MINISTRY OF EDUCATION AND SCIENCE OF UKRAINE
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AIRCRAFT SYSTEMS AND EQUIPMENT

The Summary of Lectures

Kharkiv «KhAI» 2022

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O-95

Розглянуто призначення, будову та принципи роботи авіаційних приладів та систем. Описано методи та пристрої для виконання вимірювань, що дають параметри роботи силової установки, положення літака у просторі, швидкості та перевантаження літака. Наведено інформацію про сучасні засоби забезпечення безпеки польотів.

Для студентів спеціальностей «Авіаційні та аерокосмічні технології» та «Авіаційний транспорт».

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The purpose, structure and principles of operation of aviation instruments and systems are considered. The methods and devices for performing measurements are described, which give the parameters of the operation of the power plant, the position of the aircraft in space, the speed and overload of the aircraft. Information on modern means of ensuring flight safety is presented.

For students of specialties **«**Aerospace Engineering**»** and **«**Aviation Transport**»**.

Figs 88. Bibliogr. : 12 items

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CONTENT

INTRODUCTION

Since the beginning of manned flight, it has been recognized that supplying the pilot with information about the aircraft and its operation could be useful and lead to safer flight. The Wright Brothers had very few instruments on their Wright Flyer, but they did have an engine tachometer, an anemometer (wind meter), and a stopwatch. They were obviously concerned about the aircraft's engine and the progress of their flight. From that simple beginning, a wide variety of instruments have been developed to inform flight crews of different parameters. Instrument systems now exist to provide information on the condition of the aircraft, engine, components, and the aircraft's attitude in the sky, weather, cabin environment, navigation, and communication.

The ability to capture and convey all of the information a pilot may want, in an accurate, easily understood manner, has been a challenge throughout the history of aviation. As the range of desired information has grown, so too have the size and complexity of modern aircraft, thus expanding even further the need to inform the flight crew without sensory overload or over cluttering the cockpit. As a result, the old flat panel in the front of the cockpit with various individual instruments attached to it has evolved into a sophisticated computer-controlled digital interface with flat-panel display screens and prioritized messaging.

There are usually two parts to any instrument or instrument system. One part senses the situation and the other part displays it. In analog instruments, both of these functions often take place in a single unit or instrument (case). These are called direct-sensing instruments. Remote-sensing requires the information to be sensed, or captured, and then sent to a separate display unit in the cockpit. Both analog and digital instruments make use of this method. The relaying of important bits of information can be done in various ways. Electricity is often used by way of wires that carry sensor information into the cockpit. Sometimes pneumatic lines are used.

In complex, modern aircraft, this can lead to an enormous amount of tubing and wiring terminating behind the instrument display panel. More efficient information transfer has been accomplished via the use of digital data buses. Essentially, these are wires that share message carrying for many instruments by digitally encoding the signal for each. This reduces the number of wires and weight required to transfer remotely sensed information for the pilot's use. Flat-panel computer display screens that can be controlled to show only the information desired are also lighter in weight than the numerous individual gauges it would take to display the same information simultaneously.

An added bonus is the increased reliability inherent in these solid-state systems. It is the job of the aircraft technician to understand and maintain all aircraft, including these various instrument systems. Accordingly, in this summary of lectures, discussions begin with analog instruments and refer to modern digital instrumentation when appropriate.

Equipment of the modern aircraft can be divided into several functional groups:

– airframe and its systems;
– powerplant;
– aviation equipment;
– electronic equipment.

Aviation equipment includes:

– electrical equipment;
– instrument equipment;
– electronic equipment;
– oxygen equipment;
– high-rise equipment;
– flight data recorders ("black boxes").

The subject of the course study is: aircraft instruments, attitude and navigation systems, automatic control systems, flight and navigation complexes.

This discipline explores the principles of construction, theory, arrangement and operation of on-board measurements designed to provide the crew and some aircraft systems with the flight data needed to perform the flight. These flight data include:

1. Coordinates of the aircraft location (latitude, longitude, altitude, course angles and range to airports).
2. The angular position of the aircraft (course, roll and pitch angles) and angular velocity vector.
3. Parameters of the aircraft movement in the air (speed, overload, aerodynamic angles).
4. Parameters of power plants and individual on-board systems.

One of the goals of flight control is to bring the aircraft to a predetermined position relative to the Earth, as well as to maintain the specified value of air speed. The devices that provide the solution to such a problem are called flight instruments. These include gyro horizon, air speed indicator, Mach meter, turn and slip indicator, overload and vertical speed indicator.

Flight control also provides navigation, the purpose of which is to bring the aircraft to the designated point at a given time and speed. To solve this problem, navigation systems are used to determine coordinates, ground speed, lateral deviation from a given line of track, range to the airport and other navigational values.

It can be noted that the course, altitude and speed measuring devices are used to solve both aerobatic and navigational tasks.

A separate group consists of devices and systems to control the operation of the power plant: tachometers, gauges, thermometers, meters of quantity and fuel consumption.

The failure of aircraft devices creates an emergency situation. For example, if such aerobatic devices as overload meters, angle of attack, instrument speed, number M, it is possible to go beyond the acceptable flight modes. The failure of the air horizon can lead to the loss spatial orientation in difficult weather conditions.

The purpose of studying the discipline is to gain knowledge about the principles of operation, design and maintenance of equipment, instruments and electronic systems of aircraft for the further acquisition of practical skills in the technical operation and repair of aircraft.

The course "Instruments and Aircraft Electronic Systems" is allocated 120 hours of classes in the 6th semester. Of these: 64 hours – lectures and practical classes and 56 hours – independent work. Credits 4. The course includes 16 lectures, 9 Topics and 3 modules. The final control of the subject's knowledge is the credit.

Part I. AVIATION INSTRUMENTS AND SYSTEMS

Topic 1. AVIATION INSTRUMENTS, PROPERTIES OF THE ATMOSPHERE, MEASUREMENT METHODS

1.1. Purpose and classification of aviation instruments

Flight modes of the aircraft are set by many parameters. These parameters are continuously changed during the flight due to the impact on the aircraft of external atmospheric conditions.

To perform the flight, you need to ensure that these parameters are continuously measured. This measurement is carried out with the help of aircraft instruments.

There are three basic kinds of instruments classified by the job they perform: flight instruments, engine instruments, and navigation instruments. There are also miscellaneous gauges and indicators that provide information that do not fall into these classifications, especially on large complex aircraft. Flight control position, cabin environmental systems, electrical power, and auxiliary power units (APUs), for example, are all monitored and controlled from the cockpit via the use of instruments systems. All may be regarded as position/ condition instruments since they usually report the position of a certain movable component on the aircraft, or the condition of various aircraft components or systems not included in the first three groups.

The instruments used in controlling the aircraft's flight attitude are known as the flight instruments. There are basic flight instruments, such as the altimeter that displays aircraft altitude; the airspeed indicator; and the magnetic direction indicator, a form of compass. Additionally, an artificial horizon, turn coordinator, and vertical speed indicator are flight instruments present in most aircraft.

Over the years, flight instruments have come to be situated similarly on the instrument panels in most aircraft. This is the basic T arrangement for flight instruments. The top center position directly in front of the pilot and copilot is the basic display position for the artificial horizon even in modern glass cockpits (those with solid- state, flat-panel screen indicating systems).

Original analog flight instruments are operated by air pressure and the use of gyroscopes. This avoids the use of electricity, which could put the pilot in a dangerous situation if the aircraft lost electrical power. Development of sensing and display techniques, combined with advanced aircraft electrical systems, has made it possible for reliable primary and secondary instrument systems that are electrically operated. Nonetheless, often a pneumatic altimeter, a gyro artificial horizon, and a magnetic direction indicator are retained somewhere in the instrument panel for redundancy.

Engine instruments are those designed to measure operating parameters of the aircraft's engine(s). These are usually quantity, pressure, and temperature indications. They also include measuring engine speed(s). The most common engine instruments are the fuel and oil quantity and pressure gauges, tachometers, and temperature gauges.

Engine instrumentation is often displayed in the center of the cockpit where it is easily visible to the pilot and copilot. On light aircraft requiring only one flight crew member, this may not be the case. Multi-engine aircraft often use a single gauge for a particular engine parameter, but it displays information for all engines through the use of multiple pointers on the same dial face.

Navigation instruments are those that contribute information used by the pilot to guide the aircraft along a definite course. This group includes compasses of various kinds, some of which incorporate the use of radio signals to define a specific course while flying the aircraft en route from one airport to another. Other navigational instruments are designed specifically to direct the pilot's approach to landing at an airport.

Traditional navigation instruments include a clock and a magnetic compass. Along with the airspeed indicator and wind information, these can be used to calculate navigational progress. Radios and instruments sending locating information via radio waves have replaced these manual efforts in modern aircraft.

All aircraft Instruments are classified by purpose, by the principle of action and by the way the measured value is reproduced:

1. As intended:
– aerobatic navigation devices and systems;
– power plant control devices;
– aircraft systems control devices;
– instruments measuring environmental parameters.

2. On the principle of action:
– mechanical (manometric and gyroscopic);
– electromechanical;
– electronic;
– optical.

3. By the way the measured value is reproduced:
– with a direct calculation of the readings on the device scale;
– registration devices;
– signal sensors (used to obtain information and transmit it to control and control systems).

An aircraft instrument is a dynamic system that converts input signals into signals of the visual response of the measured parameter (on a scale) or the recording of the value of the parameter on the media, or in the electric signal (e.g., the system control signal).

The structural scheme of the aircraft instrument can be presented in the following form (Fig. 1.1).

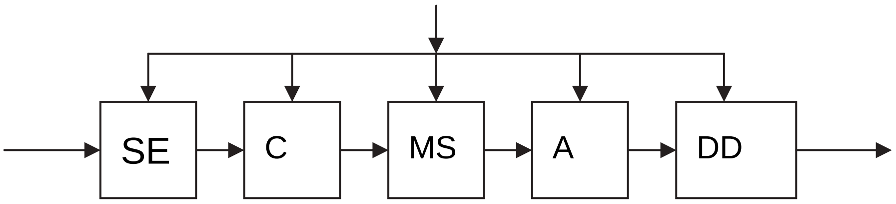


Fig. 1.1

There are SE – sensitive element, C – converter, MS – measuring scheme, A – amplifier and DD – display device in Fig. 1.1.

The input parameters of the device are measurable parameters (pressure, temperature, speed, course). These are useful signals. Let's designate them X.

But besides the useful signal the η–interference harmful signals get to the input of the sensitive element (acceleration, vibration, friction, temperature fluctuations, etc.). These signals cause the device errors when measuring.

In addition to external harmful signals internal interference of ζ elements of the device itself (dry friction, harmful noises, etc.) has a significant influence on the error of the device.

The following can be an output signal of the device:

- values proportional to inputs;
- set functions depending on inputs;
- visual or auditory images that reflect the properties of input information;
- control signals (used for automatic command and control).

Let's denote the output signal Y^*. Taking into account the error we have:

$$Y^* = Y_i + \Delta Y, \tag{1.1}$$

where Y_i is the true value of the measured value in a converted form;

ΔY is the error of the output signal.

1.2. Properties of the Earth's Atmosphere

The entire height of the atmosphere (250 to 300 km), depending on the law of temperature variaitons from height can be divided into:

- troposphere (from 0 to 10 – 12 km);
- stratosphere (up to 50 – 60 km);
- mesosphere (up to 80 km);
- thermosphere (over 80 km).

In the troposphere (from 0 to 10 – 12 km) the temperature drops by an average of 0.6 degrees Celsius per 100 m, reaching 56.5 degrees Celsius at altitude of 11 km.

The composition of air in the troposphere remains constant and amounts to 78.68% nitrogen, 20.95 % oxygen, 0.93 % argon, 0.03 % carbon dioxide. In the troposphere there is a significant change in temperature, pressure and humidity.

Look at the atmospheric structure scheme (Fig. 1.2).

In the *stratosphere* (up to 50 – 60 km) the temperature increases with an increase in height and at an altitude of 50 – 60 km is minus 12 degrees Celsius. There are no clouds in the stratosphere.

In the *mesosphere* (up to 80 km) the temperature with an increase in height decreases and at an altitude of 80 km reaches minus 88.16 degrees C.

In the *thermosphere* (over 80 km) the temperature reaches several hundred degrees, but ships moving there do not heat up, except for the heating from the sun's rays, because the number of particles colliding with the hull of the ship is extremely small.

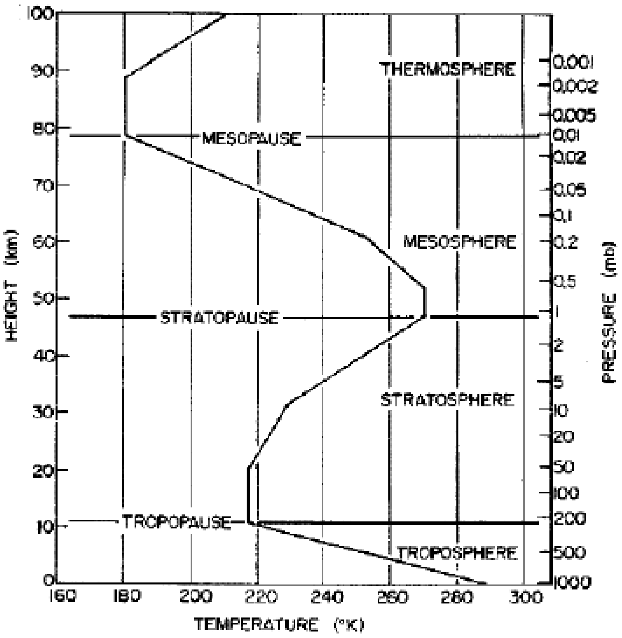


Fig. 1.2

The atmosphere is characterized by layers with a sharp degree of ionization. Up to an altitude of about 80 km the degree of air ionization is small (neutrosphere), and at altitudes of 80 – 800 km lies a layer of significant ionization. This layer is called the ionosphere.

The characteristics of the atmosphere do not remain constant and vary depending on the time of year and day, latitude of the place, meteorological conditions, solar activity, etc. So, for example, the temperature of the Earth vary at mid-latitudes by a gradient of 70 degrees Celsius. Pressure and density also do not remain constant near the Earth's surface.

All these circumstances make it difficult to solve a number of aerodynamic tasks and the gradation of the scales of the devices. Therefore, instead of the actual atmosphere introduced the concept of **a standard atmosphere**. In the standard atmosphere, actual characteristics (pressure, temperature, density, humidity, composition, speed of currents, turbulent fluctuations, degree of ionization), which are random functions of time and coordinates, are replaced by their average values (mathematical expectations) close to the averages of the same characteristics in the summer at mid-latitudes at sea level:

– air pressure of 760 mm Hg;
– temperature of 15 degrees Celsius;
– density of 1.225 kg/m;
– sound speed 340.28 m/s.

The atmosphere is characterized by air currents (winds), as well as random fluctuations in speed (turbulent oscillations). Air currents are horizontal and vertical. In the stratosphere at altitudes of 10 – 20 km, the western winds prevail, and at altitudes of 40 – 50 km – easterly winds.

Aviation instruments can be exposed to air temperatures varying from +50 to minus 60 degrees Celsius, and near heated engine parts up to +100 degrees Celsius. At high supersonic flight speeds, the additional heating of the devices from the hot parts of the aircraft can be several tens of degrees.

Since the devices are graded at 20 degrees Celsius, the readings of the devices have temperature errors caused by the influence of external temperature on the structural and circuit elements of the devices (the rigidity of springs, the size of the arms of transmission mechanisms, electrical and magnetic resistances, etc.).

The relative humidity of the air can vary from 0 to 100 %. Humidity affects aircraft devices twofold:

– at high humidity, vapor particles condense on the elements of the instruments, which often leads to corrosion;

– at zero and negative temperatures, condensed vapors can freeze, resulting in the failure of the instruments.

Due to the decrease in air density at high altitudes, the diversion of heat from the instruments deteriorates.

The evaporation of the grease of bearings increases and the insulation properties in the electrical nodes of the instruments deteriorate. The deterioration of the insulating properties of rarefied air is further exacerbated due to increasing the degree of ionization at high altitudes. All this requires special insulation of electrical nodes, and sometimes sealing of instruments.

Aviation instruments are subjected to mechanical impacts caused by overloads due to the evolution of the aircraft, overloads from turbulent vibrations of the atmosphere, impacts during takeoff and landing, as well as vibrations of aircraft parts. Overload from evolution on heavy aircraft reach 4, and on maneuverable supersonic to 10. Overloads from vibrations of aircraft parts are added to these overloads.

1.3. Methods of Measurement and Their Classification

Measuring information from an object to be controlled is transmitted to the measuring system in the form of signals of any kind of energy. When the information is transmitted from object to instrument pointer, the signals change in level and intensity and are converted from one type of energy to another. This is because primary signals are not always convenient for transmission, processing, further conversion and reproduction. For example, when measuring temperature by a device whose sensitive element is placed in a controlled environment, the perceived heat flow is difficult to transmit, much less reproduce on the instrument pointer. Therefore, when measuring non-electric values, the signals perceived by the sensitive element are converted into electrical signals.

Electrical methods of measuring non-electric values are classified by the type of connection between electrical and non-electric values and are divided into:

– parametric measurement methods;

– generator methods of measurement.

1.3.1. *Parametric Measurement Methods*

Parametric measurement methods are methods in which the input (measurable) parameter changes the parameters of the electrical circuits of the instrument.

Parametric measurement methods include:

a) resistance method;

b) the wet method;

c) the inductive method.

In parametric methods of measurement, the non-electric value is converted into a corresponding change in the parameters of electrical circuits (electric resistance, capacity, inductivity) of the measuring device, which are powered by an external voltage source. At the same time, signals proportional to the measured parameter serve only to control the energy of this external source of voltage.

The parametric method of resistance is based on the dependence of electrical resistance of resistors on the environment temperature, on pressure, on the deformation factor of the resistor material, on the geometric size of the resistors.

The wet parametric method of measurement is based on changing the distance between the capacitor electrodes or altering the effective area of the electrodes, or changing the dielectric constant.

The inductive parametric measurement method is based on the property of the inductivity coil to alter reactive (inductive) resistance, and thus inductivity, when the magnitude and area of the air gap change.

1.3.2. *Generator Methods of Measurement*

Generator measurement methods are methods in which the input measured parameter is directly converted into an electrical signal, i.e. electrical energy is generated under the influence of the measured parameter.

Generator measurement methods include:

a) photovoltaic method;
b) ionization method;
c) electromagnetic method;
d) thermoelectric method;
e) piezoelectric method.

The photovoltaic measurement method is based on the effect of electric current (photo current) when some materials are illuminated by light rays.

The ionization method of measurement is based on the use of the phenomenon of the electric current flow through ionized gas. This method is used in instruments to measure the density and speed of gas flow, pressure, for qualitative and quantitative analysis of gases.

The electromagnetic method of measuring non-electric values is based on the direct use of the law of electromagnetic induction: if a conductor moves at a constant magnetic field at a speed of υ, it induces an electric motion force equal to that of

$$e = B\,l\,\upsilon\,. \tag{1.2}$$

Thermoelectric measurement method is that in a closed chain consisting of two heterogeneous conductors (chrome-aluminium, chrome-kopel), there are currents, if the solders of these conductors have different temperatures.

The emerging thermo-electric propulsion force is a function of the difference in the temperature of the solders. Its size depends only on the kind of conductors and temperature, and does not depend on the shape and size of these conductors. The chain formed by such two conductors is called a thermocouple.

Piezoelectric measurement method is based on the use of piezoelectric effect, which is that some crystals (quartz, tourmaline, signet salt) during deformation on the surface are allocated equal and opposite on the sign electric charges, proportional deformations. This method is used to measure variable pressures, acceleration, and deformation.

Topic 2. INSTRUMENTS AND CONTROL SYSTEM OF THE POWER PLANT. FUEL MEASURING SYSTEMS

2.1. Power Plant and Control Devices of its Operation

2.1.1. *Power Plant Operating Modes*

The operating mode of an aircraft engine is a set of parameters that determine the working process in the engine, as well as the external conditions under which the aircraft engine operates (flight speed, temperature and air pressure).

The regulation of an aircraft engine is called maintaining the required mode, as well as changing it in a certain direction when external conditions change.

Aircraft engine control is the change in the operating mode of the aircraft engine in the desired direction, regardless of external conditions.

Engine modes are distinguished by thrust level:

- idle mode;
- nominal mode;
- maximum mode;
- augmented rating;
- extreme augmented rating.

The modes are controlled by means of a throttle control lever, each position of which corresponds to a certain operating mode of the aircraft engine.

In the theory of aircraft engines, the parameters that characterize the workflow are the following parameters:

- speed (frequency) of the rotor or propeller n;
- the degree of pressure increase in the compressor π_{K}^{*};
- turbine inlet gas temperature T_3;
- compressor efficiency η_K;
- efficiency of the turbine η_T;
- air mass flow rate m_B;
- fuel mass flow rate in working m_T and in afterburner m_{T_Φ} combustion chambers;
- the temperature of gases in the T_Φ afterburner.

These parameters, in turn, determine the engine thrust P, engine power N and specific fuel consumption per unit of thrust or power C.

In two-rotor (twin-shaft) and bypass jet engines, the high pressure (HP) and low pressure (LP) rotor speeds are distinguished, respectively, as well as the degree of pressure increase in HP and LP compressors.

The most advantageous combination of parameters is such that for a given thrust or power, the minimum fuel consumption is ensured and the least dynamic and thermal loads arise in the aircraft engine parts. However, to achieve this, a large number of parameters would have to be controlled simultaneously, which would create complexity and reduce the reliability of management and control systems.

The main parameters of the aviation engine, which determine the traction efficiency, economy and loads on engine parts, are:

– frequency of rotation of rotor and propeller n;

– turbine inlet gas temperature T_3;

– the temperature of gases in the afterburner T_Φ.

These parameters of the aviation engine operation are called controlled parameters. Thus, in order to change the thrust of the aircraft engine (i.e., to adjust the aircraft engine thrust), it is necessary to change the value of the indicated parameters or at least one of them.

The question arises which parameters should be changed to adjust parameters.

To regulate the thrust of the aviation engine, the influence on the regulated parameters is carried out by means of the so-called regulating factors, which include:

– fuel supply to the working combustion chambers m_T;

– fuel supply to the afterburner m_{T_Φ};

– the area of the throat of the outlet nozzle F_C;

– the area of the turbine nozzle F_{CA};

– outlet area of the expanding part of the jet nozzle F_{output};

– angles of installation of the rotary guide vanes of the compressor φ_{na};

– air bypass from the compressor to the atmosphere m_{nB};

– angles of installation of the propeller blades φ_B.

The impact on regulatory factors is carried out through the regulatory authorities.

2.1.2. *Instruments and Systems for monitoring the Operation of Power Plant*

To monitor the operating modes of aircraft engines and control them, it is necessary to have information about the temperature and pressure of gases and liquids in various engine systems, about the angular speed of rotation of the gas turbine shaft and about the fuel consumption per unit time. In addition, to determine the possible duration and range of the flight, it is necessary to have information about the fuel amount in the fuel tanks of the aircraft. These parameters are monitored by the aircraft crew using appropriate measuring instruments.

Devices for visual control of the power plant operation include:

- – gas thermometers;
- – thermometers for liquids;
- – tachometers measuring the speed of rotation of the aviation engine shaft;
- – pressure gauges measuring the pressure of liquids and gases;
- – signaling devices of pressures of liquids and gases;
- – meters of thrust (or torque);
- – fuel flow meters.

This group of instruments also includes systems for measuring the amount of fuel, although they do not directly control the operation of the power plant.

Very strict requirements are imposed on the operating errors of such devices. So the measurement error of the rotor speed of the gas turbine engine should not exceed ± 0.5 %. In this case, the error in determining the thrust of the gas turbine engine will be 3 to 5 %.

The aircraft engine operates in intense thermal conditions. Therefore, thermometers are required to increase the accuracy of measuring the temperature of the gas flow. So, at the maximum values of the measured temperatures, the error in measuring the gas temperature of a gas turbine engine should not exceed ± 0.5%.

For automatic stabilization and reconfiguration of the aircraft engine operating modes, a set of sensors is used to measure the corresponding parameters and send signals to the automatic engine control system.

2.2. Fuel Measuring Systems

2.2.1. *Fuel Meters*

Instruments that measure the volumetric or weight quantity of fuel in tanks are called ***fuel meters***.

They allow the aircraft crew at any time during the flight to determine fuel amount in the tanks and estimate the time during which the flight can be continued.

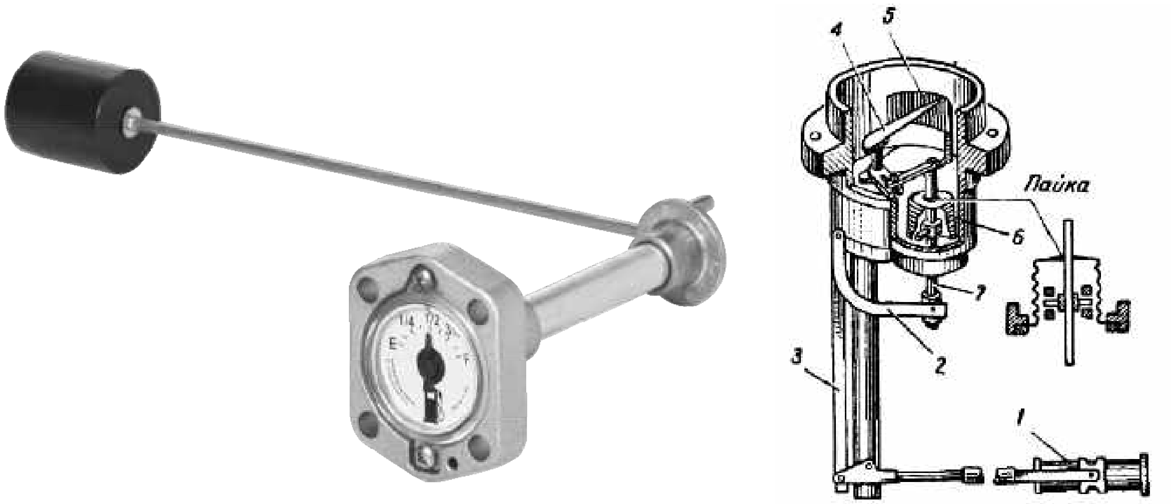
To maintain optimal engine operation, it is important to know the amount of fuel consumed by the engine per unit of time. To determine the instantaneous or average fuel consumption per unit of time, flow meters are used.

Using fuel meters, the total fuel supply in all tanks and the amount of fuel in each of them are determined separately. It is necessary to know how the fuel is distributed between the tanks in order to determine the correct sequence of fuel consumption from the tanks in order to avoid an unacceptable shift in the center of mass of the aircraft. Tank switching is controlled by automatic fuel gauges.

Most methods for measuring the amount of fuel are turn into measuring its level in the tank (the height of the liquid column). However, the scales of the fuel gauges are graduated in units of volume (liters) or in kilograms. Therefore, the calibration of the scale depends on the size and shape of the fuel tank.

Float and capacitive fuel meters are used to measure the amount of fuel in aircraft.

***Float fuel meters*.** The float fuel gauge consists of a sensor (Fig. 2.1), located in the fuel tank, and a pointer, which is placed on the dashboard.



a b

Fig. 2.1

There is a float 1 on the fuel surface. When the fuel level in the tank changes, the float moves the rod 3. The rod turns the rocker 2 and the lever 7, which moves the brush 4 along the wire potentiometer 5. Thus, the mechanical transmission converts the vertical movement of the float into an angular movement of the potentiometer brush. The sensor measures the fuel level in the tank and outputs the measurement result in the form of a DC or AC electrical signal.

The fuel gauge pointer is a ratiometer. The frames of the ratiometer r are included in the diagonal of the bridge circuit (Fig. 2.2).

Float-type fuel meters are characterized by methodological and instrumental errors.

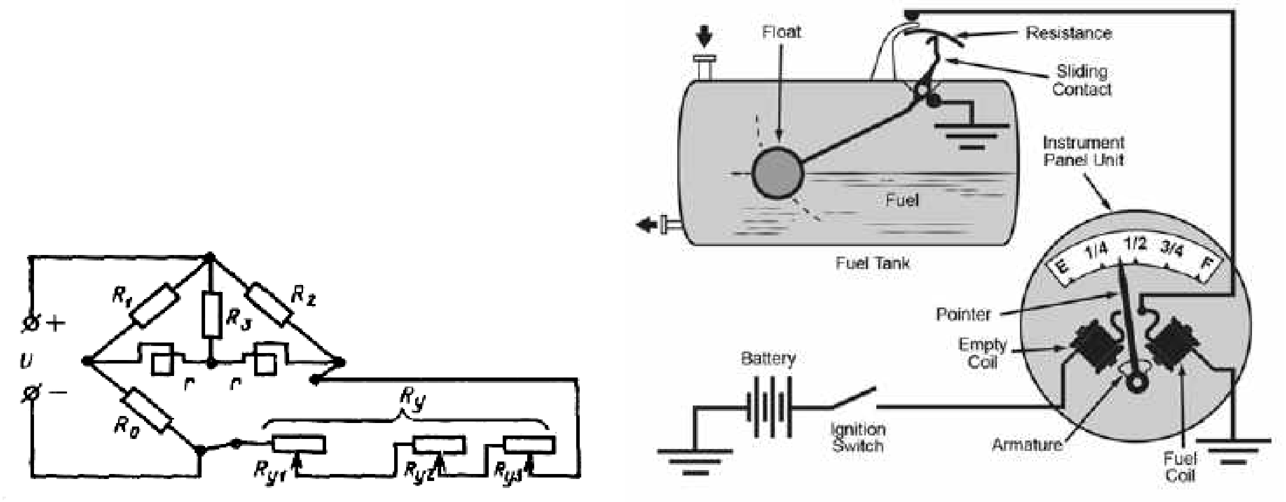


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Fig. 2.2

Methodological errors include:

a) errors due to changes in the position of the fuel in the tank during longitudinal and transverse tilts of the aircraft and when the aircraft is moving with longitudinal and lateral accelerations;

b) errors caused by the influence of temperature on the fuel level (in accordance with the coefficient of volumetric expansion of the fuel);

c) errors caused by the approximations made in the design of the device (approximation of the characteristics of rheostats, etc.).

The instrumental errors of the fuel meter are similar to the errors of other electromechanical devices. They are determined by the imperfection of mechanical and electrical elements, the presence of frictional forces, backlash, the effect of temperature on the mechanical, electrical and magnetic properties of parts and assemblies, etc.

Capacitive fuel meters. Electric capacitive fuel meters are most widely used in modern aircraft. The operation of a capacitive fuel meter is based on the dependence of the capacitance of the capacitor located in the fuel tank on the fuel level. Since the dielectric constant of the fuel differs from the dielectric constant of air, then when the level in the tank changes, the capacitance of the capacitor will also change.

Capacitive fuel gauges consist of a sensor located in the fuel tank and an indicator mounted on the dashboard of the aircraft. The set of the device also includes a measurement unit containing amplifying-converting and switching elements.

The principle of operation of a capacitive fuel meter is based on measuring the electrical capacity of a cylindrical capacitor – a sensor placed in the aircraft tank in a vertical position. The sensor (Fig. 2.3) consists of two (or more) concentric uninsulated pipes with a gap between them. The height of such a capacitor is equal to the height of the tank, and the electric capacity depends on the level, i.e. on the volume of fuel. The total capacitance of the sensor C is equal to the sum of the capacitances of the lower part of the sensor (condenser filled with fuel) with a height h_T and the upper part of the sensor (a capacitor whose dielectric is air or fuel vapors with a height $h_T - h_f$)

$$C = C_1 + C_2\,, \tag{2.1}$$

where C_1 is the capacity of the lower part of the capacitor, filled with fuel;

C_2 is the capacity of the upper part of the capacitor, the dielectric of which is air or fuel vapor.

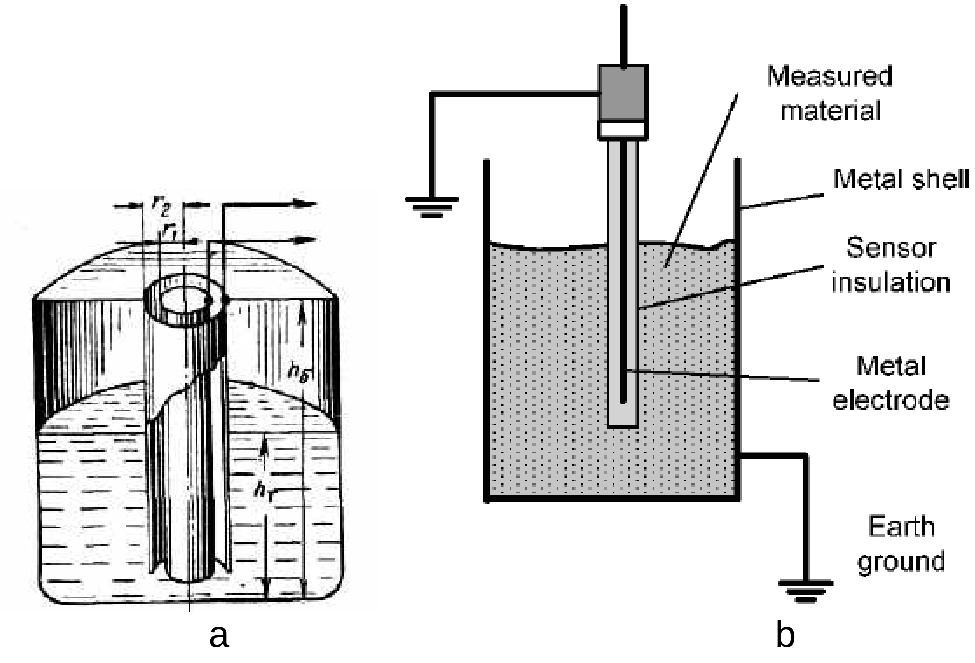

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Fig. 2.3

The capacities C_1 and C_2 depend on the fuel level h. Since the dielectric constants of air ε_B ($\varepsilon_B \approx 1$) and fuel ε_T ($\varepsilon_T \approx 2$) are different, then when the fuel level in the tank changes, the capacitance of the capacitor will also change.

The capacitance of the sensor is connected to one of the arms of the measuring bridge. When the capacitance of the sensor C changes, the balance of the bridge is disturbed and a signal appears at the input of the amplifier.

To measure the total fuel in several tanks, all sensors are connected in parallel.

Methodological errors include:

a) errors due to changes in the position of the fuel in the tank during longitudinal and transverse tilts of the aircraft and when the aircraft is moving with longitudinal and lateral accelerations;

b) errors caused by the influence of temperature on the fuel level (in accordance with the coefficient of volumetric expansion of the fuel).

Instrumental errors of a capacitive fuel meter are caused mainly by the effect of temperature on the parameters of the bridge circuit elements (capacitors, resistances).

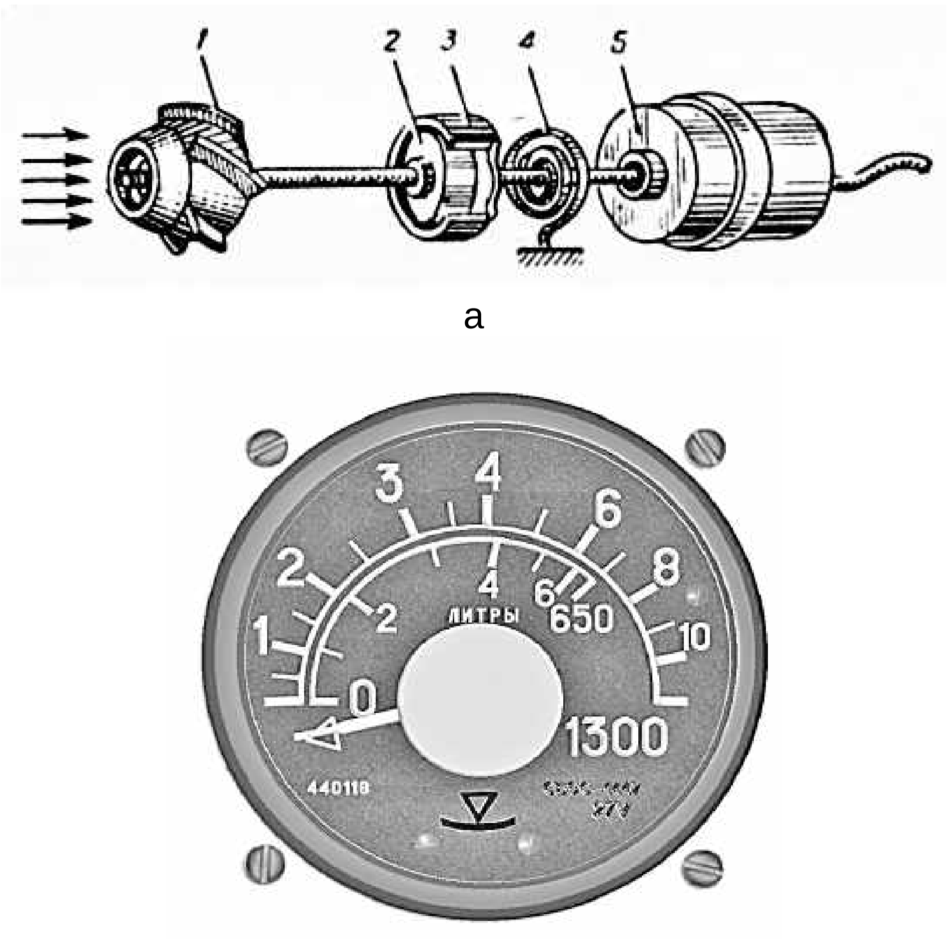
2.2.2. *Fuel Flow Meters*

To determine the instantaneous or average fuel consumption per unit of time, flow meters are used. To determine the total fuel consumption for a certain time (for example, during the flight), cumulative flow meters are used. These devices consist of a flow meter that measures the flow rate per unit of time and an integrator that adds signals proportional to the flow rates. Knowing the fuel consumption, it is possible to judge the maintenance of a given engine mode. Before the flight, the arrow of the totalizing flow meter is set to the division indicating the amount of fuel poured into the tanks, and as the fuel is used up, the device shows the stock (remaining) of fuel in the tanks (usually in kilograms).

Flow meters are more reliable than fuel meters. Their readings are independent of the maneuvering of the aircraft. On many aircraft, where combined fuel flow meters are used, the fuel meter controls the amount of fuel in the fuselage tanks, and the totalizing flow meter monitors the total amount of fuel, including in the outboard tanks (the fuel meter cannot measure the amount of fuel in the outboard tanks, since they have no sensors).

The disadvantage of cumulative flow meters is that they do not react to fuel losses when the fuel system is damaged.

The high-speed method is used to measure instantaneous fuel consumption. Flow meters, whose operation is based on the high-speed method, determine the instantaneous flow rate using an impeller (spinner) (Fig. 2.4), placed in the fuel line.



a

b

Fig. 2.4

The flowmeter pointer thus has two arrows and two scales. The internal scale shows the total fuel remaining in the aircraft fuel tanks, and the external scale shows the hourly fuel consumption.

Measurement of instantaneous fuel consumption makes it possible to set a preset engine operating mode. But the pilot needs to know the fuel consumption during the flight. This task is solved with the help of totalizing flow meters.

Topic 3. AVIATION DEVICES FOR MEASURING TEMPERATURE, PRESSURE AND ROTATION SPEED

3.1. Aviation thermometers

The devices that measure temperature are called ***thermometers***.

On aircraft, thermometers are used for remote measurement of the average temperature of exhaust gases from a gas turbine engine nozzle, exhaust gases from turbo starters, oil, coolants, outside air, etc.

As a principle of operation for the device of thermometers, one can use any physical process in which the temperature is unambiguously associated with some easily determined quantity (for example, a change in volume, electrical resistance, electromotive force, etc.).

Currently, thermometers are used based on various principles of operation, namely:

– thermometers, the work of which is based on thermal expansion of liquids and solids;

– manometric thermometers, the work of which is based on the change in pressure inside a closed volume when the temperature changes;

– electrical resistance thermometers;

– thermoelectric thermometers;

– radiation thermometers (pyrometric), etc.

A thermometer usually consists of a receiver, a pointer and connecting actuators. The receiver contains a heat sensing element.

For accurate measurement, the temperature of the sensing element must be equal or close to the temperature of the medium. The coincidence of these temperatures depends on the size, shape and material of the sensitive element, on the size, shape and material of the controlled environment and on the conditions of heat transfer to the sensitive element of the receiver. Any heat sink introduced into the controlled environment distorts its temperature field. This is one of the reasons for the temperature mismatch. The temperature of the sensing element is the closer to the temperature of the medium, the better the heat exchange between the medium and the sensing element.

By purpose, thermometers of aircraft engines can be divided into the following main types, differing in the measurement ranges:

a) Thermometers for measuring the temperature of outgoing gases in gas turbine engines with an upper measurement limit of up to 900 – 2000°C;

b) Thermometers for measuring the temperature of oil, water, air up to 150°C.

According to the measurement methods, thermometers are divided into three groups:

– mechanical;

– electrical:

a) resistance thermometers (resistor and semiconductor);
b) thermoelectric;

– radiation thermometers.

Mechanical thermometers, due to their inherent operational shortcomings (susceptibility to mechanical stress, the complexity of remote transmission of readings), are not used to control the operation of aircraft engines.

Electrical thermometers (resistance and thermoelectric thermometers) are used to control the operation of aircraft engines.

Radiation thermometers are not yet used on aircraft.

***Resistance thermometers*.** Electrical resistance thermometers are used on aircraft to measure the temperature of oil, coolant, air and other media in the range from - 50 to + 150°C.

Resistance thermometers are based on the temperature dependence of the electrical resistance of conductors or semiconductors. A thermistor serves as a sensitive element, i.e. measuring temperature is reduced to measuring resistance.

The dependence of the electrical resistance R of the thermistor on the temperature T is determined by the material of the thermistor. The higher is sensitivity of a thermistor, the greater (in absolute value) its temperature coefficient of resistance α.

Among conductors, platinum, copper, nickel have the highest value of the temperature coefficient α.

For aviation thermometers, the sensors are made of nickel wire with a diameter of 0.05 mm. The choice of this material is due to the fact that nickel, in comparison with copper, has a higher heat resistance and, accordingly, a higher measurement limit (up to 300°C), as well as a large value of the temperature coefficient α (0.00635 Ohm/°C).

Platinum is rarely used due to the fact that it is an expensive material, although its heat resistance is higher than that of nickel.

The aviation resistance thermometer circuit is an unbalanced DC bridge, one arm of which is the resistance of sensor, and the resistances of the other arms are constant. The diagonal of the bridge includes a tester, which is used as a magnetoelectric ratiometer. The sensitive element of the sensor is a nickel wire wound on a mica framework, which is installed in a stainless steel case.

The errors of resistance thermometers are the sum of the errors of sensor, the electrical measuring circuit and the pointer.

Sensor errors include:

1. Error due to losses from heat radiation and thermal conductivity. This error is due to the fact that part of the heat is removed from the thermistor to the place of its attachment and to the walls of the chamber or pipeline, inside which the sensor is located.

2. Error due to gas flow deceleration. This error arises when measuring the temperature of rapidly moving gas streams and is due to the transition of

the kinetic energy of the gas into heat when the gas is decelerated by the sensor.

3. Error due to heating of the sensitive element when current flows through it.

4. Dynamic error, which is caused by the lag of the temperature of thermistor in relation to the temperature of the medium due to the fact that the sensor has heat capacity.

The error of the electrical measuring circuit is due to the fact that when the ambient temperature changes, the electrical resistances of the ratiometer frames change, as a result of which the ratio of the currents flowing within this frame changes.

The ratiometer errors are due to:

– friction in the supports of the moving system of the ratiometer;

– imbalance of the moving system;

– the forces of interaction between the screen and the moving magnet of the ratiometer;

– backlash in the supports of the movable system;

– magnetic hysteresis in the screen material during its magnetization reversal.

Thermoelectric thermometers. The principle of operation of thermoelectric thermometers is based on measuring the thermoelectromotive force arising when heating the common junction point of two wires 1 and 2 (Fig. 3.1) from different metals – "hot" junction.

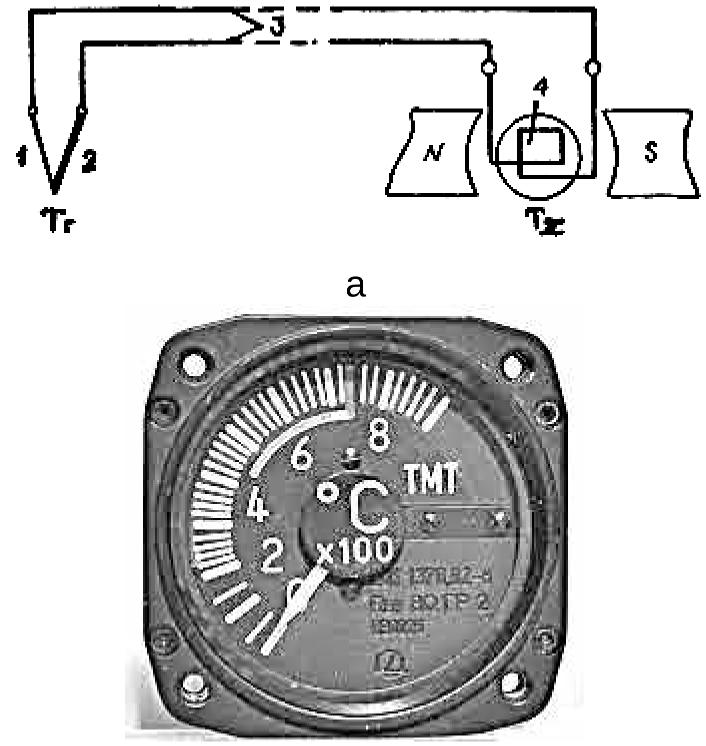
The cold junction is the frame of the measurer – millivoltmeter 4. The cold junction and the hot junction are connected by wires 3. The thermocouple 1-2 is the sensitive element of such thermometers.

The thermo-electromotive force measured by the device is the difference of two functions

$$e = e_h - e_c, \tag{3.1}$$

where e_h **–** thermoelectromotive force of "hot" junction;

e_c – thermoelectromotive force of "cold" junction.



a

b

Fig. 3.1

The most commonly used materials for thermoelectrodes are alloys based on chromel, alumel, nickel-cobalt (NC) alloy, and special alumel (SA).

The temperature of the exhaust gases at different points of the jet nozzle of a gas turbine engine may not be the same, and therefore temperature measurements at several points give more accurate results. The thermocouples are connected in series with the galvanometer.

The errors of thermoelectric thermometers are made up of the errors of a sensor, electrical measuring circuit and a pointer.

The sensor has most of the errors due to losses from thermal conductivity and heat radiation, due to deceleration of the gas flow, a dynamic error.

Errors in the electrical measuring circuit are caused by a change in the resistance of the electrical circuit when the ambient temperature changes.

Static errors of the pointer (galvanometer) are caused by the action of harmful frictional forces in the supports and by unbalance of the moving system; temperature change, which affects the modulus of elasticity of spring and the magnetic induction in the gap.

3.2. Aviation Pressure Gauges

In aviation, pressure gauges are used to measure the pressures of gases and liquids in power plants, in hydraulic systems of aircraft, in the fuel system, in the air system, in the pressurization system of pressurized cabins, in the oxygen system, etc.

The principle of operation of all pressure gauges is based on comparing the force of the measured pressure with the elastic force of the sensing element. The following are used as sensitive elements of pressure gauges:

- manometric tubes,
- gauge boxes,
- membranes,
- corrugated tubes – bellows,
- twisted tubular springs.

Gauge tubes (Fig. 3.2) are the simplest in design, reliable in operation and compact, but have low sensitivity.

Manometric boxes (Fig. 3.3) have high sensitivity, but are more difficult to manufacture and have a non-linear dependence of deformation on the measured pressure.

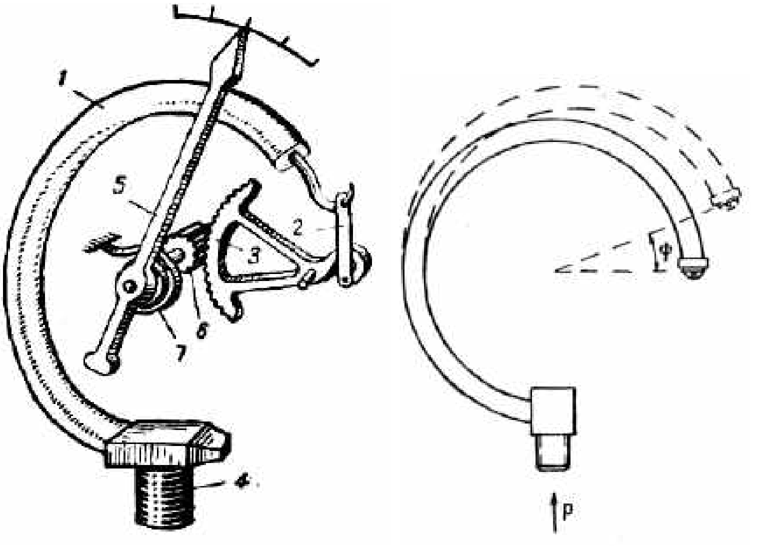


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Fig. 3.2

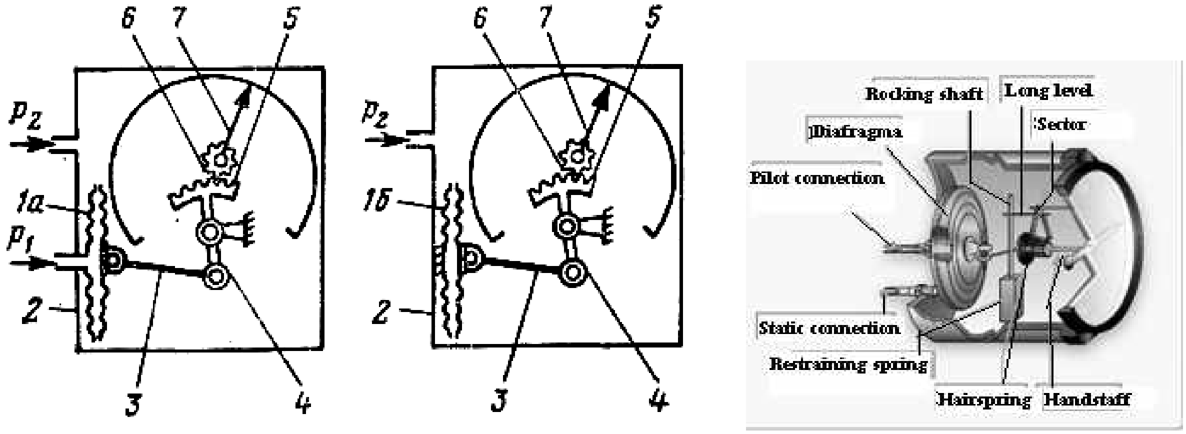

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Fig. 3.3

Bellows (Fig. 3.4) have significant sensitivity and an almost linear dependence of deformation on the measured pressure.

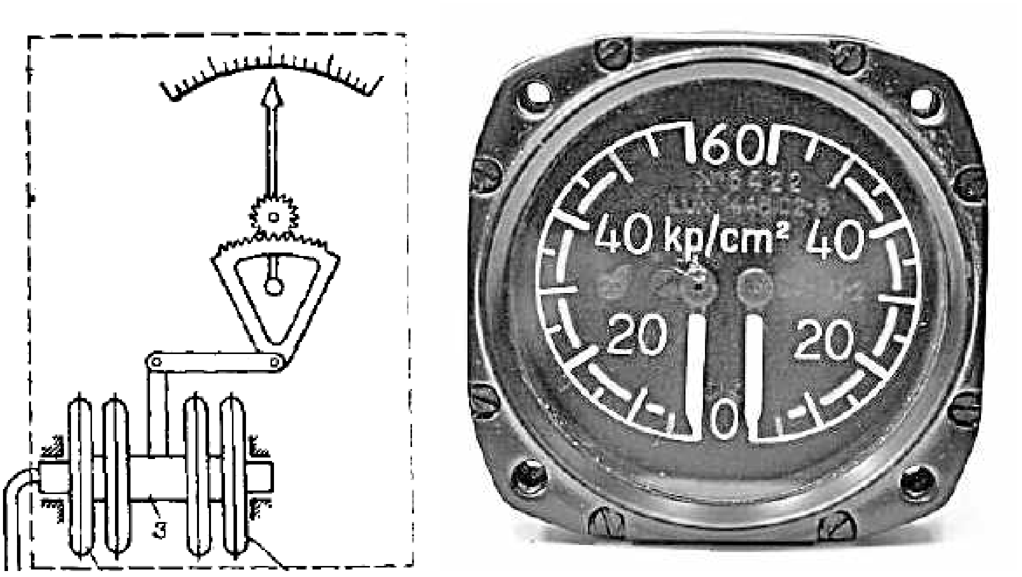

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Fig. 3.4

Diaphragms (Fig. 3.5) and coil tubular springs are used to measure high pressures.

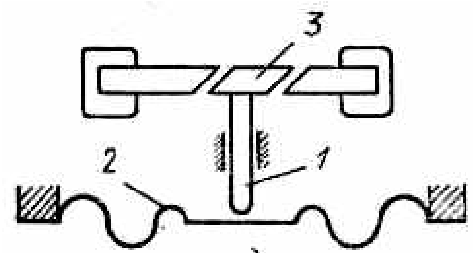

Fig. 3.5

By designation, aviation pressure gauges are divided into:

– pressure gauges measuring absolute pressure;
– pressure gauges measuring the pressure difference (differential);
– pressure gauges measuring the ratio of two pressures.

According to the methods of measuring pressure, pressure gauges are divided into:

– mechanical:
  a) liquid,
  b) weight,
  c) spring;
– electromechanical, based on the same principles as mechanical, but differing in that a mechanical sensing element is combined with an electrical remote transmission;
– electrical:
  a) electronic;
  b) gas-discharge;
  c) radioactive;

d) thermal;
e) piezoresistive.

Mechanical liquid pressure gauges are practically unsuitable for measuring pressure on moving objects due to changes in their readings when tilting and in the presence of accelerations.

Mechanical weighing gauges are beam scales in which the force developed by a piston or bellows is balanced by a reference weight. Weighing gauges are not applicable to moving objects.

Mechanical spring-loaded pressure gauges are based on the deformation of an elastic sensing element (diaphragm, bellows, tubular spring, etc.), which occurs under the action of the measured pressure.

Electromechanical remote pressure gauges can be divided into spring and power.

In a spring pressure gauge, an electrical signal is obtained from the transformation of displacement, and in a power one – from a transformation of force. The force or movement that is generated by the sensing element is converted into an electrical signal using a transmission-multiplier mechanism. This mechanism changes one of the electrical quantities – *R, L, C, U* (Fig. 3.6).

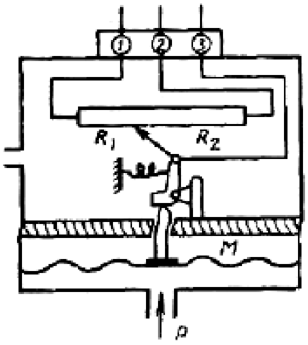

Fig. 3.6

The measuring instrument is a ratiometer, the scale of which is graduated in pressure units. The ratiometer is housed in an indicator housing that is mounted on the dashboard.

3.3. Aviation tachometers

The instruments that measure the speed of rotation are called tachometers.

Aviation tachometers are used to measure the rotational speed of the turbine shaft of a gas turbine engine (up to 25.000 rpm).

According to the principle of operation of sensitive elements, tachometers are divided into:

– centrifugal, in which the centrifugal forces of inertia are balanced by the force of elastic deformation of the spring;

– generator, based on the dependence of the magnitude of the electromotive force generated in the winding on the rotational speed of the inductor (tachogenerators);

– magnetic induction;

– pulse-frequency, in which the dependence of the frequency of the electromotive force of the synchronous generator on the frequency of its rotation is used. It is possible to use photoelectric, induction and other interrupters instead of a synchronous generator to form a pulse sequence.

In aviation, magnetic induction and frequency-pulse tachometers are used to measure the rotational speed of the power plant shaft. Frequency-pulse sensors are used to measure the speed of rotation of the impeller of instantaneous and totalizing fuel flow meters.

The operation of ***magnetic induction tachometers*** is based on the measurement of forces arising from the interaction of a rotating magnetic field with induction currents induced by this field in a solid metal rotor.

The sensor of the remote magnetic induction tachometer is a synchronous generator with a rotor in the form of a permanent magnet 1. A sensing element 2 located in a rotating magnetic field in the form of a solid thin-walled metal cylinder is fixed on a separate axis and is held by a spring 3 (Fig. 3.7).

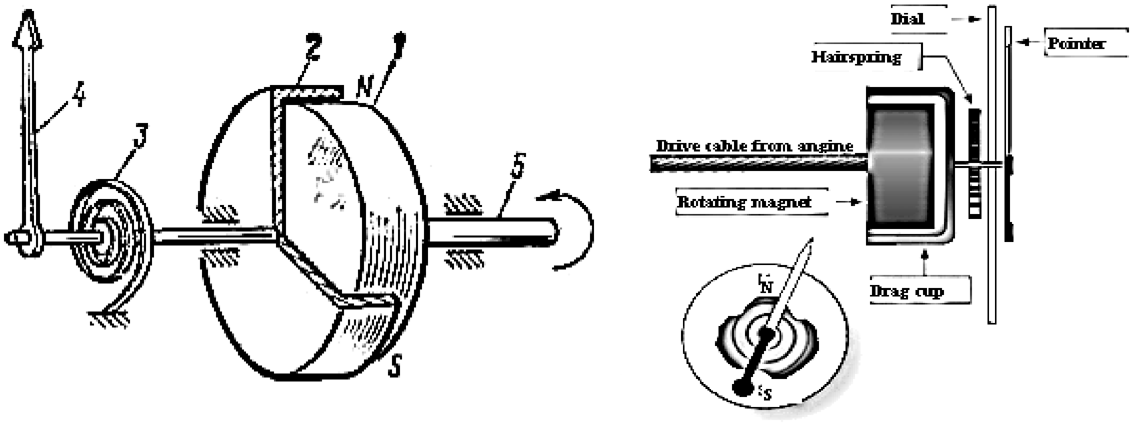

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Fig. 3.7

A permanent magnet is usually made multi-pole 1 (Fig. 3.8). The cylinder of the sensing element is often made in the form of a metal disk 2.

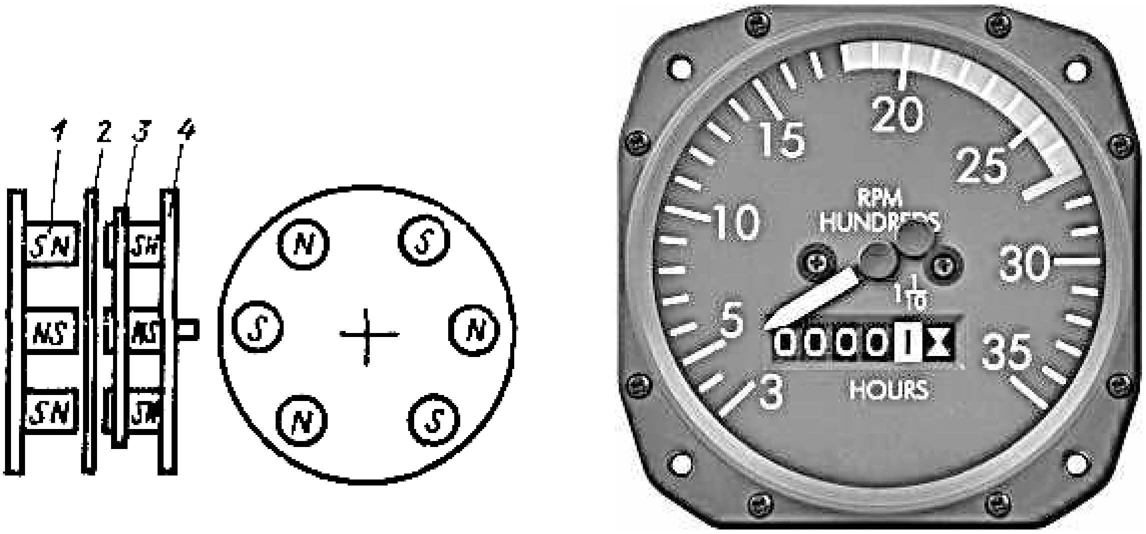

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Fig. 3.8

When the magnetic system rotates with an angular velocity ω, an electromotive force is induced in the cylinder walls, which causes a current which is closed in the cylinder body.

Interacting with the magnetic field, this current creates a torque proportional to the rotation speed of the magnet and tends to drag the cylinder along with the rotating magnetic system.

Under the action of the torque, the cylinder rotates and twists the spiral spring 3, which creates a counter moment proportional to the angle of twist (see Fig. 3.7). The deflection angle of the arrow 4 will be proportional to the angular velocity of the permanent magnet 1.

Topic 4. HEIGHT MEASURING DEVICES

4.1. Purpose of aerometric instruments

Aerometric instruments and systems are designed to measure physical quantities that characterize the movement of aircraft relative to the air environment. These values include:

- barometric altitude and the its rate of change;
- true air and indicated speed;
- mach number;
- dynamic pressure;
- angles of attack α and sliding β;
- outside air temperature T_H.

The listed physical quantities refer to aerodynamic quantities. Information about them is necessary to control the altitude and speed of flight, Mach number, the attitude of the aircraft relative to the ram air, the input devices of aircraft engines, etc. This information is used in the corresponding manual and automatic control loops.

The accuracy of aerometric measuring devices has a direct impact on the flight characteristics of an aircraft, i.e., they are related to flight safety devices.

4.2. Pitot Static Tubes

The output signals of most aerometric instruments and systems are generated from indirect measurements. The input quantities are static P_s and total P_t pressure of the ram air flow, which are functionally related to the measured values.

On modern aircraft, aerometric instrument power systems are used, which include:

1) static probes;
2) pitot tubes;
3) pitot static tubes - total and static pressure;
4) pipelines;
5) moisture traps;
6) switching valves.

To measure static P_s and total P_t pressures at subsonic flight speeds, pitot tubes are used (Fig. 4.1). The pitot tube housing is installed on the aircraft exactly in the direction of flight.

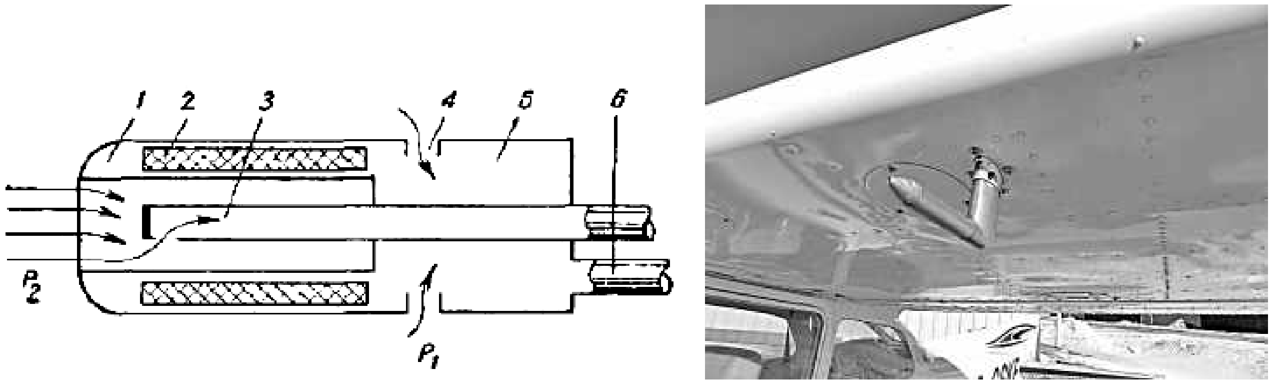

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Fig. 4.1

In the housing 1 of the pitot static tube there is a tube 3, which is open from the end of the housing towards the air flow and serves to take up the total pressure P_t. The body also contains a cavity 5 of static pressure P_s. The static pressure cavity is connected to the atmosphere through holes 4 on the surface of the pitot static tube housing. These holes are located in the zone of undistorted static pressure P_s. To aerometric instruments, the total pressure P_t is supplied through pipe 3, and static P_s through pipe 6.

To prevent icing, all pitot static tubes are electrically heated by a heating element 2 of nichrome wire.

In order to increase reliability, two or more pitot static tubes are installed on the aircraft.

At supersonic flight speeds, a shock wave appears in front of the pitot tube, distorting the static pressure. To measure the flight speed and Mach number at supersonic speeds, it is necessary to measure the total pressure behind the shock wave, and the static pressure in the undisturbed flow.

A conventional pitot static tube that combines a static and dynamic tube in one housing is not quite suitable in this case, since the holes located on its surface will take up the pressure distorted by the shock wave. To measure the static pressure in a supersonic flow, a pitot static tube with far-back (~ 10D) receiving holes is used, or separate independent pitot tubes and pitot static tubes are used. The pitot static tubes has only a total pressure tube (no static tube). The static probe is made in the form of holes located directly on the aircraft fuselage skin. The pressure is supplied to the devices through a special pipeline from the holes.

On supersonic aircraft, combined static and total pressure tubes of the **ПВД-18** type are widely used (Fig. 4.2).

In the outer tube of the tube, there are openings of three chambers C_1, C_2 and C_3 to take up static pressure. The openings of the chambers C_1 and C_2 are located on a cylindrical surface. The openings of the C_3 camera are located on a conical surface, which is an aerodynamic compensator designed to equalize the static pressure on the tube surface under certain flight conditions.

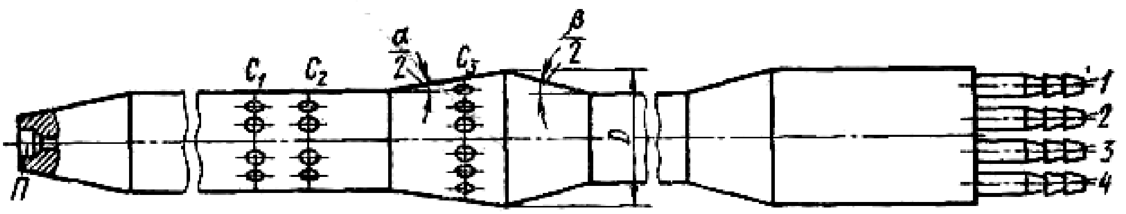

Fig. 4.2

The number of holes and their location are selected in order to ensure the independence of the pressures in the chambers from the angles of attack α and slip angle β. Static pressure measurement accuracy depends on the geometry and dimensions of the compensating circuit, as well as the distance between the pitot tube and the aircraft. Therefore, the pitot tube are produced in various modifications for different aircraft.

When flying at a subsonic speed ($M < 1$), the pressure in the C_3 chamber is close to the static one, and in the C_1 and C_2 chambers it differs significantly from it. When flying at supersonic speed ($M > 1$), the pressure in chamber C_3 differs significantly from the static one, but at the same time in chambers C_1 and C_2 it is close to static.

Therefore, when flying at subsonic speeds, the C3 camera is used, and at supersonic speeds, the C_2 and C_1 cameras are used. Switching the static pressure line to the holes of a particular chamber is done automatically using a pneumatic switch of the **ПП1** type (Fig. 4.3), which is triggered when airplane turns to the speed of sound.

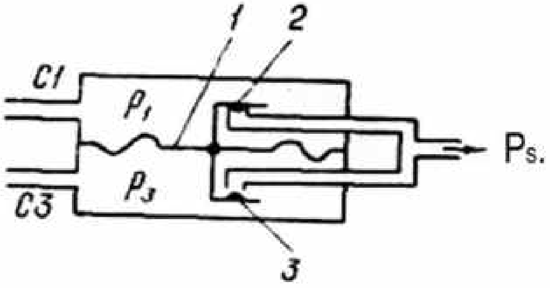

Fig. 4.3

Membrane 1, with the movable center of which valves 2 and 3 are connected, reacts to the pressure difference P_1 and P_2 in chambers C_1 and C_3. As a result, a chamber with a lower pressure is connected to the instrument line.

Chamber C_2 is always connected to the coarse static pressure line and is connected to instruments that do not require precise static pressure.

4.3. The Principle of Operation and Construction of the Barometric Altimeter.

Flight altitude is a geometric height characterizing the vertical distance between certain levels (points) of the height reference point and the aircraft (its center of mass).

Let us explain the types of heights in Fig. 4.4. Distinguish:

– the absolute height **h** is the height relative to sea level;

– true height $\mathbf{h_t}$ is the height above the terrain flown;

– the relative altitude h_r is the altitude relative to the take-off or landing site.

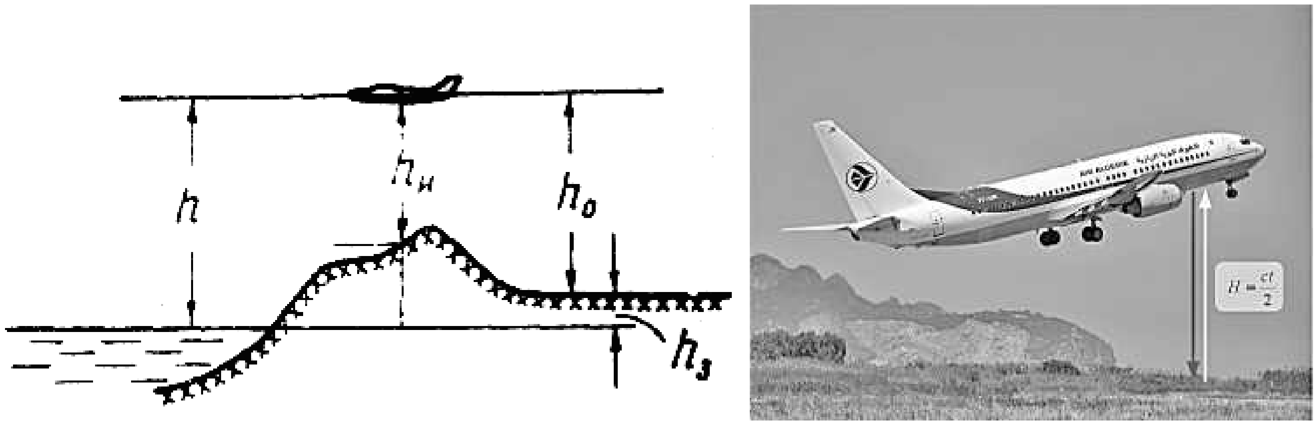

Fig. 4.4

In aviation, radio engineering and barometric methods for measuring flight altitude are used. The first corresponds to radio altimeters, which directly measure the true altitude. However, they are generally unsuitable for determining the relative altitude $\mathbf{h_r}$, as well as for ensuring vertical flight separation. These two tasks are solved using barometric altimeters and altitude channels of air signal systems implementing the barometric method.

The barometric method for measuring flight altitude is based on the dependence of the absolute pressure P in the atmosphere on the altitude H.

The atmospheric air parameters are not deterministic, i.e. unchanged. They are characterized by random spatial and temporal variations. The barometric method for measuring altitude is based on the averaged dependences of changes in air parameters over height – mathematical models of the atmosphere, the combination of which is called the "standard atmosphere".

The main parameters of a standard atmosphere are:

– pressure P_s = 101325 Pa (760 mm Hg);

– temperature T_s = 288.15 K (t = 15°C);

– density ρ_s = 1.225 kg / m^3;

– acceleration of gravity g_c = 9.80665 m / s^2;

– sound speed a_s = 340.294 m / s.

It has been established that the average temperature in the atmosphere up to the altitude of 11000 m is a linear function of altitude

$$T = T_o + \tau H, \tag{4.1}$$

where T_o = 288 °C is the average absolute temperature at sea level;
τ = 0.0065 deg / m – temperature gradient;
and the pressure is changed according to the following law

$$P = P_0(1 - \frac{\tau}{T_0} H), \tag{4.2}$$

where P_o = 760 mm Hg. Art. (101325 Pa).

Formula (4.2) is called the standard barometric formula. If we solve this equation for the height H, then we get the height:

$$H = \frac{T_0}{\tau}[1 - (\frac{P}{P_0})^{R\tau}], \tag{4.3}$$

where R = 29.27 m / deg. is a gas constant.

From formula (4.3) it can be seen that the height is a function of four parameters

$$H = f\,(P_H, P_0, T_0, \tau).$$

For a standard atmosphere, the parameters P_0, T_0, τ are constant. Thus, the height is unambiguously dependent on pressure. Therefore, by measuring pressure, the scale of the device can be calibrated in meters. A device based on this principle is called a ***barometric altimeter***.

ВД-type altimeters are widely used in aircraft. A schematic diagram of such an altimeter is shown in Fig. 4.5.

The static pressure Pc from the pitot static tube is supplied to the sealed case of the device 1, under the action of which the aneroid box 5 is deformed. It is a sensitive element of the barometric altimeter. The aneroid box consists of two corrugated elastic membranes, soldered to one another along the outer circumference. To increase the sensitivity of the altimeter, two or more aneroid boxes can be installed.

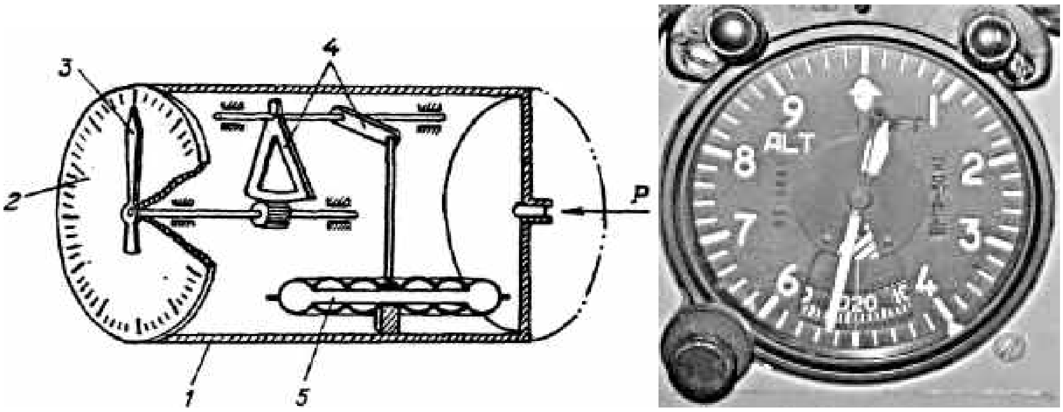

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Fig. 4.5

With an increase in flight altitude, atmospheric pressure decreases and the aneroid box expands. The deformation of the box by means of the transmission mechanism 4 moves the arrow 3. The altimeter has two arrows: the big arrow shows the height from 0 to 1000 meters, the small arrow shows the height in kilometers. Accordingly, the device has an external scale and an internal scale 2.

The parameters of the aneroid box and the transmission mechanism are chosen so that when lifting to a height of 1000 m, the large arrow makes one revolution, and the small one moves one division. At the same time, the outer scale of the device is graduated in hundreds and tens of meters, and the inner scale is in kilometers. To obtain a slow motion of the small arrow, a gear transmission mechanism is used.

The altimeter has a barometric scale in the cutout in the side of the instrument scale, a rack and altitude indices. The barometric scale is used to set the origin of the barometric pressure reading. When set with the help of the rack of the arrows to 0, the barometric pressure at the take-off aerodrome is measured on the barometric scale. If the barometric pressure scale is set to 760 mm Hg. Art., then the altimeter arrows show the absolute height (the height of the airfield in relation to sea level).

Altitude indices serve as an extension of the barometric scale, but not all altimeters have them.

The error of modern altimeters at medium and high altitudes at normal temperatures does not exceed ± 2 %. At low altitudes, the errors are: at zero height ± 10 m, at an altitude of 500 m – ± 20 m.

Topic 5. FLIGHT SPEED MEASURING INSTRUMENTS

5.1. Principle of Operation and the Structure of Variometers

A ***variometer*** is a device for measuring the vertical speed of an aircraft, i.e. the rate of ascent or descent.

The variometer is used as flight instruments and vertical speed sensors.

In fact, it does not measure the geometric vertical velocity, but the time derivative of the barometric altitude, since the signal of its sensing element is determined by the derivative of the barometric pressure.

The principle of operation of the variometer is based on measuring the delay in the change in the static pressure in the instrument case compared to the pressure in the sensitive element – the manometric box.

To clarify the principle of operation of the variometer, consider its arrangement (Fig. 5.1).

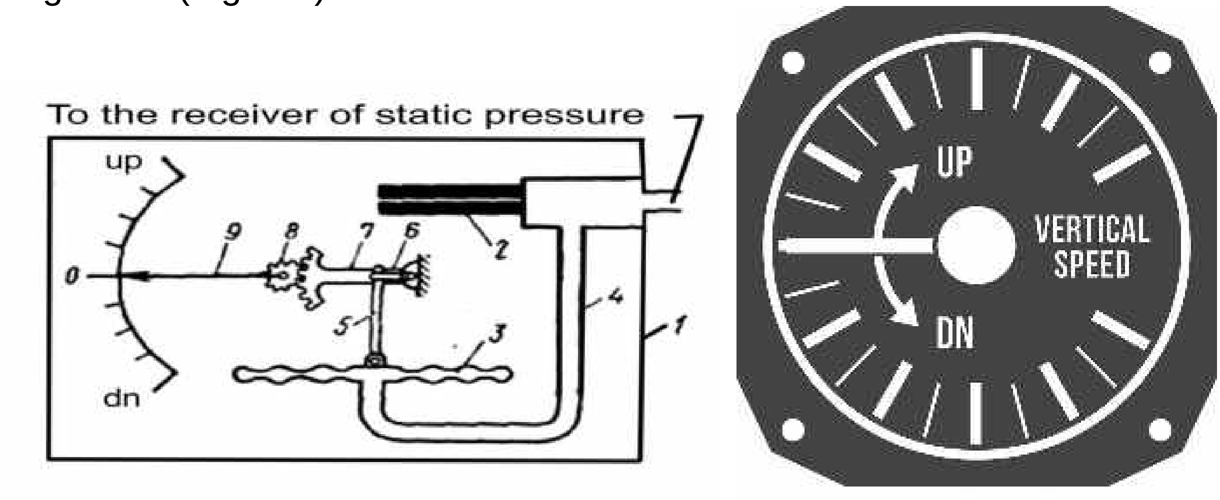

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Fig. 5.1

There are 1 – sealed housing; 2 – capillary; 3 – gauge box; 4 – pipeline; 5, 6, 7, 8 – transfer-multiplier mechanism (5 – rod, 6 – crank, 7 – sector, 8 – tube); 9 –arrow in Fig. 5.1.

The variometer works as follows: the static pressure P_c is supplied from the pitot static tube to the sealed body of the device 1 through the capillary 2, and into the manometric box 3 from the inlet fitting through the pipeline 4. When the flight altitude changes, the static pressure P_c is changed. Inside the manometric box, the pressure P_c is set almost instantly, and in the device body, due to the resistance of the capillary, the pressure P_{hous} differs from the static $\mathbf{P_c}$. The greater the vertical flight speed of the aircraft, the greater the pressure difference $P_c – P_{house}$. Under the influence of this difference, the manometric box 3 is deformed. Its deformation is transmitted to arrow 9 by means of a transfer-multiplier mechanism (5, 6, 7, 8).

In level flight (at constant altitude), the pressure difference $P_c - P_{hous}$ is zero, i.e. the pressure inside the housing 1 is atmospheric and the arrow 9 points to zero.

With increasing altitude, the atmospheric pressure continuously decreases and air from the housing 1 flows out through the capillary 2, the pressure in the housing decreases. However, due to the resistance of the capillary, the pressure inside the body does not have time to become equal to the atmospheric one instantly. Therefore, an excess pressure is formed, the magnitude of which is greater the faster the aircraft gains altitude. Under the influence of the resulting pressure difference, the gauge box 3 is compressed and, with the help of the transfer-multiplier mechanism, moves the arrow 9 upwards from zero.

As soon as the aircraft stops climbing, the atmospheric pressure stops changing and the pressure in the device body after a while will equal the atmospheric pressure. The instrument arrow will return to zero.

Since the vertical speed can change sign (descent – ascent), the scale of the device has a graduation with zero in the middle. In this case, with a decrease in P_c, air flows out through the capillary from the device body, and with an increase, it enters the body through it. In the first case, the pressure in the body will be greater than the external one, in the second – less.

Basically, the variometer is used to maintain level flight or a given low vertical speed.

Variometers are produced with different measuring ranges: from ± 10 m / s to ± 300 m / s of the following types: **BP-10, BAP-30, BAP-75, BAP-150** and **BAP-300**, where the number indicates the measurement limits.

5.2. Flight Speed and Methods of its Measurement

The aircraft's speed is measured relative to the air and relative to the earth. This distinguishes:

- true airspeed (**V**);
- mach number (M);
- indicated (indicator) airspeed (V_{IAS});
- ground speed (W).

True airspeed V is the speed of the aircraft relative to the air.

Information about the true airspeed is used to solve navigation problems: dead reckoning, determining the time of arrival at a given route point. The accuracy of measuring the true airspeed must ensure the arrival of the aircraft to the aerodrome or a given target with time accuracy Δt = 15 – 30 s.

The ***Mach number*** is the ratio of the true airspeed V to the speed of sound a

$$M = \frac{V}{a} . \tag{5.1}$$

Mach number is a dimensionless quantity.

Mach number information is used to prevent an aircraft or engine from entering critical controllability modes. In addition, the control laws and characteristics of manual, semi-automatic and automatic flight control systems are changed in the function of the Mach number. The highest accuracy in measuring the Mach number is required by the problem of preventing the aircraft from reaching critical flight conditions. For a reliable solution of this problem, a signal of approaching the critical regime is given with a margin of 1 – 2 % up to M_{cr}.

The ***indicated (indicator) airspeed*** (IAS) V_{IAS} is a physical quantity that is a function of only the dynamic pressure P_d and has the dimension of speed

$$P_d = P_f - P_{st}. \tag{5.2}$$

In other words, the indicated airspeed is the dynamic pressure expressed in units of speed.

Indicated airspeed characterizes the aerodynamic forces acting on the aircraft. As you know, the aerodynamic forces acting on the aircraft in flight are proportional to the velocity head of the oncoming air flow $\rho V^2/2$. Information about it is used primarily to set the minimum admissible speed under the conditions of a stall into a spin and the maximum admissible speed under the conditions of the aircraft strength. Thus, the pilot uses the indicated airspeed to carry out aerobatic missions.

The indicated (indicator) airspeed coincides with the true air speed under normal atmospheric conditions during flights near the Earth (air density $\rho = 1.225$ kg / m^3). With an increase in flight altitude, the true airspeed increases in comparison with the indicator airspeed and differs significantly from it.

Ground speed *W* is the horizontal component of the flight speed of the aircraft relative to the Earth.

In magnitude and direction, the ground speed is determined by the geometric sum of the vectors of the true airspeed *V* and wind speed *U* (Fig. 5.2).

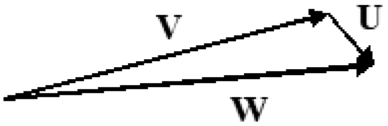


Fig. 5.2

Airspeed is *a* vector quantity that requires knowledge of module and direction to determine. The direction of the airspeed vector in the coordinate system rigidly connected with the aircraft axes is determined by the angles of attack and slip. Thus, for a complete determination of the airspeed vector, it is necessary to measure the vector modulus and the angles of attack and slip.

To measure the speed of an aircraft relative to the air, devices are used, which are called speed indicators as well as instruments for measuring angles of attack and sliding.

There are the following methods for measuring speed:

– aerometric;

– inertial;

– doppler.

The aerometric method is based on measuring the velocity (dynamic) air pressure, functionally related to the flight speed.

The inertial method is based on the measurement of accelerations and a single integration of the received signals.

The Doppler flight speed technique measures the Doppler frequency shift of the radio signal reflected from the Earth.

5.3. Principle of Operation and Structure of Airspeed Indicators

Aviation uses indirect methods for measuring the Mach number, true airspeed and indicated airspeed. They are based on the use of a functional relationship between the parameters P_d**,** P_{st} and T of the incoming air flow and its speed ($P_d = P_f - P_{st}$ – dynamic pressure).

The value of the dynamic pressure P_d is a function of the true airspeed V:

$$P_d = \frac{\rho V^2}{2}, \tag{5.3}$$

where ρ – air density;

The indicator of the indicated airspeed measures the dynamic pressure of the oncoming air flow $q = \rho V^2/2$, expressed in units of flight speed. Its scale is calibrated at normal air density, so the readings of the device correspond to the true airspeed when flying near the Earth.

As you know, the aerodynamic forces acting on an aircraft in flight are proportional to the dynamic pressure. So the lift of the aircraft is also a function of the dynamic pressure. Thus, the indicated airspeed indicator essentially provides lift information at any aircraft altitude. Therefore, in order to maintain the balance of forces acting on the aircraft, when piloting, it is important to know not the true airspeed, but the indicated flight speed.

The indicated airspeed indicator can be used not only as a flight instrument, but also as a navigation instrument to determine the true airspeed. In this case, a number of corrections are introduced into the readings of the device.

The device (Fig. 5.3) consists of a pitot static tube 1, a static pressure probe 2 and an indicator 3.

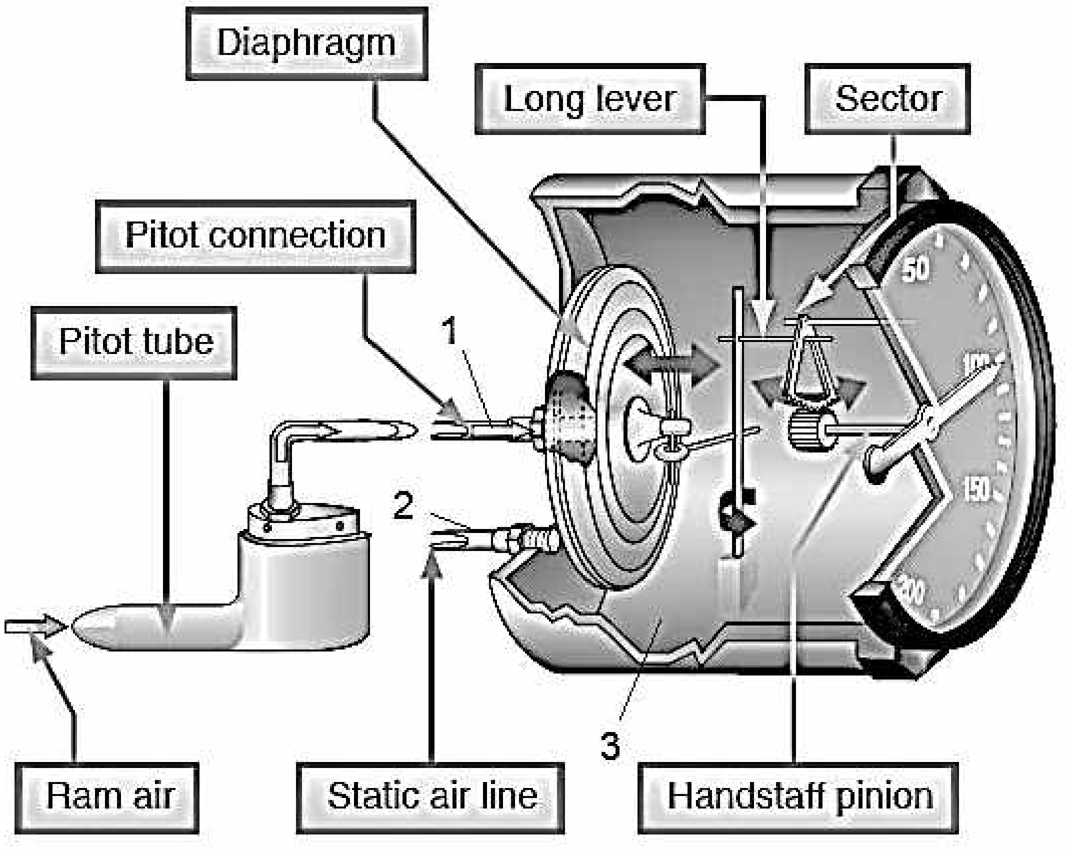


Fig. 5.3

During the flight the oncoming air flow approaching the pitot static tube is decelerated and its relative speed is made equal to zero. In this case the kinetic energy of air particles is converted into potential energy as a result of which excess pressure is created in the tube.

The total air pressure P_t equals the sum of static P_{st} and dynamic P_d pressures. It enters the cavity of the membrane box 4 which is the sensitive element.

$$P_t = P_{st} + P_d \,. \tag{5.4}$$

Static pressure P_{st} is supplied to the internal cavity of the device. Thus, the elastic sensitive element 4 will be affected by the difference between the total and static pressures

$$P_t - P_{st} = P_d = f(V). \tag{5.5}$$

Instrumental errors are inherent in the indicated speed indicators:

– the temperature instrumental error of the indicated speed indicator is caused by the influence of temperature on the elastic sensitive element and the transmission mechanism. The main source of temperature error is the change in the elastic modulus of the sensing element. To reduce this error, a bimetallic compensator is used;

– the error caused by different linear expansion of the mechanism parts is insignificant.

The true airspeed indicator, like the indicated airspeed indicator, is based on the measurement of the dynamic pressure of the oncoming air flow. The difference is that the true airspeed indicator automatically corrects for changes in air density, or rather, for changes in temperature and air pressure at flight altitude.

True airspeed indicators consist of two measuring elements manometric 1 and aneroid 4 (Fig. 5.4). The displacement of the center of aneroid box 4 is kinematically summed up with the displacement of the center of gauge box 1. This takes into account the change in pressure.

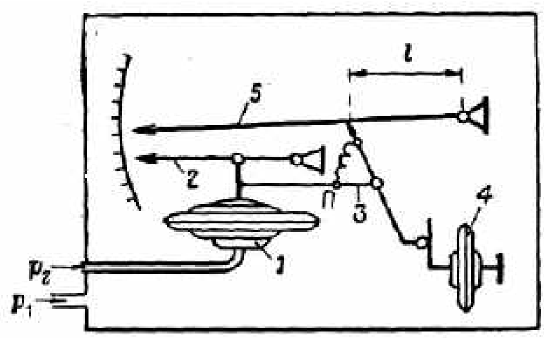


Fig. 5.4

An outside air thermometer can be used to measure the temperature. However, it is not possible to measure the temperature at the flight altitude with sufficient accuracy, therefore, instead of the outside air temperature, a compensation correction is introduced into the indication of the true airspeed indicator according to the standard atmosphere.

The true airspeed indicator has errors similar to the indicated airspeed indicator. Some difference lies in the fact that instrumental errors caused by a change in the modulus of elasticity of the material of gauge and aneroid boxes are partially mutually compensated. Therefore, temperature compensation is not used in true airspeed indicators.

The combined airspeed indicator combines two instruments – the indicated airspeed indicator and the true airspeed indicator with incomplete temperature compensation.

The device has a single scale and two arrows (see Fig. 5.4), one of which shows the indicated airspeed (wide arrow), and the other shows the true airspeed (narrow arrow).

When flying at low altitude, the indicated and true airspeed coincide and both arrows move along the scale together. With climbing, the true airspeed exceeds the indicated air speed and the arrows diverge, forming a "fork". The gap of this fork increases with increasing altitude, with the true airspeed being reckoned with the arrow, the reading of which is greater.

The pilot uses the indicated airspeed arrow (wide) mainly during takeoff and landing. The true airspeed arrow (narrow) is for navigation purposes. In addition, this arrow can be used to indirectly determine the Mach number, since at a certain air temperature, each Mach number corresponds to a completely specific true air speed.

Topic 6. MACH NUMBER INDICATORS AND AEROMETRIC SYSTEMS

6.1. Principle of Operation and Structure of the Mach Number Indicators

Mach number is the ratio of the aircraft's true airspeed *V* and the speed of sound.

The speed of sound at sea level under standard atmospheric conditions is 1224 km/h.

At flight speeds close to the speed of sound, the nature of the air flow around the aircraft wing changes sharply. As a result of this, there are significant changes in the coefficients of lift C_y and drag C_x. Therefore, for correct piloting of an aircraft at these speeds, it is necessary to know not only the indicated airspeed, but also the Mach number.

Changes in the coefficients C_y and C_x lead to a change in the controllability characteristics of the aircraft. In order for the pilot to cope with the control with changing characteristics, they must know the values of the Mach number at which these changes occur.

To obtain the Mach number, it is necessary to measure the dynamic pressure *q*

$$q = \frac{\rho V^2}{2},$$

which determines the dynamic pressure P_d as well as the static pressure at the flight altitude.

It is known that the Mach number is a function of the ratio of dynamic pressure and static pressure. Therefore, to measure the Mach number, you can use the true airspeed indicator (Fig. 6.1) with a graduated scale in the Mach number units.

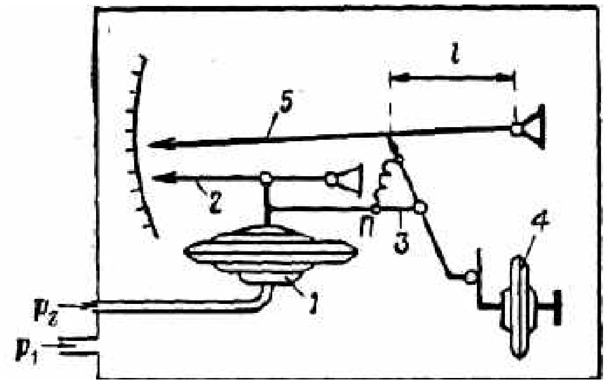


Fig. 6.1

To measure the static pressure, an aneroid block 4 is introduced into the device circuit, which takes up a change in static pressure (P_1). The displacement of the center of aneroid box 4 is kinematically summed up with the displacement of the center of gauge box 1, into which the dynamic pressure

(P_2) is supplied. This takes into account the change in pressure. Therefore, the Mach number indicators consist of two measuring elements, manometric 1 and aneroid 4.

An outside air thermometer can be used to measure the temperature. However, it is not possible to measure the temperature at the flight altitude with sufficient accuracy, therefore, instead of the outside air temperature, a compensation correction is introduced into the indication of the true airspeed indicator according to the standard atmosphere.

As flight instruments, mechanical Mach number meters are used as independent instruments, or as part of combined true airspeed and Mach number indicators of the **УИСМ** type. The Mach number indicator has a uniform scale with divisions from 0.4 to 2.5 with a division value of 0.02 M.

Mach number indicators are installed on modern aircraft: **M-1.5**, **MC-1.5**, **M-2.5**, **MC-2.5**. The letter C indicates that the instrument has a critical Mach number alarm.

6.2. Measuring Angles of Attack and Slip

The speed of an aircraft is a vector quantity, to determine which it is necessary to know its module and direction in space. If we talk about the airspeed vector, then in the coordinate system rigidly connected with the aircraft axes, its direction is determined by the angles of attack and slip.

The angle of attack α is the angle between the longitudinal axis of the aircraft Ox_1 and the projection of the airspeed vector onto the plane of symmetry Ox_1y_1 (Fig. 6.2).

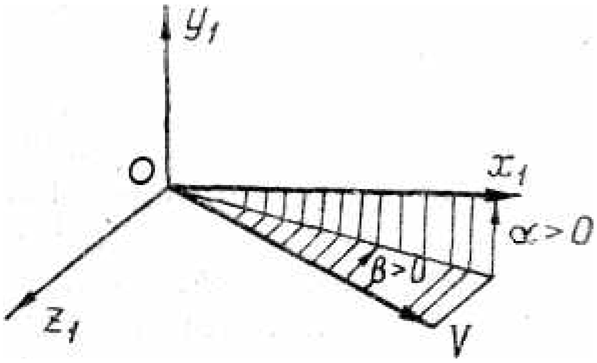


Fig. 6.2

The angle of attack is positive if the projection of the airspeed vector deviates from the longitudinal axis towards the negative end of the normal axis Oy_1.

The slip angle β is the angle between the airspeed vector and the plane of symmetry of the Ox_1y_1 aircraft (see Fig. 6.2).

The slip angle is positive if the airspeed vector is deflected from this plane towards the right wing.

Information about the angles of sliding and attack is used to solve the problems of piloting, shooting and bombing. A pilot or an autopilot has the task of maintaining a zero slip angle, at which the plane is symmetrically flown around by the incoming air flow, has a lower drag and less aerodynamic disturbances.

The information about the angle of attack is used to prevent the aircraft from stalling into a tailspin, which occurs after exceeding α_{cr} – the critical angle of attack corresponding to the maximum lift.

To maximize the dynamic capabilities of the aircraft, flights should be allowed at angles close to critical. However, for reasons of flight safety, it is necessary to leave a certain margin for the angle of attack

$$\Delta\alpha = \alpha_{cr} - \alpha.$$

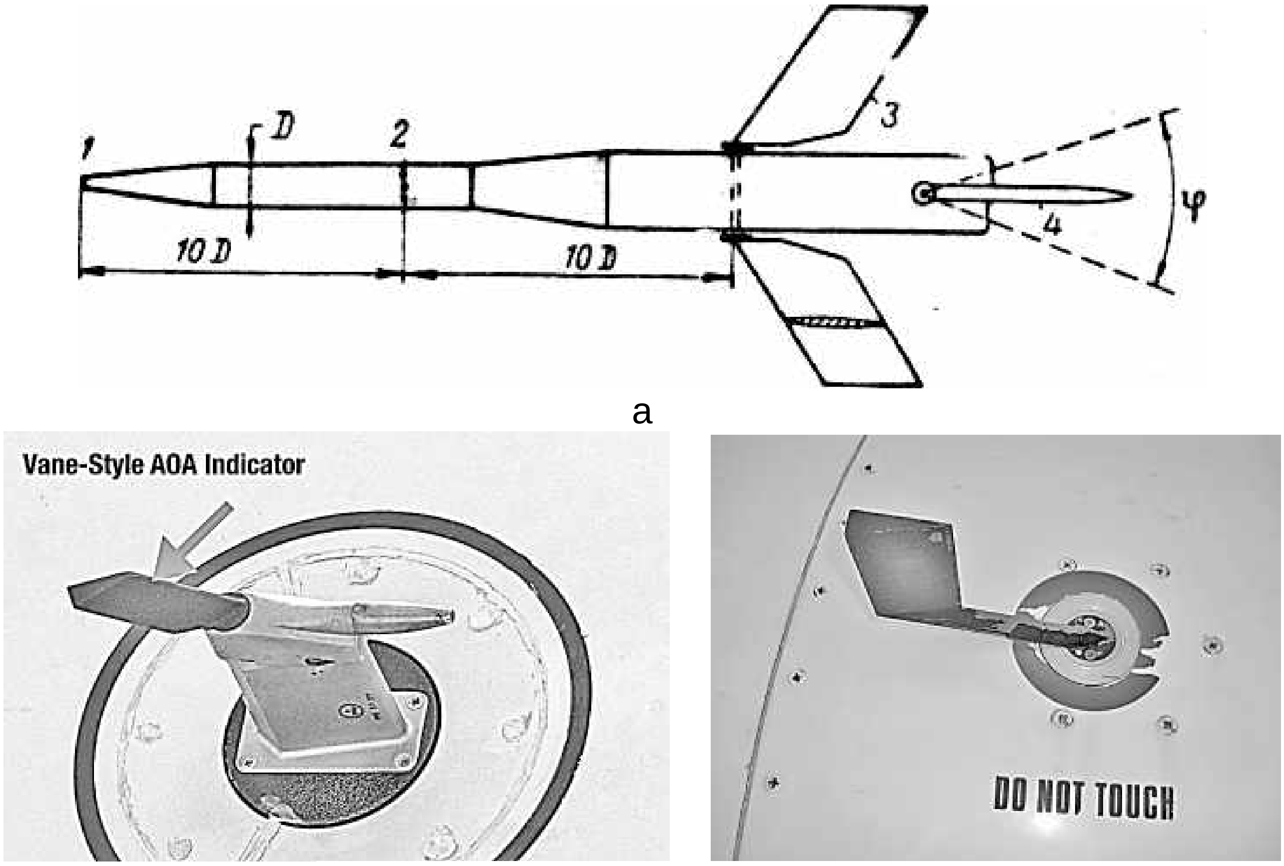
In this regard, the problem arises of indicating not only the angle of attack, but also its critical value. On landing, it is required to maintain a given landing angle of attack, which ensures effective deceleration of the aircraft and maintains the required lift.

In modern automatic control systems, signals of the angles of attack and slip are introduced into the systems of increasing the stability of the aircraft.

Measuring the angles of attack and slip is an urgent and rather complex technical problem. Its complexity lies in the fact that the air flow in front of the flying aircraft is disturbed, and in order to measure the true air angle of attack and the true slip angle, the sensor has to be carried far enough forward on special rods, as a rule, combined with receivers such as a pitot static tube.

To measure the angles of attack and slip, the aerometric method is used, on the basis of which autonomous onboard instruments are created. Consider two options for such devices.

The simplest sensitive element of the angle of attack and slip meters is the weather vane, that is, the streamlined element of the symmetrical profile, which can freely rotate around the axis 0. If the direction of the flow changes, then the weather vane will be installed along the flow. Usually, vane meters of the angles of attack and slip are combined with pitot static tubes (Fig. 6.3).



a

b

c

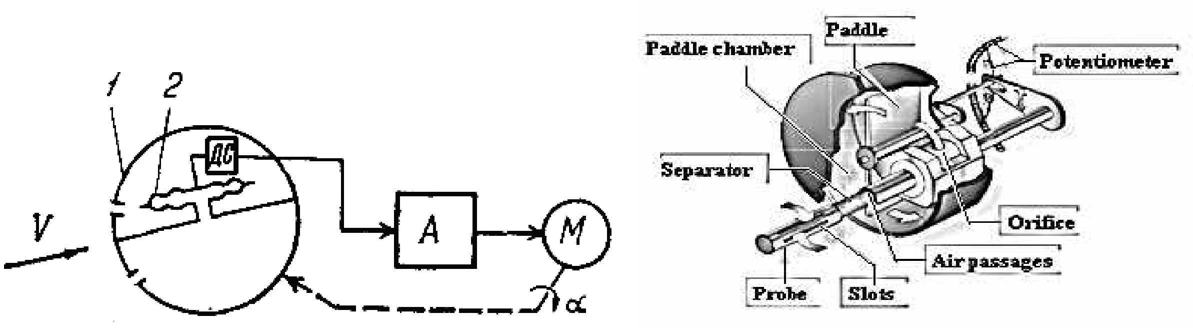
Fig. 6.3

For this, two pairs of interconnected weather vane are placed on the receiver, and the axis of one pair is perpendicular to the axis of the other pair. This arrangement of wind vane provides measurement of the angles of attack and slip angles of the weather vane. The brushes of potentiometers included in the bridge circuit are connected to the axes of the weather vane inside the receiver housing. With the middle position of the weather vanes, the brushes are in the middle position and the bridge is balanced. When deviating from the middle position, a signal appears on the measuring diagonal, which, after amplification, is processed by a **ДИД-0.5** type engine.

Thus, the output of the device is a signal proportional to the angle of attack or slip. Note that only potentiometers with brushes are located inside the pitot tube. The rest of the device elements (two other bridge arms, amplifier, test engine, etc.) are located in the aircraft cockpit.

At present, weather vane sensors placed on the fuselage of an aircraft have become widespread.

Compensation-type probes located in the same place are also used. The compensation probe (Fig. 6.4) is a cylinder divided into two paddle chambers and equipped with two groups of orifices. The cylinder axis is perpendicular to the angle measurement plane (α or β).



a b

Fig. 6.4

The cylinder is rotated in the incoming flow by the follower system until the pressures in the chambers are equalized. The latter corresponds to a symmetrical arrangement of orifices with respect to the flow, as shown in the figure. The differential pressure is sensed by the gauge box 2 and converted into an electrical signal by the signal transmitter. The angle of rotation of the shaft of motor M corresponds to the measured angle.

The disadvantage of fuselage-mounted sensors is that they measure local angles of attack and slip, which can be very different from true angles.

When using vane probes, the angle of attack is indicated in the simplest case using a galvanometric indicator. The pointer is connected to a potentiometer, the slider of which moves when the vane blades are turned. An example is angle of attack indicators of the **УУА** type, on the scales of which the zones of warning areas α and the zone of unacceptable values α are highlighted. There is also a system for signaling the approach to the critical angle α_{cr} by flashing a light bulb with a frequency of 2 – 4 Hz, taking into account the rate of change α.

6.3. Air Signal Systems

Most aerometric instruments do not have an electrical outlet, i.e. do not give an electrical signal. Therefore, in addition to them, devices for issuing data in the form of electrical signals to various consumers are used. A number of aircraft have a significant number of such devices (altitude correctors, altitude sensors, airspeed sensors, etc.). This leads to the cumbersomeness of both the complex of flight and navigation devices and communication channels with consumers. In order to reduce the weight of the complex, in such conditions it is especially important to achieve the minimum dimensions of individual devices. This is usually in conflict with requirements for improved accuracy.

These reasons led to the widespread introduction of unified systems for calculating the main aerodynamic parameters of the flight and issuing signals about them to all consumers. Such aerometric systems are called air signal systems. They are an important component of modern flight and navigation systems.

From the theory of aerometric instruments, it follows that to calculate the main aerodynamic parameters (M, V_t, V_t, H) it is enough to measure only three input quantities: static pressure P_{st}, dynamic pressure P_d and temperature of stalled air T_b. The volume of calculations of the true airspeed V can be reduced taking into account the dependence $V = f\,(M, T_b)$. These circumstances form the basis for the construction of air signal systems.

All air signal systems calculate M, V, altitude H and / or relative barometric altitude h_0. In some systems, V_i and ambient air temperature T also are calculated.

The output signals of the air signal system are fed to various on-board systems (automatic control system, navigation systems, etc.), as well as to electromechanical indicators.

Some air signal systems also give to consumers (automatic control system) signals of deviations from the set (constant) or programmed values of H, M and V_i. To increase the reliability of the flight-navigation complex and flight safety, the units for calculating such signals, called corrector-masters, operate independently of the main computing system of air signals.

The peculiarities of the circuit and arrangement of air signal systems are largely determined by the type of computing devices. Domestic air signal systems use analog electromechanical or electronic computing devices.

The central unit of the system is a block-calculator of speed, Mach number and altitude, in which the pressure sensors P_{st}, P_d and temperature T_t are also located.

The height indicator with command index H_{const} shows the values of H and H_{const}.

Signals H_{const} and M_{const} are issued by the radio guidance system.

Topic 7. GYROSCOPIC MEASURING INSTRUMENTS

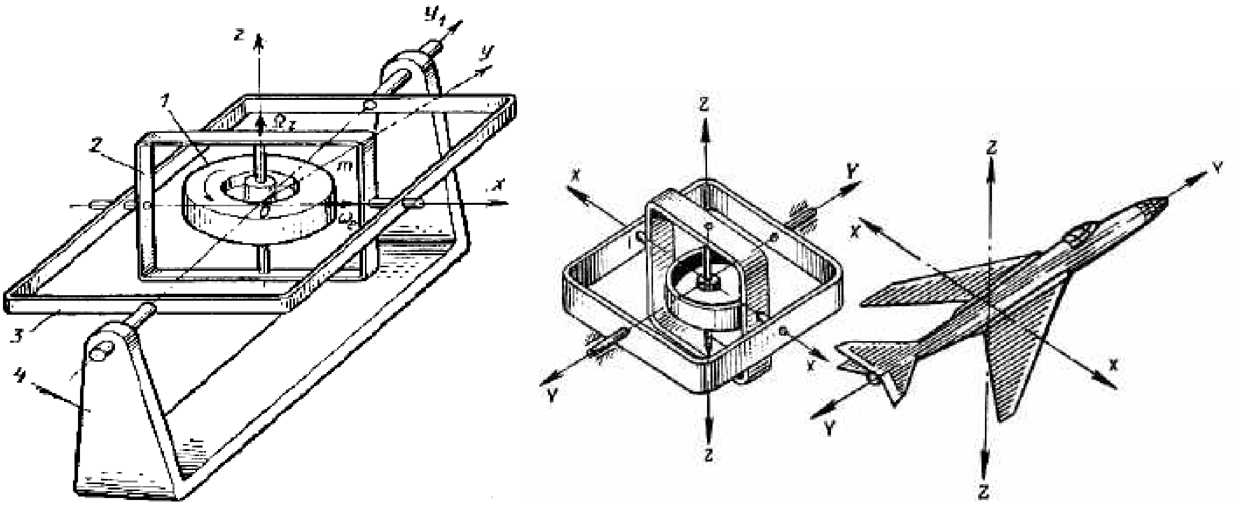
7.1. Gyroscopic Measuring Devices: Purpose, Classification

Everyone knows the remarkable properties of a fast spinning top. An immobile top invariably falls on its side under the influence of its own weight. The fast spinning top balances calmly on the tip of its axis.

The first serious use of the remarkable properties of the top was an experiment staged by the physicist L. Foucault in 1852. Foucault demonstrated a "gyroscope" device built by him, the main part of which was a rapidly rotating rotor (flywheel). This device made it possible for the first time to detect the Earth's daily rotation by direct laboratory observation.

The term "gyroscope" is derived from the Greek words "gyros" – rotation and "scopo" – watching.

A gyroscope is a body rotating around an axis of symmetry with a high angular velocity, one of the points of which (point 0) is stationary (Fig. 7.1).



a b

Fig. 7.1

There are 1 – gyroscope rotor; 2 – inner frame of the gimbal; 3 – outer frame of the gimbal; 4 – base; ω_e – vector of the transportation angular velocity; Ω_z – own angular velocity of the gyroscope rotor rotation in Fig. 7.1.

The *z*-axis of symmetry of rotor 1 is called the figure-axis or the gyroscope rotor axis.

In most gyroscopic devices, to ensure freedom of rotation of the gyroscope rotor around a fixed point *O*, gimbals are used, which consist of two frames 2 and 3. The rotor 1 of the gyroscope with a high angular velocity Ω_z rotates around the O_z axis relative to the inner frame 2, which can rotate around the axis O_x relative to the frame 3, and the latter – around the axis O_{y1} relative to the fixed support 4. The O_y axis is perpendicular to the O_z and O_x axes.

All three axes O_x, O_y, O_z intersect at one point O, which is a fixed point. This point is also called the ***point of suspension or support***.

The axis of rotation of gyroscope rotor (O_z) is called ***the main axis of gyroscope***.

The gimbal provides the gyroscope rotor with freedom of rotation about three axes (O_z, O_x and O_{y1}). Therefore, a gyroscope mounted on a gimbal is called a ***three-degree-of-freedom gyroscope***.

If the center of mass of the gyroscope coincides with the point of intersection of the axes of gimbal, then such a gyroscope is called ***astatic gyroscope***.

An ideal gyroscope, which is not affected by any external moments, is called ***free gyroscope***.

The implementation of such a gyroscope requires, firstly, the exact alignment of the center of mass of gyroscope with the point of intersection of the rotor and suspension axes in order to exclude the influence of the moments of gravitational forces and inertial forces on the gyroscope. Secondly, there must be no friction in the suspension supports. It is clear that any real gyroscope can be a model of free gyroscope only with a certain degree of accuracy.

The main characteristic of the gyroscope is the angular momentum ($\overline{H}$). It is numerically equal to the product of the axial moment of inertia I and the angular speed of the rotor's own rotation Ω:

$$\overline{H} = I \cdot \overline{\Omega}. \tag{7.1}$$

Strictly speaking, the gyroscope itself is its rotor. However, usually the whole device shown in Fig. 7.1, that is, a rotor with suspension frames is called a gyroscope. The internal frame is usually absent in an explicit form – its function is performed by the casing (housing) of the rotor. The casing with the rotor enclosed in it forms a ***gyro unit***.

The behavior of a three-degree-of-freedom gyroscope when exposed to external moments differs significantly from the motion of a non-rotating rigid body. At $\Omega = 0$, rotation of the base 4 (see Fig. 7.1) around the axis of the outer frame will cause a corresponding rotation of frames with the rotor due to friction in the bearings of the axis of outer frame. Applying a moment to one of the frames will cause accelerated rotation around the axis of this frame. Impact on the frame will also rotate it.

If the rotor rotates at a sufficiently high speed Ω then the rotation of the base, as well as the impact, will not cause a noticeable change in the position of the frames and the axis of rotation of the rotor.

The application of the moment ***M*** to one of the frames will cause the gyroscope to rotate the axis of the other frame. This gyroscope specific motion is called ***precession***.

The precession rule is formulated as follows: under the action of the external moment M, the gyroscope moves (precesses) with an angular velocity ω, trying to combine the axis of its rotation (vector Ω) along the shortest distance with the vector of the external moment M (Fig. 7.2).

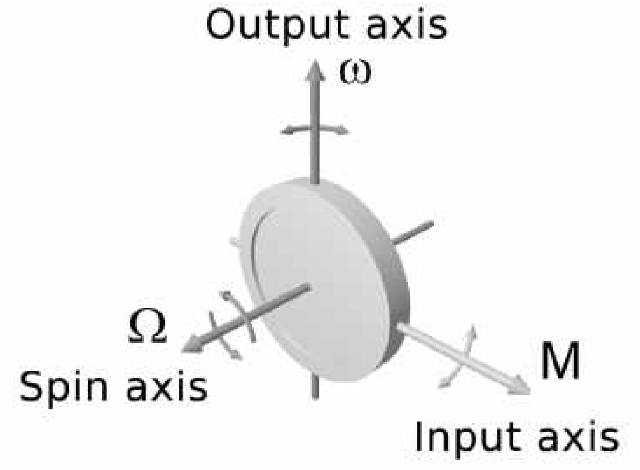


Fig. 7.2

The precession appears immediately after the application of external torque and disappears after its removal.

The "unusual" direction of the precession is explained by the rotation of its own rotor. If the external moment is equal to zero, then the angular velocity of the precession is also equal to zero. Consequently, the rotor axis of such a gyroscope remains unchanged in absolute space. This property is often referred to as ***the basic property of a gyroscope***. However, a real gyroscope cannot be free (for example, due to friction in the suspension axes and unbalance).

Three-degree-of-freedom gyroscopes are used in aircraft gyroscopic devices to measure the angular position of an object (vertical gyro and gyrocompass).

Two-degree-of-freedom gyroscopes are used in high-speed gyroscopes for measuring angular velocities, as well as in power gyroscopic stabilization systems as integrating gyroscopes.

Devices and systems, the main part of which is a gyroscope, are called ***gyroscopic devices and systems***.

With the help of gyroscopic systems, the direction of the meridian and true vertical is determined, angular velocities and accelerations are measured.

According to the principle of operation, gyroscopic devices and systems are divided into the following main groups:

- gyroscopes with two and three degrees of freedom;
- course gyroscopic systems;
- gyroscopic stabilizers;
- gyroscopic sensors of the true vertical direction;
- inertial systems.

In gyroscopes for aircraft gyroscopic systems, rotors are usually driven by induction motors at a speed of 22.000 – 30.000 rpm.

7.2. Angular Velocity Sensors

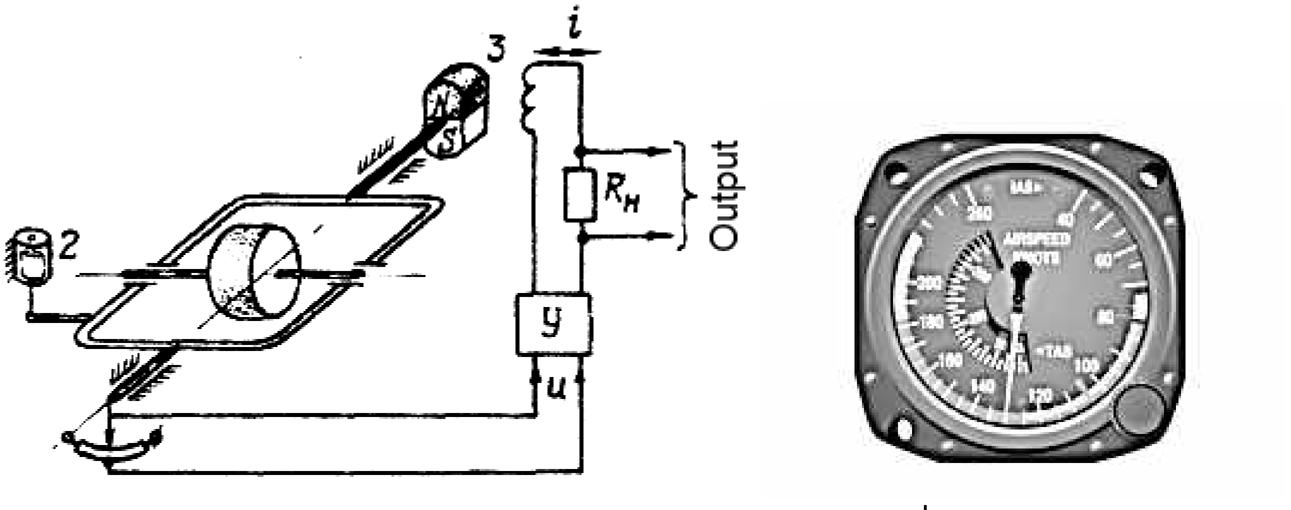
For manual and automatic control of aircraft, in addition to determining their angular position, it is necessary to measure the angular speeds of rotation relative to the axes of aircraft. For this purpose, gyroscopic angular velocity meters (angular velocity sensors) are widely used.

Angular velocity sensors are used as sensing elements of autopilots, strapdown attitude control systems and inertial navigation systems of airplanes and helicopters.

The heart of any yaw rate sensor is a two-degree-of-freedom gyroscope (speed gyroscope). The rotations of the gyroscope frame are converted by a potentiometric angle sensor or an inductive proximity sensor into an electrical signal proportional to the measured angular velocity.

The angular velocity sensor has two degrees of freedom, one of which is the degree of freedom of the frame and has an elastic limitation in the form of a mechanical spring. In addition to mechanical springs, so-called electrical springs are used. Fig. 7.3 shows angular velocity sensors with an electric spring, which is a system of potentiometer 1, amplifier **У** and magnetoelectric torque sensor 3.

Since the voltage *u* taken off the potentiometer is proportional to the value of the angle β of the frame rotation, the current *i* of the winding, and hence the moment M_{PR} of the sensor 3 will also be proportional to the angle β.



a b

Fig. 7.3

Voltage $u_H = i\,R_H$, which is proportional to ω_c, is supplied to consumers. To damp the oscillations of the frame in the transient process of setting the angle β, a pneumatic damper 2 is used. The damper consists of a piston connected to the frame and a cylinder fixed to the device body. The cylinder has calibrated holes through which air flows in or out when the piston moves.

As a result, a resistance force F_d and a damping moment M_d are created, which is proportional to the speed of rotation of the frame. This moment is added to the M_{PR}.

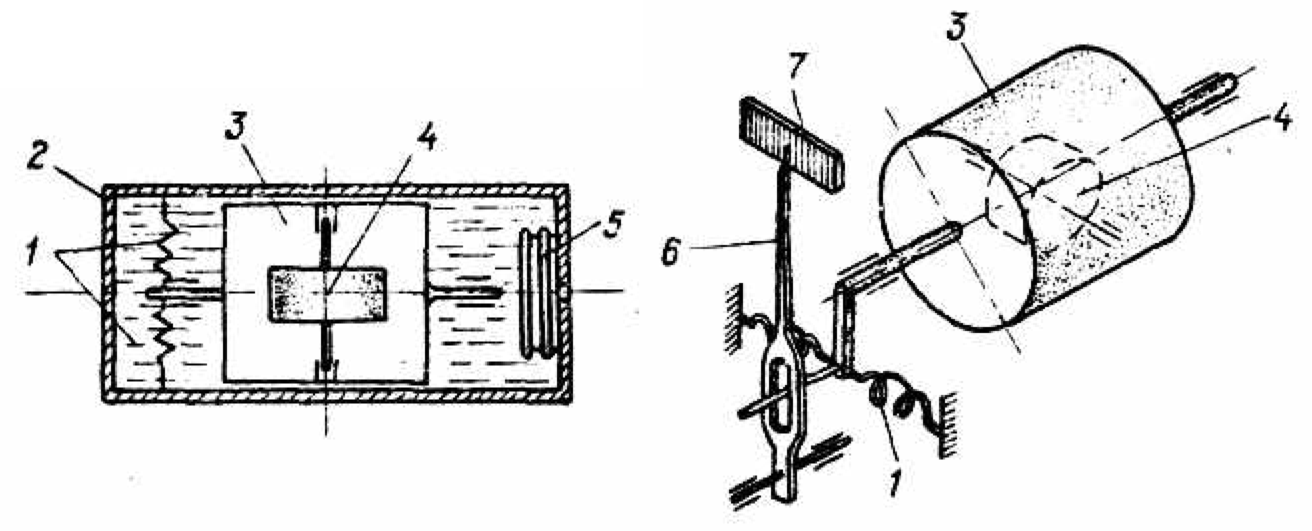
The moments of friction and imbalance relative to the axis of the frame cause errors in measuring the angular velocity.

A significant reduction in friction moments is achieved in high-speed float gyroscopes. An example of such a gyroscope is a high-speed gyroscope of an angular velocity sensor of the **ДУС-М** type, used in autopilots and aircraft vibration dampers.

The gyroscope circuit (without a signal pickup device) is shown in Fig. 7.4, a. The sealed case 2 of the device is filled with a special liquid with a high density (1.5 – 2 g/cm^3). The gyro unit is a float 3, which acts as a frame, with a rotor 4 enclosed inside.

The interaction of the float 3 with the brush 6 of the output potentiometer 7 and with the opposing springs 1 are explained in Fig. 7.4, b. If the weight of the float is equal to the weight of the displaced liquid, the bearings of the float axes are unloaded from the forces of normal pressure and the frictional moment becomes insignificant. The fluid also provides damping of the gyro unit vibrations.

The sensitivity of **ДУС**-type float gyroscopes is several hundredths of *a* deg / s.



a b

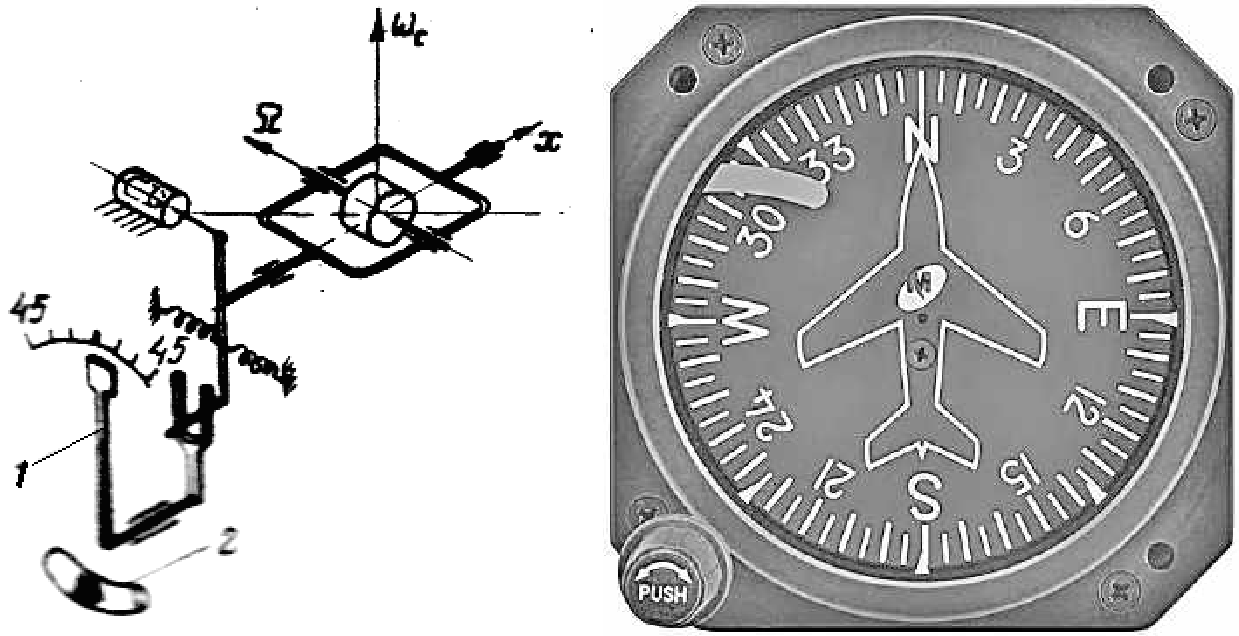
Fig. 7.4

There are 1 – mechanical opposing springs, 2 – sealed body, 3 – float, 4 – gyroscope rotor, 5 – bellow (to compensate for changes in fluid temperature), 6 – potentiometer brush, 7 – potentiometer in Fig. 7.4.

7.3. Direction Indicators

Of all the angular velocities of the aircraft during manual piloting, only the angular velocity ω_C of the turn (bank) is directly controlled. Devices for visual assessment of this speed are called direction indicators.

The kinematic diagram of an electric direction indicator of the **ЭУП-53** type is shown in Fig. 7.5.



a b

Fig. 7.5

The axis of the frame is parallel to the longitudinal *x*-axis of the aircraft. If the bend angle is γ, then the frame will also rotate through the angle γ under the action of the springs. In this case, the angle of rotation of frame β (indication of arrow 1), due to the speed of the turn (bend) ω_C, depends on both ω_C and the angle of bank γ. Consequently, the readings of the device are not uniquely determined by the angular velocity of the bank. Therefore, the electric direction indicator serves only for an approximate estimate of the value of the bank angular speed.

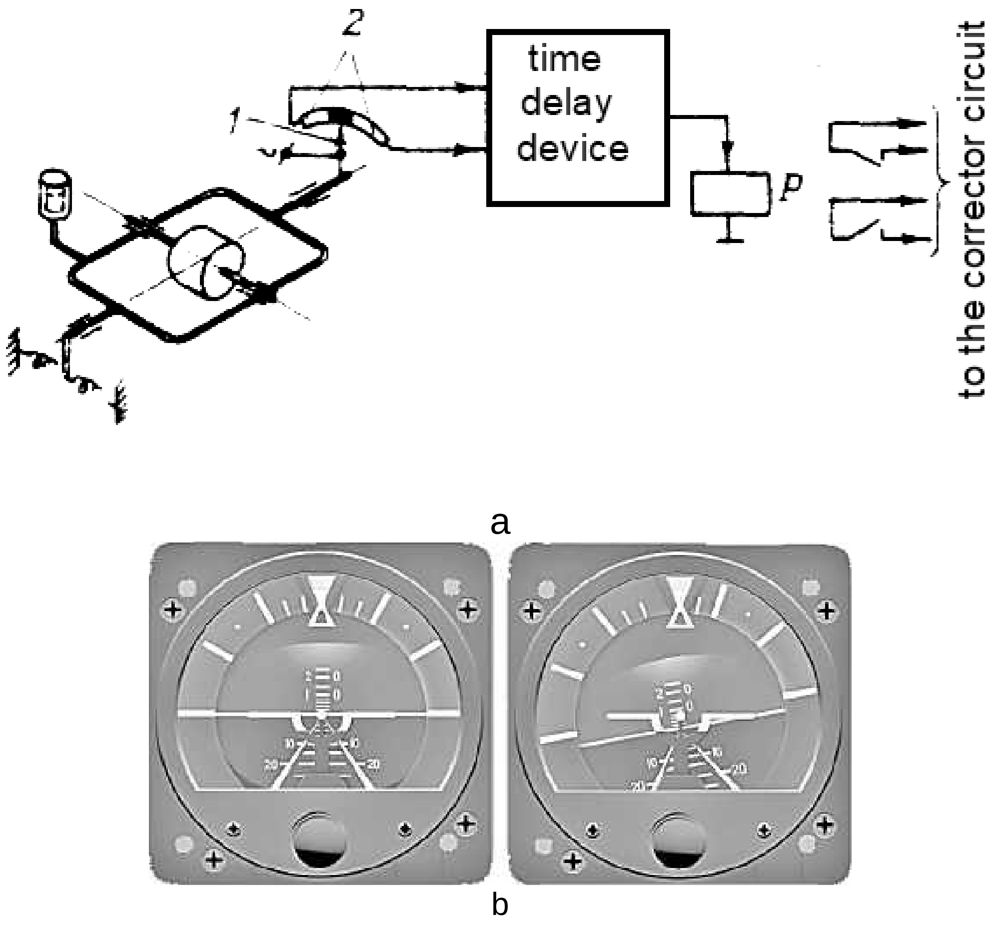
Note that the **ЭУП-53** scale is not graduated in units of angular velocity, but in the values of the bank angles. In this case, it is assumed that for a certain average flight speed *V*, the angle β_{set} will be a single-valued function of the bank γ. For **ЭУП-53** this speed value is taken to be 500 km/h. The value of the calibration speed is usually chosen approximately equal to the speed of the aircraft during the pre-landing maneuvers. Therefore, with a correct bank at such a flight speed, an electric direction indicator can also serve for rough control of the artificial horizon. There is a slip indicator 2 on the instrument face.

The direction indicator, which is similar to the electric direction indicator, is a part of the **ДА-200** type artificial horizon backup. It combines a **ВАР-200** variometer and a slip indicator in one housing. Comparison of the readings of these three instruments makes it possible to determine the deviations of the flight mode from the rectilinear horizontal.

7.4. Gyroscopic Correction Switches

A conventional high-speed gyroscope is used in gyroscopic correction switches of the **ВК-53РБ** type. Correction gyro switches are intended for commutation of lateral correction circuits in artificial horizons, vertical gyroscopes and course gyroscopes, depending on the value of the angular speed ω_B of the bank.

The correction switch is a gyroscopic device (Fig. 7.6), similar to the direction indicator discussed above. The only difference is that the gyroscope moves here not the arrow, but the contact of the lamellar device. This contact closes the override cutout circuit as soon as the swing angle reaches the set value.



a

b

Fig. 7.6

In the body of the correction switch, in addition to the high-speed gyroscope, there is an electromechanical time delay device. The aircraft can perform rapid oscillatory yaw movements in flight around an axis normal to the plane of wings. In this case, the override switch will respond not only to the angular rate of the superelevation, but additionally to the angular rate of yaw. This can lead to unreasonable disabling of the correction of gyroscopic devices. In order to avoid this phenomenon, the signal from the potentiometer is fed to the time delay device. In this case, the time delay device after 5 – 15 s closes the power supply circuit of the relay group *P*. Normally closed and normally open relay contacts are included in the circuits of lateral correction of artificial horizons and vertical gyro, as well as in the circuit of azimuthal and horizontal corrections of heading systems.

Part II. FLIGHT-NAVIGATION DEVICES AND SYSTEMS

Topic 8. GEOSTEERING INFORMATION

8.1. Geosteering Information: Characteristics of Geophysical Fields and Parameters of the Earth's Motion

Geophysical parameters are certain data on the properties of the Earth as a celestial body.

These include: the geometrical characteristics of the shape and surface of the Earth, the parameters of the gravitational and magnetic fields, the peculiarities of the atmosphere with rise to altitude, the parameters of the radiation belts.

For correct orientation during flight over the Earth's surface, it is necessary to know the shape of the Earth, its geometric dimensions, have detailed maps, and also know the characteristics of the magnetic field surrounding the globe.

The Earth has the shape of an ellipsoid compressed from the poles. The length of the major axis of the ellipsoid (equatorial diameter) is taken to be 12,756,490 m, and the length of the minor axis (the distance between the poles) is equal to 12,713,726 m.

The ratio of the difference between the semi-axes of the ellipsoid to the value of its semi-major axis is very small (0.0033876), and therefore in aeronautical navigation, the shape of the Earth is usually taken as a regular sphere, the volume of which is equal to the volume of the Earth's ellipsoid.

The radius of this ball is 6371 km. In this case, the errors in determining the distances on the Earth's surface will not exceed 0.5 %, and the errors in determining the directions will not exceed 12 arc minutes.

The length of the Earth's meridian is 40,000 km.

The arc length of 1º is 111.2 km.

The arc length of any great circle corresponding to 1' is taken with a high degree of accuracy equal to 1852 m (1 nautical mile).

The arc length of 1'' is 30.9 m.

The Earth turns by 360º per day, while rotating around its axis from west to east. The angular velocity of the Earth's rotation is 15.04107 º/hour.

The Earth's orbit is in the shape of an ellipse, with the Sun in focus. A complete revolution is completed in 365.25036 days. The axis of the Earth's daily rotation is inclined to the plane of the orbit of the annual rotation at an angle of 66º 33 '.

The position of any point on the earth's surface (for example, point A in Fig. 8.1) is determined by its geographical coordinates – longitude and latitude.

The longitude of the place λ is the angle between the plane of the true meridian of the place and the plane of the initial (zero) meridian.

Longitude is measured in angular degrees.

The Greenwich meridian is taken as the initial meridian, and to the west of it the longitude is considered to be west, and to the east is east, ranging from 0 to 180°.

The latitude of a place φ is the angle in degrees between the plane of the equator and the vertical passing through the given place.

North of the equator latitude is considered north, and south of the equator latitude is considered south in the range from 0 to 90°.

The shortest route of flight should be in an arc of a great circle, called an orthodrome (Fig. 8.2).

Flying along the orthodrome is practically hampered by the fact that the plane must continuously change course (ψ_1, ψ_2, ψ_3, etc.), since the orthodrome crosses the meridians at different angles.

If an airplane flies with a constant angle ψ in relation to the meridians, then it moves along a curved line called a loxodrome (see Fig. 8.2).

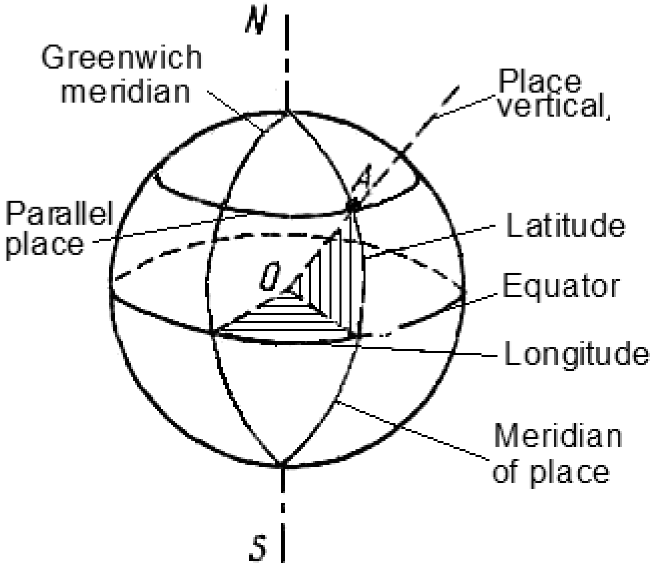


Fig. 8.1

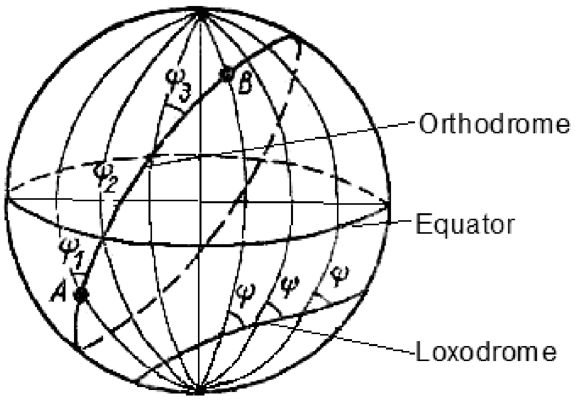


Fig. 8.2

Long-distance flight along the orthodromy shortens the distance by hundreds and even thousands of kilometers. For long-distance flights, the flight path is divided into a number of sections, within which the flight is performed along the orthodromies of these sections.

There is a magnetic field around the globe. The earth's magnetic poles do not exactly match the geographic ones. The magnetic north pole is located in Canada, northwest of Hudson Bay (78.5° north latitude and 69° west longitude), and the south pole is located in Antarctica, in the northern part of the Victoria mainland (78.5° south latitude and 111° east longitude). The names of the north and south magnetic poles, as well as similar names of the poles of the magnetic needle, are conditional.

The North magnetic pole_of the Earth is called the pole located in the northern hemisphere, but attracting the north end of the magnetic needle.

Almost everywhere (with the exception of the so-called magnetic equator), the Earth's magnetic field is inclined to the horizon. The arrow, freely rotating around the center of mass, would occupy a vertical position at the magnetic pole, and horizontal at the equator. For all other latitudes, the arrow is located at a certain angle Θ (for Kharkiv, this angle is approximately 53°), called the ***inclination angle or inclination*** (Fig. 8.3).

The total strength of the Earth's magnetic field T can be decomposed into a vertical component Z and a horizontal component H. Under the action of the horizontal component of strength H, the movable magnetic needle is set to the North-South direction, and the vertical component Z tilts the arrow down. The horizontal component of the earth's field strength is

$$H = T \cos \Theta, \tag{8.1}$$

and the vertical component is

$$Z = T \sin \Theta. \tag{8.2}$$

In the northern hemisphere, under the action of the vertical component of the field, the magnetic compass needle, mounted on the tip, tends to tilt with the northern end down. To reduce this slope, the southern end of the arrow in compasses is made heavier. In the southern hemisphere, the north end of the magnetic needle should be weighted down.

The horizontal component H does not coincide with the direction of the geographic meridian of the given place. This discrepancy is small.

The direction of the horizontal component of the Earth's magnetic field strength H, along which the magnetic needle is established, is called the magnetic meridian of the given place.

The angle Δ_M between the magnetic meridian N_M and the geographic (true) meridian N_I is called the ***declination angle***, or simply ***declination*** (Fig. 8.4).

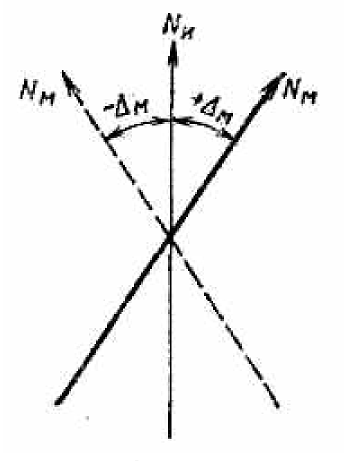


Fig. 8.3

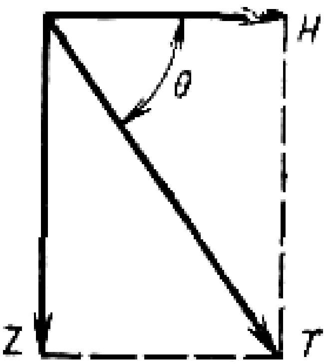


Fig. 8.4

Declination is considered positive (“+” sign) if the arrow deviates from the north end east of the geographic meridian, and negative (“-” sign) if the arrow deviates to the west. For different points of the globe, declinations are different in magnitude and sign. The declination is not constant, it changes over time, and the magnitude of the change usually does not exceed 10' per year. The declination values are determined from special maps of magnetic declination, on which lines are plotted connecting places of equal magnetic declination, these lines are called isogons. The same maps indicate the magnitude of annual changes. The time for which the isogons are plotted on the map is called the epoch of the magnetic map.

The total strength of the Earth's magnetic field is different for different latitudes. At the equator, it is equal to ≈ 0.39 Oe (0.0049 A/m), in middle latitudes it is about 0.4 – 0.5 Oe (0.0050 – 0.0062 A/m), and at the poles it reaches a maximum of ≈ 0.75 Oe (0.0094 A/m).

The magnetic field strength decreases with distance from the Earth's surface, roughly inversely proportional to the cube of the distance from the center of the Earth to the measurement point.

8.2. Methods of Air Navigation

Any flight of an aircraft is carried out for a specific purpose. At the same time, ways of solving the flight problem are being developed, a route, speed, and flight profile for the implementation of air navigation are selected.

Aeronavigation (air navigation) is a complex of works performed by the crew and aircraft systems and ensuring the achievement of the assigned flight task.

Aeronavigation includes:

– Flying (pilotage, piloting) an aircraft;

– Navigation.

Piloting is maintaining a given position of an aircraft in space and pre-calculated flight parameters for performing a flight task.

Navigation is a complex of works that ensures the achievement of a given point on the earth's surface or a certain object at a specified time and at a certain height.

Navigation includes:

– determination of flight goals and the program of movement of the aircraft in space and time, taking into account possible conditions and circumstances;

– use of methods for obtaining information about the space-time position of the aircraft and the parameters of its flight mode;

– the use of technical means for collecting, processing and displaying information about the flight;

– using the information received for orientation and flight control.

The main task of navigation is to determine the current coordinates of the location of the moving object. The size of the object itself in navigation is usually neglected and the aircraft is considered a material point.

Let us first briefly explain the basic concepts used in navigation.

The ***flight path*** is the spatial curve of the movement of the aircraft.

The ***track line*** is the projection of the flight path onto the surface of the Earth's ellipsoid.

A ***flight path or line of a given path*** is a projection of a given (programmed) trajectory onto the surface of the Earth's ellipsoid.

The location of the aircraft is the projection of the spatial location of the aircraft onto the Earth's surface.

The velocity vector of the relative motion, directed tangentially to the flight path, is called **the *trajectory (ground) velocity*** W_t.

The horizontal projection of the relative motion velocity vector onto the surface of the earth's ellipsoid is called the ***ground speed*** W.

The coordinates of a moving object can be determined in various means, namely:

a) by direct calculation of coordinates from geometric relationships, when the initial information is the ranges, azimuths or course angles to points on the earth's surface with known coordinates or the heights and azimuths of celestial bodies observed from the object;

b) by calculating the line of motion (trajectory) according to the data on the velocity vector and coordinates of the point of origin of movement;

c) by comparing the visible terrain with the map.

The first mean for determining the coordinates of an aircraft is based on methods of short-range and long-range radio navigation and methods for determining the coordinates of an aircraft by sighting landmarks (points) with the help of airborne radar, thermal or optical stations, the coordinates of which are known. Bridges, river mouths, individual islands and rocks on the sea, etc. can act as such points.

The second mean for determining the coordinates of the aircraft can be carried out by calculating the flight trajectory carried out using an on-board computer according to the indications of directional instruments, Doppler ground speed meters, airspeed meters, as well as according to the indications of inertial navigation systems. This method is also called flight dead reckoning.

The third mean for determining the coordinates of the aircraft is most often implemented by visual observations of the crew outside the cockpit space or screens of observation onboard stations.

Based on these means, the following navigation methods can be distinguished:

– radio engineering methods of short-range and long-range navigation;

– determination of the coordinates of the aircraft by sighting landmarks, the coordinates of which are known, using onboard radar, thermal and optical stations;

– dead reckoning methods:

  a) air;
  b) Doppler;
  d) inertial;

– overview and comparative method;

– correlation-extreme method;

– astronomical.

Using the known coordinates of ground objects such as bridges, river mouths, individual large factories, TV towers, individual islands in the sea, etc., the coordinates of the aircraft are determined by sighting these objects with the help of onboard radar, thermal and optical stations.

Dead reckoning methods are based on calculating the flight path in onboard computers according to the readings of directional devices, Doppler ground speed meters, airspeed meters and indicators of inertial navigation systems. Inertial navigation systems determine the vector of the ground speed of an object using calculations that are performed on signals from accelerometers (devices that measure the acceleration of an aircraft along the axes of a coordinate system associated with the aircraft).

The survey-comparative method is based on comparing the terrain visible from the cockpit with a map. This method is most often implemented by visual observations of the terrain or by screens of onboard observation stations.

Recently, the correlation-extremal navigation method has been widely used. However, this method cannot independently determine the coordinates of the object. It is used to correct other navigation systems, most often dead reckoning systems.

Distinguish between autonomous and non-autonomous navigation methods.

Autonomous navigation methods are methods and their corresponding navigation systems that do not require the use of off-board technical means.

Radiotechnical methods are non-autonomous, because require radio communication with ground stations. All other navigation methods are standalone.

Topic 9. FLIGHT AND NAVIGATION ELEMENTS

9.1. Flight and Navigation Parameters of Aircraft Movement

The flight and navigation parameters of the aircraft flight include:

1) flight and navigation information;
2) navigation coordinate systems;
3) methods for converting information from one coordinate system to another;
4) methods for measuring time in different regions of the Earth;
5) features of displaying geophysical information on maps.

Flight and navigation information includes:

– parameters of movement of the center of mass of object:
  a) coordinates of the aircraft relative to different coordinate systems,
  b) ground speed W, indicated airspeed V_{is} and true airspeed V;
  c) vertical velocity;
  d) lateral velocity V_Z;
  e) linear accelerations a_x, a_y, a_z;
– parameters that determine motion relative to the center of mass:
  a) angles of pitch ϑ, bank γ, yaw ψ, attack α and slip β;
  b) angular rates of bank ω_X, yaw ω_Y and pitch ω_Z and their derivatives;
– environmental parameters:
  a) outside temperature T_O;
  b) atmospheric pressure P_O;
  c) density ρ;
– parameters that determine the movement of the aircraft relative to other objects;
– parameters of transition from one coordinate system to another:
  a) direction cosines;
  b) quaternions;
  c) differential equations of Rodrigues-Hamilton, Kelly-Klein.

Movement Parameters of the Object's Center of Mass

We considered that the parameters of motion of the center of mass of an object include:

1) coordinates of the aircraft relative to different coordinate systems;
2) ground speed W, indicated airspeed V_{is} and true airspeed V;
3) vertical velocity;
4) lateral velocity V_Z;
5) linear accelerations a_x, a_y, a_z.

The parameters that determine the motion of the center of mass of aircraft are measured relative to any coordinate system taken as the origin. The coordinates of the aircraft relative to different coordinate systems are positional flight and navigation elements. Such reference systems make it possible to determine the location of aircraft. For ease of use, the navigation coordinate systems are associated with any bodies, trajectories of bodies or fields of various physical nature.

The main coordinates include latitude φ, longitude λ and the length of radius vector **R**, flight altitude H (true; absolute, relative).

The airspeed of an aircraft is the speed of a relatively undisturbed environment.

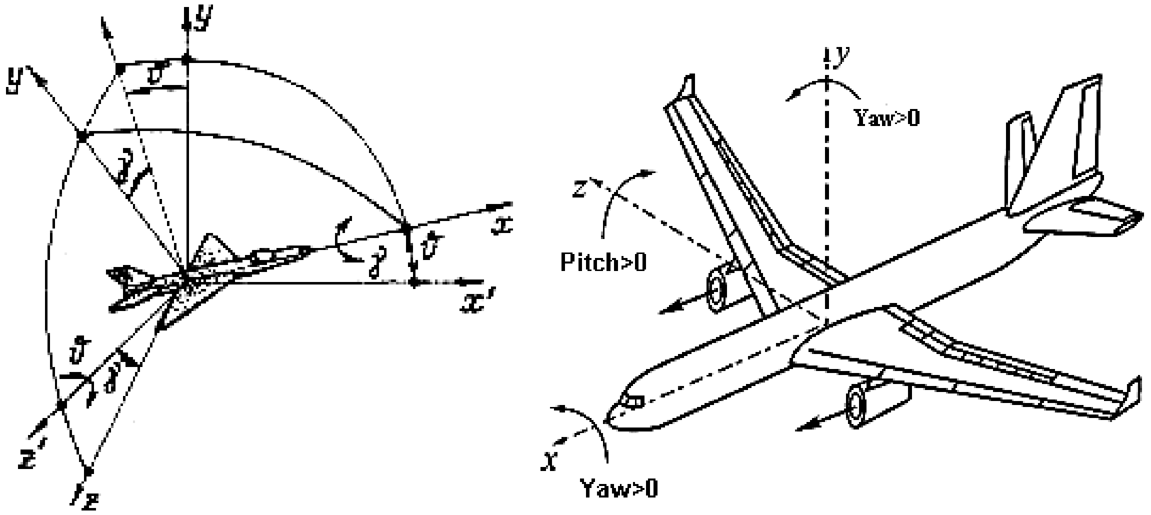
Earth speed is the speed of an aircraft relative to any of the Earth's coordinate systems.

Ground speed is the horizontal component of the speed of movement of the aircraft relative to the Earth's surface.

Parameters that Determine Motion about the Center of Mass

The ***pitch angle*** ϑ is the angle between the horizontal plane *x`Oy`* and the longitudinal axis of the aircraft *Ox* (Fig. 9.1).

The pitch angle varies from - 90 ° to + 90 °.



a b

Fig. 9.1

The ***bank angle*** γ is the angle between the vertical plane passing through the longitudinal axis *Ox* of the aircraft and the plane of symmetry of the aircraft *xOy* (see Fig. 9.1).

The bank angle varies from - 180 ° to + 180 °.

The ***yaw angle*** ψ is the angle between the north direction and the projection of the longitudinal axis *Ox* onto the horizontal plane.

The yaw angle of the aircraft varies from 0 to 360 °.

The ***angle of attack*** α is the angle between the longitudinal axis *Ox* of the aircraft and the projection of the airspeed vector onto the plane of symmetry.

The ***slip angle*** β is the angle between the airspeed vector and the plane of symmetry *xOy*.

The angular velocities of the aircraft relative to the axes *Ox*, *Oy*, *Oz* are, respectively: the angular velocity of the bank ω_X; angular pitch velocity ω_z; angular yaw rate ω_y.

***Environment parameters*:**

1) outdoor temperature T_O;
2) atmospheric pressure P_O;
3) density ρ.

Environmental parameters are of great practical importance for determining navigation parameters, especially the speed and altitude of an aircraft.

Parameters that Determine the Movement of Aircraft Relative to Other Objects

The parameters that determine the movement of the aircraft relative to other objects are important when performing interplane navigation tasks, for example, when flying in a group of aircraft.

Parameters for Transition from One Coordinate System to Another

The transition from one coordinate system to another is carried out using tables of trigonometric functions. These tables are called direction cosine matrices. It should be noted that at some values of the pitch angles (for example, at $\upsilon = \pm 90$ °), the transition matrices exhibit features associated with the need to divide by zero. In this case, the transitions are carried out using the differential equations of Rodrigues-Hamilton or Kelly-Klein. These equations are quite complex and are considered when solving special problems.

9.2. Navigation Coordinate Systems

To determine the parameters of motion of the aircraft relative to the center of mass, we introduce the following coordinate systems.

1. ***The associated coordinate system*** *Oxyz* is a moving coordinate system, the axes of which are the longitudinal axis *Ox*, the normal axis *Oy* and the transverse axis Oz, fixed relative to the aircraft.

The direction of the longitudinal axis can be selected both along the base axis of the aircraft, wing or fuselage, and along the main axis of inertia.

The position of the associated coordinate system relative to the normal terrestrial coordinate system is determined by three Euler angles: course $\boldsymbol{\psi}$, bank $\boldsymbol{\gamma}$ and pitch ϑ (see Fig. 9.1).

2. ***Local Coordinate System***. The origin of coordinates *O* is located at the observation point of the aircraft (Fig. 9.2). The *Oz* axis is directed vertically upward, the *Oy* axis is directed to the North Pole. The *Ox* axis forms the right triplet of vectors. The *xOy* plane is located in the horizon plane. *A* is the azimuth, *R* is the radius vector and h is the angle between the horizontal plane and point *M* (object location).

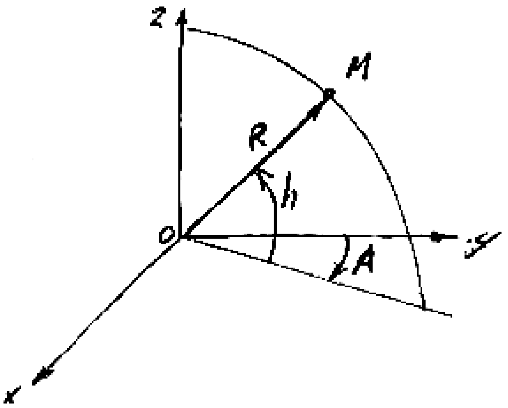


Fig. 9.2

3. For ease of use, navigation coordinate systems are associated with any bodies (for example, with a point on the Earth's surface, or the center of the Earth, or stars) or with the trajectories of bodies. All these coordinate systems can be divided into inertial and non-inertial. The most commonly used are inertial coordinate systems.

An ***inertial coordinate system*** is called a right-angled coordinate system. Its origin O_I is placed at some point in space, or moves at a constant speed, and the direction of the axes relative to the stars is invariable.

When solving problems of air navigation of an aircraft, the terrestrial coordinate systems associated with the earth's surface or with the center of the earth are often taken as inertial reference systems. In this case, the curvature of the surface and the daily rotation of the Earth are neglected.

4. To determine the coordinates of the location of points on the surface of the Earth, a geographic coordinate system is used.

The coordinates of the location of the object are determined by latitude φ and longitude λ.

A ***geographic coordinate system*** is a system, the origin of which *O* is aligned with the center of the Earth, the *Ox* and *Oz* axes are located in the equatorial plane, the *Oy* axis is directed from the center of the Earth towards the North Pole (Fig. 9.3). 1 is the equatorial plane and 2 is the the zero (Greenwich) meridian.

The coordinates of the location of the object are determined by latitude φ and longitude λ.

Geographic latitude φ is the angle between the equatorial plane and the plumb line lowered from point M (object) to the surface of the Earth (see Fig. 9.3).

Latitude is measured from the equator to the poles (from 0º to 90º) and is called, respectively, north or south.

Geographic longitude λ is the dihedral angle between the planes formed by the Greenwich meridian and the meridian that passes through the point where the object is located (point M in the figure).

Longitude is measured on both sides of the Greenwich meridian from 0º to 180º and is referred to as west and east, respectively.

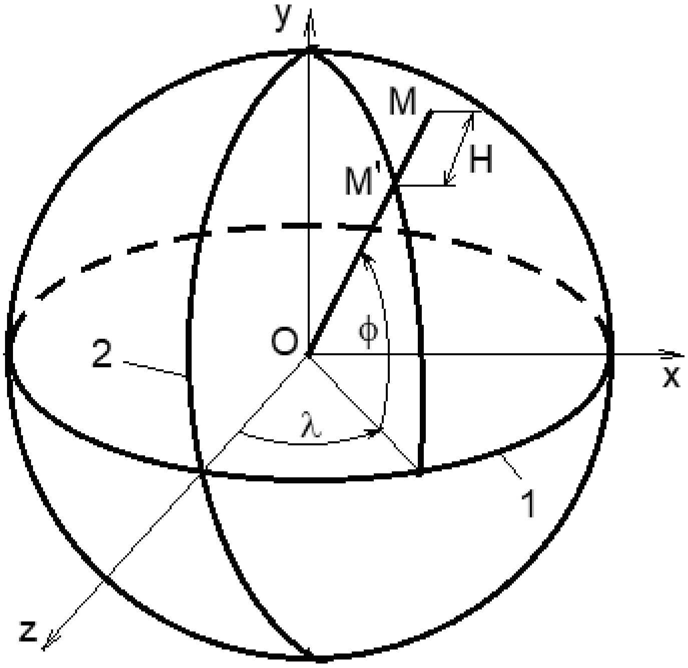


Fig. 9.3

5. The ***normal terrestrial coordinate system*** $O_oX_gY_gZ_g$ is a coordinate system fixed with respect to the Earth, the O_oY_g axis of which is directed upward along the local vertical (opposite to the direction of gravity at the O_o point), the O_oX_g axis is considered parallel to the tangent to the meridian passing through the O_o point and directed to the North.

6. The ***normal coordinate system*** $OX_gY_gZ_g$ is a moving coordinate system, the origin of which *O* is placed on the aircraft, usually at the center of

mass. The OY_g axis is directed up the local vertical, the OX_g axis is directed to the North, and the OZ_g axis to the East. The X_gOZ_g plane is parallel to the local horizon plane.

7. The ***starting coordinate system*** $O_OX_CY_CZ_C$ is a coordinate system in which the origin of coordinates O_O coincides with the characteristic point of the aircraft at the initial moment of movement, the O_OY_C axis is directed upward along the local vertical, and the direction of the OX_C and OZ_C axes is selected in accordance with the task.

8. When performing flights, it is sometimes more convenient to choose a coordinate system so that the movement of the aircraft along the line of a given path corresponds to a change in one of the coordinates (for example, longitude λ). Then the second coordinate will be constant. Such systems include the orthodromic coordinate system. Orthodromy is the line of shortest distance between two points on the earth's surface.

An ***orthodromic coordinate system*** is a coordinate system in which the origin is aligned with the center of the Earth, the plane of the orthodromic equator tilts to the plane of the geographic equator at a certain angle, selected in accordance with the navigation problem being solved.

The flight between the starting and ending points of the route, as a rule, is performed along the line of the orthodromic equator, which is called the orthodromy.

9. When the shape of the Earth is modeled by an ellipsoid, a geodetic coordinate system is used for navigation.

Topic 10. VERTICAL LINE AND METHODS FOR DETERMINING ITS DIRECTION

10.1. Types of Vertical Lines

When an airplane is flying in the absence of visibility of landmarks and celestial bodies, instruments are needed that would indicate the position of the aircraft relative to the natural horizon, i.e., the bank γ and pitch ϑ angles. In such a flight, the pilot is unable to determine the position of the aircraft relative to the horizon plane with the help of muscular perception, touch and hearing, although on the ground, in a dark room, a person determines his position in relation to the vertical direction with an accuracy of 1 – 2°.

A person determines their position with the help of the otolith apparatus of the inner ear, which is sensitive to linear accelerations, and the semicircular canals, which are sensitive to angular accelerations.

In flight, when the human body is exposed to external forces, a person can, with the help of his own sensations, determine only the direction of the "apparent vertical" and cannot distinguish a correct turn from a straight flight.

When flying in the clouds, pilots who have no experience in piloting an aircraft using instruments (or in the absence of instruments) lose orientation after 2 – 4 minutes and "fall out" of the clouds in the most arbitrary positions. Even birds do not have a special "flying instinct" and, when flying blindfolded, go into a tailspin or randomly fall. In flight without landmarks, a pilot who does not use instruments, as a rule, takes the plane into a correct turn, descending in a spiral, turning into a tailspin. On high-speed aircraft at night, even if the ground is visible, flight without an artificial horizon is unacceptable.

To study instruments and systems for controlling the spatial orientation of an aircraft, one should first define the very concept of vertical line.

In aviation navigation systems, the following concepts are used:

– geodetic vertical;
– geocentric vertical;
– gravitational vertical;
– true vertical;
– apparent vertical.

10.1.1. *Geodetic Vertical*

For the subsequent study of instruments for controlling the spatial orientation of an aircraft, it is necessary to first define the very concept of a horizontal coordinate system. For this purpose, it is advisable to first consider some questions of the geodetic plan.

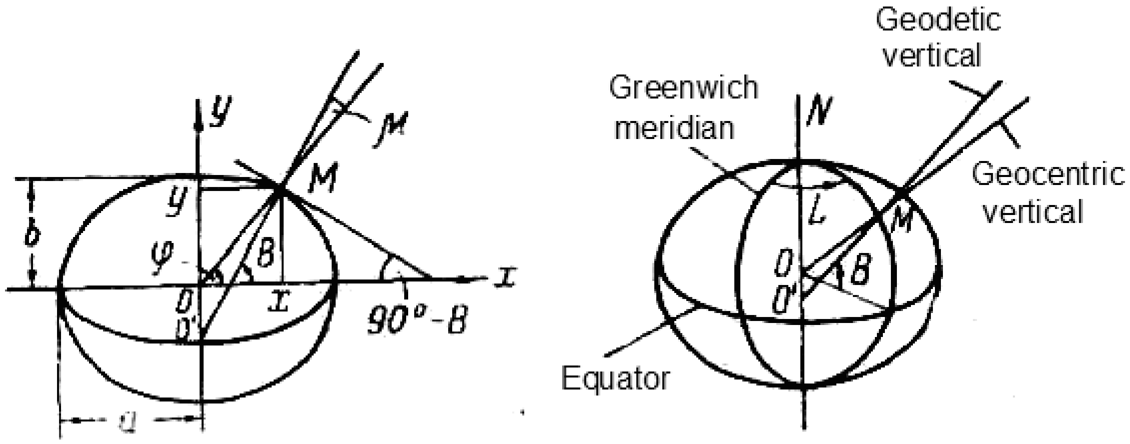
In geodesy, a body bounded by a surface at all points normal to the gravity vector g_t and coinciding with the surface of the world ocean in its calm state is taken as the initial model of the Earth's figure. Such a body is called a **geoid**. The geoid surface has a complex geometrical shape.

To solve practical problems, the geoid is replaced by an ellipsoid of revolution (Fig. 10.1), which best approximates the geoid in a given area. In Ukraine, for navigation calculations, an ellipsoid of F. N. Krasovsky was adopted with the following parameters:

– major semi-axis (radius of the equator) a = 6 378 245 m;
– minor semi-axis b = 6 356 863 m;
– eccentricity (degree of flattening)

$$e = \sqrt{a^2 - b^2}\Big/a = 0.08181. \tag{10.1}$$

The normal $O'M$ to the ellipsoid at point M represents the direction of the geodesic vertical (see Fig. 10.1). Thus, the geodesic vertical is the normal to the surface of the ellipsoid.



a b

Fig. 10.1

The angle B between the equatorial plane and $O'M$ is called geodetic latitude. The normal always belongs to the plane ONM of the meridian. The dihedral angle L between the Greenwich meridian and the meridian of point M is called geodetic longitude. Thus, the position of point M on the surface of the earth's ellipsoid is determined by geodetic latitude B and longitude L. The geodetic coordinate system is the main one in cartography, for navigation calculations when programming a route.

10.1.2. *Geocentric Vertical*

For a number of navigation calculations, a geocentric coordinate system is convenient, the origin of which O is located in the center of the Earth (see Fig. 10.1).

The geocentric vertical is defined by the line passing through the center of the Earth and the point M of the location of the aircraft.

Geocentric latitude φ is defined as the angle between the equatorial plane and the line OM, which defines the geocentric vertical. Geocentric longitude λ is equal to geodesic longitude *L*.

The angle μ (see Fig. 10.1) characterizes the difference in the directions of the geodesic and geocentric verticals.

10.1.3. *Gravitational Vertical*

The direction of the gravitational vertical is determined by the vector *g* of the intensity of the Earth's gravitational field (Fig. 10.2). If the Earth were a ball with a uniform distribution of masses, then the gravitational vertical would coincide with the geocentric one. In reality, the angle ε between these verticals does not exceed 5.6 arc minutes.

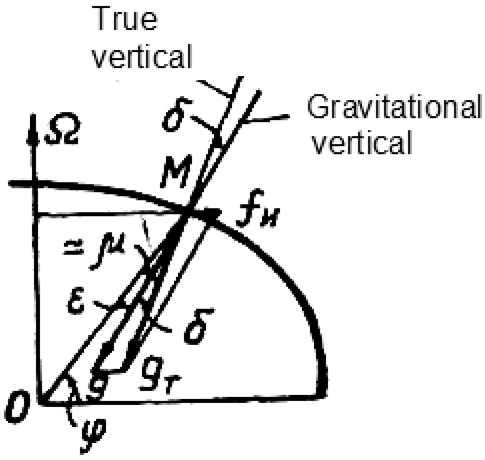


Fig. 10.2

10.1.4. *True Vertical*

The true vertical is set by the direction of the Earth's specific gravity vector g_t (see Fig. 10.2).

The direction of the true vertical on a stationary object relative to the Earth can be directly determined using a plumb bob (a weight suspended on a string) or a level bubble. It is given by the specific gravity vector g_t. It is the resultant of the intensity of the gravitational field *g* and the specific centrifugal force of inertia f_i due to the rotation of the Earth (see Fig. 10.2), and

$$F_i = \Omega^2 R \cos \varphi, \qquad (10.2)$$

where Ω is the angular velocity of the Earth's rotation (Ω = 15.04197 deg / hour);

R is the geocentric distance *OM*.

A plane perpendicular to the direction of the true vertical is called the **horizon plane**. The starting direction for all maps is the true vertical direction.

10.1.5. *Apparent Vertical*

On an aircraft, even in smooth and straight flight, the true vertical does not match the direction indicated by the plumb line or level. This is due to the fact that neither the plumb line nor the level have a natural property to be set in the direction of the true vertical due to the occurrence of Coriolis acceleration, the appearance of which is associated with the portable angular velocity of the Earth's daily rotation. The plumb line is set in the direction of the resultant of all accelerations (the resultant of all inertial forces F_i), with which the point of its suspension moves (Fig. 10.3).

The direction of such a resultant is called the direction of the apparent vertical. It is set by the resultant of the specific gravitational force g and the specific inertial force F_i and is directed along the line of the overload vector n.

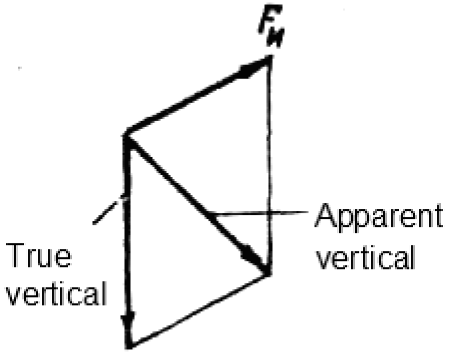


Fig. 10.3

The deviation of the apparent vertical from the true one during the evolution of the aircraft (bend, loop, etc.) can be almost as large as you like, and with a loop it can reach 180°.

The apparent vertical on an object at rest relative to the Earth is set in the direction of the specific gravity vector g_t, i.e. coincides with the true vertical. In the case of weightlessness (with free fall), the apparent vertical is not defined at all.

Of the verticals listed above, only the true vertical lends itself to physical determination: on an object stationary relative to the Earth, the plumb line is always set according to the specific gravity vector g_t.

10.2. Methods for Determining the Direction of Verticals on Board the Aircraft

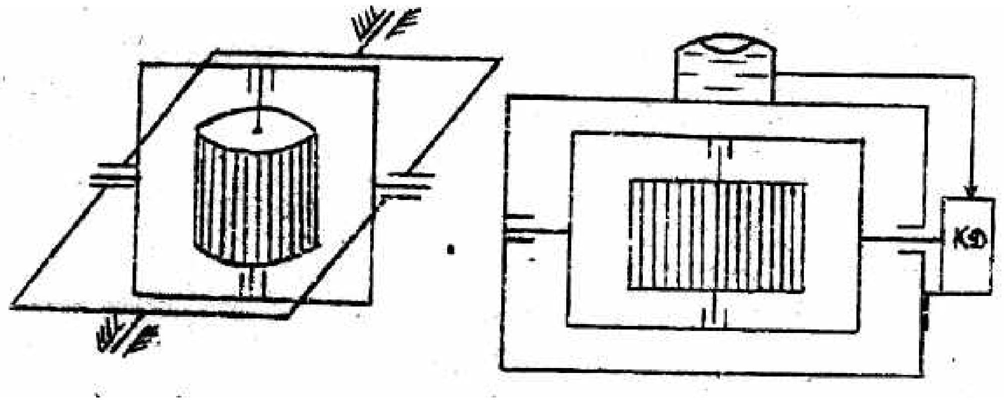
Determining the position of the aircraft relative to the horizontal plane, or thus relative to the vertical direction, is a difficult task. Since the direction of the true vertical coincides with the gravity vector **G**, a plumb or a liquid level can in principle be used to determine this direction. However, on an aircraft, the flight of which almost always occurs with accelerations, the goal will not be achieved, since the plumb will tend to be set according to the resultant forces of gravity G and inertia F_i, that is, in the direction of the so-called apparent vertical (see Fig. 10.3).

In conditions of stationary or uniform and rectilinear movement of the aircraft, simple means of determining the vertical can be:

- plumb line;
- physical pendulum;
- liquid level;
- accelerometer;
- slide indicator.

However, if such devices for determining the vertical are placed on an arbitrarily moving object, then they will be installed in the direction of the apparent vertical. To determine the true vertical with these instruments, it is necessary at the time of taking readings to know the direction and magnitude of the aircraft accelerations and take into account the angle between the gravitational and true verticals, which is a complex technical problem.

A ***free gyroscope*** can be used as a guardian of any inertial direction (Fig. 10.4, a), but the true vertical is not an inertial direction. Due to the rotation of the Earth and the movement of the aircraft relative to it, the position of the vertical relative to the gyroscope will change.



a b

Fig. 10.4

If you set the measuring axis of the gyroscope rotor in the direction of the true vertical, then at the equator, due to the daily rotation of the Earth, this axis will deviate from the direction of the true vertical with an angular velocity of

0.25 deg/min and after 10 minutes will make an angle with the direction of the true vertical equal to 2° 30 '.

The deviation from the vertical of a real gyroscope will additionally be caused by deviations (precession) due to the moments of friction and imbalance forces. Therefore, gyroscopic verticals can be used only in short-term flight conditions and for small distances compared to the Earth's radius.

For these reasons, the gyroscope as an indicator of the vertical direction can be used on an aircraft only for a short time.

Therefore, it turned out to be expedient to construct complex systems that use the positive properties of a gyroscope and a physical pendulum. Such systems are gyro-verticals, which are gyroscopes with pendulum correction of their position (see Fig. 10.4, b). Thus the main device for determining the direction of the true vertical is a gyroscope, and a physical pendulum acts as a correcting device.

During the evolution of the aircraft, the errors of the physical pendulum as an indicator of the direction of the true vertical are practically equal to the angles of deviation of the apparent vertical from the true. This happens because the physical pendulum, when flying with acceleration, practically follows the direction of the apparent vertical.

The gyroscope, on the other hand, is not very susceptible to linear accelerations, and therefore its readings turn out to be more stable in any aircraft evolution. The gyroscope's rotor axis keeps its direction unchanged with respect to the stars.

In aviation gyroscopic verticals, the following properties are simultaneously used:

1) in straight and uniform flight, the physical pendulum is installed in the direction of the true vertical;

2) in the presence of accelerations, the gyroscope keeps the direction of the rotor axis in space unchanged.

The axis of the gyro-vertical rotor is held in the direction of the true vertical by means of pendulum correcting devices.

Consider the principle of operation of the pendulum correcting device (Fig. 10.5).

The pendulum correcting device is a fluid pendulum. The pendulum corrector with the Ox axis is oriented in the direction of flight of the aircraft. Copper sealed casing 1 is filled with a conductive liquid-electrolyte 2. The role of the pendulum is played by an air bubble 5. Four symmetrically located contacts 4 are pressed into the cover of the casing of the pendulum corrector, isolated from the casing 1.

Fig. 10.5 shows only the inclusion of contacts located along the Ox axis in the circuit of the control windings W_1 and W_2 of the longitudinal correction motor D_2. The field winding W_3 is powered by a voltage U_2, which has a 90 ° phase shift with respect to the voltage $\mathbf{U_1}$ supplied to the body and the common point of the windings W_1 and W_2. In the same way, the other two contacts of the lateral correction circuit are connected.

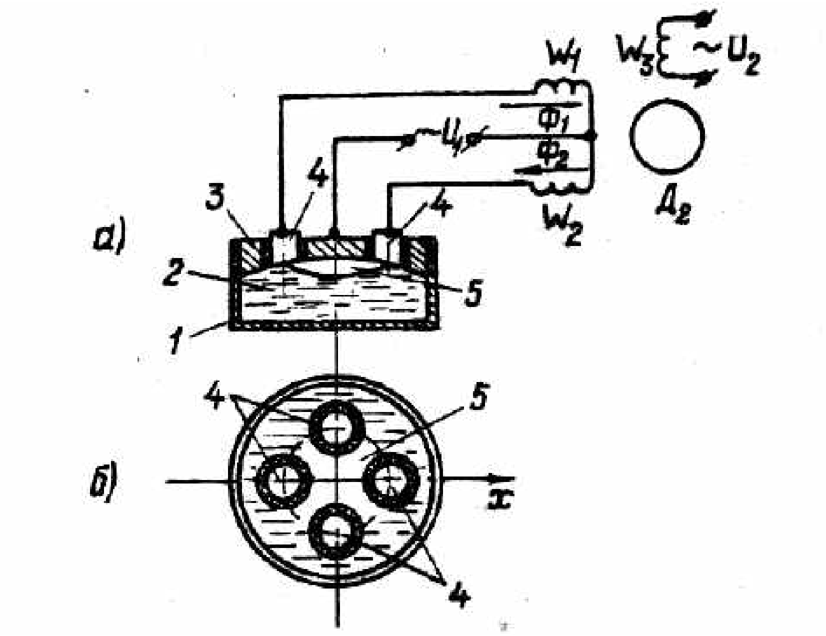


Fig. 10.5

If the measuring axis of the gyroscope is vertical, the bubble takes a central position, so that the contact surfaces of all contacts 4 with liquid 2 are the same. Consequently, the resistances between each of these contacts and the case are the same. Therefore, the magnetic fluxes F_1 and F_2 of the oppositely connected windings W_1 and W_2 will be equal, and the resulting motor torque will be zero.

The gyroscope is tilted from vertical when the aircraft is accelerating or decelerating. With longitudinal deviations of the gyroscope from the vertical, the bubble will shift along the Ox axis. As a result, the areas of contact with the liquid of the two contacts located along this axis will become unequal. This will lead to a change in the fluxes F_1 and F_2 and a corrective torque of the motor M_2 will appear, under the influence of which the gyroscope will precess to the vertical. Thus, the gyroscope axis will return to vertical.

The chain of lateral correction along the Oy axis works in a similar way.

The pendulum correcting device is attached directly to the gyroscope body, the direction of the axis of which must be corrected.

Topic 11. ARTIFICIAL HORIZONS AND VERTICAL GYRO

11.1. Artificial Horizons

The artificial horizon is a gyroscopic device designed to measure the bank angle γ and the pitch angle ϑ of aircraft.

For manual control of the flight, it is necessary to provide a visual indication of the aircraft position relative to the horizontal plane. Modern artificial horizons provide the accuracy of determining the roll and pitch angles of the order of 15 – 30' (if you do not take into account the errors in picking up signals).

The artificial horizon refers to instruments that directly affect flight safety. Therefore, increased requirements are imposed on its reliability.

The basic element of the artificial horizon is the true vertical builder for defining the horizontal plane.

A pendulum mounted on an aircraft cannot perform such a function (we established the reason for this in the last lecture).

A free gyroscope, the rotor of which maintains a constant orientation in the inertial coordinate system, is also not suitable for this. The position of vertical relative to the gyroscope will change due to the rotation of the Earth and the movement of the aircraft relative to it. Therefore, over time, the gyroscope rotor axis will deviate from the vertical (we assume that the rotor axis was originally installed vertically).

For these reasons, vertical systems are used in artificial horizons based on a three-degree gyroscope with pendulum correction.

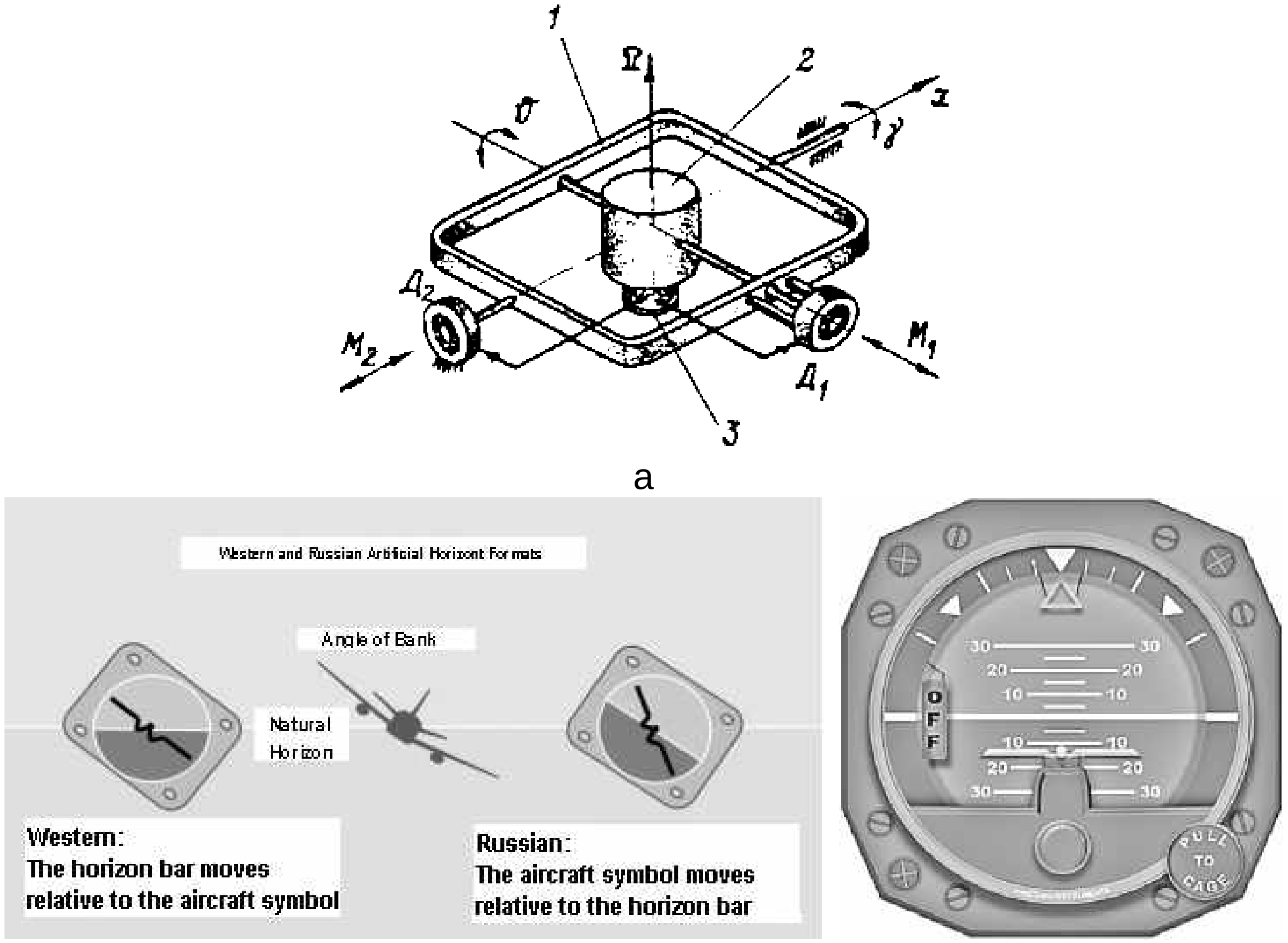
At high aircraft accelerations, the artificial horizon correction circuits are disabled by a liquid pendulum switch. After that, only the gyroscope is used to determine the vertical. In this case, the accuracy of the gyroscope depends on the duration of operation without correction.

The gyro sensors of the artificial horizons of low-maneuverable airplanes and helicopters have the simplest kinematics, for example, of the **АГБ-3** type (installed on the **Mi-8** helicopter). Let's consider the principle of operation of such an artificial horizon (Fig. 11.1).

The axis of the outer frame 1 is parallel to the longitudinal axis of the aircraft. In the lower part of the gyro unit 2 there is a pendulum device 3. The signals of this device in case of deviations of the gyro-vertical rotor axis are fed to the two-phase electric motors D_1 and D_2 of transverse and longitudinal correction, respectively.

Correcting moments M_1 and M_2 of these electric motors act on the gyroscope and ensure its restoration to the vertical. As a result, the axis of the gyro (inner frame) is kept in the horizontal plane and is the axis for measuring the pitch angle ϑ.

The axis of the outer frame serves as the axis of measurement of the bank angle γ.



a

b c

Fig. 11.1

The pendulum device is a liquid pendulum (Fig. 11.2). We examined its principle of operation in the previous lecture.

Let me remind you that liquid pendulums react to gravitational and inertial forces, which means that when flying with accelerations, the pendulum correction will tend to set the gyro-vertical along the apparent vertical. Therefore, when flying with acceleration, the correction is disabled.

Note that the artificial horizons of the considered kinematics are not suitable for aircraft of unlimited maneuvering. So, at pitch angles ±90°, the concepts of bank and course of the aircraft lose their meaning, since in this case the longitudinal axis of the aircraft is projected onto a horizontal plane to a point. Therefore, in the artificial horizon, the measurement of roll and heading at pitch angles from ±20° to ±90° is stopped. Consumers are given the roll angle and heading that the plane had at the time of transition to vertical maneuver mode.

To eliminate this disadvantage, gyro sensors of more complex kinematics, for example, of the **АГД-1** type, are used in the artificial horizons of fighter aircraft. It consists of a vertical gyro (gyro sensor) and one or two artificial horizon indicators.

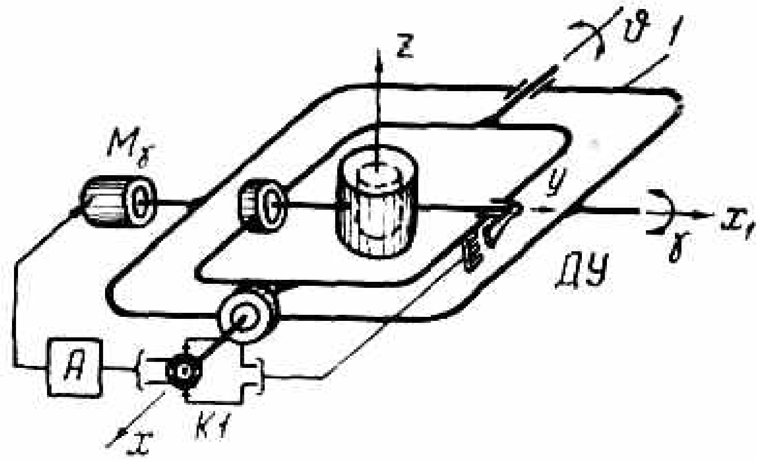


Fig. 11.3

The kinematic diagram of the **АГД-1** attitude gyro sensor is shown in Fig. 11.3. The axle supports of the outer frame are installed in an additional tracking frame 1, the axis of which is parallel to the longitudinal axis of the aircraft. Longitudinal and lateral correction systems are not shown in Fig. 11.3. They are similar to the systems discussed above and ensure the vertical position of the gyroscope rotor axis when the gyro unit suspension axis is horizontal. The rotor of the induction angle sensor is fixed on the gyro unit axis. The stator of this sensor is connected to the outer frame. When the aircraft bank, turns of the tracking frame are possible, which lead to corresponding turns of the outer frame around the axis of the gyro assembly. Such turns result in a non-zero angle sensor signal. After amplification, this signal is fed to the M_γ electric motor, which turns the tracking frame in the direction opposite to the bank.

As a result, the orthogonality of the rotor axes and the outer frame is always maintained.

Thus, the axis of the outer frame is kept horizontal and serves as the axis for measuring the pitch angle.

The measuring axis of the roll angle is the axis of the follower frame. In the signal circuit of the angle sensor, a lamellar switch K1 is installed on the axis of the outer frame. It changes the phase of the signal when $|\vartheta|>90^\circ$. This is necessary to maintain the correct direction of rotation of the follower frame motor when maneuvering in a vertical plane.

On the front side of the artificial horizon there is a caging button. Setting the gyro assembly axes parallel to the corresponding aircraft axes is achieved by caging the gyro assembly before turning on the attitude indicator. Caging ensures quick readiness of the vertical gyro for operation after switching on the attitude indicator.

The upper part of the pitch scale (above the horizon line) is colored blue, and the lower part is brown. If the scale is illuminated with red light, the upper part is painted gray, and the lower one – black. As a result, a clear indication of the spatial position of the aircraft is provided.

In level flight, the horizon line on the pitch scale can be offset relative to the silhouette of the aircraft. This misalignment can be eliminated by turning the ratchet on the pointer. In this case, the scale will rotate until the horizon line is aligned with the aircraft silhouette, and the index will indicate the value of the angle of attack.

On the bottom of the front of the indicator there is a sliding indicator, which is a curved glass tube filled with toluene. A plastic ball is placed in the tube which in straight flight and such with a correct turn occupies a central position. When turning with a slip, the ball is displaced in the direction opposite to the slip.

The artificial horizon readiness time at power-up does not exceed 1.5 minutes. To reduce the readiness time of the artificial horizon, the gyro sensor has an electromechanical caging. It turns on automatically when the power is turned on and ensures that the gyroscope rotor axis is quickly aligned with the plane's normal axis. When the aircraft is parked, small deviations of the rotor axis from the vertical may appear. These deviations are eliminated by a correction pendulum system.

Caging can also be performed in horizontal straight flight to quickly eliminate post-roll errors, as well as errors after prolonged longitudinal accelerations. To do this, briefly press the red button on the front of the indicator.

The artificial horizon is the most important flight instrument. Therefore, in flight, the pilot is obliged to monitor its serviceability by comparing the attitude indicator readings with such instruments as turn-and-slip indicators, a variometer, an altimeter and a speed indicator. On some aircraft there are so-called artificial horizon backups of type **ДА-200**.

11.2. Vertical Gyro

The aircraft has a large number of instruments, automation and control systems, for the operation of which information on the angular positions of the aircraft is required. The use of individual sensors of the bank and pitch angles in this case turns out to be inappropriate, since their total weight and volume are large, and the accuracy of measuring the angles by each sensor is low.

On some aircraft, a single gyroscopic sensor is used – a vertical gyro, which provides all consumers with bank and pitch signals.

Fig. 11.4 shows the electro-kinematic diagram of the **ЦГВ** gyro-vertical.

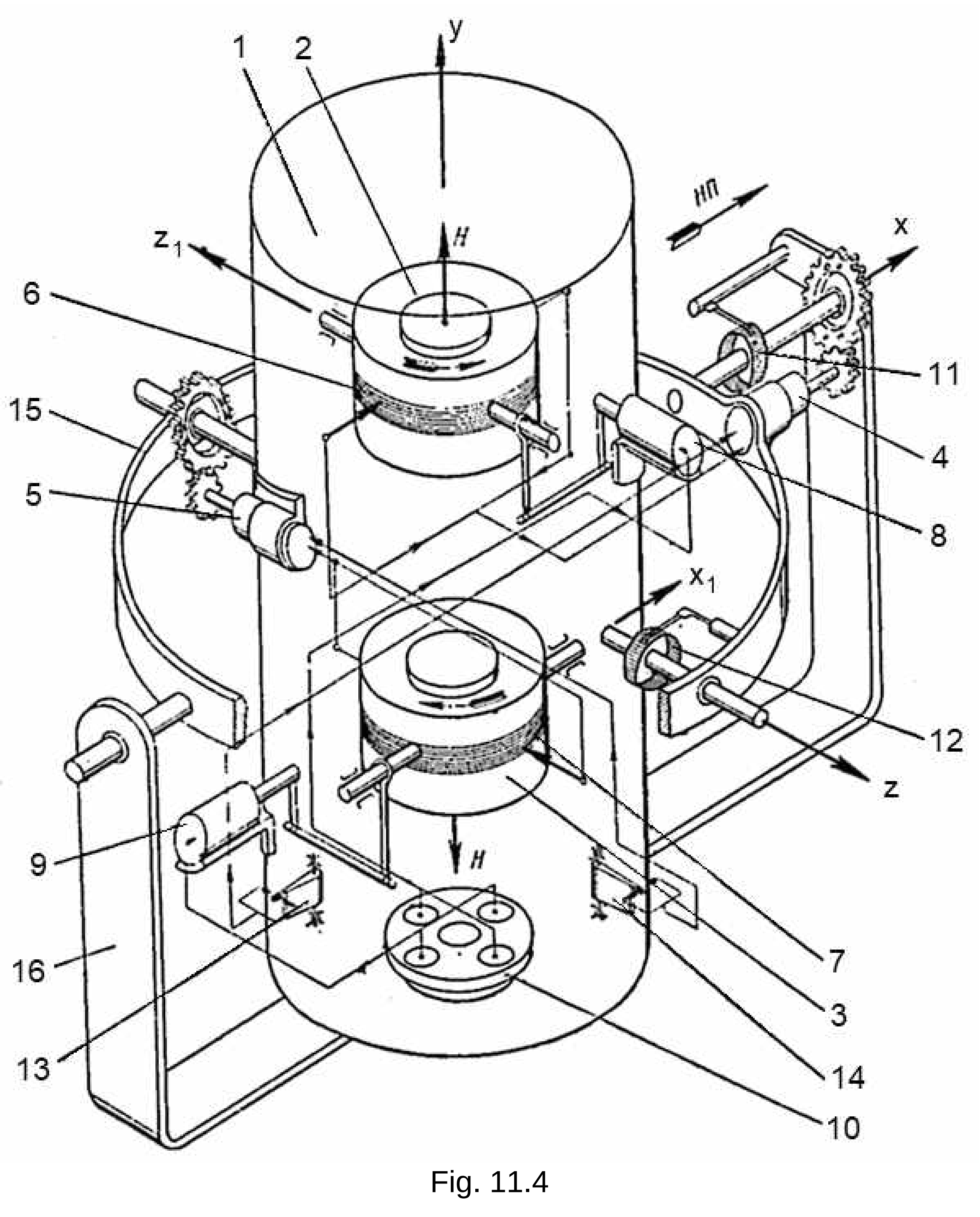


Fig. 11.4

The outer frame 15 of the gimbal is fixed in the bearings of the housing 16 of the device. The inner frame is a cylindrical platform 1. The axis of the platform 1 is stabilized vertically. If the *x* axis is directed parallel to the longitudinal axis of the aircraft, and the *z* axis is parallel to its transverse axis, then the roll angle signals are picked up from the potentiometer 11, and the pitch angles from the potentiometer 12.

Inside the platform 1 there are two gyroscopes 2 and 3 with two degrees of freedom.

The power gyro stabilization system includes potentiometers 6 and 7, electric motors 5 and 4.

Power gyro stabilization works as follows. Let the external moment M_x be applied to the x-axis of the frame 15. Through the bearings of the frame 15, the platform body 1 and the z_1 axis, the moment M_x is applied to the rotor axis of the gyroscope 2. The axis of this gyroscope begins to precess so that the angular momentum vector H moves towards alignment with the vector M_x. In this case, the potentiometer 6 is displaced relative to its brushes and a voltage appears on them, the phase and value of which are determined by the direction and magnitude of the deflection of the gyroscope 2.

From the brushes of the potentiometer 6 the voltage is supplied to the electric motor 4. It creates a torque, the direction of which is opposite to the external torque $M_\mathbf{x}$. As a result, the total moment applied to the axis of the gyroscope 2 decreases. The gyroscope precesses until the moment of the electric motor 4 equals the moment M_x. As a result, the axis of the rotor of the gyroscope 2 will deviate from the vertical y, but the axis at the platform (together with the potentiometer 12) will remain stationary. After removing the external moment M_x under the influence of the moment of the electric motor 4, the gyroscope 2 precesses to the vertical y until the voltage on the brushes of the potentiometer 6 becomes equal to zero.

If an external moment is applied to the z axis, then through the platform 1 and the x_1 axis it is transmitted to the gyroscope 3. It precesses about the x_1 axis, and the voltage from the brushes of its potentiometer 7 is supplied to the electric motor 5. The external moment is compensated about the z axis in the same way as in the action of the moment M_x.

The presence of force compensation for external moments does not eliminate the drift of the gyro-vertical as a result of the Earth's rotation. Elimination of the influence of the Earth's rotation is provided by a correction system consisting of a liquid pendulum 10 and correction motors 9 and 8.

When the vertical gyro is launched, the platform 1 can be in any position. Mechanical pendulums 1 and 15 are used to quickly set it to a vertical position. Their electrical circuits are closed through the contacts of the button, which is located on the dashboard. If the platform is deflected at an angle greater than 1.5–2°, then pendulums 13 and 14 close their contacts and supply voltage to electric motors 4 and 5. These electric motors set the platform 1 vertically with an accuracy of 1.5–3 °. After that, pendulums 13 and 14 open their contacts. Further installation of the vertical gyro is provided with the help of a liquid pendulum and correction electric motors.

Topic 12. COURSE INSTRUMENTS

12.1. The Ways to Determine Aircraft Course

An important flight and navigation parameter is the aircraft heading. Accurate heading knowledge ensures correct orientation of the longitudinal axis of the aircraft in the horizontal plane.

The course (heading angle) of the aircraft is the angle between any given direction relative to the Earth's surface in the horizontal plane and the projection of the longitudinal axis of the aircraft on the horizontal plane.

The course is measured by means of heading devices and systems and is counted from a given direction clockwise (from 0º to 360º). It is designated by the letter Ψ.

Depending on what is taken as a given direction in the horizontal plane, they are distinguished (Fig. 12.1):

– true Ψ_T;
– magnetic Ψ_M;
– compass Ψ_C**;**
– orthodromic Ψ_{ORT};
– gyroscopic Ψ_G;
– conditional and other courses.

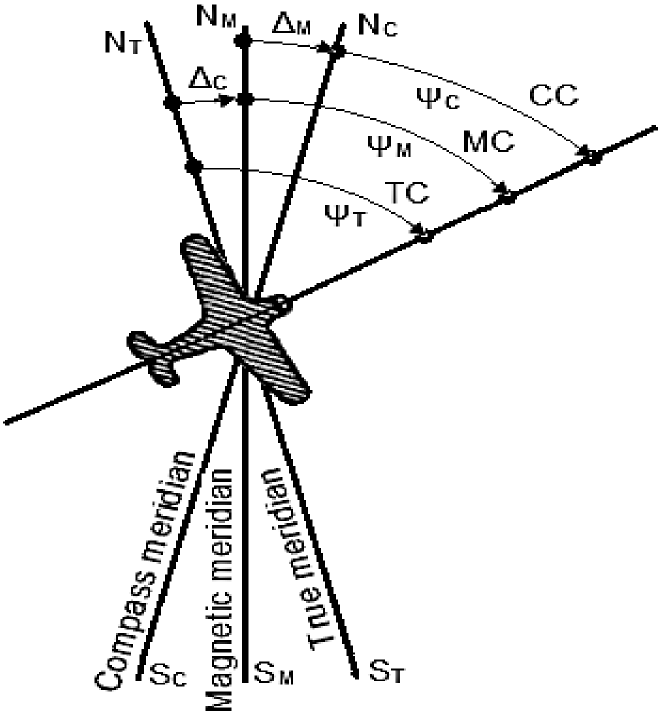


Fig. 12.1

Magnetic North Pole – 78° 30′ North latitude, 69° West longitude.
Magnetic South Pole – 78° 30′ South latitude, 111° East longitude.

The distance from the North Pole to the Magnetic North Pole is 1278.760 km.

The true (geographical) course Ψ_T is measured from the north direction of the true (geographical) meridian N (see Fig. 12.1).

The magnetic course Ψ_M is measured from the north direction of the magnetic meridian N_M (see Fig. 12.1).

The compass course Ψ_C is measured from the north direction of the compass meridian N_C (see Fig. 12.1).

The discrepancy between the compass and magnetic meridians is explained by the fact that the magnetic compass needle deviates from the plane of the magnetic meridian under the influence of steel and iron parts of the aircraft, as well as under the influence of magnetic and electromagnetic fields generated by the equipment. This leads to a deviation of the horizontal vector component of the strength of the Earth's magnetic field. The resulting direction of the horizontal component of this vector is called the ***compass meridian***. This direction is indicated by the magnetic needle on board the aircraft.

The angle Δ_C between the directions of the magnetic N_M and compass N_C meridians is called the ***magnetic deviation*** of the compass (Fig. 12.2).

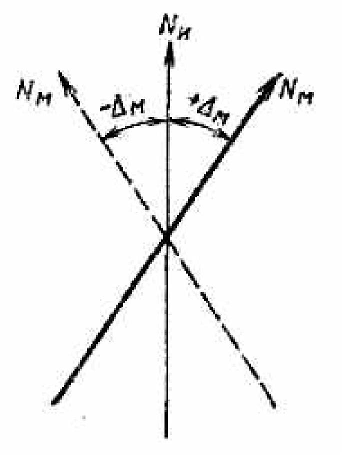


Fig. 12.2

The magnitude of the magnetic deviation is not the same at different courses of the aircraft.

Orthodromic course Ψ_{ORT} is measured from the orthodromy. Orthodromy is the line of intersection of the earth's sphere with a plane passing through the center of the Earth and two specified points on its surface (the starting point of the route and the final destination of the route, or intermediate points of the route).

The gyroscopic course Ψ_G is measured from the direction of the main axis of the gyroscope, which is set in the horizontal plane.

There are several methods for determining the course.

1. ***The magnetic method*** is based on the property of a permanent magnet freely suspended in the earth's magnetic field to be located in the

direction of the magnetic meridian (magnetic compasses).

2. ***The induction method*** is based on the induction of the electromotive force by the Earth's magnetic field in a choke with a permalloy core, the magnetic permeability of which is periodically changed using an alternating magnetic field (induction compasses).

3. ***The gyroscopic method*** is based on the property of the gyroscope to maintain a given direction in space (gyrocompass and flight gyroscope).

4. ***Gyromagnetic and gyro-induction methods*** are based on maintaining the axis of the three-degree gyroscope in the direction of the magnetic meridian by correcting it with a magnetic or induction sensitive element (gyromagnetic and gyro-induction compasses).

5. ***The astronomical method*** is based on the direction finding of celestial bodies (astrocompasses).

6. ***The radio engineering method*** is based on the direction finding of ground radio stations or moving radiation sources (radio compasses).

When measuring the course by the astronomical or radio engineering method, it is necessary to measure the heading angle β on board the aircraft.

The heading angle is the angle between the horizontal projection of the longitudinal axis of the aircraft and the horizontal projection of the direction onto an object (Fig. 12.3). It is counted clockwise (from 0° to 360°).

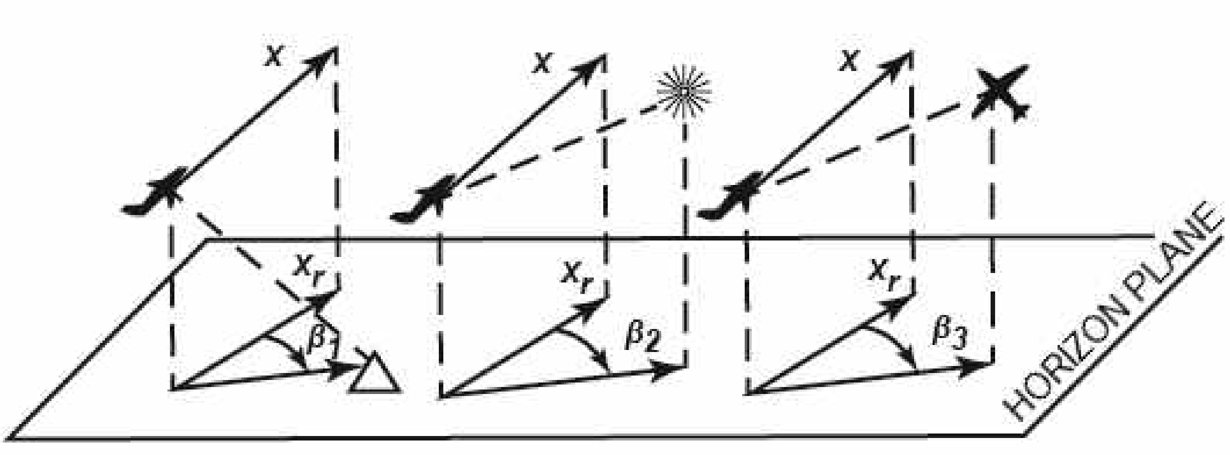


Fig. 12.3

There are distinguished:

– heading angle of the radio station β_1 for radio engineering systems;

– heading angle β_2 for astronomical systems,

– target heading angle β_3 for weapons systems.

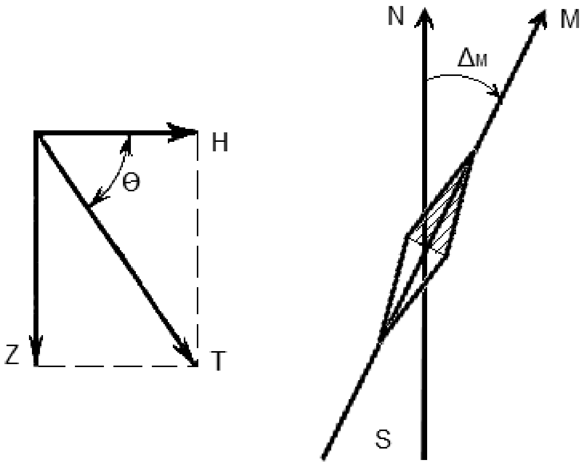
Since only the heading angle can be measured on board the aircraft, then to determine the course it is necessary to know the ***azimuth***, which is the angle between the given direction in the horizontal plane (meridian, orthodrome) and the direction to the pole. Then

$$\Psi_T = A - \beta . \qquad (12.1)$$

12.2. The Structure and Elements of the Earth's Magnetic Field

The Earth's magnetic field, like any other magnetic field, is characterized by its intensity. In general, it is a complex vector function of location coordinates and time.

The direction of the magnetic meridian is set by the horizontal component *H* of the vector of intensity *T* of the Earth's magnetic field (Fig. 12.4, a). And the simplest device for determining the direction of the Earth's magnetic meridian is a magnetic needle (Fig. 12.4, b).



a b

Fig. 12.4

The characteristic angles for the vectors of the Earth's magnetic field are Δ_M and Θ_M (see Fig. 12.4):

1. Magnetic declination Δ_M is the angle between the northern direction of the geographic meridian and the direction of the horizontal component of the Earth's magnetic field vector H (see Fig. 12.4, b). The magnitude and sign of Δ_M are necessary for the transition to the true course. They are determined by special maps.

2. Magnetic inclination Θ_M is the angle between the horizontal plane and the vector of the Earth's magnetic field *T* (see Fig. 12.4, a) which is counted clockwise.

12.3. Magnetic Compasses

Since ancient times, the Earth's magnetic field has been used mainly only to determine the direction to the Earth's magnetic poles using a compass, and if the magnetic declination is known, the direction of the geographical meridian.

Devices for determining the direction of flight (course) of an aircraft using the earth's magnetic field are called ***magnetic compasses***.

The operation of a magnetic compass is based on the property of a permanent magnet freely suspended in the Earth's magnetic field to be located in the direction of the magnetic meridian (Earth's magnetic pole).

The direction of the horizontal component *H* of the intensity vector *T* of the Earth's magnetic field is taken as the northern direction of the magnetic meridian.

In aviation magnetic compasses such as **КИ-12** and **КИ-13** (Fig. 12.5), the role of a magnetic needle is played by a moving system called a card. The card consists of one or more pairs of cylindrical permanent magnets 1 and a dial (scale) 2, in which they are fixed. The scale is graduated from 0º to 360º. The card has a spire 4, which rests on a cushioned thrust bearing 5. The sealed compass case 3 is filled with liquid (ligroin), thereby damping the vibrations of the card and reducing friction in the support due to partial weighing of the card (like a float). A membrane chamber 7 serves to compensate for temperature changes in the volume of liquid. There is a vertical line on the front glass 6 for reading the course on the scale of the card.

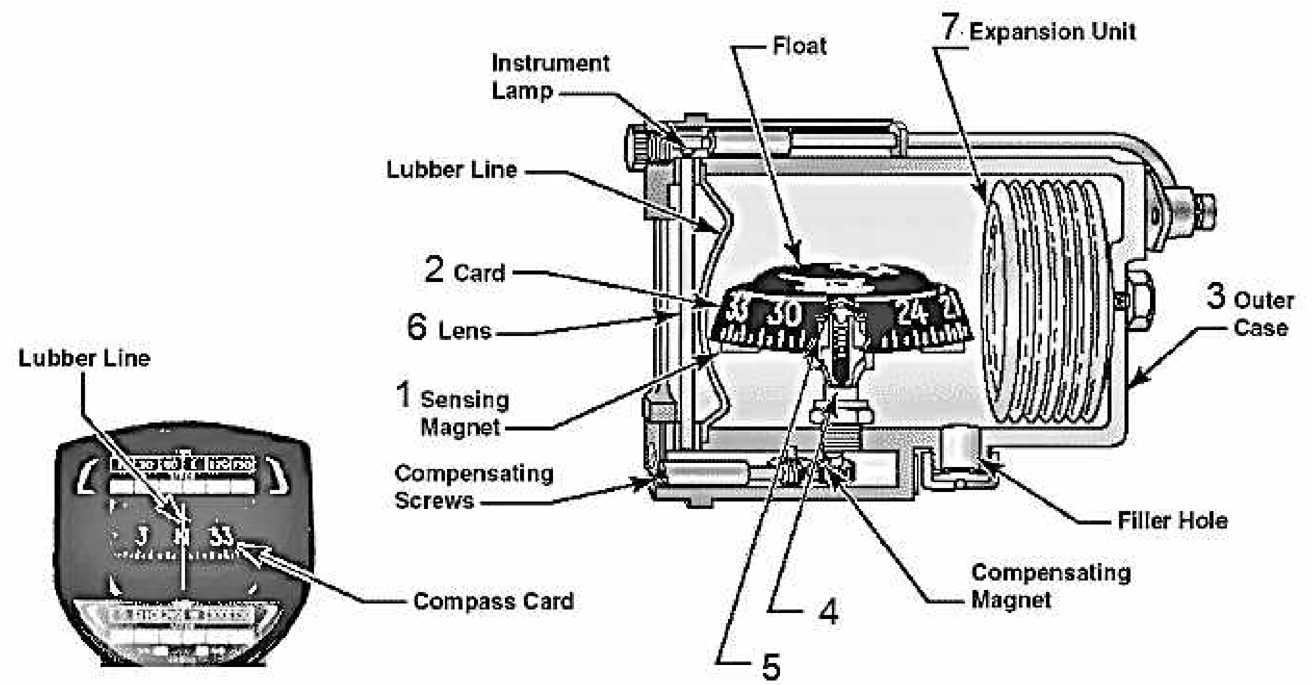


Fig. 12.5

Aviation magnetic compasses are used as backup devices, as they have low accuracy (up to ± 3º).

A significant disadvantage of permanent magnet compasses are:

– errors of stagnation (due to friction in the supports of movable magnets);

– the influence on the compass readings of the acceleration of aircraft movement (bank, turn errors);

– the influence of disturbances in the Earth's magnetic field;

– the influence of deviation.

The magnetic compass determines the direction of the horizontal component of the Earth's magnetic field. The direction of the meridian is obtained by adding corrections for magnetic declination and deviation (compass errors from the aircraft's own magnetic fields) to the compass readings.

12.4. Induction Compasses

The accuracy of magnetic compasses is very low. Therefore, designs of compasses were created, in which there is no moving magnetic system, and the direction of the earth's magnetic field is determined by the induction method.

The operation of an induction compass is based on the guidance of the electromotive force by the Earth's magnetic field in a choke with a permalloy core. The magnetic permeability of the core is periodically changed by an alternating magnetic field created by a special winding.

The induction compass has a sensitive element (Fig. 12.6) in the form of two permalloy cores 3 with oppositely connected primary windings 1 and 2, so that their fluxes Φ_1 and Φ_2 in the cores are equal in magnitude and opposite in direction. Each of these windings covers one core and is supplied with alternating current with voltage U_1 and frequency f.

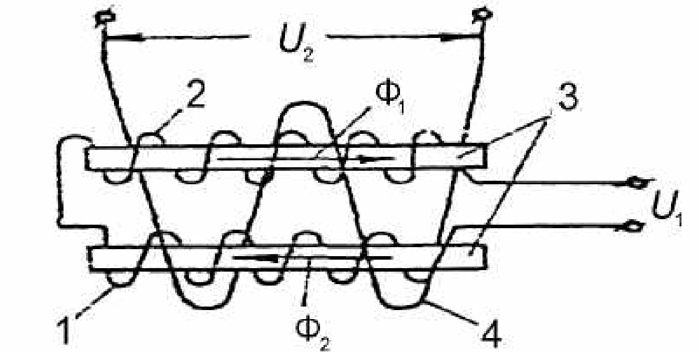


Fig. 12.6

The secondary signal winding 4 covers both cores. Often two signal windings are used, wound over the primary and connected in parallel. In the absence of external magnetic fields, these flows do not induce an electromotive force in the winding 4, since the total magnetic flux of the primary windings 1 and 2 $\Phi_1 + \Phi_2$ is zero. Those, there is no current in the signal winding.

Changes in alternating current **i** in windings 1 and 2 will lead to periodic changes in the inductions B_1 and B_2 in the cores. With an increase of induction, the magnetic resistance of the core R_M increases, and the magnetic permeability $\mu=1/R_M$ decreases. At saturation ($R_M = \infty$), the permeability is zero.

Thus, the primary windings are used to periodically change the magnetic permeability of the cores.

In the presence of a constant external magnetic field, an additional magnetic flux Φ_3 appears in the cores, pulsating in time along with the magnetic permeability μ. The intensity H of the Earth's magnetic field will be projected on the axis of cores in the form of a component $H\,\mathrm{Cos}\psi_M$. This component of the field strength will determine the induction B_3 in the cores:

$$B_3 = \mu\, H\, \mathrm{Cos}\Psi_M \qquad (12.2)$$

and magnetic flux

$$\Phi_3 = B_3\, S = S\, \mu\, H\, \mathrm{Cos}\Psi_M\ , \qquad (12.3)$$

where S is the cross-sectional area of the core.

This additional flux Φ_3 induces a voltage U_2 in the signal winding, the frequency of which is twice the frequency of the supply voltage. If the sensitive element is placed in the horizontal plane, then the voltage U_2 of the secondary winding will be proportional to the projection $H \bullet \mathrm{Cos}\Psi_M$ of the vector of the strength H of the Earth's magnetic field on the axis of the primary windings.

If the induction element is positioned so that its sensitivity axis is in the north-south direction, then the voltage U_2 will be the highest, but if in the west-east direction, the voltage U_2 will be zero. So the voltage U_2 will be determined by the value of the magnetic course Ψ_M and the value of the strength H of the Earth's field, depending on the location.

The angle between the longitudinal axis of the aircraft and the axis of the sensing induction element can be transmitted to the pointer using a remote transmission.

Fig. 12.7 shows another version of the construction of an induction compass. Three induction sensitive elements are arranged in a triangle on a common base 6. Their primary windings are wound on each pair of cores in antiphase and connected in series. Their secondary windings are connected in a triangle pattern, the tops of which are connected by a three-wire line with the stator windings of selsyn 2, which is located in the indicator.

The angles between the axis of each secondary winding of the sensing element and the direction of the vector H of the Earth's magnetic field differ from each other by 120°. The values of the voltages U_2 of these windings will be different and will depend on the magnetic course Ψ_M of the aircraft. The voltages U_2 create the corresponding currents in the selsyn stator windings. The direction of the vector of the resulting magnetic flux of the Φ_C stator due to these currents is uniquely determined by the value of the magnetic course Ψ_M.

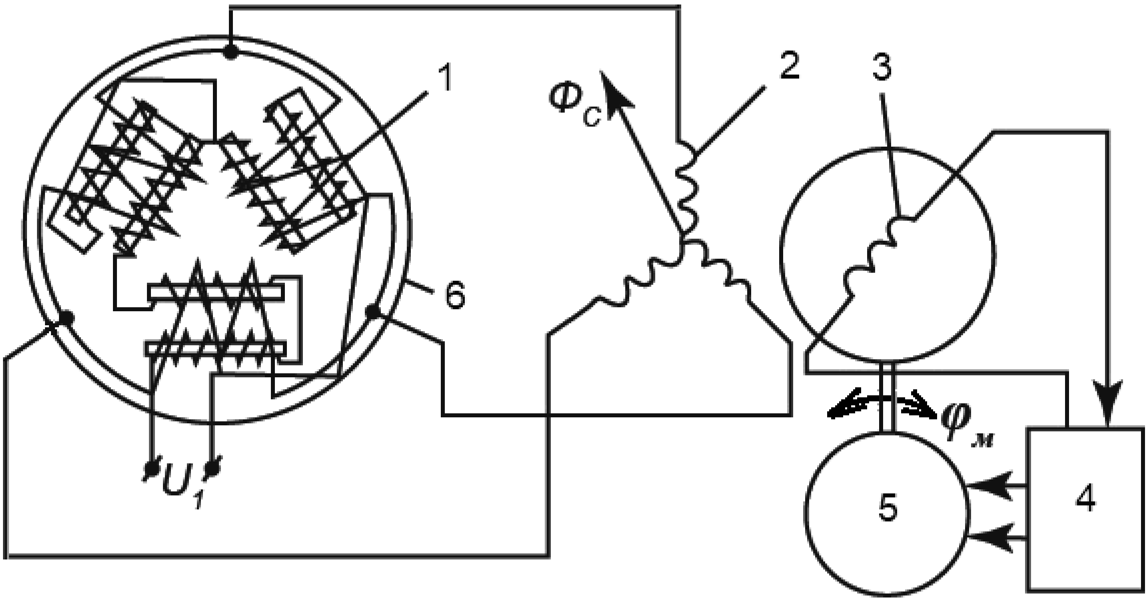


Fig. 12.7

If the axis of the winding 3 of the selsyn rotor is not perpendicular to the Φ_C vector, then the resulting winding voltage after amplification by the amplifier 4 is supplied to the electric motor 5. The latter turns the selsyn rotor to an agreed position corresponding to the magnetic course Ψ_M.

Induction heading sensors are not used as independent devices. They are widely used in magnetic correction systems of modern heading systems due to their higher instrumental accuracy compared to sensors with a moving magnet system.

Topic 13. AVIONICS SYSTEMS. AUTOFLIGHT

13.1 Automatic Pilot System

An aircraft automatic pilot system controls the aircraft without the pilot directly maneuvering the controls. The autopilot maintains the aircraft's attitude and/ or direction and returns the aircraft to that condition when it is displaced from it. Automatic pilot systems are capable of keeping aircraft stabilized laterally, vertically and longitudinally. On a large aircraft, the autopilot is engaged in auto flight mode. Automatic flight control systems encompass autopilot and related systems such as auto throttle and auto land. A discussion of the basics of an auto pilot systems follows.

The primary purpose of an autopilot system is to reduce the work strain and fatigue of controlling the aircraft during long flights. Most autopilots have both manual and automatic modes of operation. In the manual mode, the pilot selects each maneuver and makes small inputs into an autopilot controller. The autopilot system moves the control surfaces of the aircraft to perform the maneuver. In automatic mode, the pilot selects the attitude and direction desired for a flight segment. The autopilot then moves the control surfaces to attain and maintain these parameters.

Autopilot systems provide for one, two, or three axis control of an aircraft. Those that manage the aircraft around only one axis control the ailerons.

They are single-axis autopilots, known as wing leveler systems, usually found on light aircraft (Fig. 13.1). In this figure the wing leveler system on a small aircraft is a vacuum-operated single-axis autopilot. Only the ailerons are controlled. The aircraft's turncoordinator is the sensing element. Vacuum from the instrument vacuum system is metered to the diaphragm cable actuators to move the ailerons when the turn coordinator senses roll.

Other autopilots are two axis systems that control the ailerons and elevators. Three axis autopilots control the ailerons, elevators, and the rudder. Two and three axis autopilot systems can be found on aircraft of all sizes.

There are many autopilot systems available. They feature a wide range of capabilities and complexity. Light aircraft typically have autopilots with fewer capabilities than high performance and transport category aircraft. Integration of navigation functions is common, even on light aircraft autopilots. As autopilots increase in complexity, they not only manipulate the flight control surfaces, but other flight parameters as well.

Some modern small aircraft, high-performance, and transport category aircraft have very elaborate autopilot systems known as automatic flight control systems (AFCS). These three axis systems go far beyond steering the aeroplane. They control the aircraft during climbs, descents, cruise, and approach to landing. Some even integrate an auto throttle function that automatically controls engine thrust that makes auto landings possible. For further automatic control, flight management systems have been developed.

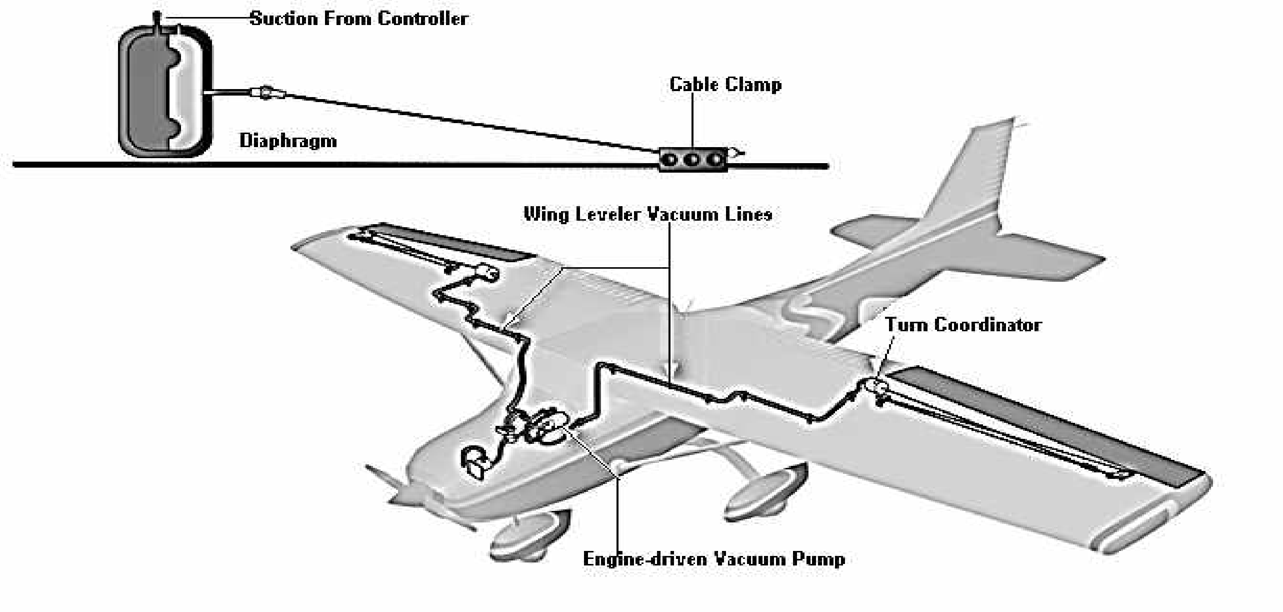


Fig. 13.1

Through the use of computers, an entire flight profile can be programmed ahead of time allowing the pilot to supervise its execution. An FMS computer coordinates nearly every aspect of a flight, including the autopilot and auto throttle systems, navigation route selection, fuel management schemes, and more.

13.2. Basis for Autopilot Operation

The basis for autopilot system operation is error correction. When an aircraft fails to meet the conditions selected, an error is said to have occurred. The autopilot system automatically corrects that error and restores the aircraft to the flight attitude desired by the pilot. There are two basic ways modern autopilot systems do this. One is position based and the other is rate based. A position based autopilot manipulates the aircraft's controls so that any deviation from the desired attitude of the aircraft is corrected. This is done by memorizing the desired aircraft attitude and moving the control surfaces so that the aircraft returns to that attitude. Rate based autopilots use information about the rate of movement of the aircraft, and move control surfaces to counter the rate of change that causes the error. Most large aircraft use rate based autopilot systems. Small aircraft may use either.

13.3. Autopilot Components

Most autopilot systems consist of four basic components, plus various switches and auxiliary units. The four basic components are: sensing elements, computing element, output elements, and command elements. Many advanced autopilot systems contain a fifth element: feedback or follow up. This refers to signals sent as corrections are being made by the output elements to advise the autopilot of the progress being made (Fig. 13.2).

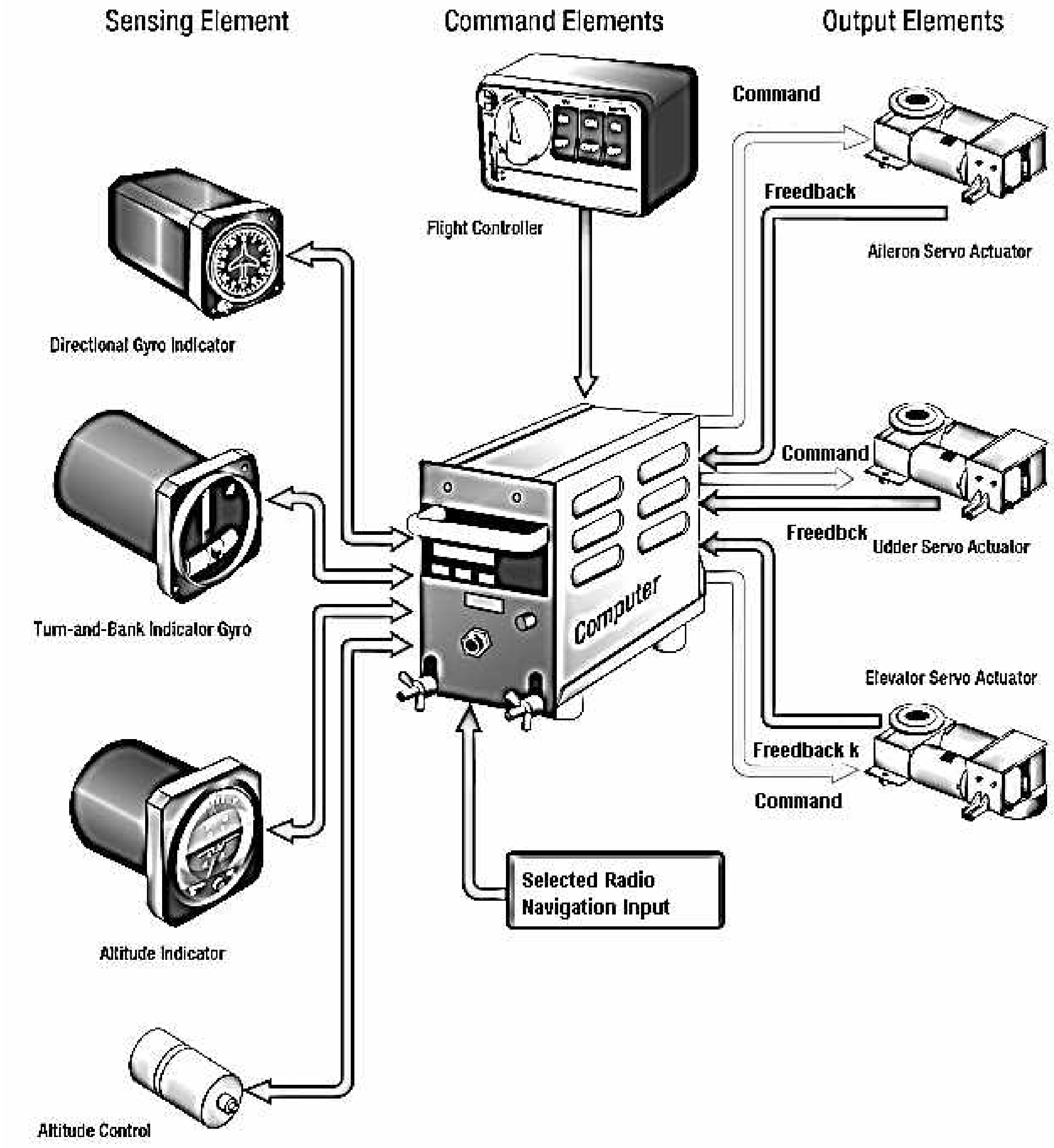


Fig. 13.2

Sensing Elements

The attitude and directional gyros, the turn coordinator, and an altitude control are the autopilot sensing elements. These units sense the movements of the aircraft. They generate electric signals that are used by the autopilot to automatically take the required corrective action needed to keep the aircraft flying as intended. The sensing gyros can be located in the cockpit mounted instruments. They can also be remotely mounted. Remote gyro sensors drive the servo displays in the cockpit panel, as well as provide the input signals to the autopilot computer.

Modern digital autopilots may use a variety of different sensors. MEMS gyros may be used or accompanied by the use solid state accelerometers and magnetometers. Rate based systems may not use gyros at all. Various input sensors may be located within the same unit or in separate units that transfer information via digital data bus. Navigation information is also integrated via digital data bus connection to avionics computers.

Computer and Amplifier

The computing element of an autopilot may be analog or digital. Its function is to interpret the sensing element data, integrate commands and navigational input, and send signals to the output elements to move the flight controls as required to control the aircraft. An amplifier is used to strengthen the signal for processing, if needed, and for use by the output devices, such as servo motors. The amplifier and associated circuitry is the computer of an analog autopilot system. Information is handled in channels corresponding to the axis of control for which the signals are intended (i.e., pitch channel, roll channel, or yaw channel). Digital systems use solid state microprocessor computer technology and typically only amplify signals sent to the output elements.

Output Elements

The output elements of an autopilot system are the servos that cause actuation of the flight control surfaces. They are independent devices for each of the control channels that integrate into the regular flight control system. Autopilot servo designs vary widely depending on the method of actuation of the flight controls. Cable- actuated systems typically utilize electric servo motors or electro-pneumatic servos. Hydraulic actuated flight control systems use electro-hydraulic autopilot servos.

Digital fly-by-wire aircraft utilize the same actuators for carrying out manual and autopilot maneuvers. When the autopilot is engaged, the actuators respond to commands from the autopilot rather than exclusively from the pilot. Regardless, autopilot servos must allow unimpeded control surface movement when the autopilot is not operating.

Aircraft with cable actuated control surfaces use two basic types of electric motor operated servos. In one, a motor is connected to the servo output shaft through reduction gears. The motor starts, stops, and reverses direction in response to the commands of autopilot computer. The other type of electric servo uses a constantly running motor geared to the output shaft through two magnetic clutches. The clutches are arranged so that energizing one clutch transmits motor torque to turn the output shaft in one direction; energizing the other clutch turns the shaft in the opposite direction (Fig. 13.3). This figure shows a reversible motor with capstan and bridle cable (left), and a single-direction constant motor with clutches that drive the output shafts and control cable in opposite directions (right).

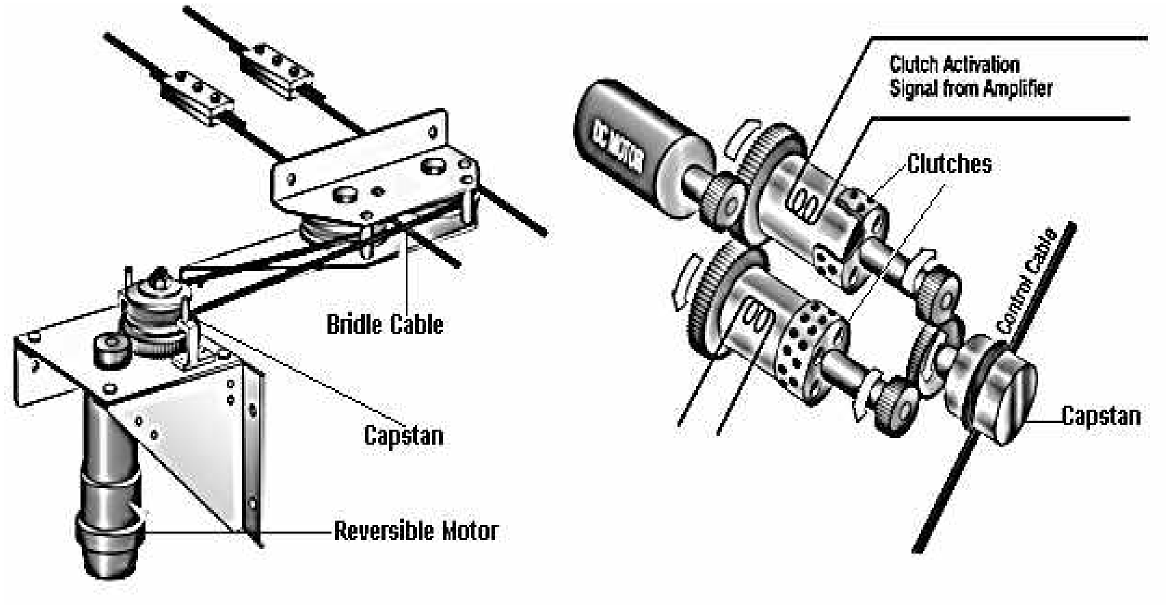


Fig. 13.3

Electro-pneumatic servos can also be used to drive cable flight controls in some autopilot systems. They are controlled by electrical signals from the autopilot amplifier and actuated by an appropriate air pressure source. The source may be a vacuum system pump or turbine engine bleed air.

Each servo consists of an electromagnetic valve assembly and an output linkage assembly. Aircraft with hydraulically actuated flight control systems have autopilot servos that are electro hydraulic. They are control valves that direct fluid pressure as needed to move the control surfaces via the control surface actuators.

They are powered by signals from the autopilot computer. When the autopilot is not engaged, the servos allow hydraulic fluid to flow unrestricted in the flight control system for normal operation. The servo valves can incorporate feedback transducers to update the autopilot of progress during error correction.

Command Elements

The command unit, called a flight controller, is the human interface of the autopilot. It allows the pilot to tell the autopilot what to do. Flight controllers vary with the complexity of the autopilot system. By pressing the desired function buttons, the pilot causes the controller to send instruction signals to the autopilot computer, enabling it to activate the proper servos to carry out the command(s). Level flight, climbs, descents, turning to a heading, or flying a desired heading are some of the choices available on most autopilots. Many aircraft make use of a multitude of radio navigational aids. These can be selected to issue commands directly to the autopilot computer.

In addition to an on/off switch on the autopilot controller, most autopilots have a disconnect switch located on the control wheel(s). This switch, operated by thumb pressure, can be used to disengage the autopilot system should a malfunction occur in the system or any time the pilot wishes to take manual control of the aircraft.

Feedback or Follow up Element

As an autopilot maneuvers the flight controls to attain a desired flight attitude, it must reduce control surface correction as the desired attitude is nearly attained so the controls and aircraft come to rest on course. Without doing so, the system would continuously over correct. Surface deflection would occur until the desired attitude is attained. But movement would still occur as the surface(s) returned to pre-error position. The attitude sensor would once again detect an error and begin the correction process all over again.

Various electric feedback, or follow up signals, are generated to progressively reduce the error message in the autopilot so that continuous over correction does not take place. This is typically done with transducers on the surface actuators or in the autopilot servo units.

Feedback completes a loop as illustrated in Fig. 13.4, which shows basic function of an analog autopilot system including follow up or feedback signal.

A rate system receives error signals from a rate gyro that are of a certain polarity and magnitude that cause the control surfaces to be moved. As the control surfaces counteract the error and move to correct it, follow up signals of opposite polarity and increasing magnitude counter the error signal until the aircraft's correct attitude is restored. A displacement follow up system uses control surface pickups to cancel the error message when the surface has been moved to the correct position.

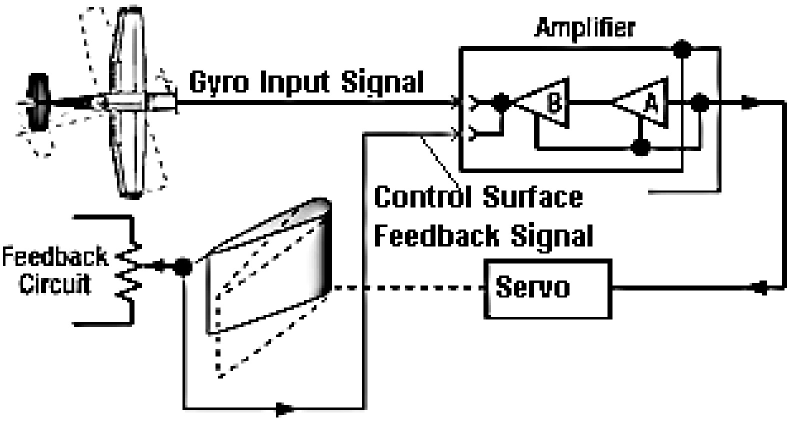


Fig. 13.4

13.4. Autopilot Functions

The following autopilot system description is presented to show the function of a simple analog autopilot. Most autopilots are far more sophisticated; however, many of the operating fundamentals are similar.

The automatic pilot system flies the aircraft by using electrical signals developed in gyro-sensing units. These units are connected to flight instruments that indicate direction, rate of turn, bank, or pitch. If the flight attitude or magnetic heading is changed, electrical signals are developed in the gyros. These signals are sent to the autopilot computer/amplifier and are used to control the operation of servo units.

A servo for each of the three control channels converts electrical signals into mechanical force, which moves the control surface in response to corrective signals or pilot commands. The rudder channel receives two signals that determine when and how much the rudder moves. The first signal is a course signal derived from a compass system.

As long as the aircraft remains on the magnetic heading it was on when the autopilot was engaged, no signal develops. But, any deviation causes the compass system to send a signal to the rudder channel that is proportional to the angular displacement of the aircraft from the preset heading. The second signal received by the rudder channel is the rate signal that provides information anytime the aircraft is turning about the vertical axis. This information is provided by the turn-and-bank indicator gyro. When the aircraft attempts to turn off course, the rate gyro develops a signal proportional to the rate of turn, and the course gyro develops a signal proportional to the amount of displacement. The two signals are sent to the rudder channel of the amplifier, where they are combined and their strength is increased.

The amplified signal is then sent to the rudder servo. The servo turns the rudder in the proper direction to return the aircraft to the selected magnetic heading.

As the rudder surface moves, a follow up signal is developed that opposes the input signal. When the two signals are equal in magnitude, the servo stops moving. As the aircraft arrives on course, the course signal reaches a zero value, and the rudder is returned to the streamline position by the follow up signal.

The aileron channel receives its input signal from a transmitter located in the gyro horizon indicator. Any movement of the aircraft about its longitudinal axis causes the gyro-sensing unit to develop a signal to correct for the movement. This signal is amplified, phase detected, and sent to the aileron servo, which moves the aileron control surfaces to correct for the error. As the aileron surfaces move, a follow up signal builds up in opposition to the input signal. When the two signals are equal in magnitude, the servo stops moving.

Since the ailerons are displaced from the streamline, the aircraft now starts moving back toward level flight with the input signal becoming smaller and the follow up signal driving the control surfaces back toward the streamline position. When the aircraft has returned to level flight roll attitude, the input signal is again zero. At the same time, the control surfaces are streamlined, and the follow up signal is zero.

The elevator channel circuits are similar to those of the aileron channel, with the exception that the elevator channel detects and corrects changes in pitch attitude of the aircraft. For altitude control, a remotely mounted unit containing an altitude pressure diaphragm is used. Similar to the attitude and directional gyros, the altitude unit generates error signals when the aircraft has moved from a preselected altitude. This is known as an altitude hold function. The signals control the pitch servos, which move to correct the error. An altitude select function causes the signals to continuously be sent to the pitch servos until a preselected altitude has been reached. The aircraft then maintains the preselected altitude using altitude hold signals.

Yaw Dampening

Many aircraft have a tendency to oscillate around their vertical axis while flying a fixed heading. Near continuous rudder input is needed to counteract this effect. A yaw damper is used to correct this motion. It can be part of an autopilot system or a completely independent unit. A yaw damper receives error signals from the turn coordinator rate gyro. Oscillating yaw motion is counteracted by rudder movement, which is made automatically by the rudder servo(s) in response to the polarity and magnitude of the error signal.

13.5. Automatic Flight Control System

An aircraft autopilot with many features and various autopilot related systems integrated into a single system is called an automatic flight control system (AFCS).

These were formerly found only on high-performance aircraft. Currently, due to advances in digital technology for aircraft, modern aircraft of any size may have AFCS.

AFCS capabilities vary from system to system. Some of the advances beyond ordinary autopilot systems are the extent of programmability, the level of integration of navigational aids, the integration of flight director and autothrottle systems, and combining of the command elements of these various systems into a single integrated flight control human interface (Fig. 13.5).

It is at the AFCS level of integration that an autothrottle system is integrated into the flight director and autopilot systems with glide scope modes so that auto landings are possible. Small general aviation aircraft being produced with AFCS may lack the throttle-dependent features.

Modern general aviation AFCS are fully integrated with digital attitude heading and reference systems (AHRS) and navigational aids including glideslope.

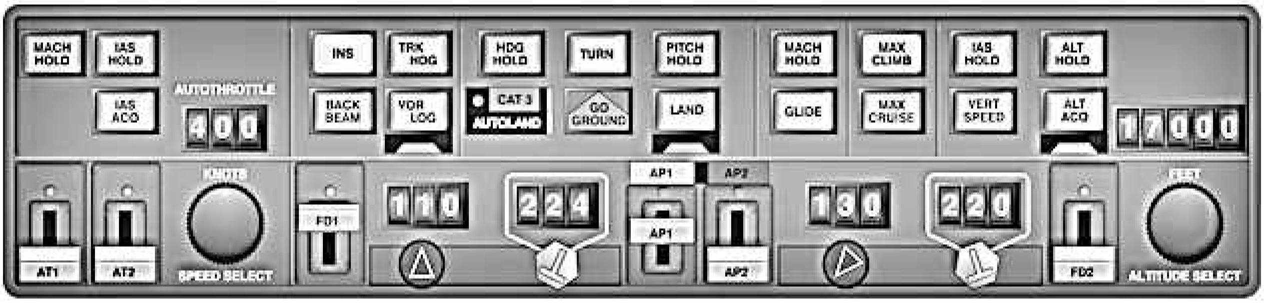


Fig. 13.5

They also contain modern computer architecture for the autopilot (and flight director systems) that is slightly different than described above for analog autopilot systems. Functionality is distributed across a number of interrelated computers and includes the use of intelligent servos that handle some of the error correction calculations. The servos communicate with dedicated avionics computers and display unit computers through a control panel, while no central autopilot computer exists. In Fig. 13.6 the automatic flight control system (AFCS) of a Garmin G1000 glass cockpit instrument system for a general aviation aircraft is shown.

13.6. Flight Director Systems

A flight director system is an instrument system consisting of electronic components that compute and indicate the aircraft attitude required to attain and maintain a preselected flight condition. A command bar on the aircraft's attitude indicator shows the pilot how much and in what direction the attitude of the aircraft must be changed to achieve the desired result. The computed command indications relieve the pilot of many of the mental calculations required for instrument flights, such as interception angles, wind drift correction, and rates of climb and descent. Essentially, a flight director system is an autopilot system without the servos.

All of the same sensing and computations are made, but the pilot controls the aeroplane and makes maneuvers by following the commands displayed on the instrument panel. Flight director systems can be part of an autopilot system or exist on aircraft that do not possess full autopilot systems. Many autopilot systems allow for the option of engaging or disengaging a flight director display. Flight director information is displayed on the instrument that displays the aircraft's attitude. The process is accomplished with a visual reference technique. A symbol representing the aircraft is fit into a command bar positioned by the flight director in the proper location for a maneuver to be accomplished. The symbols used to represent the aircraft and the command bar vary by manufacturer. Regardless, the object is always to fly the aircraft symbol into the command bar symbol (Fig. 13.7). The fight director command bar signals the pilot how to steer the aircraft for a maneuver. By flying the aircraft so the triangular aeroplane symbol fits into the command bar, the pilot performs the maneuver calculated by the flight director. The instrument shown on the left is commanding a climb while the aeroplane is flying straight and level. The instrument on the right shows that the pilot has accomplished the maneuver.

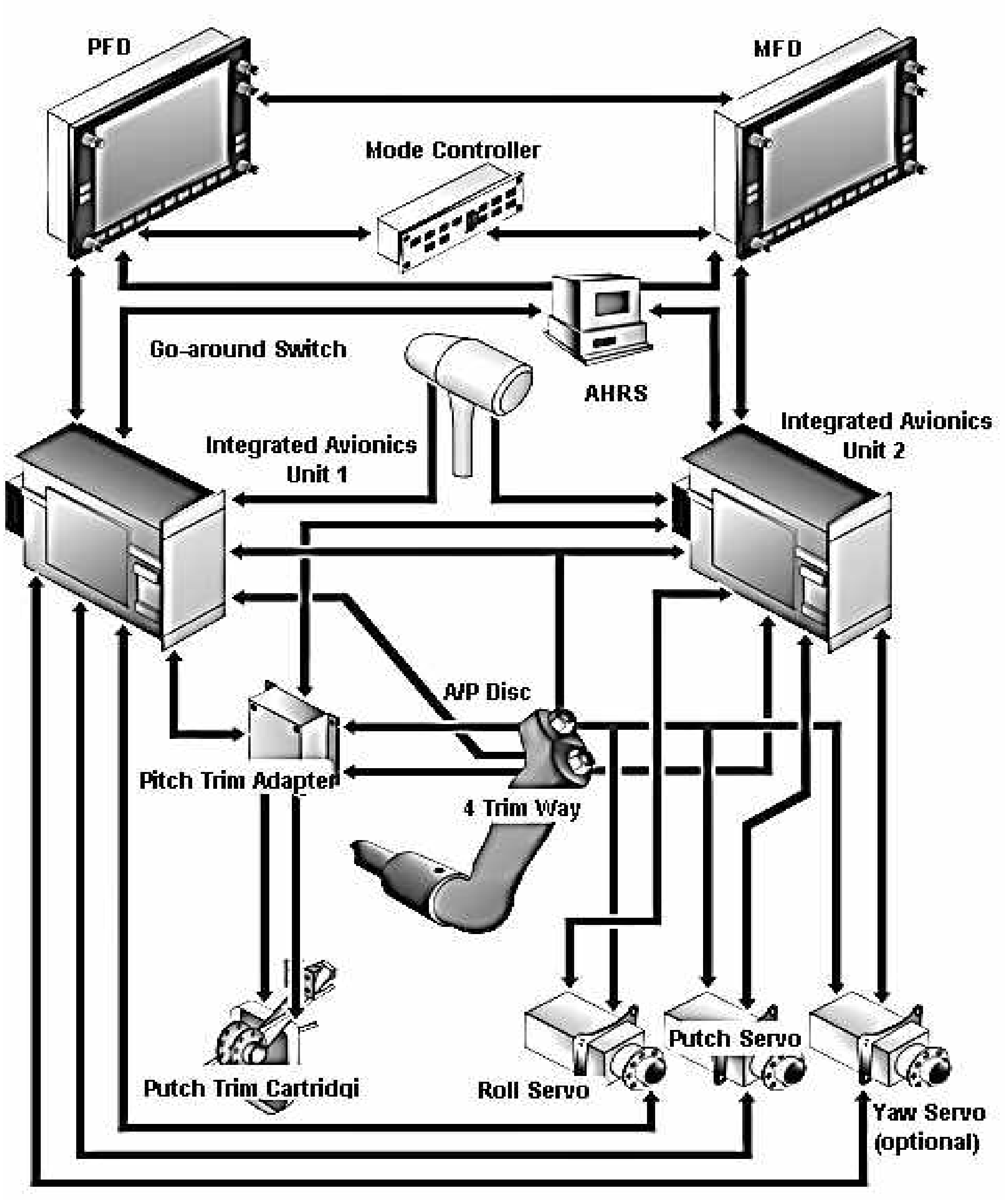
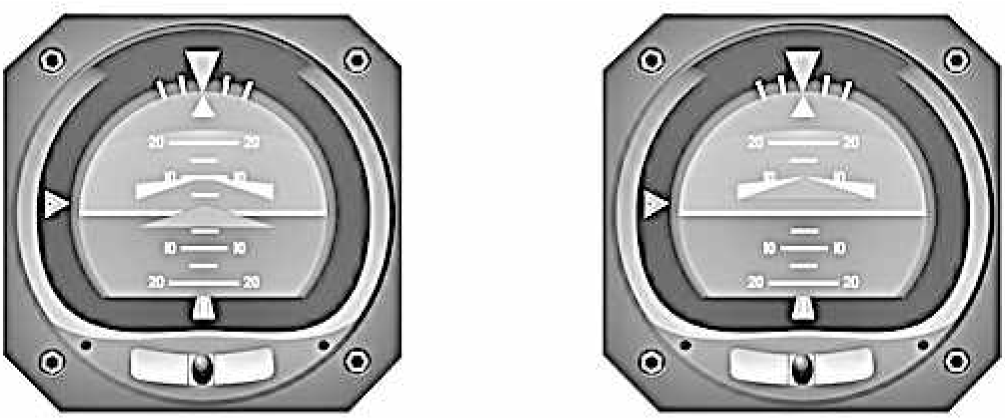


Fig. 13.6

The instrument that displays the flight director commands is known as a flight director indicator (FDI), attitude director indicator (ADI), or electronic attitude director indicator (EADI). It may even be referred to as an artificial horizon with flight director. This display element combines with the other primary components of the flight director system. Like an autopilot, these consist of the sensing elements, a computer, and an interface panel. Integration of navigation features into the attitude indicator is highly useful. The flight director contributes to this usefulness by indicating to the pilot how to maneuver the aeroplane to navigate a desired course. Selection of the VOR function on the flight director control panel links the computer to the omnirange receiver. The pilot selects a desired course and the flight director displays the bank attitude necessary to intercept and maintain this course. Allocations for wind drift and calculation of the intercept angle is performed automatically.



a b

Fig. 13.7

Flight director systems vary in complexity and features. Many have altitude hold, altitude select, pitch hold, and other features. But flight director systems are designed to offer the greatest assistance during the instrument approach phase of flight. ILS localizer and glideslope signals are transmitted through the receivers to the computer and are presented as command indications. This allows the pilot to fly the aeroplane down the optimum approach path to the runway using the flight director system. With the altitude hold function engaged, level flight can be maintained during the maneuvering and procedure turn phase of an approach. Altitude hold automatically disengages when the glideslope is intercepted.

Once inbound on the localizer, the command signals of the flight director are maintained in a centered or zero condition. Interception of the glideslope causes a downward indication of the command pitch indicator. Any deviation from the proper glideslope path causes a fly-up or fly-down command indication. The pilot needs only to keep the aeroplane symbol fit into the command bar.

Topic 14. NAVIGATION SYSTEMS

14.1. The Purpose of Navigation Systems

In the early years of aviation, a compass, a map, and dead reckoning were the only navigational tools. These were marginally reassuring if weather prevented the pilot from seeing the terrain below. Voice radio transmission from someone on the ground to the pilot indicating that the aircraft could be heard overhead was a preview of what electronic navigational aids could provide. For aviation to reach fruition as a safe, reliable, consistent means of transportation, some sort of navigation system needed to be developed. Early flight instruments contributed greatly to flying when the ground was obscured by clouds. Navigation aids were needed to indicate where an aircraft was over the earth as it progressed towards its destination. In the 1930's and 1940's, a radio navigation system was used, that was a low frequency, four-course radio range system. Airports and selected navigation waypoints broadcast two Morse code signals with finite ranges and patterns.

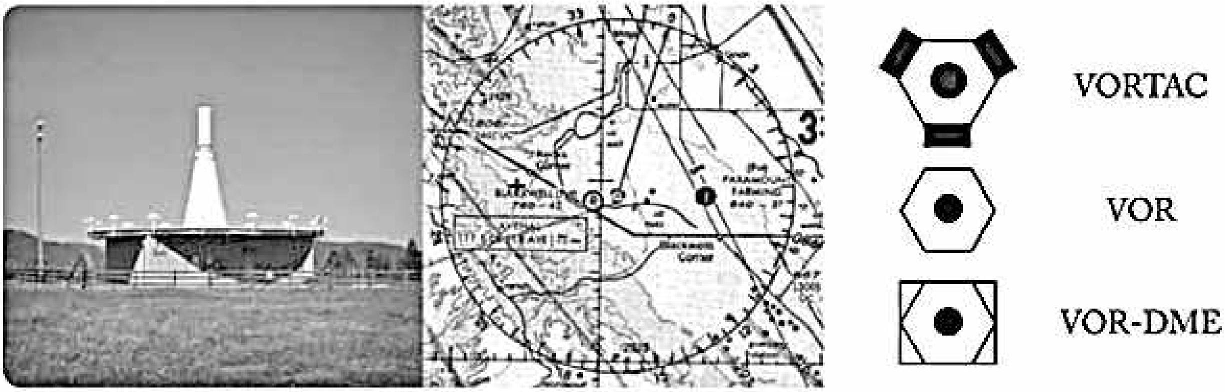
Pilots tuned to the frequency of the broadcasts and flew in an orientation pattern until both signals were received with increasing strength. The signals were received as a blended tone of the highest volume when the aircraft was directly over the broadcast area. From this beginning, numerous refinements to radio navigational aids developed.

Radio navigation aids supply the pilot with intelligence that maintains or enhances the safety of flight. As with communication radios, navigational aids are avionics devices, the repair of which must be carried out by trained technicians at certified repair stations. However, installation, maintenance and proper functioning of the electronic units, as well as their antennas, displays, and any other peripheral devices, are the responsibilities of the airframe technician.

14.2. VOR Navigation System

One of the oldest and most useful navigational aids is the VOR system. The system was constructed after WWII and is still in use today. It consists of thousands of land-based transmitter stations, or VORs, that communicate with radio receiving equipment on board aircraft. Many of the VORs are located along airways. The Victor airway system is built around the VOR navigation system. Ground VOR transmitter units are also located at airports where they are known as TVOR (terminal VOR). The U.S. Military has a navigational system known as TACAN that operates similarly to the VOR system. Sometimes VOR and TACAN transmitters share a location. These sites are known as VORTACs.

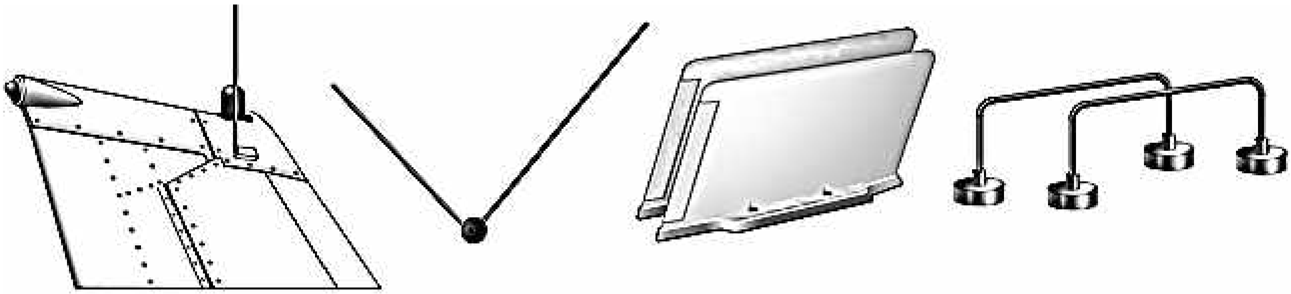
The position of all VORs, TVORs, and VORTACs are marked on aeronautical charts along with the name of the station, the frequency to which an airborne receiver must be tuned to use the station, and a Morse code designation for the station. Some VORs also broadcast a voice identifier on a separate frequency that is included on the chart. In Fig. 14.1 a VOR ground station is shown.



a b c

Fig. 14.1

VOR uses VHF radio waves (108-117.95 MHz) with 50 kHz separation between each channel. This keeps atmospheric interference to a minimum but limits the VOR to line-of-sight usage. To receive VOR VHF radio waves, generally a V-shaped, horizontally polarized, bi-pole antenna is used. A typical location for the V dipole is in the vertical fin. Other type antennas are also certified. Follow the manufacturer’s instructions for installation location. In Fig. 14.2 a V-shaped, horizontally polarized, bi-pole antennas are commonly used for VOR and VOR/glideslope reception. All antenna shown are VOR/glideslope antenna.



a b c d

Fig. 14.2

The signals produced by a VOR transmitter propagate 360° from the unit and are used by aircraft to navigate to and from the station with the help of an onboard VOR receiver and display instruments. A pilot is not required to fly a pattern to intersect the signal from a VOR station since it propagates out in every direction.

The radio waves are received as long as the aircraft is in range of the ground unit and regardless of the aircraft's direction of travel. A VOR transmitter produces signals for 360° radials that an airborne receiver uses to indicate the aircraft's location in relation to the VOR station regardless of the aircraft's direction of flight. The aircraft shown is on the 315° radial even though it does not have a heading of 315° (Fig. 14.3).

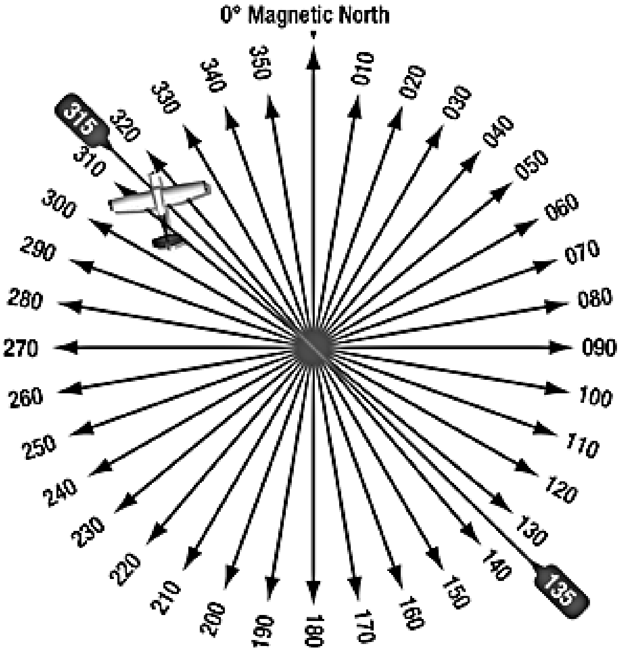
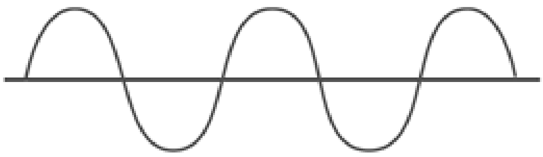


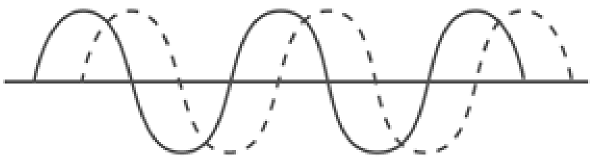
Fig. 14.3

A VOR transmitter produces two signals that a receiver on board an aircraft uses to locate itself in relation to the ground station. One signal is a reference signal. The second is produced by electronically rotating a variable signal. The variable signal is in phase with the reference signal when at magnetic north, but becomes increasingly out of phase as it is rotated to 180°. As it continues to rotate to 360° (0°), the signals become increasingly in phase until they are in phase again at magnetic north. The receiver in the aircraft deciphers the phase difference and determines the aircraft's position in degrees from the VOR ground based unit. The phase relationship of the two broadcast VOR signals is shown in Fig. 14.4.

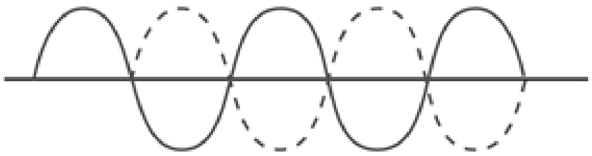
Most aircraft carry a dual VOR receiver. Sometimes, the VOR receivers are part of the same avionics unit as the VHF communication transceiver(s). These are known as NAV/COM radios. Internal components are shared since frequency bands for each are adjacent.



0° Magnetic Radial



90° Magnetic Radial



180° Magnetic Radial

270° Magnetic Radial

Fig. 14.4

Large aircraft may have two dual receivers and even dual antennas. Normally, one receiver is selected for use and the second is tuned to the frequency of the next VOR station to be encountered en route. A means for switching between NAV 1 and NAV 2 is provided as is a switch for selecting the active or standby frequency.

VOR receivers are also found coupled with instrument landing system (ILS) receivers and glideslope receivers. A VOR receiver interprets the bearing in degrees to (or from) the VOR station where the signals are generated. It also produces DC voltage to drive the display of the deviation from the desired course centerline to (or from) the selected station. Additionally, the receiver decides whether or not the aircraft is flying toward the VOR or away from it. These items can be displayed a number of different ways on various instruments. Older aircraft are often equipped with a VOR gauge dedicated to display only VOR information. This is also called an omni-bearing selector (OBS) or a course deviation indicator (CDI).

A separate gauge for the VOR information is not always used. As flight instruments and displays have evolved, VOR navigation information has been integrated into other instruments displays, such as the radio magnetic indicator (RMI), the horizontal situation indicator (HSI), an EFIS display or an electronic attitude director indicator (EADI). Flight management systems and automatic flight control systems are also made to integrate VOR information to automatically control the aircraft on its planned flight segments. Flat panel MFDs integrate VOR information into moving map presentations and other selected displays. The basic information of the radial bearing in degrees, course deviation indication, and to/from information remains unchanged however. A mechanical HSI (left) and an electronic HSI (right) both display VOR information are shown in Fig. 14.5.

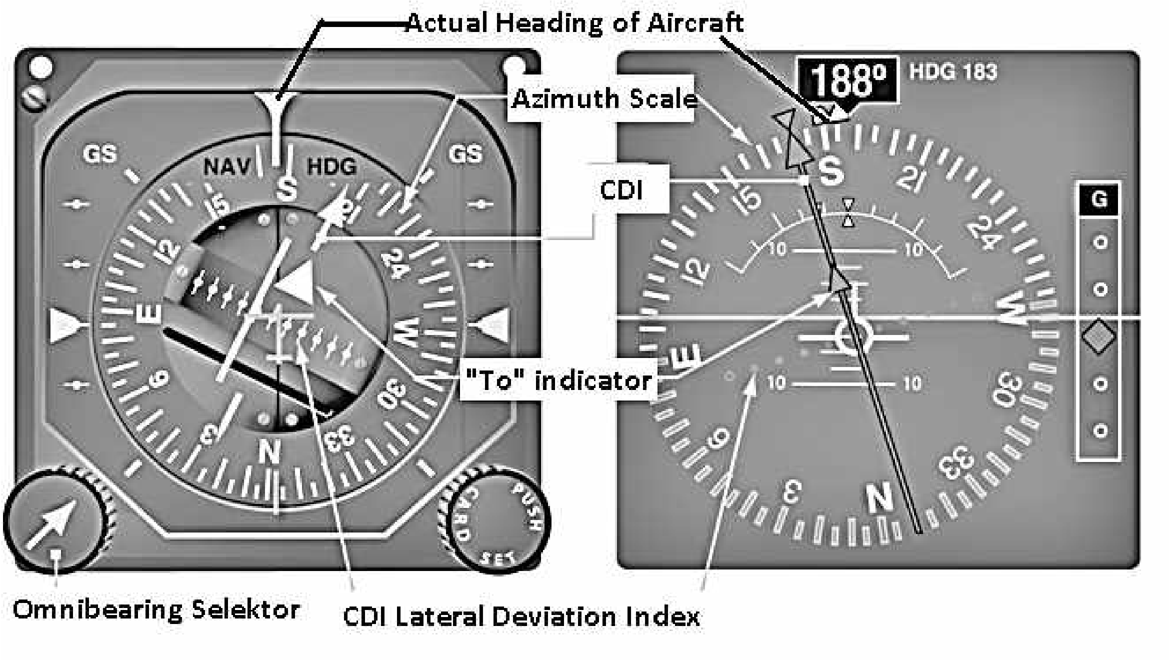


Fig. 14.5

At large airports, an instrument landing system (ILS) guides the aircraft to the runway while on an instrument landing approach. The aircraft's VOR receiver is used to interpret the radio signals. It produces a more sensitive course deviation indication on the same instrument display as the VOR CDI display. This part of the ILS is known as the localizer and is discussed below. While tuned to the ILS localizer frequency, the VOR circuitry of the VOR/ILS receiver is inactive.

It is common at VOR stations to combine the VOR transmitter with distance measuring equipment (DME) or a nondirectional beacon (NDB) such as an ADF transmitter and antenna. When used with a DME, pilots can gain an exact fix on their location using the VOR and DME together. Since the VOR indicates the aircraft's bearing to the VOR transmitter and a co-located DME indicates how far away the station is, this relieves the pilot from having to fly over the station to know with certainty his or her location. These navigational aids are discussed separately in the following sections.

Functional accuracy of VOR equipment is critical to the safety of flight. VOR receivers are operationally tested using VOR test facilities (VOT). These are located at numerous airports that can be identified in the Airport Facilities

Directory for the area concerned. Specific points on the airport surface are given to perform the test. Most VOTs require tuning 108.0 MHz on the VOR receiver and centering the CDI. The OBS should indicate 0° showing FROM on the indicator or 180° when showing TO. If an RMI is used as the indicator, the test heading should always indicate 180°. Some repair stations can also generate signals to test VOR receivers although not on 108.0 MHz. Contact the repair station for the transmission frequency and for their assistance in checking the VOR system.

A logbook entry is required. Note that some airborne testing using VOTs is possible by the pilot. An error of ±4° should not be exceeded when testing a VOR system with a VOT. An error in excess of this prevents the use of the aircraft for IFR fight until repairs are made. Aircraft having dual VOR systems where only the antenna is shared may be tested by comparing the output of each system to the other. Tune the VOR receivers to the local ground VOR station. A bearing indication difference of no more than ±4° is permissible.

14.3. Automatic Direction Finder

An automatic direction finder (ADF) operates off of a ground signal transmitted from a NDB. Early radio direction finders (RDF) used the same principle.

A vertically polarized antenna was used to transmit LF frequency radio waves in the 190 kHz to 535 kHz range. A receiver on the aircraft was tuned to the transmission frequency of the NDB. Using a loop antenna, the direction to (or from) the antenna could be determined by monitoring the strength of the signal received. This was possible because a radio wave striking a loop antenna broadside induces a null signal. When striking it in the plane of the loop, a much stronger signal is induced. The NDB signals were modulated with unique Morse code pulses that enabled the pilot to identify the beacon to which he or she was navigating.

With RDF systems, a large rigid loop antenna was installed inside the fuselage of the aircraft. The broadside of the antenna was perpendicular to the aircraft’s longitudinal axis. The pilot listened for variations in signal strength of the LF broadcast and maneuvered the aircraft so a gradually increasing null signal was maintained. This took them to the transmitting antenna. When over flown, the null signal gradually faded as the aircraft became farther from the station. The increasing or decreasing strength of the null signal was the only way to determine if the aircraft was flying to or from the NDB. A deviation left or right from the course caused the signal strength to sharply increase due to the loop antenna’s receiving properties.

The ADF improved on this concept. The broadcast frequency range was expanded to include MF up to about 1 800 kHz. The heading of the aircraft no longer needed to be changed to locate the broadcast transmission antenna. In early model ADFs, a rotatable antenna was used instead. The antenna rotated to seek the position in which the signal was null. The direction to the broadcast antenna was shown on an azimuth scale of an ADF indicator in the flight deck. This type of instrument is still found in use today. It has a fixed card with 0° always at the top of a non-rotating dial. A pointer indicates the relative bearing to the station. When the indication is 0°, the aircraft is on course to (or from) the station (Fig. 14.6).

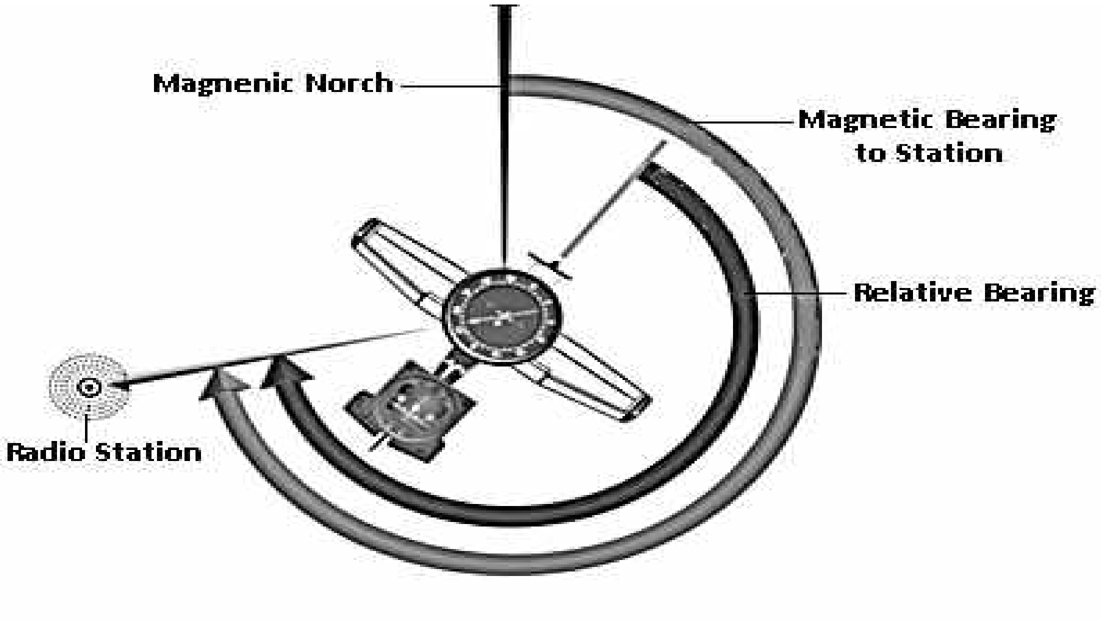


Fig. 14.6

As ADF technology progressed, indicators with rotatable azimuth cards became the norm. When an ADF signal is received, the pilot rotates the card so that the present heading is at the top of the scale. This results in the pointer indicating the magnetic bearing to the ADF transmitter. This is more intuitive and consistent with other navigational practices.

In modern ADF systems, an additional antenna is used to remove the ambiguity concerning whether the aircraft is heading to or from the transmitter. It is called a sense antenna. The reception field of the sense antenna is omnidirectional. When combined with the fields of the loop antenna, it forms a field with a single significant null reception area on one side. This is used for tuning and produces an indication in the direction toward the ADF station at all times.

The onboard ADF receiver needs only to be tuned to the correct frequency of the broadcast transmitter for the system to work. The loop and sense antenna are normally housed in a single, low profile antenna housing (Fig. 14.7). The reception fields of a loop and sense antenna combine to create a field with a sharp null on just one side. This removes directional ambiguity when navigating to an ADF station.

Any ground antenna transmitting LF or MF radio waves in range of the aircraft receiver's tuning capabilities can be used for ADF. This includes those from AM radio stations. Audible identifier tones are loaded on the NDB carrier waves. Typically a two character Morse code designator is used.

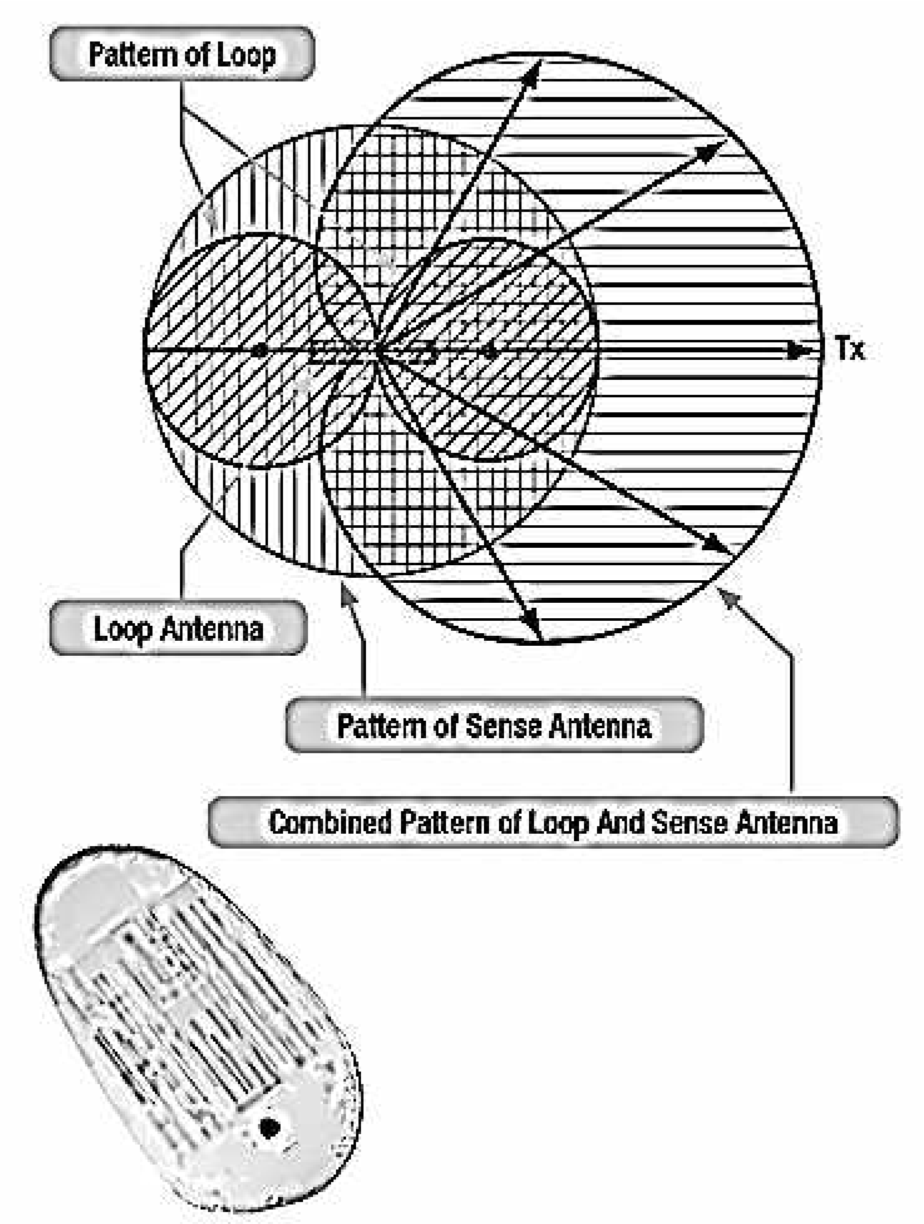


Fig. 14.7

With an AM radio station transmission, the AM broadcast is heard instead of a station identifier code. The frequency for an NDB transmitter is given on an aeronautical chart next to a symbol for the transmitter. The identifying designator is also given. Nondirectional broadcast antenna in the LF and medium frequency range are used for ADF navigation.

ADF receivers can be mounted in the flight deck with the controls accessible to the user. This is found on many general aviation aircraft. Alternately, the ADF receiver is mounted in a remote avionics bay with only the control head in the flight deck. Dual ADF receivers are common. ADF information can be displayed on the ADF indicators mentioned or it can be digital. Modern, flat, multipurpose electronic displays usually display the ADF digitally. When ANT is selected on an ADF receiver, the loop antenna is cut out and only the sense antenna is active. This provides better multi-directional reception of broadcasts in the ADF frequency range, such as weather or AWAS broadcasts.

When the best frequency oscillator (BFO) is selected on an ADF receiver/controller, an internal beat frequency oscillator is connected to the IF amplifier inside the ADF receiver. This is used when an NDB does not transmit a modulated signal. Continued refinements to ADF technology have brought it to its current state. The rotating receiving antenna is replaced by a fixed loop with a ferrite core. This increases sensitivity and allows a smaller antenna to be used. The most modern ADF systems have two loop antennas mounted at 90° to each other. The received signal induces voltage that is sent to two stators in a resolver or goniometer. The goniometer stators induce voltage in a rotor that correlates to the signal of the fixed loops. The rotor is driven by a motor to seek the null. The same motor rotates the pointer in the flight deckindicator to show the relative or magnetic bearing to the station (Fig. 14.8).

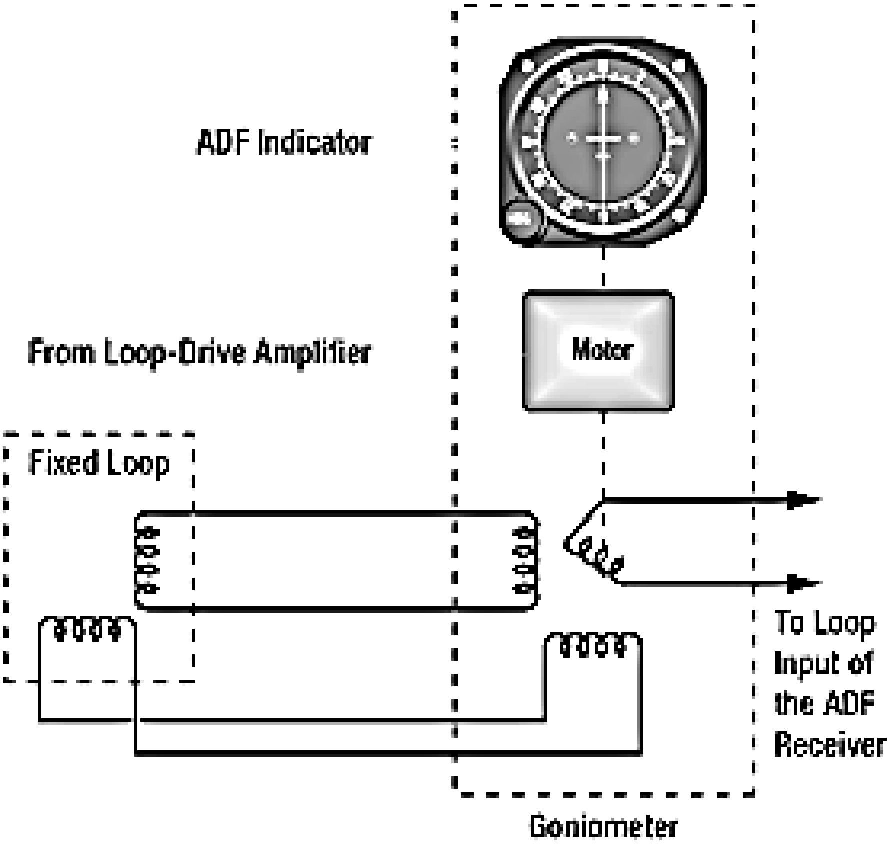


Fig. 14.8

Technicians should note that the installation of the ADF antenna is critical to a correct indication since it is a directional device. Calibration with the longitudinal axis of the fuselage or nose of the aircraft is important. A single null reception area must exist in the correct direction. The antenna must be oriented so the ADF indicates station location when the aircraft is flying toward it rather than away. Follow all manufacturer’s instructions. Radio Magnetic Indicator (RMI) To save space in the instrument panel and to consolidate related information into one easy to use location, the radio magnetic RMI combines indications from a magnetic compass, VOR, and ADF into one instrument.

The azimuth card of the RMI is rotated by a remotely located flux gate compass. Thus, the magnetic heading of the aircraft is always indicated. The lubber line is usually a marker or triangle at the top of the instrument dial. The VOR receiver drives the solid pointer to indicate the magnetic direction TO a tuned VOR station. When the ADF is tuned to an NDB, the double, or hollow pointer, indicates the magnetic bearing TO the NDB.

Since the flux gate compass continuously adjusts the azimuth card so that the aircraft heading is at the top of the instrument, pilot workload is reduced. The pointers indicate where the VOR and ADF transmission stations are located in relationship to where the aircraft is currently positioned. Push buttons allow conversion of either pointer to either ADF or VOR for navigation involving two of one type of station and none of the other.

Topic 15. INSTRUMENT LANDING SYSTEMS

15.1. Radio Navigation Systems

An ILS is used to land an aircraft when visibility is poor.

This radio navigation system guides the aircraft down a slope to the touch down area on the runway. Multiple radio transmissions are used that enable an exact approach to landing with an ILS. A localizer is one of the radio transmissions. It is used to provide horizontal guidance to the center line of the runway. A separate glideslope broadcast provides vertical guidance of the aircraft down the proper slope to the touch down point. Compass locator transmissions for outer and middle approach marker beacons aid the pilot in intercepting the approach navigational aid system.

Marker beacons provide distance from the runway information. Together, all of these radio signals make an ILS a very accurate and reliable means for landing aircraft.

Localizer

The localizer broadcast is a VHF broadcast in the lower range of the VOR frequencies (108 MHz – 111.95 MHz) on odd frequencies only. Two modulated signals are produced from a horizontally polarized antenna complex beyond the far end of the approach runway.

They create an expanding field that is 2½° wide (about 1500 feet) 5 miles from the runway. The field tapers to runway width near the landing threshold. The left side of the approach area is filled with a VHF carrier wave modulated with a 90 Hz signal.

The right side of the approach contains a 150 MHz modulated signal. The aircraft's VOR receiver is tuned to the localizer VHF frequency that can be found on published approach plates and aeronautical charts. The circuitry specific to standard VOR reception is inactive while the receiver uses localizer circuitry and components common to both. The signals received are passed through filters and rectified into DC to drive the course deviation indicator.

If the aircraft receives a 150 Hz signal, the CDI of the VOR/ILS display deflects to the left. This indicates that the runway is to the left. The pilot must correct course with a turn to the left. This centers course deviation indicator on the display and centers the aircraft with the centerline of the runway. If the 90 Hz signal is received by the VOR receiver, the CDI deflects to the right. The pilot must turn toward the right to center the CDI and the aircraft with the runway center line.

Glideslope

The vertical guidance required for an aircraft to descend for a landing is provided by the glideslope of the ILS. Radio signals funnel the aircraft down to the touchdown point on the runway at an angle of approximately 3°. The transmitting glideslope antenna is located off to the side of the approach runway approximately 1 000 feet from the threshold. It transmits in a wedge-like pattern with the field narrowing as it approaches the runway (Fig. 15.1).

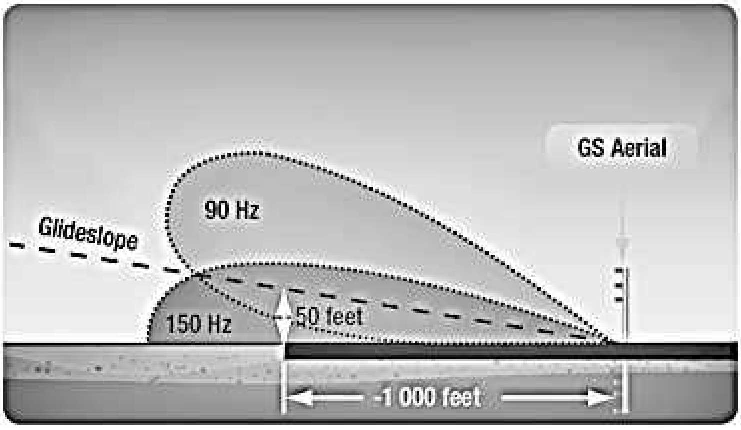


Fig. 15.1

The glideslope transmitter antenna is horizontally polarized. The transmitting frequency range is UHF between 329.3 MHz and 335.0 MHz. The frequency is paired to the localizer frequency of the ILS.

When the VOR/ILS receiver is tuned for the approach, the glideslope receiver is automatically tuned. Like the localizer, the glideslope transmits two signals, one modulated at 90 Hz and the other modulated at 150 Hz. The aircraft's glideslope receiver deciphers the signals similar to the method of the localizer receiver. It drives a vertical course deviation indicator known as the glideslope indicator. The glideslope indicator operates identically to the localizer CDI only 90° to it. The VOR/ ILS localizer CDI and the glideslope are displayed together on whichever kind of instrumentation is in the aircraft.

The UHF antenna for aircraft reception of the glideslope signals comes in many forms. A single dipole antenna mounted inside the nose of the aircraft is a common option. Antenna manufacturers have also incorporated glideslope reception into the same dipole antenna used for the VHS VOR/ILS localizer reception. Blade type antennas are also used.

Compass Locators

It is imperative that a pilot be able to intercept the ILS to enable its use. A compass locator is a transmitter designed for this purpose. There is typically one located at the outer marker beacon 4–7 miles from the runway threshold. Another may be located at the middle marker beacon about 3 500 feet from the threshold. The outer marker compass locator is a 25 watt NDB with a range of about 15 miles. It transmits omnidirectional LF radio waves (190 Hz to 535 Hz) keyed with the first two letters of the ILS identifier.

The ADF receiver is used to intercept the locator so no additional equipment is required. If a middle marker compass locator is in place, it is similar but is identified with the last two letters of the ILS identifier. Once located, the pilot maneuvers the aircraft to fly down the glidepath to the runway.

Marker Beacons

Marker beacons are the final radio transmitters used in the ILS. They transmit signals that indicate the position of the aircraft along the glidepath to the runway. As mentioned, an outer marker beacon transmitter is located 4–7 miles from the threshold. It transmits a 75 MHz carrier wave modulated with a 400 Hz audio tone in a series of dashes. The transmission is very narrow and directed straight up. A marker beacon receiver receives the signal and uses it to light a blue light on the instrument panel. This, plus the oral tone in combination with the localizer and the glideslope indicator, positively locates the aircraft on an approach.

A middle marker beacon is also used. It is located on approach approximately 3 500 feet from the runway. It also transmits at 75 MHz. The middle marker transmission is modulated with a 1 300 Hz tone that is a series of dots and dashes so as to not be confused with the all dash tone of the outer marker. When the signal is received, it is used in the receiver to illuminate an amber- colored light on the instrument panel Various marker beacon instrument panel display lights is shown in Fig. 15.2.

Fig. 15.2

Some ILS approaches have an inner marker beacon that transmits a signal modulated with 3 000 Hz in a series of dots only. It is placed at the land-or-go-around decision point of the approach close to the runway threshold. If present, the signal when received is used to illuminate a white light on the instrument panel. The three marker beacon lights are usually incorporated into the audio panel of a general aviation aircraft or may exist independently on a larger aircraft. Electronic display aircraft usually incorporate marker lights or indicators close to the glideslope display near attitude director indicator.

ILS radio components can be tested with an ILS test unit. Localizer, glideslope, and marker beacon signals are generated to ensure proper operation of receivers and correct display on flight deck instruments (Fig. 15.3).

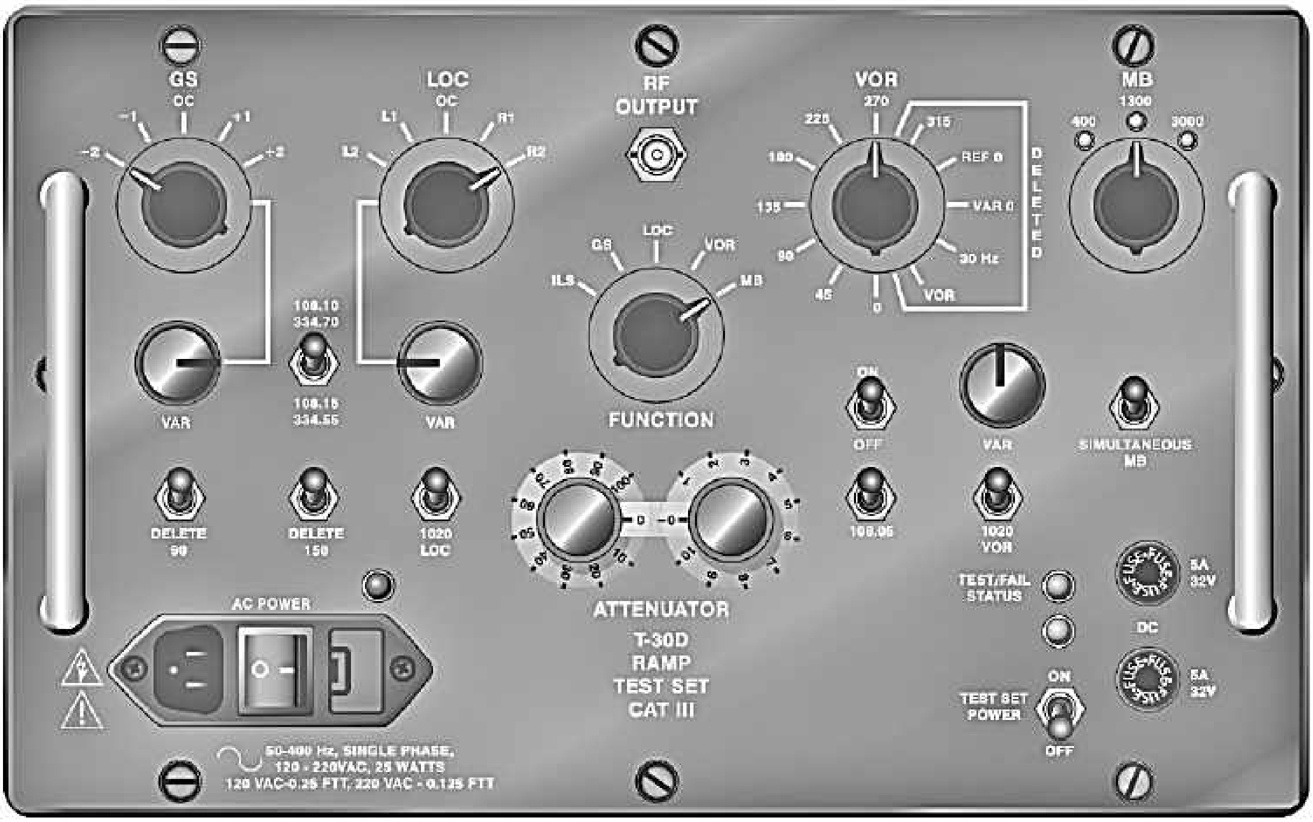


Fig. 15.3

15.2. Distance Measuring Equipment

Many VOR stations are co-located with the military version of the VOR station, which is known as TACAN. When this occurs, the navigation station is known as a VORTAC station. Civilian aircraft make use of one of the TACAN features not originally installed at civilian VOR stations – distance measuring equipment (DME). A DME system calculates the distance from the aircraft to the DME unit at the VORTAC ground station and displays it on the flight deck. It can also display calculated aircraft speed and elapsed time for arrival when the aircraft is traveling to the station.

DME ground stations have subsequently been installed at civilian VORs, as well as in conjunction with ILS localizers. These are known as VOR/DME and ILS/ DME or LOC/DME. The latter aid in approach to the runway during landings. The DME system consists of an airborne DME transceiver, display, and antenna, as well as the ground based DME unit and its antenna (Fig. 15.4).

Fig. 15.4

The DME is useful because with the bearing (from the VOR) and the distance to a known point (the DME antenna at the VOR), a pilot can positively identify the location of the aircraft. DME operates in the UHF frequency range from 962 MHz to 1 213 MHz. A carrier signal transmitted from the aircraft is modulated with a string of integration pulses. The ground unit receives the pulses and returns a signal to the aircraft. The time that transpires for the signal to be sent and returned is calculated and converted into nautical miles for display. Time to station and speed are also calculated and displayed. DME readout can be on a dedicated DME display or it can be part of an EHSI, EADI, EFIS, or on the primary flight display in a glass cockpit.

The DME frequency is paired to the co-located VOR or VORTAC frequency. When the correct frequency is tuned for the VOR signal, the DME is tuned automatically. Tones are broadcast for the VOR station identification and then for the DME. The hold selector on a DME panel keeps the DME tuned in while the VOR selector is tuned to a different VOR. In most cases, the UHF of the DME is transmitted and received via a small blade type antenna mounted to the underside of the fuselage centerline.

A traditional DME displays the distance from the DME transmitter antenna to the aircraft. This is called the slant distance. It is very accurate. However, since the aircraft is at altitude, the distance to the DME ground antenna from a point directly beneath the aircraft is shorter. Some modern DMEs are equipped to calculate this ground distance and display it (Fig. 15.5).

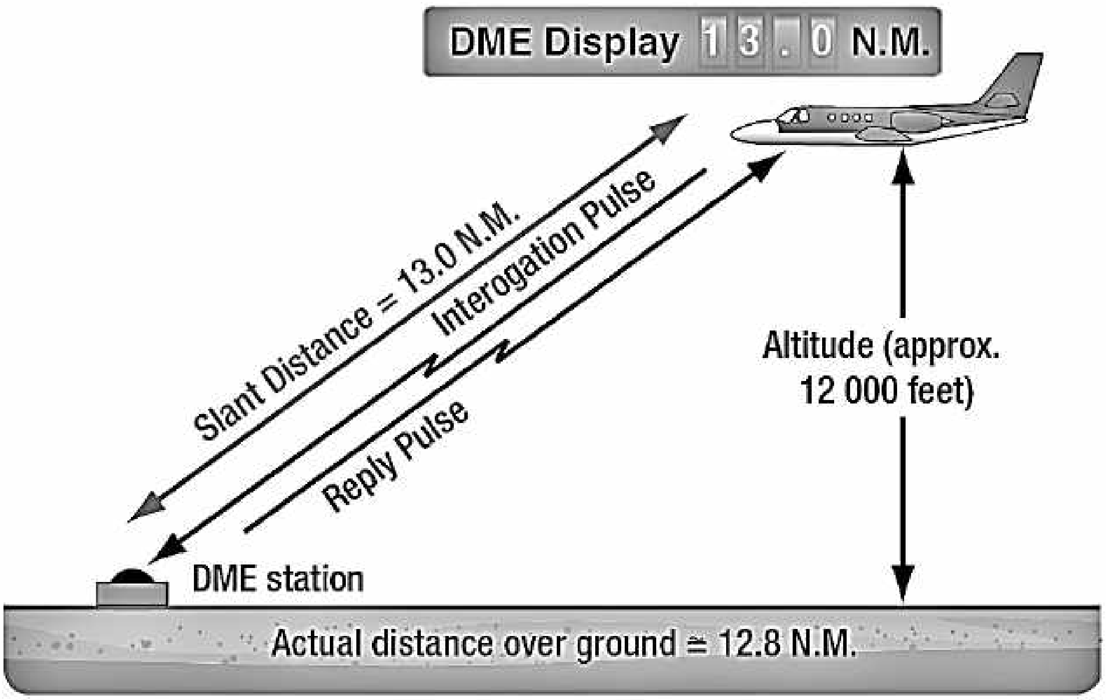


Fig. 15.5

15.3. Area Navigation

Area navigation (RNAV) is a general term used to describe the navigation from point A to point B without direct over flight of navigational aids, such as VOR stations or ADF nondirectional beacons. It includes VORTAC and VOR/DME based systems, as well as systems of RNAV based around LORAN, GPS, INS, and the FMS of transport category aircraft. However, until recently, the term RNAV was most commonly used to describe the area navigation or the process of direct flight from point A to point B using VORTAC and VOR/DME based references which are discussed in this section.

All RNAV systems make use of waypoints. A waypoint is a designated geographical location or point used for route definition or progress-reporting purposes. It can be defined or described by using latitude/longitude grid coordinates or, in the case of VOR based RNAV, described as a point on a VOR radial followed by that pointís distance from the VOR station (i.e., 200/25 means a point 25 nautical miles from the VOR station on the 200° radial).

Fig. 15.6 illustrates an RNAV route of flight from airport A to airport B. The VOR/DME and VORTAC stations shown are used to create phantom waypoints that are overflown rather than the actual stations. This allows a more direct route to be taken. The phantom waypoints are entered into the RNAV course-line computer (CLC) as a radial and distance number pair. The computer creates the waypoints and causes the aircraft's CDI to operate as though they are actual VOR stations. A mode switch allows the choice between standard VOR navigation and RNAV.

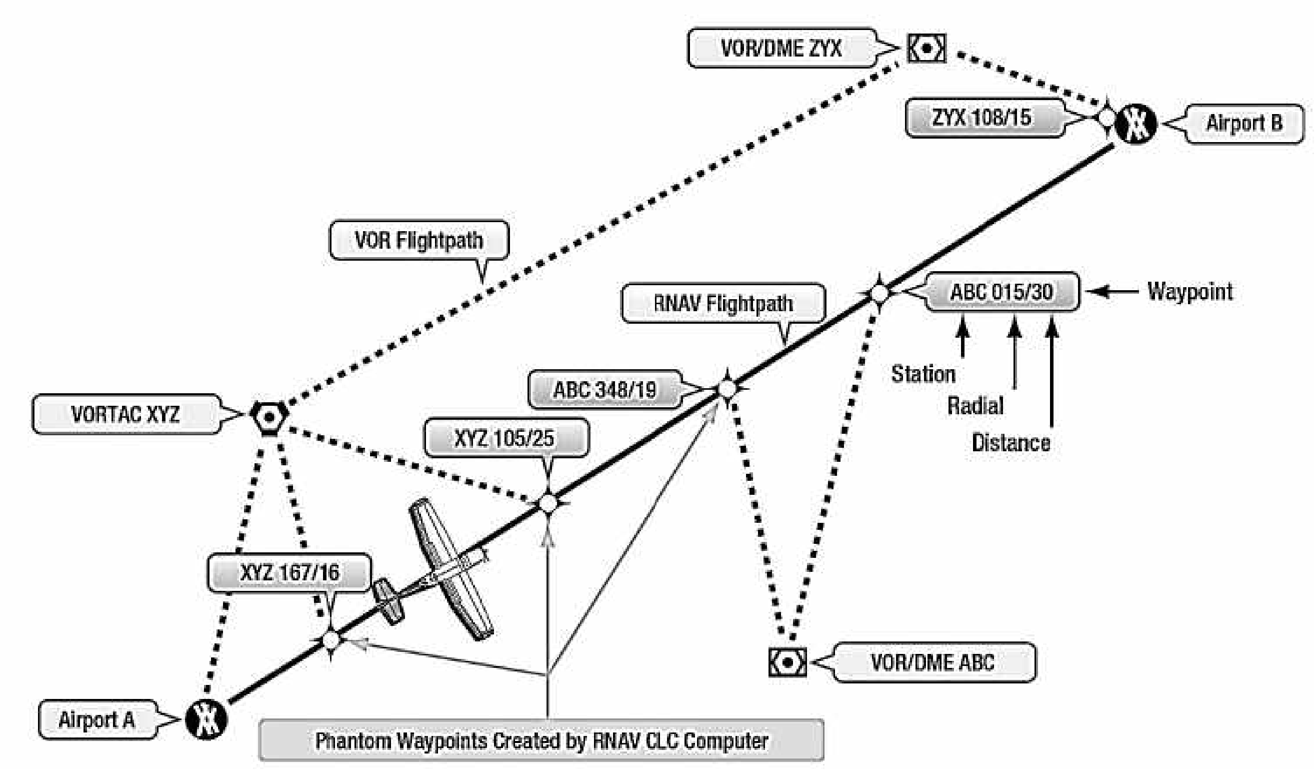


Fig. 15.6

VOR based RNAV uses the VOR receiver, antenna, and VOR display equipment, such as the CDI. The computer in the RNAV unit uses basic geometry and trigonometry calculations to produce heading, speed, and time readouts for each waypoint. VOR stations need to be within line-of sight and operational range from the aircraft for RNAV use.

RNAV has increased in flexibility with the development of GPS. Integration of GPS data into a planned VOR RNAV flight plan is possible as is GPS route planning without the use of any VOR stations.

15.4. Radar Beacon Transponder

A radar beacon transponder, or simply, a transponder, provides positive identification and location of an aircraft on the radar screens of ATC. For each aircraft equipped with an altitude encoder, the transponder also provides the pressure altitude of the aircraft to be displayed adjacent to the on-screen blip that represents the aircraft.

Radar capabilities at airports vary. Generally, two types of radar are used by air traffic control (ATC). The primary radar transmits directional UHF or SHF radio waves sequentially in all directions. When the radio waves encounter an aircraft, part of those waves reflect back to a ground antenna. Calculations are made in a receiver to determine the direction and distance of the aircraft from the transmitter.

A blip or target representing the aircraft is displayed on a radar screen also known as a plan position indicator (PPI). The azimuth direction and scaled distance from the tower are presented giving controllers a two-dimensional fix on the aircraft (Fig. 15.7).

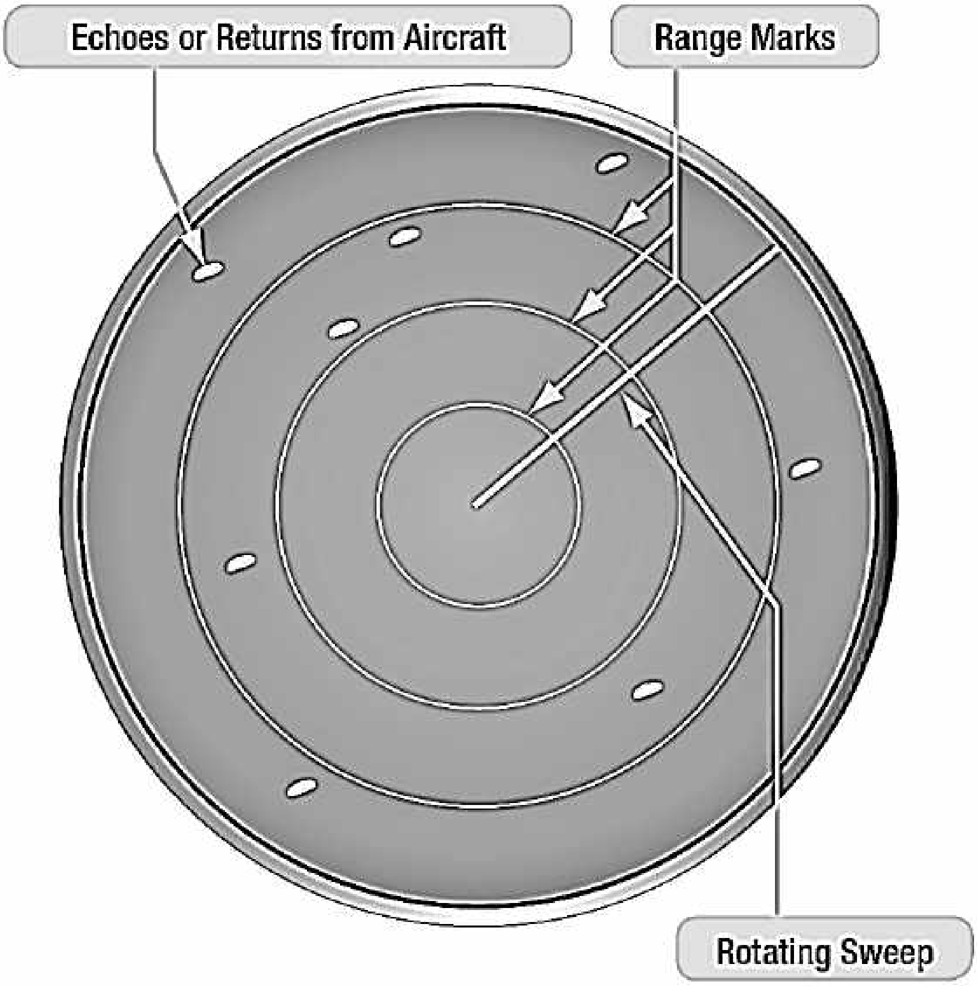


Fig. 15.7

A secondary surveillance radar (SSR) is used by ATC to verify the aircraft’s position and to add the third dimension of altitude to its location. SSD radar transmits coded pulse trains that are received by the transponder on board the aircraft. Mode 3/A pulses, as they are known, aid in confirming the location of the aircraft.

When verbal communication is established with ATC, a pilot is instructed to select one of 4 096 discrete codes on the transponder. These are digital octal codes. The ground station transmits a pulse of energy at 1 030 MHz and the transponder transmits a reply with the assigned code attached at 1 090 MHz. This confirms the aircraft's location typically by altering its target symbol on the radar screen. As the screen may be filled with many confirmed aircraft, ATC can also ask the pilot to ident.

By pressing the IDENT button on the transponder, it transmits in such a way that the aircraft’s target symbol is highlighted on the PPI to be distinguishable.

To gain altitude clarification, the transponder control must be placed in the ALT or Mode C position. The signal transmitted back to ATC in response to pulse interrogation is then modified with a code that places the pressure altitude of the aircraft next to the target symbol on the radar screen. The transponder gets the pressure altitude of the aircraft from an altitude encoder that is electrically connected to the transponder. Typical aircraft transponder antennas are illustrated in Fig. 15.8. Aircraft radar beacon transponder antennas transmit and receive UHF and SHF radio waves.

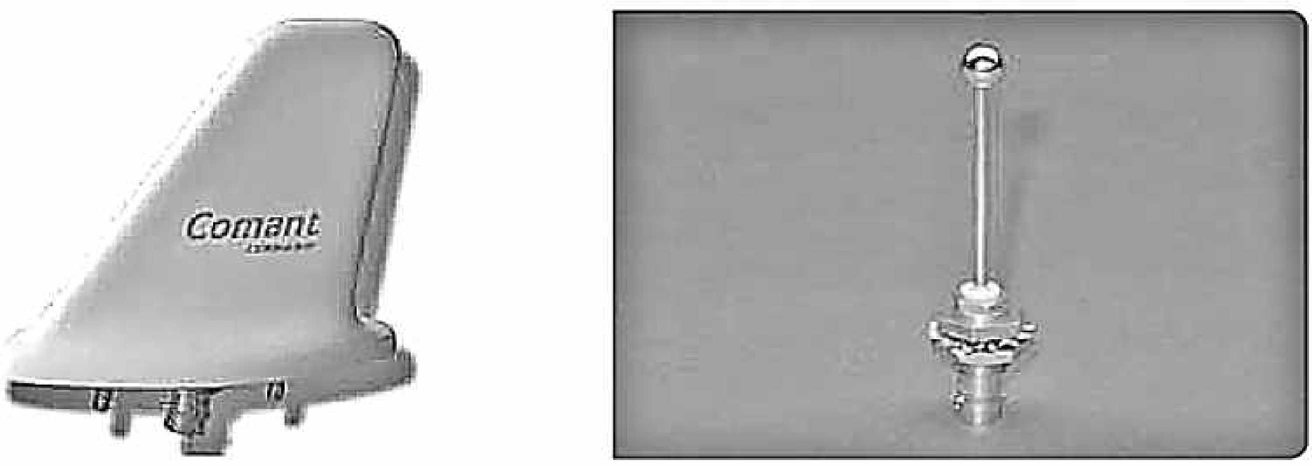


Fig. 15.8

The ATC/aircraft transponder system described is known as Air Traffic Control Radar Beacon System (ATCRBS). To increase safety, Mode S altitude response has been developed. With Mode S, each aircraft is pre-assigned a unique identity code that displays along with its pressure altitude on ATC radar when the transponder responds to SSR interrogation. Since no other aircraft respond with this code, the chance of two pilots selecting the same response code on the transponder is eliminated.

A modern flight data processor computer (FDP) assigns the beacon code and searches flight plan data for useful information to be displayed on screen next to the target in a data block for each aircraft.

Mode S is sometimes referred to as mode select. It is a data packet protocol that is also used in onboard collision avoidance systems. When used by ATC, Mode S interrogates one aircraft at a time. Transponder workload is reduced by not having to respond to all interrogations in an airspace. Additionally, location information is more accurate with Mode S. A single reply in which the phase of the transponder reply is used to calculate position, called monopulse, is sufficient to locate the aircraft. Mode S also contains capacity for a wider variety of information exchange that is untapped potential for the future. At the same time, compatibility with older radar and transponder technology has been maintained.

Transponder Tests and Inspections

Because of the danger involved should a transponder malfunction and, for example, report the wrong altitude information, the functional condition of all transponders is of great concern to aviators. In the U.S., for years the FAA has required transponder testing every 24 months by a certified repair station approved to do such testing. Relatively recent data suggests that such testing may not affect the number of transponder malfunctions. Widespread testing may be more of a problem in that, if not performed with strict adherence to manufacturer's testing guidelines, transponder radio signals may be transmitted into the atmosphere. Errant signal may cause aircraft to take evasive action or divert the attention of the flight crew from other critical matters.

Technician should follow the requirements for periodic testing of transponders issued by the NAA of the country of registration of the aircraft. They should also be sure to comply with any airworthiness directive of that country, EASA, and the country of aircraft manufacture. As with many radio-electronic devices, test equipment exists to test airworthy operation of a transponder.

Operating a transponder in a hangar or on the ramp does not immunize it from interrogation and reply. Transmission of certain codes reserved for emergencies or military activity must be avoided. The procedure to select a code during ground operation is to do so with the transponder in the OFF or STANDBY mode to avoid inadvertent transmission. Code 0000 is reserved for military use and is a transmittable code. Code 7500 is used in a hijack situation and 7600 and 7700 are also reserved for emergency use. Even the inadvertent transmission of code 1200 reserved for VFR flight not under ATC direction could result in evasion action. All signals received from a radar beacon transponder are taken seriously by ATC.

Altitude Encoders

Altitude encoders convert the aircraft's pressure altitude into a code sent by the transponder to ATC. Increments of 100 feet are usually reported. Encoders have varied over the years.

Some are built into the altimeter instrument used in the instrument panel and connected by wires to the transponder. Others are mounted out of sight on an avionics rack or similar out of the way place. These are known as blind encoders. On transport category aircraft, the altitude encoder may be a large black box with a static line connection to an internal aneroid. Modern general aviation encoders are smaller and more lightweight, but still often feature an internal aneroid and static line connection. Some encoders use microtransistors and are completely solid-state including the pressure sensing device from which the altitude is derived. No static port connection is required. Data exchange with GPS and other systems is becoming common.

When a transponder selector is set on ALT, the digital pulse message sent in response to the secondary surveillance radar interrogation becomes the digital representation of the pressure altitude of the aircraft. There are 1 280 altitude codes, one for each 100 feet of altitude between 1 200 feet mean sea level (MSL) and 126 700 feet MSL. Each altitude increment is assigned a code. While these would be 1280 of the same codes used for location and IDENT, the Mode C (or S) interrogation deactivates the 4 096 location codes and causes the encoder to become active. The correct altitude code is sent to the transponder that replies to the interrogation. The SSR receiver recognized this as a response to a Mode C (or S) interrogation and interprets the code as altitude code.

Topic 16. COLLISION AVOIDANCE SYSTEMS, RADIO ALTIMETER, WEATHER RADAR, EMERGENCY LOCATOR TRANSMITTER, GLOBAL POSITIONING SYSTEM, INERTIAL NAVIGATION / INERTIAL REFERENCE SYSTEM

16.1. Collision Avoidance Systems

The ever increasing volume of air traffic has caused a corresponding increase in concern over collision avoidance. no longer adequate in today's increasingly crowded skies. Onboard collision avoidance equipment, long a staple in larger aircraft, is now common in general aviation aircraft. New applications of electronic technology combined with lower costs make this possible.

Traffic Collision Avoidance Systems

Traffic collision avoidance systems (TCAS) are transponder based air-to-air traffic monitoring and alerting systems. There are two classes of TCAS. TCAS I was developed to accommodate the general aviation community and regional airlines. This system identifies traffic in a 35–40 mile range of the aircraft and issues Traffic Advisories (TA) to assist pilots in visual acquisition of intruder aircraft. TCAS I is mandated on aircraft with 10 to 30 seats.

TCAS II is a more sophisticated system. It is required internationally in aircraft with more than 30 seats or weighing more than 15 000 kg. TCAS II provides the information of TCAS I, but also analyzes the projected flightpath of approaching aircraft. If a collision or near miss is imminent, the TCAS II computer issues a Resolution Advisory (RA). This is an aural command to the pilot to take a specific evasive action (i.e., DESCEND). The computer is programmed such that the pilot in the encroaching aircraft receives an RA for evasive action in the opposite direction (if it is TCAS II equipped).

The transponder of an aircraft with TCAS is able to interrogate the transponders of other aircraft nearby using SSR technology (Mode C and Mode S). This is done with a 1030 MHz signal. Interrogated aircraft transponders reply with an encoded 1090 MHz signal that allows the TCAS computer to display the position and altitude of each aircraft. Should the aircraft come within the horizontal or vertical distances an audible TA is announced. The pilot must decide whether to take action and what action to take. TCAS II equipped aircraft use continuous reply information to analyze the speed and trajectory of target aircraft in close proximity. If a collision is calculated to be imminent, an RA is issued.

TCAS target aircraft are displayed on a screen on the flight deck. Different colors and shapes are used to depict approaching aircraft depending on the imminent threat level. Since RAs are currently limited to vertical evasive maneuvers, some stand-alone TCAS displays are electronic vertical speed indicators. Most aircraft use some version of an electronic HSI on a navigational screen or page to display TCAS information.

A multifunction display may depict TCAS and weather radar information on the same screen (Fig. 16.1). TCAS information is displayed on a multifunction display. An open diamond indicates a target; a solid diamond represents a target that is within 6 nautical miles of 1 200 feet vertically. A yellow circle represents a target that generates a TA (25-48 seconds before contact). A red square indicates a target that generates an RA in TCAS II (contact within 35 seconds). A (+) indicates the target aircraft is above and a (-) indicates it is below. The arrows show if the target is climbing or descending.

A TCAS control panel and computer are required to work with a compatible transponder and its antenna(s). Interface with EFIS or other previously installed or selected display(s) is also required. TCAS may be referred to as airborne collision avoidance system (ACAS), which is the international name for the same system. TCAS II with the latest revisions is known as Version 7. The accuracy and reliability of this TCAS information is such that pilots are required to follow a TCAS RA over an ATC command.

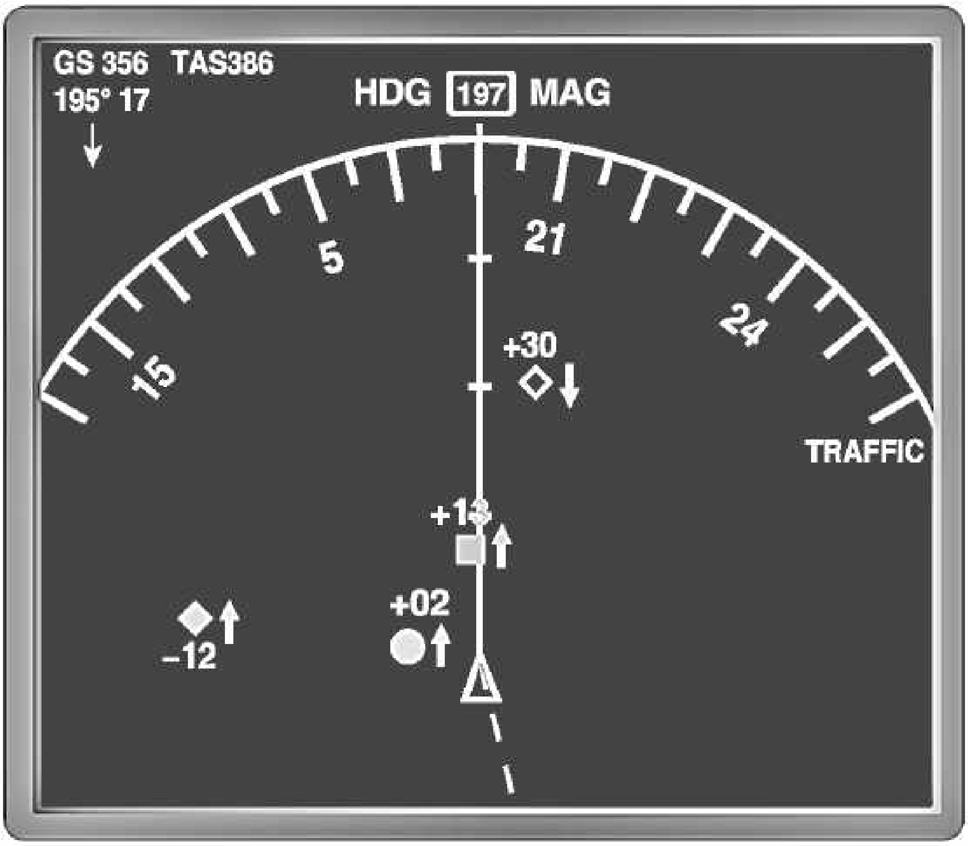


Fig. 16.1

ADS-B

Collision avoidance is a significant part of the FAA's NextGen plan for transforming the U.S. National Airspace System (NAS). Increasing the number of aircraft using the same quantity of airspace and ground facilities requires the implementation of new technologies to maintain a high level of performance and safety. The successful proliferation of global navigation satellite systems (GNSS), such as GPS, has led to the development of a collision avoidance system known as automatic dependent surveillance broadcast (ADS-B). ADS-B is an integral part of NextGen program. The implementation of its ground and airborne infrastructure is currently underway. ADS-B is active in parts of the United States and around the world (Fig. 16.2). Low power requirements allow remote ADS-B stations with only solar or propane support. This is not possible with ground radar due to high power demands which inhibit remote area radar coverage for air traffic purposes.

Fig. 16.2

ADS-B is considered in two segments: ADS-B OUT and ADS-B IN. ADS-B OUT combines the positioning information available from a GPS receiver with on-board flight status information, i.e. location including altitude, velocity, and time. ADS-B OUT uses satellites to identify the position aircraft. It then broadcasts this information to other ADS-B equipped aircraft and ground stations (Fig. 16.3).

Two different frequencies are used to carry these broadcasts with data link capability. The first is an expanded use of the 1 090 MHz Mode-S transponder protocol known as 1 090 ES. The second, largely being introduced as a new broadband solution for general aviation implementation of ADS-B, is at 978 MHz. A 978 universal access transceiver (UAT) is used to accomplish this. An omni- directional antenna is required in addition to the GPS antenna and receiver. Airborne receivers of an ADS-B broadcast use the information to plot the location and movement of the transmitting aircraft on a flight deck display similar to TCAS (Fig. 16.4).

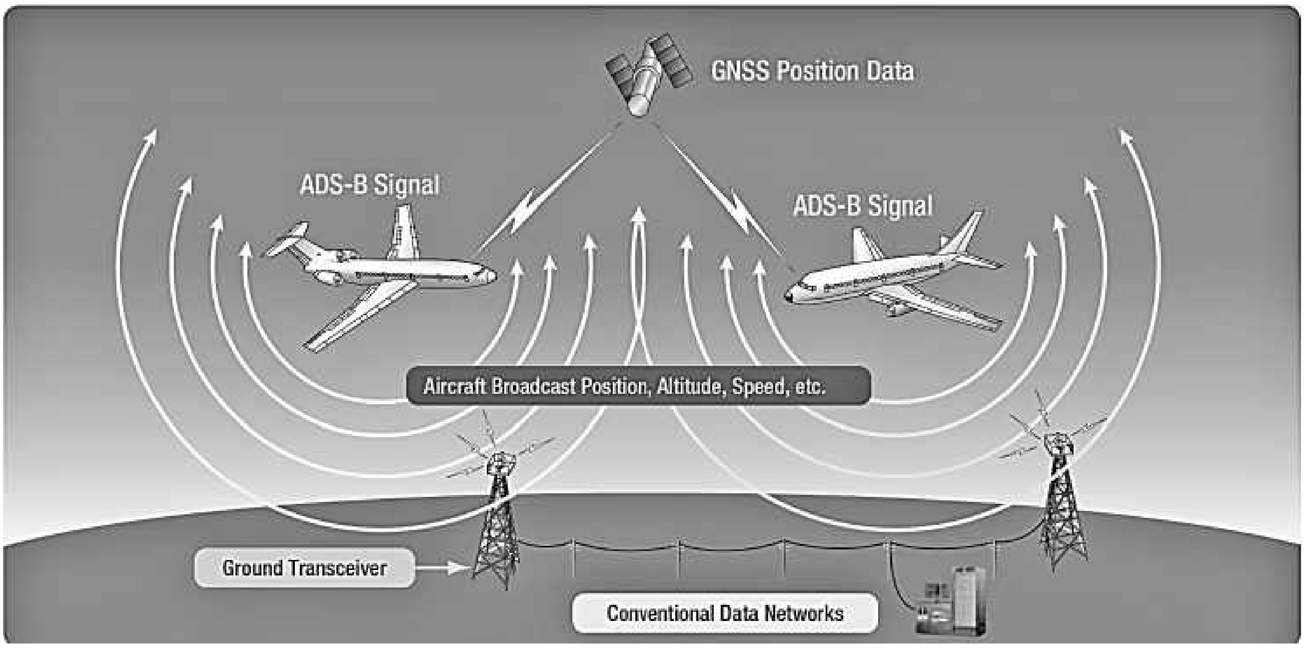


Fig. 16.3

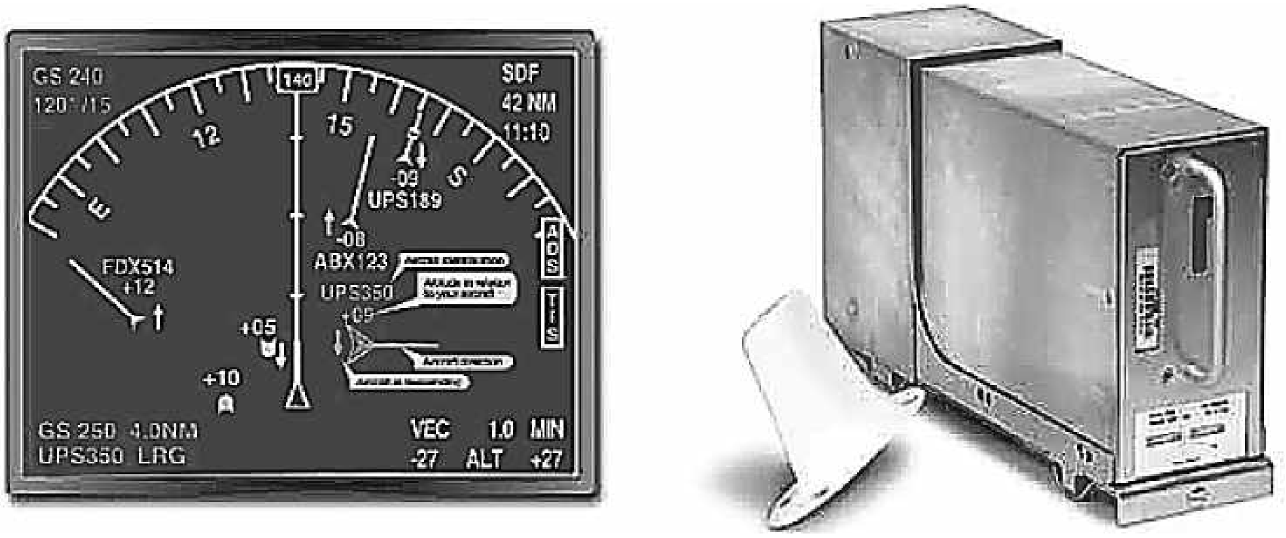


Fig. 16.4

Inexpensive ground stations (compared to radar) are constructed in remote and obstructed areas to proliferate ADS-B. Ground stations share information from airborne ADS-B broadcasts with other ground stations that are part of the air traffic management system (ATMS). Data is transferred with no need for human acknowledgement. Microwave and satellite transmissions are used to link the network.

For traffic separation and control, ADS-B has several advantages over conventional ground-based radar. The first is the entire airspace can be covered with a much lower expense. The aging ATC radar system that is in place is expensive to maintain and replace. Additionally, ADS-B provides more accurate information since the vector state is generated from the aircraft with the help of GPS satellites. Weather is a greatly reduced factor with ADS-B. Ultra high frequency GPS transmissions are not affected. Increased positioning accuracy allows for higher density traffic flow and landing approaches, an obvious requirement to operate more aircraft in and out of the same number of facilities.

The higher degree of control available also enables routing for fewer weather delays and optimal fuel burn rates. Collision avoidance is expanded to include runway incursion from other aircraft and support vehicles on the surface of an airport. ADS-B IN offers features not available in TCAS. Equipped aircraft are able to receive abundant data to enhance situational awareness.

Traffic information services-broadcast (TIS-B) supply traffic information from non-ADS-B aircraft and ADS-B aircraft on a different frequency. Ground radar monitoring of surface targets, and any traffic data in the linked network of ground stations is sent via ADS-B IN to the flight deck. This provides a more complete picture than air-to-air only collision avoidance. Flight information services-broadcast (FIS-B) are also received by ADS-B IN. Weather text and graphics, ATIS information, and NOTAMS (notices to airmen) are able to be received in aircraft that have 987 UAT capability (Fig. 16.5).

ADS-B test units are available for trained maintenance personnel to verify proper operation of ADS-B equipment. This is critical since close tolerance of air traffic separation depends on accurate data from each aircraft and throughout all components of the ADS-B system.

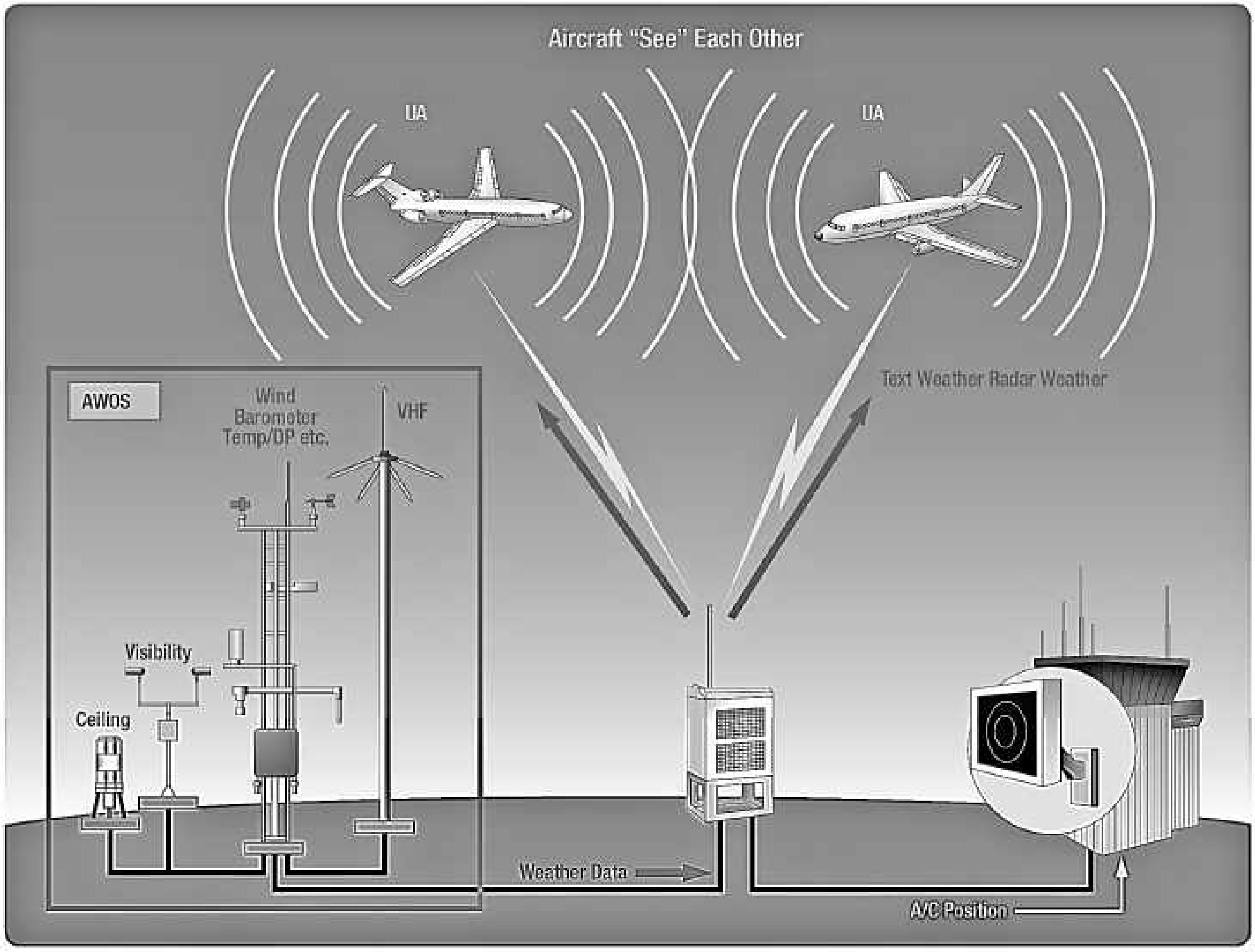


Fig. 16.5

16.2. Radio Altimeter

A radio altimeter, or radar altimeter, is used to measure the distance from the aircraft to the terrain directly beneath it. It is used primarily during instrument approach and low level or night flight below 2500 feet. The radio altimeter supplies the primary altitude information for landing decision height. It incorporates an adjustable altitude bug that creates a visual or aural warning to the pilot when the aircraft reaches that altitude. Typically, the pilot will abort a landing if the decision height is reached and the runway is not visible.

Using a transceiver and a directional antenna, a radio altimeter broadcasts a carrier wave at 4.3 GHz from the aircraft directly toward the ground. The wave is frequency modulated at 50 MHz and travels at a known speed. It strikes surface features and bounces back toward the aircraft where a second antenna receives the return signal. The transceiver processes the signal by measuring the elapsed time the signal traveled and the frequency modulation that occurred. The display indicates height above the terrain also known as above ground level (AGL).

A radar altimeter is more accurate and responsive than an air pressure altimeter for AGL information at low altitudes. The transceiver is usually located remotely from the indicator. Multifunctional and glass cockpit displays typically integrate decision height awareness from the radar altimeter as a digital number displayed on the screen with a bug, light, or color change used to indicate when that altitude is reached.

Large aircraft may incorporate radio altimeter information into a ground proximity warning system (GPWS) which aurally alerts the crew of potentially dangerous proximity to the terrain below the aircraft. A decision height window (DH) displays the radar altitude on the EADI in Fig. 16.6. The decision height, DH200, in the lower right corner of this EADI display uses the radar altimeter as the source of altitude information.

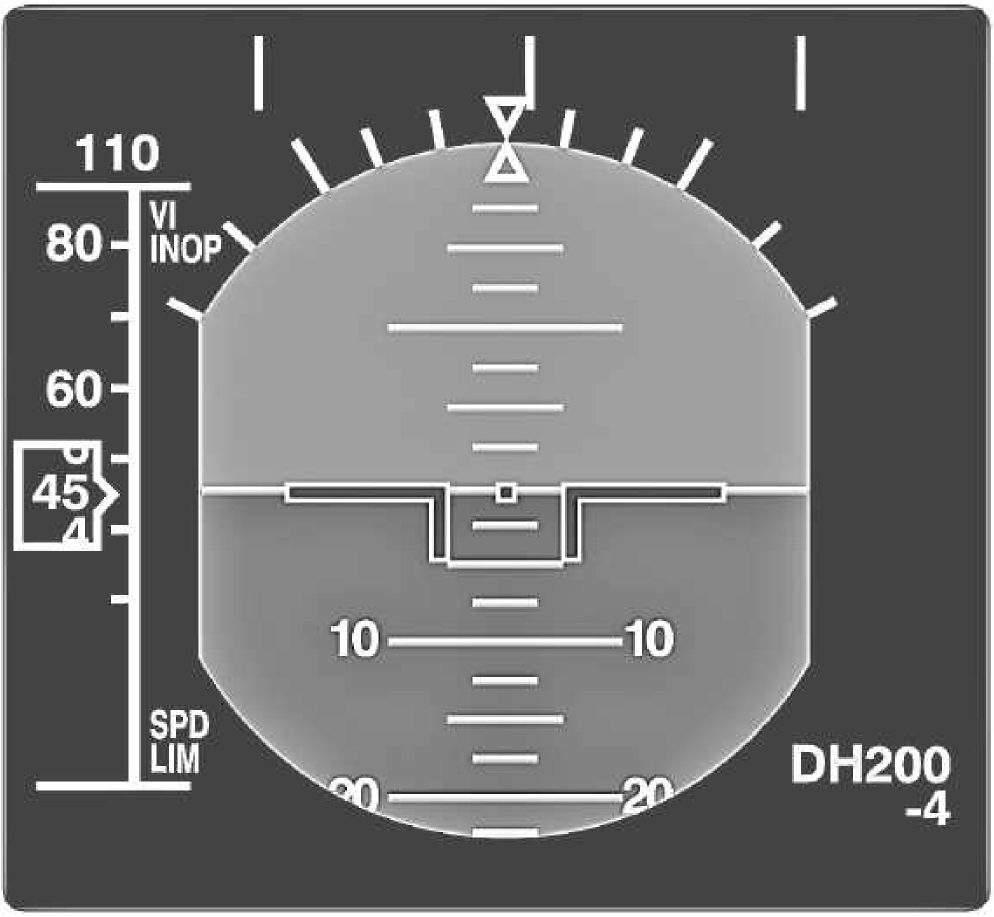


Fig. 16.6

16.3. Weather Radar

There are three common types of weather aids used in an aircraft flight deck that are often referred to as weather radar:

1. Actual on-board radar for detecting and displaying weather activity;
2. Lightning detectors; and
3. Satellite or other source weather radar information that is uploaded to aircraft from an outside source.

On-board weather radar systems can be found in aircraft of all sizes. They function similar to ATC primary radar except the radio waves bounce off of precipitation instead of aircraft. Dense precipitation creates a stronger return than light precipitation. The on-board weather radar receiver is set up to depict heavy returns as red, medium return as yellow and light returns as green on a display in the flight deck. Clouds do not create a return. Magenta is reserved to depict intense or extreme precipitation or turbulence. Some aircraft have a dedicated weather radar screen. Most modern aircraft integrate weather radar display into the navigation display(s).

Radio waves used in weather radar systems are in the SHF range such as 5.44 GHz or 9.375 GHz. They are transmitted forward of the aircraft from a directional antenna usually located behind a nonmetallic nose cone. Pulses of approximately one microsecond in length are transmitted. A duplexer in the radar transceiver switches the antenna to receive for about 2500 micro seconds after a pulse is transmitted to receive and process any returns. This cycle repeats and the receiver circuitry builds a two-dimensional image of precipitation for display. Gain adjustments control the range of the radar. A control panel facilitates this and other adjustments (Fig. 16.7). A typical on-board weather radar system for a high performance aircraft uses a nose-mounted antenna that gimbals. It is usually controlled by the inertial reference system (IRS) to automatically adjust for attitude changes during maneuvers so that the radar remains aimed at the desired weather target. The pilot may also adjust the angle and sweep manually as well as the gain. A dual mode control panel allows separate control and display on the left or right HSI or navigational display.

Severe turbulence, wind shear, and hail are of major concern to the pilot. While hail provides a return on weather radar, wind shear and turbulence must be interpreted from the movement of any precipitation that is detected. An alert is annunciated if this condition occurs on a weather radar system so equipped. Dry air turbulence is not detectable. Ground clutter must also be attenuated when the radar sweep includes any terrain features. The control panel facilitates this. Special precautions must be followed by the technician during maintenance and operation of weather radar systems.

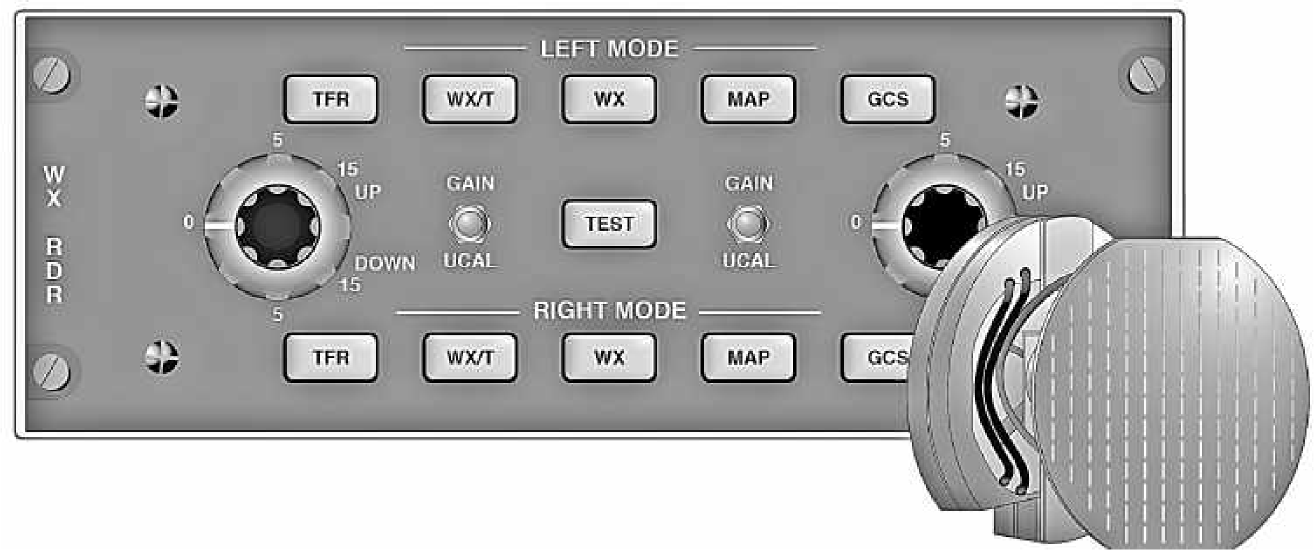


Fig. 16.7

The radome covering the antenna must only be painted with approved paint to allow the radio signals to pass unobstructed. Many radomes also contain grounding strips to conduct lightning strikes and static away from the dome. When operating the radar, it is important to follow all manufacturer instructions. Physical harm is possible from the high energy radiation emitted, especially to the eyes and testes. Do not look into the antenna of a transmitting radar.

Operation of the radar should not occur in hangars unless special radio wave absorption material is used. Additionally, operation of radar should not take place while the radar is pointed toward a building or when refueling takes place. Radar units should be maintained and operated only by qualified personnel.

Lightning detection is a second reliable means for identifying potentially dangerous weather. Lightning gives off its own electromagnetic signal. The azimuth of a lightning strike can be calculated by a receiver using a loop type antenna such as that used in ADF. Some lightning detectors make use of the ADF antenna. The range of the lightning strike is closely associated with its intensity. Intense strikes are plotted as being close to the aircraft.

Stormscope is a proprietary name often associated with lightning detectors. There are others that work in a similar manner. A dedicated display plots the location of each strike within a 200 mile range with a small mark on the screen. As time progresses, the marks may change color to indicate their age. Nonetheless, a number of lightning strikes in a small area indicates a storm cell, and the pilot can navigate around it. Lightning strikes can also be plotted on a multifunctional navigation display.

A third type of weather radar is becoming more common in all classes of aircraft. Through the use of orbiting satellite systems and/or ground up-links, such as described with ADS-B IN, weather information can be sent to an aircraft in flight virtually anywhere in the world. This includes text data as well as real-time radar information for overlay on an aircraft's navigational display(s). Weather radar data produced remotely and sent to the aircraft is refined through consolidation of various radar views from different angles and satellite imagery.

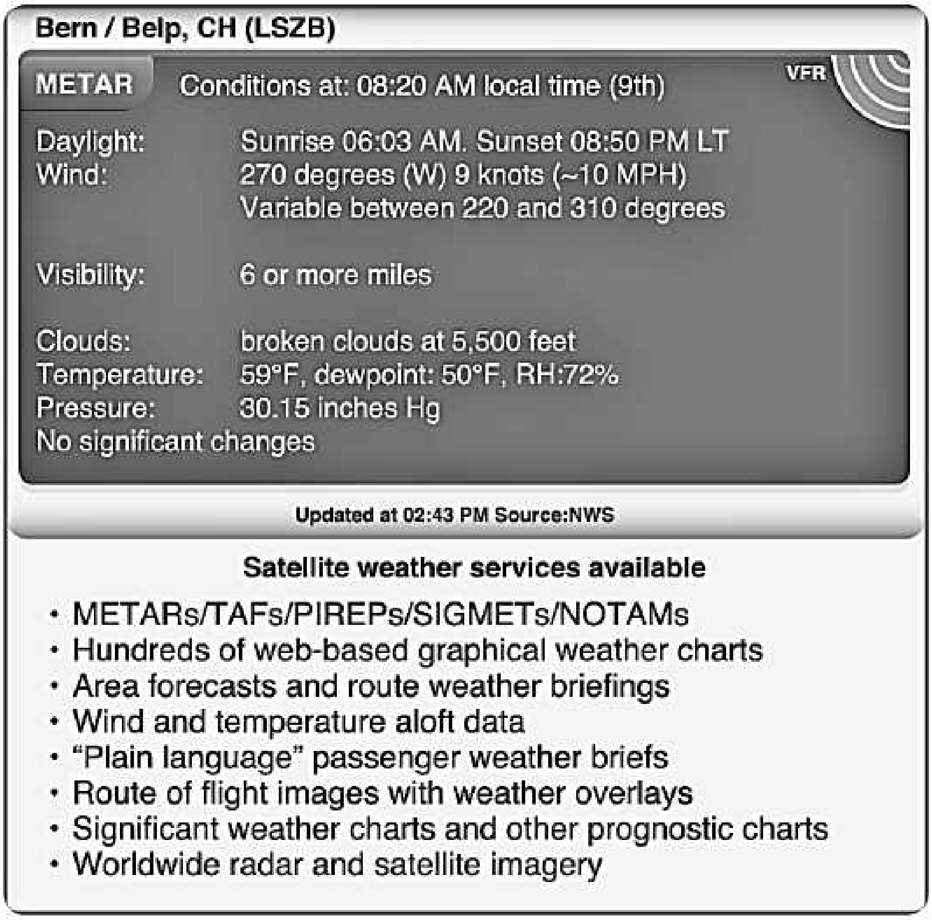
This produces more accurate depictions of actual weather conditions. Terrain databases are integrated to eliminate ground clutter. Supplemental data includes the entire range of intelligence available from the National Weather Service (NWS) and the National Oceanographic and Atmospheric Administration (NOAA).

Fig. 16.8 illustrates a plain language weather summary received in an aircraft along with a list of other weather information available through satellite or ground link weather information services. As mentioned, to receive an ADS-B weather signal, a 1090 ES or 970 UAT transceiver with associated antenna needs to be installed on board the aircraft. Satellite weather services are received by an antenna matched to the frequency of the service. Receivers are typically located remotely and interfaced with existing navigational and multifunction displays. Handheld GPS units also may have satellite weather capability.

16.4. Emergency Locator Transmitter

An emergency locator transmitter (ELT) is an independent battery powered transmitter activated by the excessive G-forces experienced during a crash. It transmits a digital signal every 50 seconds on a frequency of 406.025 MHz at 5 watts for at least 24 hours.

The signal is received anywhere in the world by satellites in the COSPAS-SARSAT satellite system. Two types of satellites, low earth orbiting (LEOSATs) and geostationary satellites (GEOSATs) are used with different, complimentary capability. The signal is partially processed and stored in the satellites and then relayed to ground stations known as local user terminals (LUTs). Further deciphering of a signal takes place at the LUTs, and appropriate search and rescue operations are notified through mission control centers (MCCs) set up for this purpose. NOTE: Maritime vessel emergency locating beacons (EPIRBs) and personal locator beacons (PLBs) use the exact same system. The United States portion of the COSPAS-SARSAT system is maintained and operated by NOAA.



Bern / Belp, CH (LSZB)

METAR Conditions at: 08:20 AM local time (9th)

Daylight: Sunrise 06:03 AM. Sunset 08:50 PM LT
Wind: 270 degrees (W) 9 knots (~10 MPH)
Variable between 220 and 310 degrees

Visibility: 6 or more miles

Clouds: broken clouds at 5,500 feet
Temperature: 59°F, dewpoint: 50°F, RH:72%
Pressure: 30.15 inches Hg
No significant changes

Updated at 02:43 PM Source:NWS

Satellite weather services available

- METARs/TAFs/PIREPs/SIGMETs/NOTAMs
- Hundreds of web-based graphical weather charts
- Area forecasts and route weather briefings
- Wind and temperature aloft data
- "Plain language" passenger weather briefs
- Route of flight images with weather overlays
- Significant weather charts and other prognostic charts
- Worldwide radar and satellite imagery

Fig. 16.8

ELTs are required to be installed in aircraft according to FAR 91.207. This encompasses most general aviation aircraft not operating under Parts 135 or 121.

ELTs must be inspected within 12 months of previous inspection for proper installation, battery corrosion, operation of the controls and crash sensor, and the presence of a sufficient signal at the antenna. Built-in test equipment facilitates testing without transmission of an emergency signal. The remainder of the inspection is visual. Technicians are cautioned to not activate the ELT and transmit an emergency distress signal. Inspection must be recorded in maintenance records including the new expiration date of the battery. This must also be recorded on the outside of the ELT.

ELTs are typically installed as far aft in the fuselage of an aircraft as is practicable just forward of the empennage. The built-in G-force sensor is aligned with the longitudinal axis of the aircraft. Helicopter ELTs may be located elsewhere on the airframe. They are equipped with multidirectional activation devices. Follow ELT and airframe manufacturer’s instructions for proper installation, inspection, and maintenance of all ELTs.

Use of Doppler technology enables the origin of the 406 MHz ELT signal to be calculated within 2 to 5 kilometers. Second generation 406 MHz ELT digital signals are loaded with GPS location coordinates from a receiver inside the ELT unit or integrated from an outside unit. This reduces the location accuracy of the crash site to within 100 meters. The digital signal is also loaded with unique registration information. It identifies the aircraft, the owner, and contact information, etc. When a signal is received, this is used to immediately research the validity of the alert to ensure it is a true emergency transmission so that rescue resources are not deployed needlessly. ELTs with automatic G-force activation mounted in aircraft are easily removable. They often contain a portable antenna so that crash victims may leave the site and carry the operating ELT with them. A flight deck mounted panel is required to alert the pilot if the ELT is activated. It also allows the ELT to be armed, tested, and manually activated if needed.

Modern ELTs may also transmit a signal on 121.5 MHz. This is an analog transmission that can be used for homing. Prior to 2009, 121.5 MHz was a worldwide emergency frequency monitored by the CORPAS- SARSAT satellites. However, it has been replaced by the 406 MHz standard. Transmission on 121.5 MHz are no longer received and relayed via satellite. The use of a 406 MHz ELT has not been mandated by the FAA.

An older 121.5 MHz ELT satisfies the requirements of FAR Part 91.207 in all except new aircraft. Thousands of aircraft registered in the United States remain equipped with ELTs that transmit a 75 watt analog 121.5 MHz emergency signal when activated.

The 121.5 MHz frequency is still an active emergency frequency and is monitored by over flying aircraft and control towers. Technicians are required to perform an inspection/test of 121.5 MHz ELTs within 12 months of the previous one and inspect for the same integrity as required for the 406MHz ELTs mentioned above. However, older ELTs often lack the built-in test circuitry of modern ELTs certified to TSO C-126. Therefore, a true operational test may include activating the signal. This can be done by removing the antenna and installing a dummy load.

Any activation of an ELT signal is required to only be done between the top of each hour and 5 minutes after the hour. The duration of activation must be no longer than three audible sweeps. Contact of the local control tower or flight service station before testing is recommended.

It must be noted that older 121.5 MHz analog signal ELTs often also transmit an emergency signal on a frequency of 243.0 MHz. This has long been the military emergency frequency. Its use is being phased out in favor of digital ELT signals and satellite monitoring. Improvements in coverage, location accuracy, identification of false alerts, and shortened response times are so significant with 406 MHz ELTs, they are currently the service standard worldwide.

16.5. Global Positioning System

Global positioning system navigation (GPS) is the fastest growing type of navigation in aviation. It is accomplished through the use of NAVSTAR satellites set and maintained in orbit around the earth by the U.S. Government. Continuous coded transmissions from the satellites facilitate locating the position of an aircraft equipped with a GPS receiver with extreme accuracy. GPS can be utilized on its own for en route navigation, or it can be integrated into other navigation systems, such as VOR/RNAV, inertial reference, or flight management systems.

There are three segments of GPS: the space segment, the control segment, and the user segment. Aircraft technicians are only involved with user segment equipment such as GPS receivers, displays, and antennas. Twenty-four satellites (21 active, 3 spares) in six separate plains of orbit 12 625 feet above the planet comprise what is known as the space segment of the GPS system. The satellites are positioned such that in any place on earth at any one time, at least four will be a minimum of 15° above the horizon. Typically, between 5 and 8 satellites are in view.

Two signals loaded with digitally coded information are transmitted from each satellite. The L1 channel transmission on a 1 575.42 MHz carrier frequency is used in civilian aviation. Satellite identification, position, and time are conveyed to the aircraft GPS receiver on this digitally modulated signal along with status and other information. An L2 channel 1 227.60 MHz transmission is used by the military. The amount of time it takes for signals to reach the aircraft GPS receiver from transmitting satellites is combined with each satellite’s exact location to calculate the position of an aircraft. The control segment of the GPS monitors each satellite to ensure its location and time are precise. This control is accomplished with five ground-based receiving stations, a master control station, and three transmitting antennas.

The receiving stations forward status information received from the satellites to the master control station. Calculations are made and corrective instructions are sent to the satellites via the transmitters.

The user segment of the GPS is comprised of the thousands of receivers installed in aircraft as well as every other receiver that uses the GPS transmissions. Specifically, for the aircraft technician, the user section consists of a control panel/display, the GPS receiver circuitry, and an antenna. The control, display and receiver are usually located in a single unit which also may include VOR/ILS circuitry and a VHF communications transceiver. GPS intelligence is integrated into the multifunctional displays of glass cockpit aircraft.

The GPS receiver measures the time it takes for a signal to arrive from three transmitting satellites. Since radio waves travel at 186 000 miles per second, the distance to each satellite can be calculated. The intersection of these ranges provides a two-dimensional position of the aircraft. It is expressed in latitude/longitude coordinates. By incorporating the distance to a fourth satellite, the altitude above the surface of the earth can be calculated as well. This results in a three-dimensional fix. Additional satellite inputs refine the accuracy of the position.

Having deciphered the position of the aircraft, the GPS unit processes many useful navigational outputs such as speed, direction, bearing to a waypoint, distance traveled, time of arrival, and more. These can be selected to display for use. Waypoints can be entered and stored in the unit’s memory. Terrain features, airport data, VOR/RNAV and approach information, communication frequencies, and more can also be loaded into a GPS unit. Most modern units come with moving map display capability. A main benefit of GPS use is immunity from service disruption due to weather. Errors are introduced while the carrier waves travel through the ionosphere; however, these are corrected and kept to a minimum. GPS is also relatively inexpensive. GPS receivers for IFR navigation in aircraft must be built to TSO-129A. This raises the price above that of handheld units used for hiking or in an automobile.

But the overall cost of GPS is low due to its small infrastructure. Most of the inherent accuracy is built into the space and control segments permitting reliable positioning with inexpensive user equipment.

The accuracy of current GPS is within 20 meters horizontally and a bit more vertically. This is sufficient for en route navigation with greater accuracy than required. However, departures and approaches require more stringent accuracy. Integration of the wide area augmentation system (WAAS) improves GPS accuracy to within 7.6 meters and is discussed below. The future of GPS calls for additional accuracy by adding two new transmissions from each satellite. An L2C channel will be for general use in non-safety critical application. An aviation dedicated L5 channel will provide the accuracy required for category I, II, and III landings. It will enable the NEXTGEN NAS plan along with ADS-B.

The first replacement NAVSTAR satellites with L2C and L5 capability have already been launched. Full implementation is schedule by 2015.

16.6. Wide Area Augmentation System

To increase the accuracy of GPS for aircraft navigation, the wide area augmentation system (WAAS) was developed. It consists of approximately 25 precisely surveyed ground stations that receive GPS signals and ultimately transmit correction information to the aircraft. An overview of WAAS components and its operation is shown in Fig. 16.9. A WAAS enabled GPS receiver is required for its use as corrective information is sent from geostationary satellites directly to an aircraft’s GPS receiver for use.

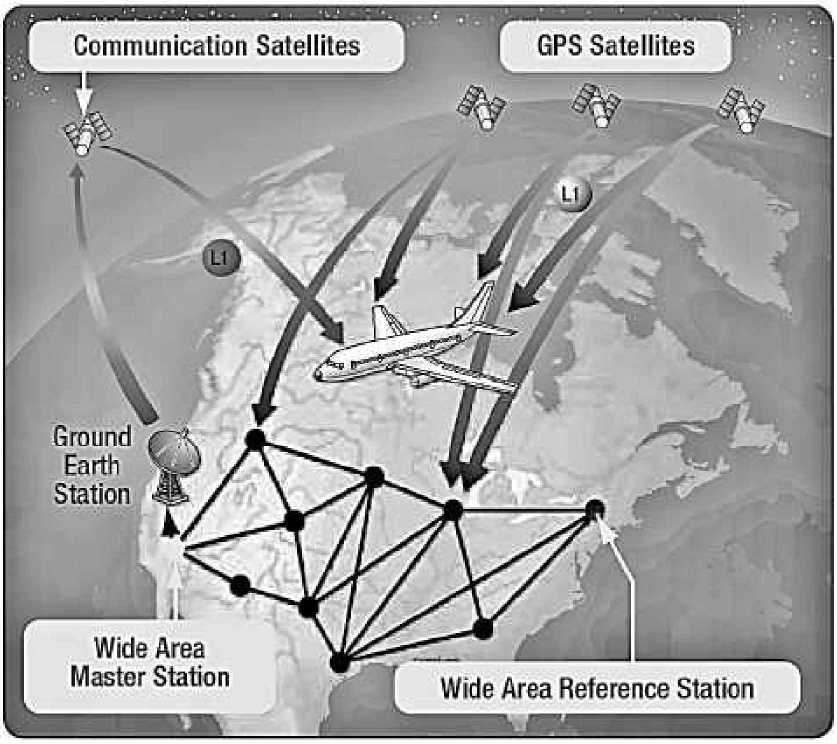


Fig. 16.9

WAAS ground stations receive GPS signals and forward position errors to two master ground stations. Time and location information is analyzed, and correction instructions are sent to communication satellites in geostationary orbit over the NAS. The satellites broadcast GPS-like signals that WAAS enabled GPS receivers use to correct position information received from GPS satellites.

A WAAS enable GPS receiver is required to use the wide area augmentation system. If equipped, an aircraft qualifies to perform precision approaches into thousands of airports without any ground-based approach equipment.

Separation minimums are also able to be reduced between aircraft that are WAAS equipped. The WAAS system is known to reduce position errors to 1-3 meters laterally and vertically.

16.7. Inertial Navigation System / Inertial Reference System

An inertial navigation system (INS) is used on some large aircraft for long range navigation. This may also be identified as an inertial reference system (IRS), although the IRS designation is generally reserved for more modern systems.

An INS/IRS is a self-contained system that does not require input radio signals from a ground navigation facility or transmitter. The system derives attitude, velocity, and direction information from measurement of the aircraft's accelerations given a known starting point. The location of the aircraft is continuously updated through calculations based on the forces experienced by INS accelerometers. A minimum of two accelerometers is used, one referenced to north, and the other referenced to east. In older units, they are mounted on a gyro-stabilized platform. This averts the introduction of errors that may result from acceleration due to gravity. An INS uses complex calculation made by an INS computer to convert applied forces into location information.

An interface control head is used to enter starting location position data while the aircraft is stationary on the ground. This is called initializing. From then on, all motion of the aircraft is sensed by the built-in accelerometers and run through the computer. Feedback and correction loops are used to correct for accumulated error as flight time progresses.

The amount an INS is off in one hour of flight time is a reference point for determining performance. Accumulated error of less than one mile after one hour of flight is possible. Continuous accurate adjustment to the gyro-stabilized platform to keep it parallel to the Earth's surface is a key requirement to reduce accumulated error. A latitude/longitude coordinate system is used when giving the location output.

INS is integrated into an airliner's flight management system and automatic flight control system. Waypoints can be entered for a predetermined flightpath and the INS will guide the aircraft to each waypoint in succession. Integration with other NAV aids is also possible to ensure continuous correction and improved accuracy but is not required. Modern INS systems are known as IRS. They are completely solid-state units with no moving parts.

Three ring, laser gyros replace the mechanical gyros in the older INS platform systems. This eliminates precession and other mechanical gyro shortcomings. The use of three solid-state accelerometers, one for each plane of movement, also increases accuracy. The accelerometer and gyro output are input to the computer for continuous calculation of the aircraft's position.

The most modern IRS integrate is the satellite GPS. The GPS is extremely accurate in itself. When combined with IRS, it creates one of the most accurate navigation systems available. The GPS is used to initialize the IRS so the pilot no longer needs to do so. GPS also feeds data into the IRS computer to be used for error correction. Occasional service interruptions and altitude inaccuracies of the GPS system pose no problem for IRS/GPS. The IRS functions continuously and is completely self contained within the IRS unit. Should the GPS falter, the IRS portion of the system continues without it. The latest electronic technology has reduced the size and weight of INS/IRS avionics units significantly.

16.8. Communication and Navigation Avionics Installations

The aircraft maintenance technician may remove, install, inspect, maintain, and troubleshoot avionics equipment. It is imperative to follow all equipment and airframe manufacturers' instruction when dealing with an aircraft's avionics.

The installation of avionics equipment is partially mechanical, involving sheet metal work to mount units, racks, antennas, and controls. Routing of the interconnecting wires, cables, antenna leads, etc. is also an important part of the installation process. When a location for the equipment is selected by the manufacturer avionics radio equipment is securely mounted to the aircraft. All mounting bolts must be secured by locking devices to prevent loosening from vibration. Adequate clearance between all units and adjacent structure is provided to prevent mechanical damage to electric wiring or to the avionic equipment from vibration, chafing, or landing shock. Combustible materials are kept away from avionics.

The performance and service life of most avionics equipment is seriously limited by excessive ambient temperatures. High performance aircraft with avionics equipment racks typically route air-conditioned air over the avionics to keep them cool. It is also common for non-air conditioned aircraft to use a blower or scooped ram air to cool avionics installations. Measures are also taken to prevent moisture from reaching the avionics equipment. The presence of water in avionics equipment areas promotes deterioration of the exposed components and could lead to failure.

Avionics equipment is sensitive to mechanical shock and vibration and is normally shock mounted to provide some protection against inf light vibration and landing shock. Vibration is a continued motion by an oscillating force. The amplitude and frequency of vibration of the aircraft structure will vary considerably with the type of aircraft. Special shock mounted racks are often used to isolate avionics equipment from vibrating structure (Fig. 16.10).

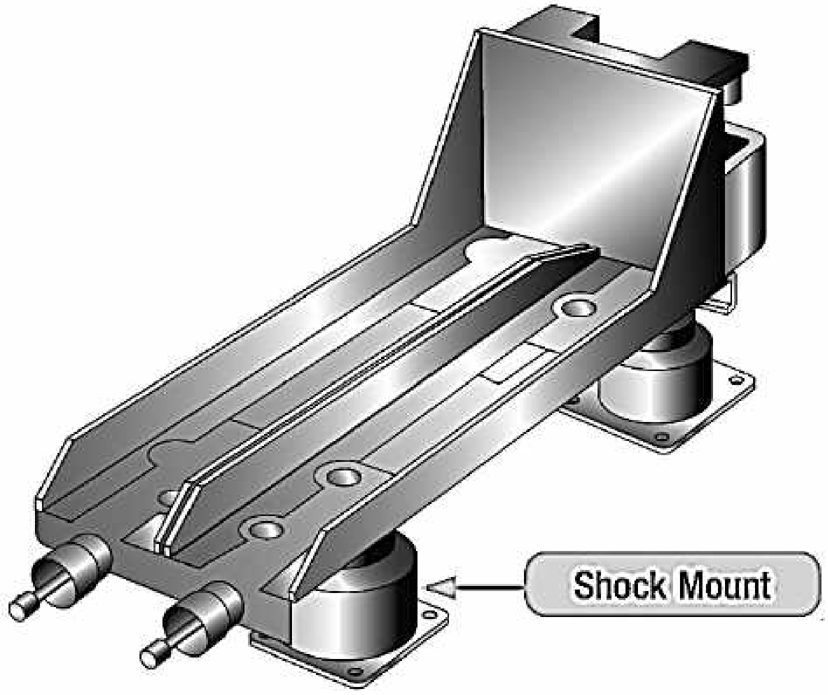


Fig. 16.10

Such mounts provide adequate isolation over the entire range of expected vibration frequencies. Periodic inspection of the shock mounts is required and defective mounts should be replaced with the proper type.

The following factors to observe during inspection are:

1) deterioration of the shock-absorbing material;

2) stiffness and resiliency of the material;

3) overall rigidity of the mount.

If the mount is too stiff, it may not provide adequate protection against the shock of landing. If the shock mount is not stiff enough, it may allow prolonged vibration following an initial shock.

Shock-absorbing materials commonly used in shock mounts are usually electrical insulators. For this reason, each electronic unit mounted with shock mounts must be electrically bonded to a structural member of the aircraft to provide a current path to ground. This is accomplished by secure attachment of a tinned copper wire braid from the component, across the mount, to the aircraft structure. Occasional bonding is accomplished with solid aluminum or copper material where a short flexible strap is not possible.

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СИСТЕМИ ТА ОБЛАДНАННЯ АВІАЦІЙНОЇ ТЕХНІКИ

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