDRAULIC CIRCUIT IS DEPRESSURIZED

COCKPIT INDICATIONS
- SINGLE CHIME SOUNDS
- MASTER CAUTION LIGHTS ILLUMINATE STEADY
- FAULT/OFF LIGHTS ILLUMINATES ON OVERHEAD PANEL
- CLR PB ILLUMINATES ON ECP



ECAM UPPER DISPLAY (E/W)

- LEFT PART
  . INDEPENDENT FAILURE AND PRIMARY FAILURE
- RIGHT PART
  .SECONDARY FAILURE



ECAM LOWER DISPLAY (S/S)

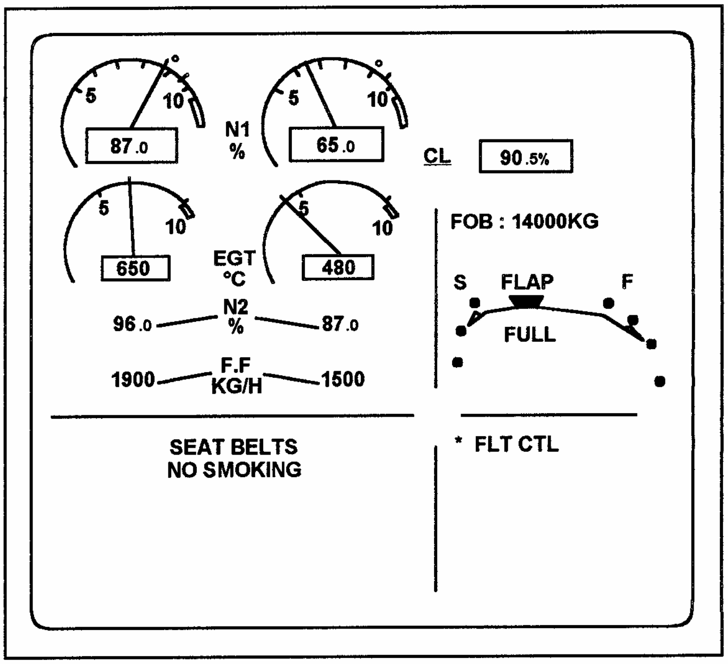
- THE SYNOPTIC OF THE SYSTEM PAGE IS CHANGED ACCORDING TO THE NEW SYSTEM CONFIGURATION
OVHT AND THE PRESSURE ARE DISPLAYED IN AMBER

NOTE : BLUE ELECT PUMP NOW DISPLAYED OFF.

Step 3

THE CREW DEPRESSES THE CLR PB

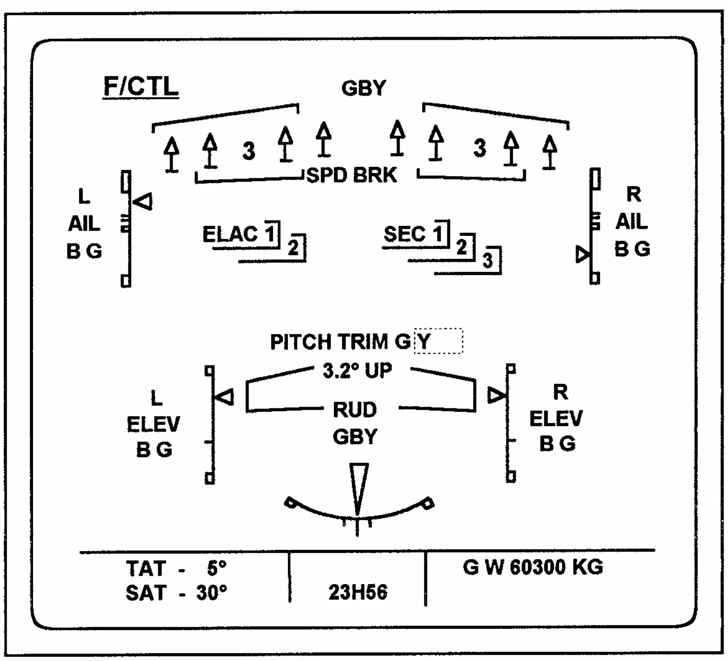
COCKPIT INDICATIONS
- **CLR PB ILLUMINATES ON ECP**
- **FAULT/OFF LIGHTS REMAIN ON**



ECAM UPPER DISPLAY (E/W)

- LEFT PART
.MEMO INFORMATION

- RIGHT PART
.SECONDARY FAILURE



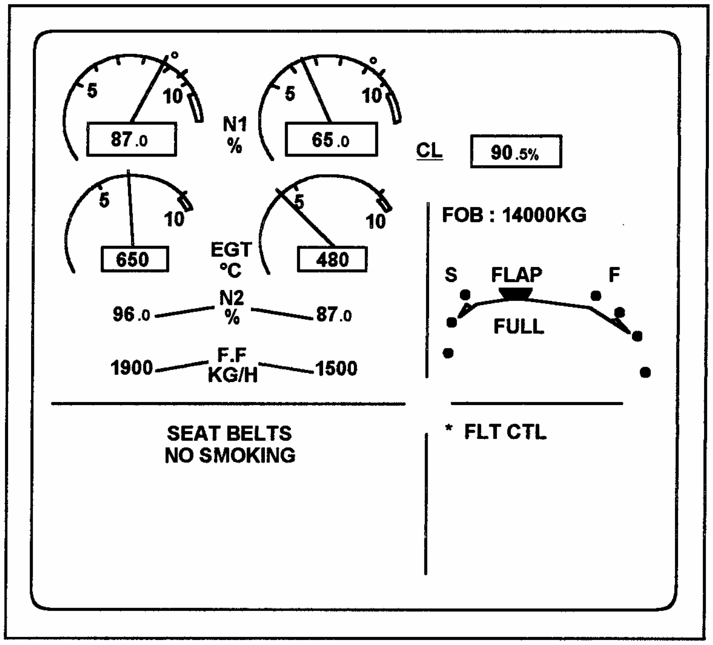
ECAM LOWER DISPLAY (S/S)

- F/CTL SYSTEM PAGE AUTOMATICALLY DISPLAYED FAULTY SPOILERS (n°3) AND SURFACE ACTUATORS PRESSURE INDICATIONS B ARE DISPLAYED IN AMBER

Step 4

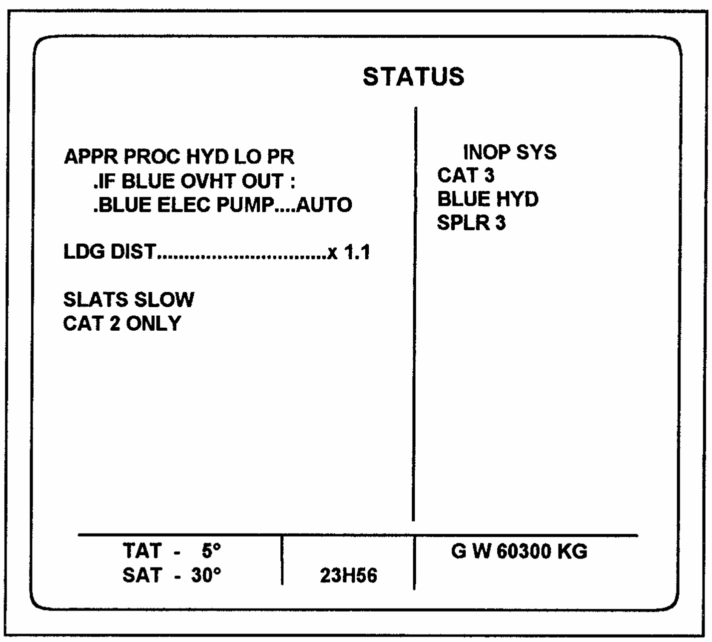
THE CREW DEPRESSES THE CLR PB A SECOND TIME

COCKPIT INDICATIONS
- **CLR AND STS PBS ILLUMINATES ON ECP**



ECAM UPPER DISPLAY (E/W)

- **FULL MEMO DISPLAYED**

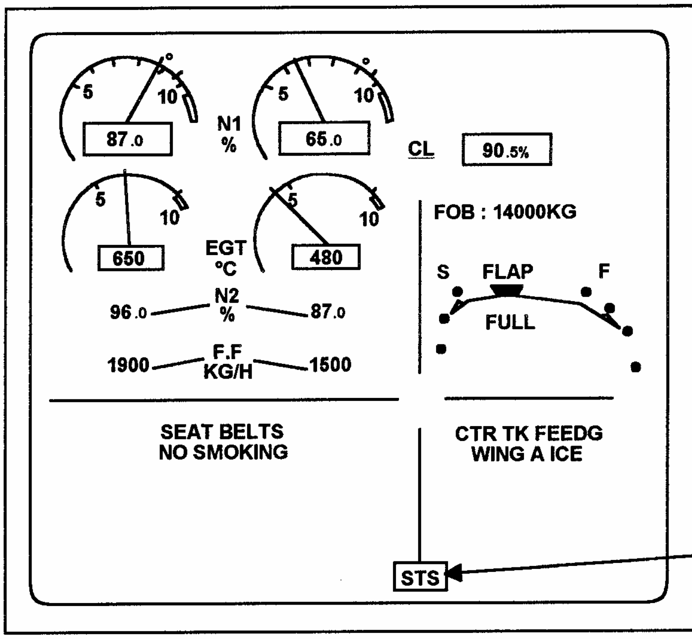


ECAM LOWER DISPLAY (S/S)

- **THE STATUS PAGE IS AUTOMATICALLY DISPLAYED TO:**
  - **.PROVIDE THE PROCEDURE TO APPLY FOR APPROACH**
  - **.PROVIDE LANDING DISTANCE FACTORS AND INFORMATION**
  - **.LIST THE INOPERATIVE SYSTEMS**

Step 5

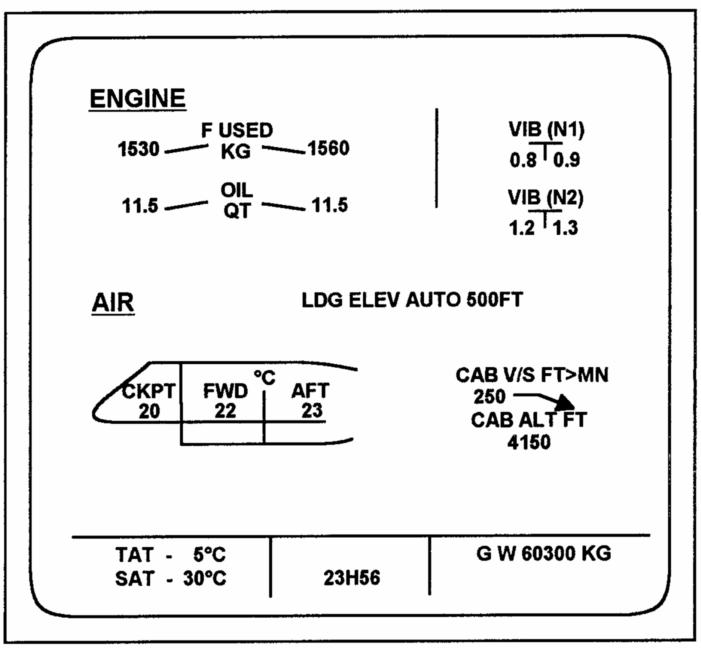
THE CREW DEPRESSES THE CLR PB A THIRD TIME



ECAM UPPER DISPLAY (E/W)

- **FULL MEMO DISPLAYED**

STATUS REMINDER



ECAM LOWER DISPLAY (S/S)

- **RETURN TO THE FLIGHT PHASE RELATED SYSTEM PAGE : CRUISE PAGE**

Step 6