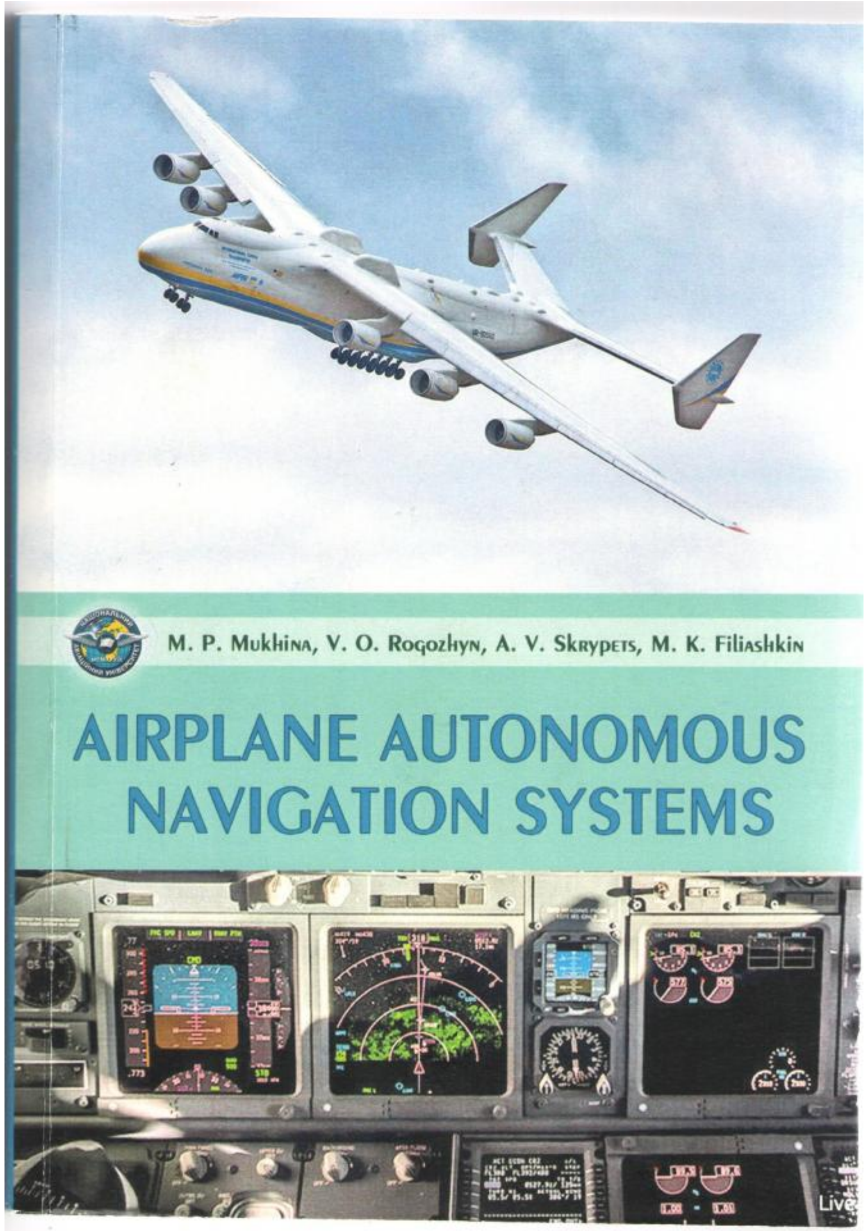

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А 28

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Систематизовано викладено принципи побудови автономних навігаційних систем та особливості їх технічної експлуатації. Розглянуто теоретичні основи навігації, де показано роль автономних навігаційних систем у складі бортового обладнання. Наведено призначення, принципи роботи, структуру побудови та класифікації існуючих систем повітряних сигналів, платформних і безплатформенних інерціальних навігаційних систем, астрономічних засобів навігації, а також радіовисотомірів, доплерівських вимірників швидкості та кута знесення і оглядово-порівняльних систем навігацій зокрема, кореляційно-екстремальних навігаційних систем.

Для студент вищих навчальних закладів. Може бути корисним інженерно-технічному складу авіаційних підприємств, а також інженерам у галузі розроблення та досліджень автономних систем навігації.

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The principles of the construction of autonomous navigation systems and the features of their technical operation are described systematically. The theoretical bases of navigation are considered, which shows the role of autonomous navigation systems in the composition of airborne equipment. The assignments, principles of operation, the structure of construction and classification of existing airdata computer systems, gimbaled and strapdown inertial navigation systems, astronomical means of navigation, as well as radio altimeters, Doppler navigators and overview-comparative navigation systems, in particular, the correlation extreme navigation systems.

For students of higher educational institutions. It may be helpful to the engineering and technical staff of aviation enterprises, as well as engineers in the field of development and research of autonomous navigation systems

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INTRODUCTION

Air navigation is a science of methods and means of guiding aircraft, helicopters, missiles, etc., as well as a set of operations to determine the parameters of the aircraft motion and their use for piloting. The principles of navigation originate from naval navigation in ancient times (from Latin *navigato* – navigation, *navis* – the ship), in particular the method of using magnetic compass and method of astronavigation were developed and used by sailors.

Air navigation provides air traffic on a given trajectory, determined by the route and flight profile, from the take-off of the aircraft in the initial waypoint and to the landing at the final waypoint in a given time. Moreover, the air navigation solves certain navigation tasks: maintaining specified distances and time intervals between the aircraft on the route, preventing collision of aircraft in flight with terrestrial obstacles, converging two aircraft in flight, for example, meeting with a tanker aircraft for refueling, etc.

Different technical means, which can be divided into autonomous and non-autonomous navigation systems, are used to determine the navigational elements (heading, drift angle, track angle, airspeed and ground speed, altitude, coordinates of the aircraft's location, etc.).

Non-autonomous navigation systems include systems which determine the navigational elements of the aircraft flight, using information from the ground aids. The radio beacons are usually used for this. Radio beacons can be ground-based and may also be on board aircraft or spacecraft.

In autonomous navigation systems, all means of measuring the flight and navigation parameters characterizing the flight of the aircraft are located on board the aircraft, so throughout the flight, the aircraft does not depend on the ground aids, airborne, or space navigational devices from outside.

This manual explains the principles of the construction, functioning and operation of autonomous navigation systems (both radio navigation and non-radio navigation systems).

Chapter 1. THEORETICAL BACKGROUND OF NAVIGATION

1.1. Navigation systems in structure of airborne equipment

Development and improvement of aircraft, complication and expansion of realized flight missions cause the corresponding development and improvement of their airborne equipment. The airborne equipment is a set of technical means (instruments, indicators, machines and so on) installed aboard of aircraft. Structurally the airborne equipment is combined in the systems intended for the solution of definite tasks. Since the essential part of practically all aircraft systems is electronics, then all airborne radio electronic equipment is usually called avionics (from aviation electronics).

The structure of avionics for various aircraft types is correspondingly different. The most represented avionics can be found at passenger international aircraft like Airbus A-320, Tu-204, Boeing 757. Avionics can be divided into the following groups: systems of measurement of aircraft engines parameters, automatic flight control systems (AFCS), radio communication systems, electronic flight indication systems (EFIS). Navigation systems comprise the separate large group.

The first group of systems measures the parameters of different general aircraft (helicopter) systems: hydraulic, fuel, conditioning electrical power supply systems and others. Sensors are under the action of physical parameters like pressure, temperature, motion, etc. Output signals of the sensors have electrical nature. By measured electrical signals, the system evaluates the parameter value at the sensor input. The systems of measurement of aviation engines parameters have the same principle of operation.

Automatic flight control systems of aircraft are intended to increase the efficiency of aircraft use and of pilot actions, to minimize the fatigability of crew members and to raise the flight safety. At modern airplanes, AFCS have been transformed from the devices used to facilitate the control process for pilot into the devices which sufficiently increase the economic efficiency and flight safety.

The functionality of AFCS is realized by the performance of the following functions:

- improvement of characteristics of aircraft stability and controllability at manual and combined control (subsystem of manual control);
- automatic stabilization of main piloting parameters (flight computer system);
- automatic and director control at typical flight stages (navigation computer system);
- control of engine thrust to stabilize or to change the flight speed (autothrottle computer system).

Radio communication systems are used for bi-directional information exchange between the aircraft crew and ground radio stations; for bi-directional information exchange between the aircraft crew and other airplanes; for inner communication between crew and passengers.

The obligatory minimum of radio communication equipment of passenger airplanes contains: radio station of microwave communication, radio station of high frequency (HF) communication, radio station of emergency communication. Also, the satellite communication station is installed at many modern airplanes.

Electronic flight indication system is used to indicate the piloting and navigation information.

The main function of ***the inboard warning system*** is to alert the crew about critical and emergency situations during flight. The system indicates the emergency and warning messages. It also uses other methods of alerting with the help of speech and audible (rings, gongs, etc.) signals and also tactile influences (e.g. shaking of control column).

Group of navigation systems includes:

- autonomous navigation systems;
- autonomous radio navigation systems;
- non-autonomous navigation systems;
- other information navigation systems.

The task of ***autonomous navigation systems*** is the measurement of flight and navigation parameters of aircraft without emitting any signals. This type of systems includes airdata computer system (ADC), inertial navigation system (INS), astrocompasses and star trackers.

<u>*Airdata computer system (ADC)*</u> determines flight parameters by performing the measurement in external air environment. It measures and evaluates the following aerometric parameters: barometrical altitude (true and height); rate of climb; indicated and true airspeeds; Mach number; outside air temperature; temperature of fully decelerated airflow; angle of attack and slip angle; static and pitot pressures; maximal allowable airspeed. ADC system also provides warning about exceeding the allowable speed and other aerometric parameters of flight.

Inertial navigation system (INS) is used to measure the angular aircraft orientation in space and to determine its position. It measures the angles: yaw, roll, and pitch; rates of change in angular parameters, linear accelerations (overloads); it also calculates the flight path angle, true heading, ground speed, vertical speed, drift angle, wind parameters, aircraft geographical coordinates. Sometimes, instead of INS for the same purpose it is preferable to use the simpler by design system - attitude and heading reference system (AHRS). AHRS is able to measure only the current angles of aircraft orientation - yaw, pitch and roll.

Astronomical compasses measure the true or great circle heading of aircraft by direction finding of celestial bodies taking into account Earth rotation and location coordinates of aircraft. With their help, it is possible to measure the heading in any regions of Earth, including areas near geographic and magnetic poles, and at all speeds and altitudes of flight.

Star trackers use the method of astronomical orientation based on astronomical measurements of parameters of two stars which are fixed by automatic sextants. In star tracker the computer determines the coordinates of the aircraft location using the relationship between astronomical and navigation coordinate systems.

Autonomous radio navigation systems include systems that operate on the radar location principle; radio altimeter; Doppler navigator; weather radar.

These devices do not use radio beacons, but they get information from their own radio signal reflected by ground surface or by meteorological formations.

Radio altimeter (RA) measures the height. There are two types of RA, for low and large heights. Radio altimeter of low heights generates the radio signal with the frequency that varies linearly in the range 4200-4400 MHz. The signal reflected by the ground or water is received back, then its frequency is measured and compared to the frequency of signal generated at this moment. The frequency difference is proportional to the measured distance.

Doppler navigator (DN) measures the parameters of aircraft velocity vector: ground speed, i.e. the speed relative to the ground, and the drift angle - the angle between the direction of the aircraft longitudinal axis and the actual direction of aircraft motion.

DN uses the directional radiation of the ground surface and determines the parameters of velocity vector by the frequency spectrum of the signal reflected by the ground. Due to Doppler phenomena there is a frequency shift of radiated and reflected signals. To improve the accuracy DN radiates not one but 3 or 4 beams in different directions.

Meteorological radar station (MRS) allows detecting areas of thunderstorm activity and avoiding them. To solve the navigation tasks there is ground-mapping mode.

During flight the MRS antenna scans leftwards and rightwards from the flight direction in the prescribed range. In the scanning range the weather radar radiates many individual radio beams; at receiving each beam is divided by points separated one after another and then for each point the level of reflected signal is measured. This level indicates the presence and density of clouds and turbulence.

To speed up the scanning the pilot can reduce the scanning range from control panel. The pilot can also set the antenna angle in the range ± 15° degrees from the aircraft horizontal axis. This allows avoiding noises and improving image sharpness, scanning the vertical structure of clouds. By the downward inclination of antenna it is possible to use MRS for scanning (mapping) the relief of ground surface for the purpose of navigation.

Autonomous radio navigation systems also include *correlation extreme navigation system*. Principles of their operation are based on scanning the surrounding area and the comparison of its image with reference map or landmarks system saved in the memory of onboard computer. Radar stations and, in particular MRS, are usually a part of correlation extreme systems.

The main disadvantage of correlation extreme navigation systems is their dependence on external information, access to which may be limited by natural and artificial factors.

Non-autonomous radio navigation systems include systems which determine the aircraft location, using signals from radio beacons. Radio beacons used by non-autonomous radio navigation systems can be ground or located on board the aircraft or spacecraft.

Ground radio beacons are used to control the aircraft motion at en-route flight and at landing approach to destination aerodrome. They are located at ground waypoints and in the aerodrome zone. The signal radiated or retransmitted by radio beacon is fixed by the onboard receiver. By measuring the signal parameters, the receiver determines the beacon direction and range or the cross-track error. Radio beacons are typically used to provide the flight to or from beacon. However, by two separated radio beacons it is possible to determine the current aircraft location.

Non-autonomous radio navigation systems include automatic direction finder; very high frequency omnidirectional range (VOR) system; distance measurement equipment (DME); instrument landing system (ILS); microwave landing system (MLS); radio system of short range navigation; satellite navigation system.

Automatic direction finder (ADF) is used for navigation by signals of marker or broadcasting radio beacons. ADF has two antennas (loop and sense antennas) and receiver. The operation principle of ADF is based on the comparison of amplitudes and phases of the signals obtained from loop and sense antennas, then this information is used to calculate the direction to the marker radio beacon, that is the relative bearing.

Very high frequency omnidirectional range (VOR) system determines the azimuth of the aircraft relative to the point of radio beacon location. The beacon antenna generates two antenna patterns: directional and non-directional. The sense antenna radiates the reference signal modulated at 30 Hz. The directional antenna pattern is rotated with frequency 30⌠/s. The aircraft receiver obtains both signals; the signal from loop antenna is modulated by amplitude (the signal maximum appears when the antenna is directed to aircraft). The phase of reference signal coincides with the phase of envelope amplitude-modulated signal in case when the azimuth equals zero. It allows measuring the current azimuth.

VOR receiver takes the signals of marker radio beacons. These signals clearly show the distance from the runway.

Distance measurement equipment (DME) is used to determine the slant distance to radio beacon. The equipment includes the interrogator block and slot antenna.

The interrogating impulse is sent from aircraft. This request is received at the ground equipment and then transmitted back as coded signal with the prescribed constant delay. By measuring the interval between signals, DME determines the slant range.

Instrument landing system (ILS) operates by radio beacons of metric wavelength of ILS ("СП" in Ukraine) type and determines the aircraft deviation from course and from glide slope during landing approach by radio beacon information. At the aerodrome two radio beacons are installed: localizer and glide slope beacons.

The localizer beacon sets the plane of landing course by equisignal method. It forms this plane as intersection of two antenna patterns in horizontal plane. The beacon is located in order to form the plane coinciding with center axis of runway. The airborne receiver measures the difference of modulation depth (DMD) of received signals. In the runway heading the DMD equals zero. It increases proportionally to the error of runway heading. Information about the error enters the indication system and is shown at display to the pilot. The error of runway heading provides information about the precision of landing approach and sign of deviation from prescribed heading direction - rightwards or leftwards.

The glide slope beacon sets the glide path, allowing the pilot to maintain the desired descending angle. Glide slope is given by equisignal method too.

The disadvantage of operation in meter wavelength range is a strong influence of the reflected signals and as a result - errors of aircraft guidance to runway. This drawback is significantly reduced in *Microwave Landing System* (MLS) working in the centimeter wavelength range. MLS performs the same function as ILS: receives signals of two MLS beacons located at the aerodrome. One of them specifies the trajectory of runway approach by elevation angle, and the second works by azimuth.

In comparison with ILS, the MLS has the following advantages: smaller dependence on relief and obstacles, wider angular size of operation zone, higher accuracy of localization .

Radio system of short range navigation (RSSRN) is an analogue of VOR/DME systems. It was used in USSR for navigation support of en-route flights and to guide aircraft in the operation zone of aircraft landing systems. Radio beacons of RSSRN provide information on polar coordinates of aircraft relative to the given beacon, namely azimuth and slant range. In comparison with VOR/DME the system additionally determines the azimuth and slant range at the ground and can be used for aircraft identification upon air traffic controller request. The action principle of range measurement channel is the same as in DME.

Besides azimuth and range the RSSRN can receive error signals from the axis of equisignal zones of localizer and glide slope beacons, and also call signals from ground radio beacons.

Satellite navigation system (SNS) provides navigation data to pilot and other systems. These data are obtained by measuring signals from navigation aerospace satellites. SNS determines three aircraft coordinates (latitude, longitude and altitude) and three components of the velocity vector. To do this, SNS is adjusted to orbital satellites grouping. Due to use of on-board atomic standards of frequency, the reciprocal synchronization of navigation radio signals is provided for orbital grouping.

The navigation measurements are based on the determination of range to satellites whose coordinates of current position are known precisely. The range determination is done by measured delay of received code relatively the same code formed in airborne equipment. The velocity determination is done by the measured Doppler frequency shift of received signal relatively the frequency of reference generator.

Besides named above systems the *radio systems of long range navigation* (RSLRN) can be used. They determine the geographical coordinates of aircraft by signals of ground phase radio navigation systems like РСДН-20, «Omega» or impulse-phase systems like РСДН-3, РСДН-10, «Loran-C». These stations work in long wavelength range and provides aircraft localization at large coverage - hundreds and thousands kilometers, that is, worldwide. However, SNS gradually replace RSLRN, receivers are not installed on new aircraft and ground stations are being unmounted.

Other information navigation systems

Collision avoidance system (CAS) determines the position of other airplanes relative the given. The purpose of such system is to avoid aircraft-to-aircraft collisions. It is possible only for those airplanes which have the same system aboard. That is why the USA and Europe have the presence of CAS of TCAS (Traffic Collision Avoidance System) type as compulsory condition to perform flights at their airspace.

Ground proximity warning system (GPWS) warns the crew about the danger of aircraft collision with ground surface. The system continuously tracks the aircraft location relatively the ground in order to indentify the dangerous tendency and to warn the crew in advance. The crew warning is generated in the following cases: when aircraft approaches the ground surface too close or to quick; when it loses the altitude during the takeoff or the go-around maneuver; when it flies near the ground surface in not-landing configuration (landing gear is not extended; flaps are not in the landing position); when it deviates downwards from glide slope path during landing approach.

The new generation of GPWS is named - EGPWS (Enhanced Ground Proximity Warning System) and uses both the information of airborne sensors and the data from digital elevation map (DEM). It allows the system to predict the collision in advance, even before reaching the dangerous relief height. Moreover, the system displays the relief image on the indicator and simplifies the aircraft control process for the pilot in the conditions of limited visibility and at night.

Critical conditions warning system alerts about reaching of aircraft exploitation limits in vertical overload, angle of attack, minimal and maximal flight speed, roll angle. Furthermore, the system can perform the following functions:

- checking the takeoff run and warning in advance about the remained length of runway if it is not enough to delay a decision about takeoff continue or its abortion;
- checking the flight level and warning the pilot about its violation;
- indication of wind shear.

Thunderstorm warning system can catch the lightning strikes and notify the pilot. One of frequently used system of this type is *Stormscope* by BFGoodrich company. It is a passive system that measures the electrical strikes at range up to 350 km. The system determines the azimuth and range to fixed point of electrical strike. Visualization of this information on the screen of navigation indicator helps the pilot to identify the incipient thunderstorm in advance and avoid it.

Eletronic clock serves to provide the precise time for the pilots and for other aircraft system. The current time and date are indicated at clock display and are transmitted to airborne consumers in serial code. The electronic clock is also used by pilots to measure the time intervals (as stopwatch), to warn the given time moment (timer), to measure and indicate the flight duration.

Fault detection and isolation system serves to organize the technical maintenance of aircraft. At flight the system fixes all faults and keeps them in nonvolatile memory, and on the ground it indicates the data to maintenance staff. During pre-flight procedure the system organizes the testing of all airborne electronic systems.

The separate navigation systems can be combined to create integrated systems (complexes), e.g. inertial and satellite navigation system, aerometric-Doppler navigation system, astro-inertial system and others. The combination of navigation systems in higher structural formation is called navigation complexes or flight and navigation system.

Further the construction principles and exploitation will be considered only for autonomous navigation systems (both radio and non-radio types).

1.2. Methods for determining navigation parameters of aircraft motion

Navigation is used to take information about the position and motion parameters of the object relatively the environment. Methods and means of navigation have been developed dependent on the tasks solved with object motion.

To determine the aircraft coordinates in particular three basic methods are used: dead (deduced) reckoning, position fix, and surveillance comparative method.

Dead reckoning method. The method is based on continuous calculation of aircraft trajectory by velocity vector data (integration of measured velocity vector in time or double of acceleration relatively the ground surface) taking into account the initial coordinates.

Information about the aircraft velocity can be obtained from inertial navigation system, from Doppler navigator or from airdata computer system.

With known aircraft heading and taking into account the wind speed and drift angle, it is possible to take the velocity components in the selected coordinate system. Their further integration gives the information about the components of traveled path.

To determine the current coordinates of aircraft the system needs the coordinates of initial waypoint from which the dead reckoning is started.

Systems of aerometric, radio location, Doppler and inertial dead reckoning are based on this method.

Position fix method. The method is based on using the surfaces of lines of position (LOP) to determine the object location.

Position surface is the geometric locus of points where possibly the object is located relatively the Earth, and where the physical parameter is constant to be measured onboard or at ground.

The mathematical description of position surface has the following form:

$$F = f(x, y, z)$$

where x, y, z are coordinates of aircraft location.

Properties of position surface are defined by the following characteristics: geometric shape $f(x, y, z)$; gradient q; offset error σ_{offset}. The geometric shape is found as a result of building the function $f(x, y, z) = \text{const}$ in the assumed coordinate system. The gradient characterizes the rate and direction of highest change in function $f(x, y, z)$. The gradient direction is always normal to the position surface. The expression of gradient has the form:

$$q = \lim_{\Delta n \to 0} \frac{\Delta f}{\Delta n},$$

where Δf is the function increment; Δn is the value of translational motion of position surface.

Root-mean-square error of position surface offset relatively the true aircraft location is determined as

$$\sigma_{\text{offset}} = \frac{\sigma_f}{q},$$

where σ_f is the root-mean-square error of measurement of parameter f, that defines the position surface.

To solve the navigation tasks, that is to determine the spatial aircraft location (the coordinates x, y, z) it is necessary to have three different position surfaces $F_i = f_i(x, y, z)$, (i =1, 2, 3) which are intersected. The intersection of two position surfaces with ground surface determines the object location of the Earth surface.

Line-of-position (LOP) is a geometric locus of points of projection of possible aircraft location on the Earth surface with constant navigation parameter. That is, LOP is created as a result of intersection between position surface and ground surface. Properties of LOPs, the same as position surfaces, are defined by geometric shape, gradient and offset error.

Use of LOPs in navigation is of long standing. Navigator determined a navigation parameter, e.g., bearing of navigation point(NP), and then built a line corresponding to the measured parameter on map. Then again the navigator did the same: measured another navigation parameter and built the second LOP. By the point of two lines intersection the navigator defined the aircraft location.

In aeronavigation the following LOPS can be used:

- curve of equal bearings,
- circle of position,
- hyperbola,
- astronomic LOP.

To LOPs it is also possible to relate a rhumb line, that is a line which crosses the geographical meridians at the same angle; and a great circle, that is an arc of great circle which is the shortest distance between any two points on the Earth surface.

In radio navigation systems the following LOPS are used frequently (Fig. 1.1):

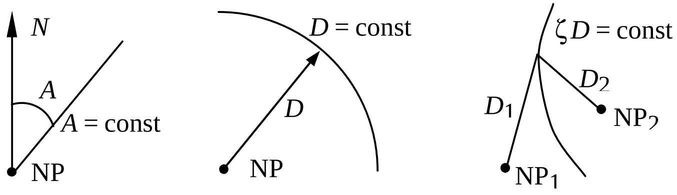


Fig. 1.1

– curves of equal bearings $(P = A = \text{const})$ are straight lines which pass through NP;

– circles of position $(P = D = \text{const})$ are circles with center in NP;

– hyperboles which are curves with equal difference in distance towards two navigation points NP_1 and NP_2 $(P = \Delta D = D_1 - D_2 = \text{const})$.

The position fix method is the base for construction of radio systems of short and long range navigation, satellite navigation system, systems operating by visual landmarks. Such landmarks can be bridges, separate buildings, oil storages, etc.

Surveillance comparative methods. Idea of surveillance comparative navigation methods is to determine the object location by comparison of land given on the map or saved in the airborne memory with its actual view observed by surveillance airborne sensors (like sights, drift sights, cameras, radar stations, etc.). If the image of observed area coincides with map data, then the object location is assumed to be recognized, and their coordinates can be found. Besides, the coordinates of landmarks, targets, aerodromes, astronomic landmarks and other objects also can be found.

Systems which realize surveillance comparative navigation methods are usually called correlation extremal systems, since extreme (maximum or minimum) of correlation function for measured and template (saved in the memory) characteristics is reached with exact correspondence of flight trajectory to the given one.

With surveillance comparative method there is no necessity to have external (relatively aircraft) radio stations; also the influence of inferences is weakened; accumulated errors are absent. However, the complexity of method limits its wide use, because it requires prior information about area characteristics along the route and large capacity of airborne memory. Also there are definite problems with calculation of correlation function.

The most frequently used is a visual orientation at all types of piloted aircraft. At flight the crew member observes the area through the cabin glass or through drift sight and then compares it with geographic map. The practical accuracy of such orientation is about 0.3-0.6 km at heights 2-3 km and 2-4 km at heights 6-10 km without special instruments. The use of optical drift sight allows determining the location with accuracy about 4% from flight altitude.

In general methods of determination of navigation parameters can be classified dependently on the nature of measured navigation parameters. For example, aerometric methods are based on the measurement of physical parameters of the Earth’s atmosphere. Geomagnetic methods use the navigation properties of magnetic field of the Earth. Astronomic methods are based on the determination of coordinates of known celestial bodies relatively the selected coordinate system and so on.

Choice of method or methods of navigation for use in the given aircraft type is explained by the following factors:

- characteristics of environment, in which aircraft is moved (water, air, space);
- range of change in navigation parameters (distance, velocity, acceleration and so on);
- required accuracy of measurement of navigation parameters;
- level of autonomy, noise immunity and reliability of navigation measurements;
- degree of physical ability to realize the navigation method (it is meant to have ability to create the navigation devices which fulfill the service conditions).

Essence of navigation process is to measure different navigation parameters by sensors of primary information. The parameters depend on the position and motion relatively the selected coordinate system related to the ground surface. Devices of information processing use the obtained data and determine the navigation elements characterizing velocity and coordinates of object location relatively the assumed coordinate system.

Thus, to provide navigation of piloted flight it is necessary to have the definite amount of information which includes data about primary navigation parameters, selected coordinate system, the Earth's shape, its magnetic and gravitation fields, aerospace in which the flight is done.

1.3. Brief characteristics of geographical fields and the Earth's atmosphere

1.3.1. Characteristics of the Earth's shape

For correct orientation during the flight above the ground surface it is necessary to know the Earth's shape, its geometric proportions, exact map and characteristics of magnetic field that covers the globe. At the same time all navigation tasks are solved at the surface of some geometric shapes (ellipsoid, sphere) or on the plane to which the Earth's surface is projected by definite rules. Thus, the development of methods of navigation problems solution, estimation of their abilities, determination of accuracy characteristics require the knowledge about geometric proportions and shape of the Earth.

The shape of the Earth is assumed to be one of the equipotential surface of gravity force which coincides with undisturbed surface of the ocean. For accurate navigation calculation the Earth's shape is assumed to be the surface of a geoid.

The geoid is a body limited by surface of earth ocean in undisturbed state. The surface of earth ocean is a surface of seas and oceans connected between each other which create total water mass. The surface of geoid has breaks, folds, or edges, that is, its curvature is changed in complex way and it is difficult to describe it mathematically. That is why the calculation on the surface is not done in strictly mathematic way.

Since the geoid surface is rather complex, then in practice the Earth's shape is approximated by simpler surfaces like ellipsoid of revolution, for example.

In Ukraine such shape is called as reference ellipsoid after the name F.M. Krasovskyi. Surface of the ellipsoid is very close to the geoid surface at the territory of Ukraine and of nearby countries. Deviations are not greater than 40 m. It is explained not only by the selected parameters of the ellipsoid, but also by its orientation in geoid.

In foreign countries there are in use the following ellipsoids: Clarke ellipsoid of 1866 (USA, Canada, Mexico), Clarke ellipsoid of 1880 (France), Bessel ellipsoid (Austria), Hayford ellipsoid (Egypt, Finland, others). The ellipsoid GRS 80 is most widely used in navigation. It is also the base for World Geodetic System WGS 84.

Krasovskyi's ellipsoid (Fig. 1.2) is formed as a result of ellipse $EP_NE'P_S$ revolution around minor semi-axis P_NP_S, and it has the fol-

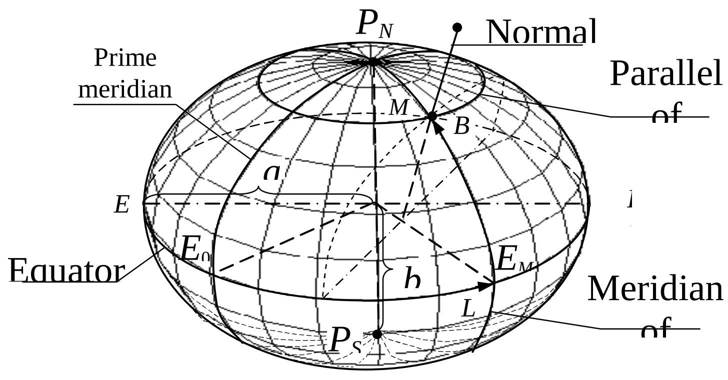


Fig. 1.2

lowing characteristics :

– major semi-axis

$$a = 6378245 \text{ м};$$

– minor semi-axis

$$b = 6356863 \text{ м};$$

– ellipticity

$$c = \frac{a - b}{b} = 0{,}00355233;$$

– 1st eccentricity

$$e = \frac{\sqrt{a^2 - b^2}}{a} = 0{,}081813334;$$

– 2nd eccentricity

$$e' = \frac{\sqrt{a^2 - b^2}}{b} = 0{,}082088521.$$

Geodetic equator divides the ellipsoid into northern and southern hemisphere; Greenwich meridian (or prime meridian) is the reference one for longitude evaluation.

The position of point *M* on the ellipsoid surface is determined by geodetic latitude *B* and geodetic longitude *L*. *Geodetic latitude* of point *M* is called the angle between the plane of geodetic equator EE_0E_ME and direction of geodetic vertical, that is the normal to ellipsoid surface in point *M*.

Geodetic longitude is called the dihedral angle between the plane of geodetic meridian $P_SE_0P_N$ (Greenwich meridian) and plane of geodetic meridian $P_SE_MMP_N$ of point *M*.

Geodetic coordinates (sometimes they are called geographic coordinates) are used for aircraft flight planning. They can be obtained from aeronautical charts of different scales or from special databases.

Ratio of difference of ellipsoid semiaxes to its major semiaxis is very small value Z 0,00355233, therefore for solution of navigation tasks without severe requirements to accuracy and also for mapping with small scales, the Earth's shape is assumed to be a sphere (Fig.1.3). The radius of sphere is 6371 km, and the equator length is 40076 km.

With such representation of the Earth's shape, the errors of aeronavigation tasks solution does not exceed 0.5% with distance determination, and error in direction determination is not greater than 12®The arc length of any great circle in 1® equals 1852 m (1 nautical mile).

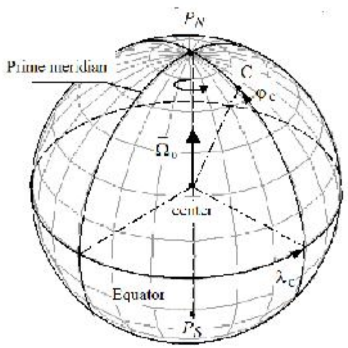


Fig. 1.3

For mapping the surface of reference ellipsoid is represented on the surface of sphere by assumed rule. The rule defines sizes of the Earth's sphere, that is, the value of its radius *R*. Professor V.V. Karvaiskyi calculated the radiuses of the Earth's sphere which provide the minimal distortions for different projecting methods.

Frequently the following values of the radiuses are used:

R = 6366707 m for normal projections;
R = 6367616 m for equal-angle projections;
R = 6371116 m for equilateral projections;
R = 6372900 m for projections, which are equilateral by meridians
R = 6378245 m for projections by central perspective.

When approximating the Earth's surface by sphere, the point position on the spherical surface is determined by its spherical coordinates.

Spherical latitude of point *C* is called the angle φ_c between the equator plane and direction to the given point from the center of the Earth.

Spherical longitude of point *C* is called the dihedral angle λ_c between the planes of prime meridian and of the meridian of the point.

Ranges and positive directions of geodetic and spherical coordinates coincide.

Range of latitude measurement is 0…90º (“ + “ corresponds to northern direction, " − " corresponds to southern direction);

Range of longitude measurement is 0…180º (" + " corresponds to eastern direction, " − " corresponds to western direction).

Pay attention that the values of geodetic and spherical longitudes at any point of ground surface coincide, and value of spherical latitude of the point depends on its position on the sphere relatively the ellipsoid surface.

When solving the aeronavigation tasks, it is necessary to provide the conversion from geodetic coordinate system (geodetic coordinates) to spherical coordinate system (or spherical coordinates)

Such conversion is realized by the following relationships:

$$\varphi_c = B - 8'39''\sin 2B, \qquad \lambda_c = L$$

and inverse conversion is done by

$$B = \varphi_c + 8'39''\sin 2\varphi_c, \qquad L = \lambda_c$$

1.3.2. Characteristics of the Earth’s gravitational field

The Earth has a mass approximately 5.976 10^{24} kg and therefore creates the gravitational field nearby (the field of gravity force). The gravitational field creates the gravity force and causes the accelerated motion of bodies.

When solving the aeronavigation tasks, the Earth’s gravity field is considered as a source of forces which cause the accelerated motion, and as a positional surface that allows us to determine the distance from an object to the ground surface.

The gravity field of the Earth can be characterized by scalar value $U_{г}(\bar{R})$, gravity potential, where $\bar{R}$ is a radius vector from the center of mass of the Earth.

For the gravitational field of the Earth as for any potential field it is possible to define the equipotential surfaces or surfaces with equal level of gravity potential.

Vector characteristic of gravitational field is a *strength vector*

$$\bar{g}_{г} = \quad U_{г}(\bar{R}),$$

where   is a symbol of function gradient (Hamilton operator).

Direction of strength vector of gravity field is a normal to equipotential surface and it coincides with direction of gravity force.

Orientation of strength vector of gravity field characterizes the direction of *gravity vertical* in any point of circumterrestrial space. Numerically the vector of gravity field strength equals the gravity force of the Earth that acts on a unit mass, and correspondingly equals the acceleration of free mass point under the action of only gravity forces. That is why the vector of gravity field strength is sometimes called *a vector of gravity acceleration*.

Due to the Earth’s rotation any body unmoved relatively ground surface anyway will be in rotational motion. The radius of circular diurnal motion of unmoved body is determined by its distance l to polar axis $P_M P_S$. A body with mass m, that moves by circular trajectory, has the centripetal acceleration $\bar{w}_{\text{centripetal}}$ (Fig. 1.4)

$$\bar{w}_{\text{centripetal}} = \frac{V^2}{l} = \vartheta_0^2 l = V\vartheta_0$$

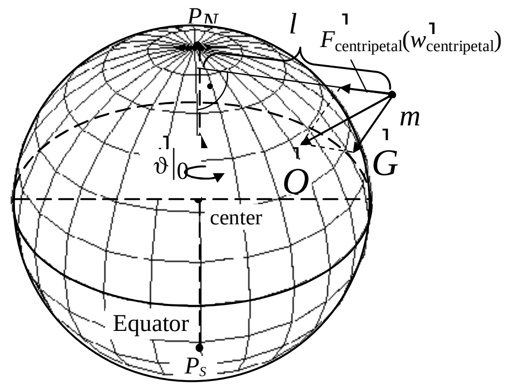


Fig. 1.4

where $V = l\Omega_0$ is a linear velocity of a body caused by diurnal rotation of the Earth with angular velocity Ω_0.

The source of centripetal force $F_{\text{centripetal}}$ (see Fig. 1.4) is a component of gravity force in the direction, perpendicular to axis $P_M P_S$. This component can be found as a vector difference between the gravity attraction force Q and gravity force G for the given body.

Note that the gravity force or weight is called the force created by a body that weighs upon a support or strains a suspension.

A field of gravity forces of unit masses in circumterrestrial space is called as geopotential gravity force. Strength lines of Earth's gravity field coincide with the direction of gravity forces of unmoved relatively ground surface bodies. Experimentally the direction of the lines can be determined by suspension filament with unmoved relatively the ground surface suspension point or by bubble level.

This direction is also called *true or astronomic vertical*, since suspension filament or by bubble level are widely used in instruments to determine the coordinates of a point on ground surface by astronomic observations.

A plane perpendicular to true vertical is called *a plane of true horizon*.

Potential characteristic of gravity force field is a scalar function $U(\bar{R})$, potential of gravity force field. Vector characteristic of the field is strength vector $\bar{g}\,X(\bar{R})$, that equals vector difference between strength vector of gravity field $\bar{g}_g\,X(\bar{R})$ and centripetal acceleration $\bar{w}_{centripetal}(\bar{R})$

$$\bar{g}(\bar{R})\,X\quad U(\bar{R})\,X\bar{g}_g(\bar{R})\,Z\bar{w}_{centripetal}(\bar{R})\,X\bar{g}_g(\bar{R})\,Z\bar{l}_0\vartheta_0^2 l$$

where $\bar{l}_0$ is a unit vector directed perpendicular to axis of the Earth's rotation from the given point.

1.3.3. Brief characteristics of the Earth's magnetic field

For aeronavigation the properties of the Earth's magnetic field are widely used. The Earth is a huge natural magnet with axis tilted to polar axis at angle of about 11⌠.

In any point of circumterrestrial space the magnetic field is charactrized by *strength vector* $\bar{T}$. The magnitude and direction of the vector are well known functions of geographic coordinates dependent on location and time of observation.

Strength vector $\bar{T}$ of magnetic field is usually resolved into two components: vertical $\bar{Z}$ and horizontal $\bar{H}$; or into three components: eastern $\bar{Y}_E$, northern $\bar{X}_N$ and vertical $\bar{Z}$.

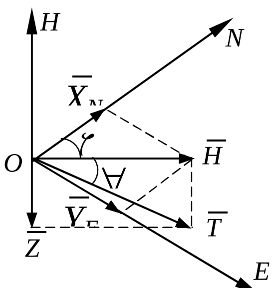


Fig. 1.4

Relationships between the components and the vector are shown in Fig. 1.4 and are the following:

$$\bar{H}\,X\bar{T}\cos\forall_M; \qquad \bar{Z}\,X\bar{T}\sin\forall_M$$

$$\bar{X}_N\,X\bar{H}\cos\zeta_M; \qquad \bar{Y}_E\,X\bar{H}\sin\zeta_M$$

Angles θ_M and Δ_M are called angle of inclination (or magnetic dip) and angle of declination (or variation), respectively. Arrows show the positive directions of angles.

In general conditions the magnitude and direction of the vector $\bar{T}$ may vary dependent on time. Periodical changes of the vector (annual, monthly, diurnal) are well investigated. Thus, information about $\bar{T}$, $\bar{H}$, θ_M i Δ_M is mapped. Usually such map contains: lines of equal magnetic variations Δ_M = const Z isogonic lines; lines of equal magnetic inclinations θ_M = const Z isoclinal lines; lines of equal strengths of magnetic field $\bar{T}$ = const Z isodynamic lines.

Relative non-stationary of characteristics of the Earth's magnetic field can be significantly taken into account by special graphs and tables dependent on season and day time, and by periodical updates (approximately each 5 years) of geomagnetic maps.

Also it is necessary to remember the possibility of random variations of geomagnetic field (e.g., due to magnetic storms). This may significantly complicate the use of the Earth's magnetic field for navigation.

Measurements of mentioned above parameters of geomagnetic field in flight and with presence of cartographic data allow solving the main task of aeronavigation, that is, determination of aircraft location coordinates. However, the most frequently solved navigation task is the determination of aircraft heading by geomagnetic field.

1.3.4. Brief characteristics of properties of earth atmosphere. Standard atmosphere

The aircraft flight is done in atmosphere - aerial layer that covers the Earth. Information about atmosphere parameters, about characteristics of air flow of aircraft and aircraft orientation in airflow has significant importance for solving the flight and navigation tasks.

In any point of circumterrestrial space the atmospheric air is characterized by composition, temperature, pressure, density and gradients of these values.

Besides, the flight region is usually characterized by meteorological elements like velocity and direction of wind, cloudiness, fog, turbulence, etc.

Atmosphere parameters and meteorological phenomena are significantly dependent on coordinates of observation and on time. Short time dependences of most of atmosphere parameters have random nature. But long time observations and study of atmosphere properties allow us to detect the definite time-spatial regularities in changes of atmosphere parameters.

By air composition the atmosphere is divided into the following layers:

Z *homosphere* (below 80...100 km) is characterized by constant chemical composition of air (78.09% of nitrogen, 20.95% of oxygen, 0.95% of other gases);

Z *heterosphere* (above 100 km) has the chemical composition of air that varies with height.

By temperature change the atmosphere is divided to troposphere (height is 10…12 km in medium latitudes, 16...18 km in the tropics, 9...10 km in polar latitudes), stratosphere (height is 40...55 km), mesosphere (up to 80 km), thermosphere (up to 800 km) and exosphere (higher than 800 km).

Exosphere has no upper limit and continually passes into interplanetary space. Among mentioned layers there are also intermediate ones which are called tropopause, stratopause, mesopause, thermopause, respectively.

Troposphere contains up to 80% of all air mass and up to 90% of all water mass. It stipulates processes which form the weather on the Earth: cloudiness formation, precipitation, thunderstorm activity, winds, air streams, etc. Temperature here decreases with altitude raise in average in 6.5î per 1 km.

Stratosphere contains approximately 20% of all air mass. Temperature in the layer up to altitude 30...35 km is almost constant, in average it is J *56î C, and then with altitude increase the temperature raises almost up to 0°C. It is caused by intensive absorption of ozone by solar radiation. Maximal concentration of ozone is in the stratosphere.*

In mesosphere the temperature rapidly decreases with altitude increase and reaches at the highest point J *88î C. Air density here is small - almost 10 times smaller than at the ground surface.*

In thermosphere the temperature increases with altitude increase and reaches values 800...1000î C at the highest point. In thermosphere there are layers with increased concentration of ions.

The average statistical regularities of change in atmospheric parameters are detected. And above all, regularity of change in temperature and air pressure dependently on change in geodetic altitude allows using information about parameters of atmospheric field in aeronavigation.

Note that solution of aeronavigation tasks by aerometrical measurement is based on use of so called standard atmosphere.

Standard atmosphere is the mathematical model of earth atmosphere in the form of dependence of atmosphere physical parameters on the true altitude. In standard atmosphere (SA) the definite values of atmosphere parameters are taken instead of their true values which are actually random functions of time and of coordinates.

These values with high accuracy correspond to mathematical expectations of atmosphere parameters at the given altitude.

Nowadays in Ukraine the dependence of SA is given by standard ГОСТ 4401-81, which corresponds to international standard. It contains the numeric values of basic atmosphere parameters for altitude ranges from Z2000 m to 1 200 000 m. For altitudes from Z2000 m to 80 000 m there are the following data: air temperature, density, atmosphere pressure, acceleration of free fall, sound speed at the given altitude and some other data. For altitudes from 80 000 m to 1 200 000 m there is reference data.

The standard has the obligatory appendix that contains basic regulations, constants, formulas and auxiliary tables required for calculations of atmosphere parameters.

Values of standard atmosphere parameters by SA-81 at zero altitude are called standard parameters. They are:

Z pressure P_0 = 760 mm of mercury column (101325 Pa);

Z temperature T_0 = 288.15 K;

Z density ∂_0= 1.225 kg/m^3;

Z acceleration of gravity force g_0 = 9.80665 m/sec^2 (corresponds to value of g_g at latitude 45⌈32®33⌊);

Z sound speed a = 340.294 m/sec.

Dependence of pressure on height by SA is based on the equation of atmosphere statics

$$dP \, X Z \partial g_g dh\,. \qquad (1.1)$$

It corresponds to balance of vertical pressure forces $(P + dP)S$; PS and gravity forces ($\partial g_g Sdh$), which act on the air column with cross area S and height h (Fig. 1.5)

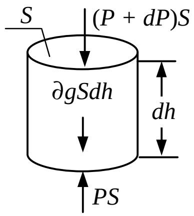


Fig. 1.5

Let's write also the equation of ideal gas

$$\partial \, X \frac{P}{RT}\,, \qquad (1.2)$$

where R = 287.05287 J/kg Z specific constant of air.

Substituting (1.2) in (1.1) it is possible to get

$$\frac{dP}{P} \, X Z \frac{g_g dh}{RT}\,. \qquad (1.3)$$

In equation (1.2) the dependence g_g on h is taken into account, that is represented with high accuracy as following:

$$g_g(h) \, X g_0 \frac{r^2}{(r \, \Gamma \, h)^2}\,, \qquad (1.4)$$

where r = 6356766 m is the conventional radius of the Earth that provides the value g_g and its gradient close to true values for latitude 45⌈32®33⌊.

To simplify solution of differential equation (1.3) and to compact the final expressions of dependence of pressure on height in (1.3) let us change from h to new independent variable Z so called geopotential height H_{gp}.

Geopotential height is used in atmosphere physics and is determined as ratio

$$H_{gp} \, X \frac{\theta}{g_0}\,, \qquad (1.5)$$

where $\theta \; X \int_0^h g_g dh$ is the potential of field of gravity force.

Geopotential height is evaluated from mean sea level (MSL). Taking into account (1.4) from (1.5) it is possible to get

$$H_{gp} \, X r^2 \int_0^h \frac{dh}{(r \, \Gamma \, h)^2} \, X h \frac{r}{r \, \Gamma \, h}\,. \qquad (1.6)$$

According to (1.6) H_{gp} ™h, and the difference

$$h \, Z H_{gp} \, X \frac{h^2}{r \, \Gamma \, h}$$

is assumed to be insignificant (for h = 10 km it is 16 m, and for h = 20 km Z 63 m). Based on (1.5) let's write

$$dH_{gp} \, X \frac{d\theta}{g_0} \, X \frac{g_g dh}{g_0}\,. \qquad (1.7)$$

With integration of equations (1.3) it is necessary to take into account the dependence of air temperature T on height (Fig. 1.6). According to SA-81 there are following dependences $T = f(H_{gp})$:

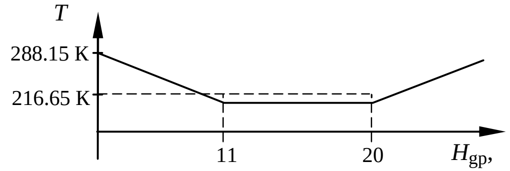


Fig. 1.6

1) H_{gp} ™11 km

$$T = T_0 + \wp H_{gp}, \qquad (1.8)$$

where $\wp X \frac{dT}{dH_{gp}}$ = Z6.5 10^{Z3} deg/m is vertical temperature gradient or so called lapse rate;

2) 11 km < H_{gp} ™21 km

$$T = T_{11}, \qquad (1.9)$$

where T_{11} = 216.65 K = const;

3) 20 km < H_{gp} ™32 km

$$T = T_{20} + \beta(H_{gp} - 20000), \qquad (1.10)$$

where $\beta = 10^{-3}$ deg/m ; $T_{20} = T_{11}$.

Taking into account (1.3), (1.7), (1.8) for $H_{gp} \le 11$ km it is possible to get

$$\int_{P_c}^{P} \frac{dP}{P} = -\frac{g_0}{R} \int_{0}^{H_{\Gamma}} \frac{dH_{gp}}{T_0 + \beta H_{gp}},$$

and to designate the calculated pressure by index 'SA'

$$P_{SA} = P_0 \left(1 + \frac{\beta}{T_0} H_{gp}\right)^{-\frac{g_0}{\beta R}}. \qquad (1.11)$$

where P_0, T_0, g_0 are standard values for $H_{gp} = 0$; $\beta = -6.5 \cdot 10^{-3}$ deg/m. Similarly taking into account (1.9), (1.10) it is possible to get:

– for $11 \text{ km} < H_{gp} \le 20$ km

$$P_{SA} = P_{11} \exp\left[-\frac{g_0}{RT_{11}}\left(H_{gp} - 11000\right)\right], \qquad (1.12)$$

where P_{11} = 22632 Pa (169.75 mm Hg); T_{11} = 216.65 K;

– for $20 \text{ km} < H_{gp} \le 32$ km

$$P_{SA} = P_{20} \left[1 + \frac{\beta}{T_{20}}\left(H_{gp} - 20000\right)\right]^{-\frac{g_0}{\beta R}}, \qquad (1.13)$$

where P_{20} = 5474.87 Pa (41.065 mm Hg); $\beta = 10^{-3}$ deg/m.

Formulas (1.11)…(1.13) regulate the dependence of atmospheric (static) pressure on geopotential height according to SA-81. Altitude from these formulas corresponds to actual static pressure P_{st}, and is called in aviation as true barometric altitude and is designated as H_{true}. From (1.11)…(1.13) let's replace H_{gp} by H_{true}, P_{SA} by P_{st} and obtain after transformations:

– for $H_{true} \le 11$ km

$$H_{true} = \frac{T_0}{\beta}\left[\left(\frac{P_{st}}{P_0}\right)^{-\frac{\beta R}{g_0}} - 1\right], \qquad (1.14)$$

where $\beta = -6{,}5 \cdot 10^{-3}$ deg/m.

– for $11 \text{ km} < H_{true} \le 20$ km

$$H_{true} = 11000 + \frac{RT_{11}}{g_0} \ln \frac{P_{11}}{P_{st}}, \qquad (1.15)$$

– for $20 \text{ km} < H_{true} \le 32$ km

$$H_{true} = 20000 + \frac{T_{20}}{\beta}\left[\left(\frac{P_{st}}{P_{20}}\right)^{-\frac{\beta R}{g_0}} - 1\right] \qquad (1.16)$$

where $\beta = 10^{-3}$ deg/m.

Formulas (1.14)…(1.16) are called hypsometric relations and are used to calibrate sensors of barometric altitude of flight.

But the measurement of altitude makes sense only in case of referencing relatively the definite datum. If the reference datum is assumed as pressure 760 mm Hg, then atmospheric pressure in the given point corresponds to true barometric altitude. With setting another referencing datum that corresponds to atmospheric pressure on destination aerodrome, for example, then the atmospheric pressure in the given point corresponds to a height (relative barometric altitude).

Aircraft flight with fixed value of barometric altitude (with constant value of atmospheric pressure) is a motion by isobaric surface in the Earth atmosphere. Usually isobaric surfaces of standard atmosphere may not precisely correspond to isobaric surfaces of real atmosphere. Moreover, the latter ones continuously change in time. But anyway isobaric surfaces of both real and standard atmospheres are never and nowhere intersected in gravitational field of the Earth. That is why airplanes piloted by different values of prescribed true barometric altitudes are assured of collisions. Piloting with control of relative barometric altitude is widely used with takeoff and landing.

Tables of SA-81 also contain the values of air density ρ, air unit weight γ and sound speed a for different altitudes, which are calculated by formulas

$$\partial \mathrm{X} \frac{P}{RT};$$

$$\uparrow \mathrm{X} \frac{G}{V_{\mathrm{g}}} \mathrm{X} \frac{mg}{V_{\mathrm{g}}} \mathrm{X} \partial g;$$

$$a \, \mathrm{X} \sqrt{\frac{RT}{\Uparrow}} \, \mathrm{X} 20.045796 \sqrt{T}.$$

Here V_{g} is the gas volume with mass m, and weight G; $\mathrm{X} \frac{c_P}{c_V} = 1.4$ Z is adiabatic coefficient, that is equal to ratio of specific heat capacities of air with constant pressure c_P and constant volume c_V; $\Uparrow$ is molar mass of air.

1.4 Basic coordinate systems

1.4.1. Basic navigation coordinate systems

The solution of any aeronavigation tasks requires the determination of flight and navigation parameters. For their numeric estimation the corresponding coordinate systems (CS) are introduced.

Navigation CSs are constructed in such a way to provide coincidence of their reference points, lines and surfaces with specific points, lines, surfaces of corresponding physical bodies and fields.

As a coordinate origin the following points are frequently selected:

Z point in the center of the Earth (center of mass of the Earth) for geocentric CS;

Z point on the ground surface for geotopical CS;

Z point onboard the aircraft for body-fixed CS.

Z With selection of CS used onboard the aircraft for solution of navigation tasks it is necessary to take into account the fact that the given CS must provide the following:

Z programming of flight trajectory by minimal time;

Z covering of territory area that is sufficient for using the single CS;

Z solving navigation tasks with required accuracy;

Z getting the simplest relationships with solution of control and navigation tasks;

Z solving special tasks connected with flight missions;

Z providing obvious information about aircraft position relatively the desired track (DSRTK) or relatively its basic waypoints;

Zsimple determining aircraft position on a map, tablet or indicator by its coordinates and easy determination of coordinates of other objects.

Use of definite CS depends on accuracy characteristics of flight and navigation systems of aircraft, pre-flight preparation of aircraft equipment, features of equipment operation during the flight, and the specifics of using navigation information by crew.

The factors like use of different sensors of flight and navigation information onboard and high number of tasks solved during pre-flight procedure and during the flight are a reason of using high amount of different CSs in modern flight and navigation complexes (FNC).

Nowadays onboard the following CSs are frequently used:

Z geocentric (geodetic, normal spherical, great circle, equatorial, horizontal);

Z geotopical (polar (spherical or cylindrical) and rectangular);

Z body-fixed (moving) CS with coordinate origin coinciding with center of mass of aircraft (local tangent plane (LTP) CS, body-fixed CS, track CS, wind axes CS).

Standard of geocentric CS in the most of countries of former USSR, Poland and others, is geocentric CS ПЗ-90, and in the most of western countries it is geocentric CS WGS-84.

Difference between CS WGS-84 and CS ПЗ-90 is in displacement of coordinate origin (WGS-84 is higher than ПЗ-90 in 4 m) and in rotation of one CS relatively the other by 0,6®Root mean square error of coordinate determination does not exceed 5 m when using different CS.

Sometimes it is possible to use some other CSs not mentioned above.

Note that the geodetic CS is related to the geocentric ones conditionally, since geodetic system has no center. In geodetic CS the point position on the surface of terrestrial ellipsoid is defined by two coordinates: latitude B and longitude L.

Normal spherical CS is CS on the surface of terrestrial sphere. The system determines the point position by spherical latitude $\nleftrightarrow$and spherical longitude $\Leftarrow$ The CS is used for direct solving of navigation tasks, and formulas of spherical trigonometry are used.

Sometimes the normal spherical CS is intermediate one between geodetic CS used for flight planning and displaying of aircraft position, and great circle CSs that are more useful and natural for some sensors of navigation information (inertial navigation systems, compass systems).

Usually, normal spherical CS is connected with terrestrial surface and rotates with it with angular velocity ϑ_3.

When solving the aeronavigation tasks, the great circle CSs are widely used. These CSs are also spherical. The feature of great circle CSs is that their initial reference planes can be changed according to the functionality of the navigation system, the type of aircraft, the nature of task, solved in flight, etc.

Indeed, in some cases it is more convenient to use such spherical CS with equator coinciding with the desired track or being close to it. It offers several advantages, including possibility to use simpler relationships for solution of navigation tasks, possibility to use formulas of plane trigonometry keeping the required accuracy of solution.

The main planes which are reference ones for evaluation of aircraft position coordinates are the plane of the great circle equator (plane of great circle) and plane of initial great circle meridian (Fig.1.6).

Great circle is the planar curve which is an arc of great circle created by intersection of sphere by a plane passing through its center. Thus, great circle is the shortest distance between two points on a sphere, and correspondingly widely used to plan a route of flight between two points on the terrestrial surface.

To create a great circle to the Earth's surface or on a map it is necessary to introduce the great circle spherical CS with the coordinate origin in the center of the Earth.

Poles of the great circle are determined as the ends of sphere diameter that is normal to the main plane - plane of the great circle. There are correspondingly northern pole P_{No} in the northern hemisphere, and the southern one P_{So} in the southern hemisphere.

Let's agree to assume *the point of ascending node of the great circle Y* as a point at which the great circle passes in the northern hemisphere during the flight along the great circle route counterclockwise when viewed from the point of northern great circle pole P_{No}.

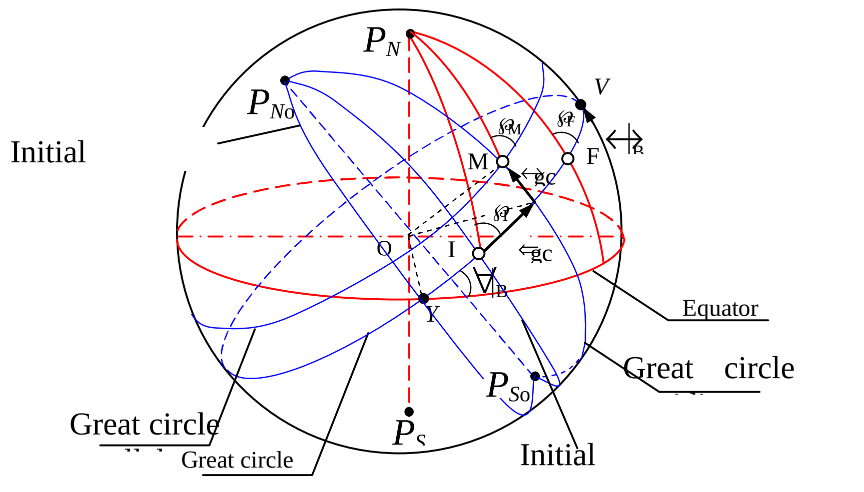


Fig. 1.6

Vertex is called the point *V* on the great circle, which is most distant from the plane of the geographical equator.

Great circle coordinates of the point M on the terrestrial sphere are: *great circle latitude* $\leftrightarrow_{gc}$, that is the angle between the plane of the great circle (plane of great circle equator) and geocentric vertical of this point; *great circle longitude* $\Leftarrow_{gc}$, that is dihedral angle between the plane of initial great circle meridian and the plane of great circle meridian of the point M.

If the position of initial meridian is not specifically mentioned, then it is usually great circle meridian that passes through the point *Y* of ascending node of great circle.

However, often the initial great circle meridian is chosen as meridian that passes through the characteristic point of the flight, namely through the fixed point of great circle, for example, through the initial waypoint I or final waypoint F of the given great circle, which coincides with the route.

With determination of aircraft position in great circle CS the angular orientation of great circle parallel relative to the geographical meridian is simultaneously determined at the same point, as a rule. This orientation is determined by true track angle of great circle $\wp_{gc}$ ($\wp_M$), that is measured in the horizontal plane and is defined as the angle between the projections of tangents to the geographic meridian and to great circle parallel on the horizontal plane. The positive direction of $\wp_{gc}$ is clockwise from the northern direction of tangent to the meridian.

The difference between the true track angles of great circle in two points of the great circle is called *meridian convergence angle* at this route section.

Position of great circle on the terrestrial sphere is frequently determined as:

- giving the geographic coordinates of two waypoints, for example, the initial I and final F waypoints of definite arc of great circle;
- giving the geographical coordinates of definite point P on great circle and the true track angle of great circle $\wp_{gc}$ $(\wp_{P})$ in the same point;
- giving the coordinates of the North Pole P_N of great circle;
- - giving the longitude of the ascending node Y of great circle and the inclination angle $\forall_B$ of the great circle to the equator plane or latitude $\leftrightarrow_B$ of vertex point B of great circle ($\forall_B = \leftrightarrow_B$).

The use of great circle CS is the most appropriate under the assumption that the value of great circle latitude during the flight will be small (less than 5 ... 10°). In this case, instead of angular great circle coordinates, the linear coordinates are used: distance *y* by the great circle and deviation *x* from the great circle.

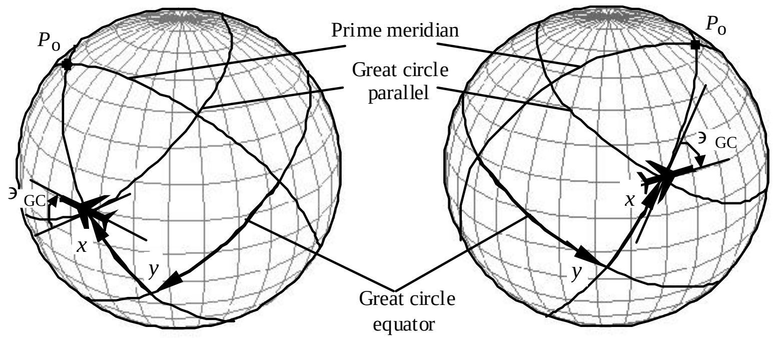
Herewith,

$$x \,\mathrm{X} \leftrightarrow_{gc} R; \quad y \,\mathrm{X} \Leftarrow_{gc} R.$$

where R is the calculated value of radius of the terrestrial sphere.

There are right and left great circle coordinate systems. In the right great circle CS the great circle latitude *x* is measured to the right from the reference direction of great circle longitude *y*, and great circle track angle $\wp_{gc}$ and great circle heading э $_{gc}$ are evaluated from the tangent to great circle parallel (Fig. 1.7, a).

In the left great circle CS, the great circle latitude is measured to the left from the reference direction of great circle longitude, and the great circle heading is evaluated from the tangent to great circle meridian (Fig. 1.7, b).



a *b*

Fig. 1.7

In the special case, the great circle equator of right great circle CS can be aligned with the geographical meridian of definite point. The great circle longitude in this case is evaluated either from the terrestrial equator or from great circle meridian of this point.

Another special case of the right great circle CS is so called *navigation leg great circle CS*. In this system, the proper right great circle CS corresponds to each flight stage. It is done by alignment of the great circle equator with desired track (DSRTK) - the great circle of each navigation stage of flight route. Such CS provides simplicity and clarity in the determination of traveled leg distance and cross-track distance relative DSRTK. The great circle longitude in this case is evaluated from the waypoint and characterizes the distance traveled from it. The great circle latitude characterizes the value of cross-track distance relative DSRTK.

The great circle coordinate systems are the basis for quasi great circle CS. Such CS differs from great circle one in that they have equator radius different from the meridian radii.

The conversion from great circle coordinates of a point to the coordinates in quasi great circle CS is done with the help of the following ratios:

$$x \,\mathrm{X} \leftrightarrow_{gc} R_x; \quad y \,\mathrm{X} \Leftarrow_{gc} R_y.$$

where $\leftrightarrow_{gc}$, $\Leftarrow_{gc}$ are angular coordinates of a point in the great circle CS which is transformed to quasi great circle CS; R_x, R_y are values of meridian and equator radii accepted in this CS, for example, radius of meridian curvature and the first vertical of the terrestrial ellipsoid.

If the equator of this CS is aligned with the geographic meridian of definite point and if the radii R_x and R_y are taken to be equal to the radius of curvature of the first vertical and of the meridian which pass through this point, then within a certain area near the point, the projection of the terrestrial ellipsoid surface will be obtained with sufficiently low distortions. This point may be, for example, the center of flight area or of departure aerodrome. The nature and magnitude of distortions in this case depend on the geodetic latitude of the selected point and on sizes of analyzed (working) area.

The *equatorial and horizontal CSs* are also spherical. They are used, for example, in astronomical sensors of navigation information and in satellite navigation systems.

Among geotopic navigation CSs, there are such coordinate systems with origin in the moving or fixed point on the Earth's surface.

As fixed coordinate origin of geotopic CS, the following points are usually selected: location point of ground means of radio navigation system; point at the airport of takeoff or of landing; initial, final and intermediate waypoints of route; landmarks and targets.

As moving coordinate origin, the point (projection of aircraft) on the ground surface is most often chosen.

Axes of *rectangular* geotopic CS can be oriented in various ways, most often they are oriented in such a way: two of them lie in the horizon plane, and the third one is aligned with the local vertical.

Azimuthal orientation of horizontal axes is determined by the nature of solved tasks, composition and functionality of technical means of aircraft navigation. These rectangular CSs in the case of fixed relative to the Earth coordinate origin are called normal terrestrial CS and designated as $O_0X_gY_gZ_g$. Most often the orientation of horizontal axes is defined by directions of geographical or great circle meridians and parallels.

Also with rectangular geotopic CS, the polar (spherical and cylindrical) coordinate systems are used.

Fig. 1.8 shows geotopic rectangular (normal terrestrial CS

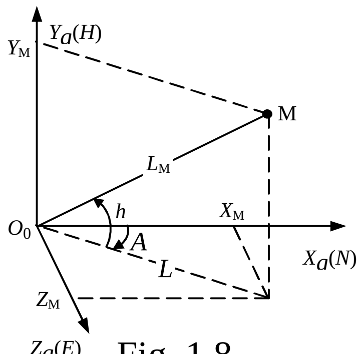


Fig. 1.8

$O_0X_gY_gZ_g$) and polar CS O_0NHE (spherical and cylindrical) with common center at point O_0 and with horizontal axes oriented in the directions of geographical parallel - axis O_0Z_g (O_0E is the eastern direction) and of geographical meridian - axis O_0X_g (O_0N is the northern direction). Axis O_0Y_g (O_0H) is directed in the direction of true vertical.

The coordinates of point M in these systems are determined by values:

- segments X_M, Y_M, Z_M in rectangular CS;
- angles A, h and segment L_M in spherical polar CS;
- angle A and segments L and Y_M (H) in cylindrical polar CS.

At the same time the angle A is called azimuth or true bearing of object M, in the point it is evaluated in the horizontal plane from the direction to north O_0N to the direction to the object O_0M' clockwise in the range from 0° to 360°.

The angle h is angular height (angular elevation) of the object and is measured from the horizontal plane within ranges from -90 ° (below the plane $X_gO_0Z_g$) to +90 ° (above the plane $X_gO_0Z_g$).

Segments L, L_M, H (Y_M) are respectively called: range (horizontal distance from the point O_0), slant range and height of the object above the horizontal plane $X_gO_0Z_g$.

In the case of orientation of horizontal axes of geotopic CS by the direction of great circle parallel (great circle equator) from this direction, the bearing B or conventional azimuth of object is evaluated. Sometimes unlike azimuth this angle is called great circle or conventional bearing of object.

Moving (body-fixed) coordinate systems are navigation reference systems with coordinate origin aligned with the aircraft center of mass.

Among the moving CSs, the special place is taken by the body-fixed rectangular CS *OXYZ*, created by longitudinal, normal and lateral axes of aircraft. The coordinate origin of this CS is aligned with the center of the aircraft mass, the axis *OX* is aligned with the direction of the longitudinal axis of the aircraft, the axis *OZ* is directed toward the starboard wing and the axis *OY* is perpendicular to the first two axes and is directed upwards and is located in the plane of aircraft symmetry.

In the body-fixed horizontal coordinate system (normal moving CS $OX_gY_gZ_g$) the coordinate origin is aligned with the center of aircraft mass, the axis OX_g is defined as the projection of the aircraft longitudinal axis on the plane of true horizon, the axis OZ_g is located in the horizon plane and directed to the right from the axis OX_g, and the axis OY_g is parallel to the normal to the horizon plane.

In the wind axes CS $OX_aY_aZ_a$ the wind axis OX_a coincides with the vector of airspeed, the axis of lift force OY_a lies in the symmetry plane and the lateral axis OZ_a complements the system to right-handed CS.

The track CS $OX_kY_kZ_k$ is moving coordinate system with the axis OX_k which coincides with the direction of ground speed vector (speed of the aircraft relative to the ground surface), and with the axis OY_k which is directed upwards from the Earth's surface and which lies in the vertical plane passing through the axis OX_k. In the absence of wind, the axis OX_k of track CS is aligned with the axis OX_a of wind axes coordinate system.

Aircraft body-fixed (moving) navigation CSs are used as a rule to obtain information from definite systems and means aboard the aircraft, for example radar systems, optic drift sights, gyroscopic systems.

Often when solving navigation problems there is a necessity of conversion from one CS to another. This conversion is performed using transition matrices (direction cosines matrices).

1.4.2. Astronomic coordinate systems

Principles of construction of astronomical navigation systems for determining the coordinates of the location and of aircraft heading are based on the geometric or analytical simulation of celestial bodies position relative navigation CS. Since the aviation astronomic devices take bearing of natural celestial bodies, then the understanding of such modeling involves knowledge of fundamentals of aviation astronomy and in particular of astronomic CSs and of problems of time measurement.

In the aviation astronomy the distances to celestial bodies are not of importance, only their angular positions are important. Therefore, for convenience of solution of positioning problems the term of celestial sphere of arbitrary radius centered at point M of aircraft location is used (Fig. 1.9). Points P'_N and $P's$, lying on the axis parallel to the axis of the Earth's own rotation, are called the north and south poles of the world, and this axis is called the celestial axis. Also the term of the plane of the celestial equator is introduced which is parallel to the plane of terrestrial equator. All celestial bodies are projected onto the celestial sphere at points C_i with keeping their relative angular orientation.

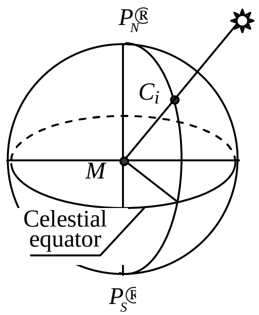


Fig.

In some cases, the center of the celestial sphere is more convenient to align with the center of the Earth, and the celestial sphere is aligned with the terrestrial, that is acceptable because of the remoteness of celestial bodies and as a rule is used in aviation astronomy. In this case, the point C_i is called geographical body position.

Depending on the solved tasks the horizontal and equatorial (moving and fixed) systems of astronomic coordinates are used. The center of celestial sphere in equatorial systems is usually connected with the center of the Earth.

Equatorial fixed system is (Fig.1.10, a), that does not rotate in the inertial space, is easy to take the relative position of celestial bodies. Planes of evaluation of angular coordinates are the plane of the celestial equator, and of *declination circle*, that is the plane passing through the celestial body C and through the celestial axis and that coincides with the plane of body meridian.

According to the name of this CS the positions of bodies are constant, while the Sun as a result of the annual motion of the Earth makes a turn by circle whose plane is called *ecliptic* and is inclined to the plane of equator at an angle of 23°27®The point of intersection of the circumference with the equator when the Sun passes to the northern hemisphere (March 21) is called the *vernal equinox* and is designated by symbol ♈.

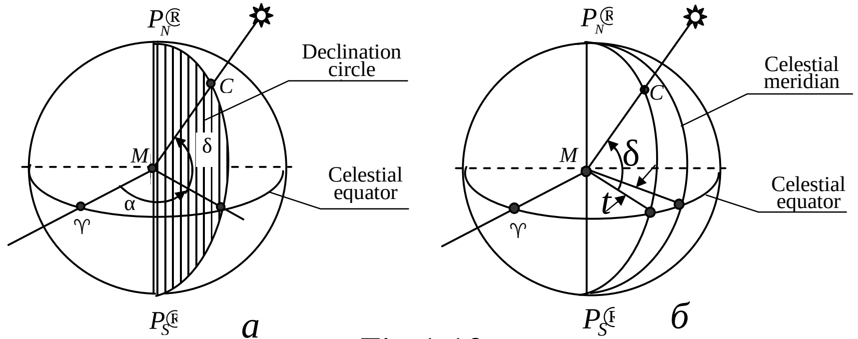


Fig 1.10

Position of celestial body in the fixed equatorial coordinate system is defined by angles ℑ and Ω The angle ℑ is so called *right ascension* and is measured along the arc of the celestial equator from the point ϒ to the circle of declination. The positive direction of measurement is shown by the arrow (see Fig. 1.10, a). The angle Ω is called the *declination* and is measured along the arc of declination circle from the celestial equator to the celestial body. When the body is in the northern hemisphere the angle Ω is considered to be positive. Naturally, the coordinates ℑ and Ω can also be defined by corresponding central angles.

Equatorial moving CS (Fig.1.10, b) is connected with the Earth and rotates relative to the fixed system with angular velocity Ω of the Earth's own rotation. It allows determining the position of celestial bodies relative the navigation CSs. Besides angle Ω, the coordinate of the body in this CS is *hour angle t*. The coordinate origin is the *celestial meridian* the plane of which coincides with the meridian of point *M* of object position. The angle *t* is measured between the planes of the celestial meridian and of body declination circle in the direction opposite to the Earth's own rotation. Note that the passing of celestial body through the celestial meridian, when *t* = 0, is called *the upper culmination*.

Local *t* and Greenwich t_{Gr} hour angles are mutually dependent

$$t\,\mathrm{X}t_{Gr}\,\Gamma\Leftarrow, \qquad (1.17)$$

where ⇐ is the eastern longitude of position.

Note that the hour angles of stars, of the Sun and of the planets vary with different angular velocities.

Since the value t_{Gr} depends not only on time, but varies from a body to another body, then the tabulation t_{Gr} for a large number of navigational stars is irrational. To do this, instead of (1.17) it is practical to use another form of expression of *t* with taking into account the relationships between moving and fixed systems of equatorial coordinates, which is explained in Fig. 1.11. This relationship is defined by sidereal time *S*, which is the hour angle of the vernal equinox. The definition of *S* implies the dependence

$$S\,\mathrm{X}t\,\Gamma\Im. \qquad (1.18)$$

By analogy with (1.17) it is possible to write down:

$$S\,\mathrm{X}S_{Gr}\,\Gamma\Leftarrow \qquad (1.18)$$

where S_{Gr} is Greenwich sidereal time.

From (1.18) and (1.18) the following expression can be obtained:

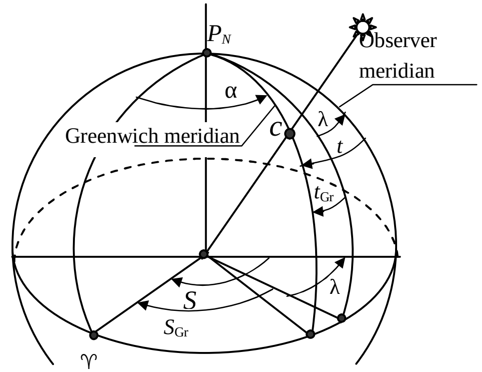


Fig.1.11

$$t\,\mathrm{X}S_{Gr}\,\Gamma\Leftarrow Z\Im. \qquad (1.19)$$

This expression is convenient for practical use, since the value of right ascension ℑ of stars is constant for many years, and S_{Gr} is tabulated as a function of Greenwich time.

In the horizontal coordinate system the position of celestial body is defined by altitude angle *h* and by azimuth *A* (Fig.1.12, *a*). Altitude angle is the angle between the horizontal plane and direction to the celestial body, and azimuth is the angle measured from the northern direction of meridian to the horizontal projection of direction to the body. To establish the relationships between the equatorial and horizontal coordinates represent the latter are placed on the celestial sphere centered at point *M* of location (Fig. 1.12, *b*). The reference plane for angles *A* and *h* is the horizontal plane and *the vertical of the body*, that is the vertical plane containing the point *C*. The point *Z* of local vertical *Z'Z* is called

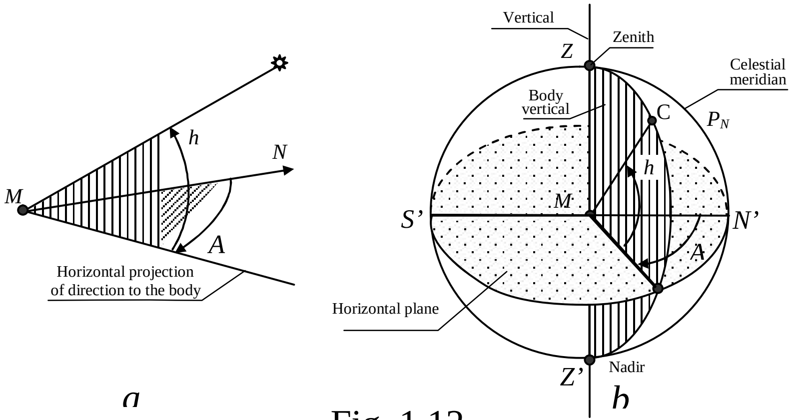


Fig. 1.12

zenith. The plane of the celestial meridian, which passes by definition through the points Z and P_N, intersects the horizontal plane at points of the north N' and the south S'. Obviously, the azimuth A can also be defined as the angle between the plane of the celestial meridian and the vertical. The arc ZC, which equals 90° Ζ h, is called the *zenith distance*.

Let's give short information about time measurement. Sidereal time in accordance with its definition varies with the angular velocity of the Earth's own rotation. *Sidereal day* is time interval between two successive upper culminations of the vernal equinox (time of one full turn of the Earth about its axis) and it is shorter than solar day by about 4 minutes because of annual rotation of the Earth. Considering the sidereal time is very important in astronavigation systems which bear stars.

In everyday life, so called *solar time* is usually used. Its unit of measurement is a second, determined as 1/86400 of mean solar day, which is equal to the time interval between two successive culminations of so called mean Sun, that is an imaginary point which uniformly moves along the equator (duration of the true solar day is not constant). Another definition of the second is 1/31556925.9747 of tropical year (for 1900), corresponding to averaging the observation data for 300 years. The accuracy of such time standard is of the order 10^{-11}. Because of increasing the requirements to the accuracy of time measurement in 1965 the atomic standard of a second was adopted, and a second is determined as a time during which there are 9192631770 transitions between two hyperfine ground states of caesium Cs 133, which gives precision about $5\ 10^{-13}$.

The origin of the solar time is the north (moment of lower culmination point of the Sun when its hour angle is ⇔ which depends on the local meridian. Therefore, the local time $T_м$, depending on the longitude ⇆ is not used in practice. For convenience, *the zone time* T_z is adopted, which varies by 1 hour relative to neighboring zones (total 24 zones) and at territory of CIS *the daylight saving time* T_d is introduced, which is greater than zone time by 1 hour ($T_d = T_z + 1$).

With astronomical measurements by T_d with taking into account a number N of zone the *Greenwich time* is determined $T_{Gr} = T_d$ Ζ N J 1.

By T_{Gr} for the measurement date it is possible to find t_{Gr} and Ω of the Sun and of planets and unified S_{Gr} of stars in the Aviation Astronomical Almanac.

1.4.3. Connection between astronomical and navigation coordinate systems

All methods of determination of the position coordinates and of aircraft heading by astronomical measurements are based on dependencies between astronomical and navigational CSs. This basic relationship is relationship between geocentric (geographical) and astronomical - equatorial (moving) and horizontal - coordinate systems, that is explained for the spherical model of the Earth in Fig. 1.13, *a* (see. also Fig. 1.10, *b* and Fig. 1.12). Vertical *ZM* is considered to be geocentric and directions to the body from point *M* and center *O* of the Earth are parallel.

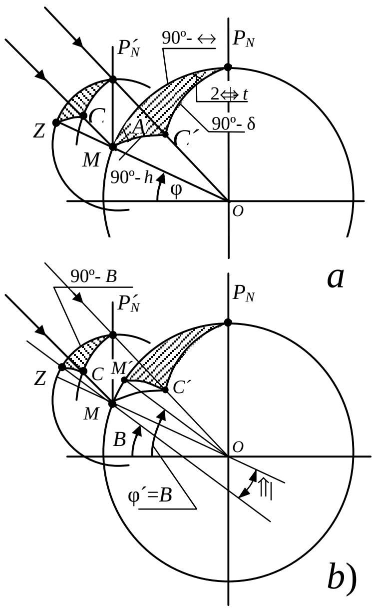


Fig.1.13

Then the spherical triangles $ZP'_N C$ on the celestial sphere and $MP_N C'$ on the Earth will be similar (triangle $ZP'_N C$ is called the *parallactic triangle*). Obviously, the arcs $ZP_N^{®}(MP_N)$, $P_N^{®}C(P_N C^{®})$ and $ZC(MC')$ are correspondingly equal to (90°Z ↔), (90°Z Ω) and (90°Z *h*), and the vertex angles $Z(M)$ and $P_N'(P_N)$ are equal to the azimuth *A* and to (2⇔J *t*).

For each of these triangles, for example $MP_N C^{\complement}$, it is possible to write the equation using formulas of spherical trigonometry which establish the relationship between the geocentric (geographical) coordinates ↔, ⇐ of point *M* and the astronomical coordinates of the body. From sines theorem it is possible to get:

$$\frac{\cos \Omega}{\sin A} X Z \frac{\cos h}{\sin t},$$

or, taking into account (1.19),

$$\sin A \, X Z \frac{\cos \Omega \sin t}{\cos h} X Z \frac{\cos \Omega \sin(S_{Gr} \, \Gamma \Leftarrow Z \Im)}{\cos h}. \qquad (1.20)$$

According to cosines theorem it is possible to write

$$\sin h \, X \sin \leftrightarrow \sin \Omega \Gamma \cos \leftrightarrow \cos \Omega \cos t \, X$$
$$X \sin \leftrightarrow \sin \Omega \Gamma \cos \leftrightarrow \cos \Omega \cos(S_{Gr} \, \Gamma \Leftarrow Z \Im). \qquad (1.21)$$

In equations (1.20) and (1.21) the parameters ℑ, Ω and S_{Gr} are known a priori (specified). Thus, with known latitude ↔ and longitude ⇐ of point *M* the solution of equations is azimuth *A* used to determine the heading. Equation (1.20) is only necessary to calculate *A*, whose value can be immediately found if longitude ⇐ is known in the case of direct measurement of *h*. The main navigational task of determining location coordinates can be solved by measuring the altitude angles of two bodies C_1 and C_2 if, similar to (1.21), a set of two equations is obtained:

$$\sin h_1 \, X \sin \leftrightarrow \sin \Omega_1 \Gamma \cos \leftrightarrow \cos \Omega_1 \cos(S_{Gr} \, \Gamma \Leftarrow Z \Im);$$
$$\sin h_2 \, X \sin \leftrightarrow \sin \Omega_2 \Gamma \cos \leftrightarrow \cos \Omega_2 \cos(S_{Gr} \, \Gamma \Leftarrow Z \Im), \qquad (1.22)$$

where indexes «1» and «2» correspond to numbers of the bodies. From these equations it is possible to find the values ↔ and ⇐.

There are other possible forms of writing the trigonometric equations of relationship between navigation and astronomical coordinates which are equivalent to those above mentioned, and there are also various methods of solving these equations.

There is a dependence of the navigation task solution using astronomical methods on the type of the vertical used in them. The fact is that the astronomical measurements are often based not on geocentric, but on the astronomical (true) vertical, which is approximated by geodesic one. In this case, the triangle $ZP_N^{®}C$ *will change since the zenith point Z is shifted, so that the arc* $ZP_N^{®}$ *will be equal to 90 °* J B, *where* B *is geodetic latitude (see Fig. 1.15, b), where for clarity the difference between angles* B *and* ¨ *is exaggerated. The triangle* $MP_N C^{\complement}$ *will be no more triangle similar to* $ZP_N^{®}C$ *. To maintain similarity it is enough to replace point* M *by point* M' *shifted to the north at angle* «*:*

$$\Uparrow X B Z \leftrightarrow$$

For each of the obtained similar triangles, replacing ↔ by *B*, it is possible to write system of equations similar to the considered one.

Thus, when measuring the altitudes h_1 and h_2 of bodies which are counted from the plane of true (more precisely, geodetic) horizon, the equation (1.22) directly determines the geodetic coordinates of point *M*: latitude *B* and longitude *L* = ⇐The value *B*, which is given on the ellipsoid, in this case, due to the application of spherical triangles (for the "earth" triangle $MP_N C'$) numerically is equal to its geocentric angle ↔ = *B*. From (1.20) the azimuth of celestial body is determined, which differs a little from angle *A*, obtained for geocentric vertical.

1.5. Flight and navigation parameters

Let's consider some of the most important flight and navigation parameters.

For solving the tasks of air navigation term "aircraft heading" is widely used.

Heading is an angle between certain reference direction in the plane of local horizon and the projection of aircraft's longitudinal axis *OX'* on the horizontal plane (Fig. 1.14).

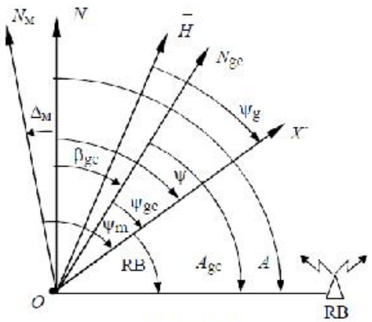


Fig. 1.14

Heading is evaluated clockwise from the chosen initial direction. Variants of the heading reference direction, more often chosen, are:

- direction to the north *ON* in the horizon plane, in this case the *true or geographical heading* э is defined;

- horizontal direction of the great circle parallel ON_{gc} (meridian) in case of determination of great circle heading э $_{gc}$;

- northern direction of the magnetic meridian ON_M defining magnetic heading э $_M$;

- horizontal direction which is measured by aircraft compass system. In this case there will be the indicated or compass heading. If the reference direction $\overline{H}$ is simulated by gyro or gyroscopic system, then determined heading is called gyroscopic one э $_g$.

The range of aircraft heading measurement is 0…360°.

Direction of the aircraft longitudinal axis projection on the horizontal plane *OX'* is also used for defining the relative bearings (RB) of certain point objects: landmarks, targets, other airplanes, radio stations and others.

Relative bearing (RB) of the reference point, for example of radio beacon, is called the angle between the direction of the aircraft longitudinal axis projection on the horizontal plane and horizontal direction to the reference point (object). Positive values of the relative bearing are defined clockwise from the horizontal projection of the aircraft longitudinal axis. The range of relative bearing measurements is 0…360°.

Azimuth of object *A* (A_{gc}) is called an angle determined in the horizontal plane from the northern direction *ON*, or from the horizontal direction of the great circle parallel ON_{gc} (for A_{gc}) to the direction to the object. Range of azimuth measurements is 0…360°.

More important flight and navigation parameters, used for solving the tasks of air navigation, are aircraft altitude and speed.

Flight altitude is geometric height, characterized by vertical distance between certain levels of reference point and aircraft.

Altitude above the ground profile is characterized as *true height* H_{tr} (Fig. 1.15), whose measurements are important for some flight stages.

If the altitude is measured relatively to any surface, for example, from the runway, this altitude is called *height* H (Fig. 1.15). It is evaluated from some datum, as a rule, from destination or departure aerodrome datums. Pressure *QFE* (*Q-code Field Elevation*) corresponds to height, i.e. atmospheric pressure at aerodrome elevation (or at runway threshold). Since during the flight above the runway true height and height coincide, international term 'height' is also used for relative height. For example, Transition Height (transition altitude, relative height, where and below it the aircraft vertical position is determined by *QFE*). This altitude, as a rule, is controlled during aircraft takeoff or landing.

If the initial reference level of the flight height is considered to be the mean sea level, then this altitude is called *true altitude* H_t (Fig. 1.15). International term for it is *Altitude*. Pressure *QNH* (*Q-code Nautical Height*) corresponds to true altitude, which indicates the pressure in the given ground point, reduced to the sea level.

To obtain pressure *QNH* it is necessary to determine pressure change on this elevation (for small altitudes, the height change in 11.2 m leads the pressure change in 1 mm Hg, or climbing on altitude of 800 m corresponds to pressure decrease to 100 GPa), knowing runway threshold elevation over sea level (common topographic parameter). Summing the pressure *QFE* (pressure at the runway threshold level) and calculated pressure change gives the pressure in the given point, reduced to the sea level (pressure *QNH*), i.e. pressure at the given point if it were located at the sea level.

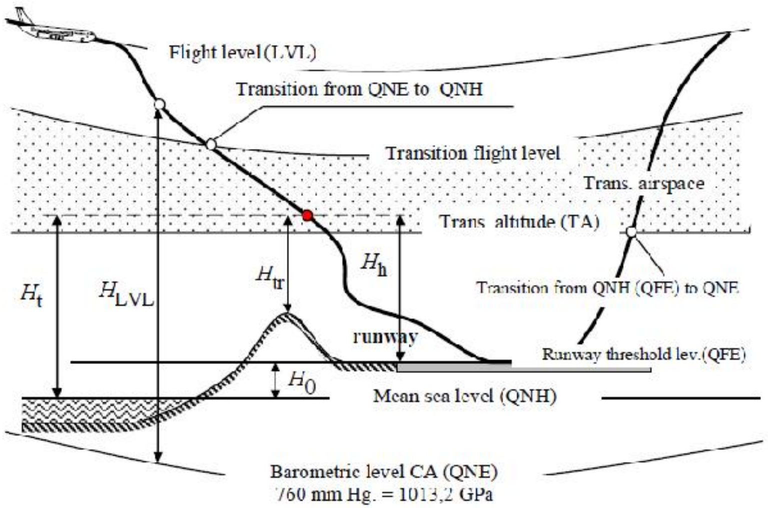


Fig. 1.15

Mean sea level (international designation MSL) in CIS countries, Russia and Poland is defined with the help of Baltic elevation system (i.e. at the Baltic Sea level in Kroonstad), and according to the ICAO standards - with the help of WGS-84 system (the ellipsoid surface GRS 80). These reference systems have some distinctions.

Obviously, the aircraft true altitude and height are connected in such a way:

$$H_h \; X H_t \; Z H_0$$

where H_0 is the geometrical height of the point, reduced as the reference point on the Earth's surface.

Before takeoff pilot sets the pressure on the altimeter indicator, which is given by dispatcher (flight operations director). For Russian and some CIS aerodromes it is pressure *QFE*, i.e. altimeter indicates zero altitude. In Ukraine and abroad before flight the pressure is set on the altimeter, reduced to the sea level, i.e. *QNH*. That's why altimeter indicates altitude of aerodrome elevation above sea level, but not zero.

One more definition of altitude level (Fig. 1.15) is flight level altitude H_{LVL} (international indication Flight Level), not confuse with flight level base altitude. For the purpose when airplanes, which are located each in its flight level (international indication LVL), are on the given interval from each other, it is necessary to make their altimeters work identically. To provide this the same initial pressure is set on each device. It is the standard atmospheric pressure 1013.2 GPa = 760 mm Hg = 29.92 inches Hg (international indication *QNE*). The altitude, which barometric pressure altimeter indicates, is usually called standard (true altitude of the standard atmosphere). It is used for provision of flight separation.

The change of the reference system from flight level to true altitude and vice versa takes place (Fig. 1.15) on the transition altitude (TA) at the climbing and on the transition flight level (TL) at the descending. Space layer between TA and TL is called transition airspace. Only climbing and descending are allowed in this layer, i.e. transition evolutions from altitude to flight level and from flight level to the altitude. Horizontal flight in transitional airspace is forbidden.

According to the ICAO norms there is standard nomenclature of the characteristic altitudes and flight levels in the aerodrome's area and during landing approach. For example, there are *Critical Height* (minimal altitude above the aerodrome); *Decision Height* (altitude of the decision making); *Transition Altitude* (true flight altitude, above and below which vertical aircraft position is determined by *QNH*); *Transition Level* (the lowest flight level, which can be used above TA, determined by *QNE*), etc.

What is more, aircraft flight altitudes lower than 200 m are called ground-hugging, from 200 to 1000 m are called low, from 1000 to 4000 m are called medium, from 4000 to 12000 m are called high, more than 12000 m are called stratospheric altitudes.

As for another flight and navigation parameter, namely speed, then depending on the chosen coordinate system, relatively to which flight speed is determined, there is true airspeed, earth-fixed flight speed and absolute speed.

True airspeed $\vec{V}$ is aircraft speed relatively to the air environment (relatively to ram airflow).

Earth-fixed flight speed $\vec{V}_f$ is aircraft speed relatively to the chosen coordinate system, which is connected to the Earth.

Projection of the aircraft earth-fixed flight speed on the horizontal plane (horizontal component of the aircraft speed relatively to the Earth) is called *ground speed* $\vec{V}_g$. Vector of the ground speed is directed as the tangent to the flight track.

Earth-fixed flight speed $\vec{V}_\mathrm{f}$ is a vector sum of the airspeed and wind speed $\vec{W}$.

Wind speed $\vec{W}$ is the speed of the air environment movement, which is not disturbed by the aircraft, relatively to the chosen coordinate system, which is connected to the Earth.

If the vectors of airspeed $\vec{V}$ and wind speed $\vec{W}$ are projected on the local horizon plane, horizontal components of the airspeed, wind speed vector and ground speed vector $\vec{V}_g$ create so-called wind triangle, which is illustrated in Fig. 1.16.

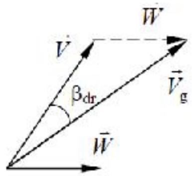


Fig. 1.16

Angle β_{dr} between the horizontal projection of the airspeed vector $\vec{V}$ and ground speed direction $\vec{V}_g$ is called drift angle.

Vertical speed $\vec{V}_y$ is the vertical component of the aircraft speed relatively to the Earth or rate of flight altitude change.

Indicated airspeed is the speed, which is shown by airspeed indicator, calibrated by the difference between total and static pressure of air.

Rectified airspeed is indicated airspeed, which takes into account instrumental error and aerodynamic correction.

Calibrated airspeed is rectified airspeed, which takes into account compressibility correction, connected with the air pressure deviation from the mean sea level pressure.

Frequently the indicated airspeed has no differences with the calibrated airspeed. Calibrated airspeed is taken into consideration during theoretical calculation. Indicated (rectified) airspeed is purely pilot-centered parameter. It is very often used in such aircraft modes as takeoff run, takeoff and landing.

On the each stage of aircraft flight the indicated airspeed is defined by airworthiness and ICAO standards with its further keeping up because of safety provision. Due to this there is standard airspeed nomenclature: minimum control speed of the takeoff run, minimum lift-off speed, decision speed, rotation speed, takeoff safety speed, lift-off speed, minimum control speed of the landing approach, maximum speed of the landing approach, stalling speed, maximum operating speed, etc.

Nowadays because of using inertial navigation systems aboard, aircraft *absolute speed* $\vec{V}_a$ is widely applied. Term “absolute” is used here with the view to emphasize that speed determination is realized in some inertial or conditionally inertial reference coordinate system. Usually aircraft motion is considered herewith and, in accordance, absolute linear speed relatively to geocentric coordinate system, which axes do not rotate in the inertial space, is determined. In this case value of the aircraft absolute speed vector $\vec{V}_a$ can be represented as the sum of its earth-fixed flight speed $\vec{V}_f$ and transferring speed $\vec{V}_{tr}$, i.e. the linear speed of point in the airspace, where aircraft is located. The value and direction of transferring speed are determined by the Earth rotary translational motion. Herewith,

$$\vec{V}_a = \vec{V}_f + \vec{V}_{\mathrm{tr}} = \vec{V}_f + \vec{\vartheta}_0 \times \vec{R},$$

where $\vec{\vartheta}_0$ is earth angular velocity; $\vec{R}$ is radius-vector, which is directed from the Earth center to the aircraft location point.

The value of the transferring speed in this case can be calculated by the formula:

$$\vec{V}_{tr} \text{ X} \vartheta_0 R \cos \leftrightarrow \text{X} \vartheta_0 l,$$

where ↔ is geocentric latitude, l is distance between aircraft and polar axis.

Transferring speed vector is directed from the west to the east as the tangent to the geographic parallel.

Using the term “absolute linear speed” it is necessary to note that it cannot be accepted successful in terms of the principle of Galilean relativity, which sets the equality of all inertial reference coordinate systems.

Information about the aircraft flight speed is used for solution of navigation tasks: dead reckoning, determining the time of aircraft arrival to the given waypoint.

For prevention of aircraft or engine critical regimes, the information about Mach number M is used.

Mach number M is the ration between true airspeed $\vec{V}$ and the sound speed a:

$$M \text{ X} \frac{V}{a} \text{X} \frac{V}{\sqrt{kRT}},$$

where k is constant of the adiabatic process; R = 287,05287 (J/kg K) is specific air constant; T is air temperature.

Aircraft angular orientation in the airflow is defined by the attack and slip angles.

Angle of attack α is an angle between aircraft longitudinal axis and projection of the true airspeed vector on the aircraft’s symmetrical plane.

Slip angle β is an angle between true airspeed vector and the aircraft’s symmetrical plane.

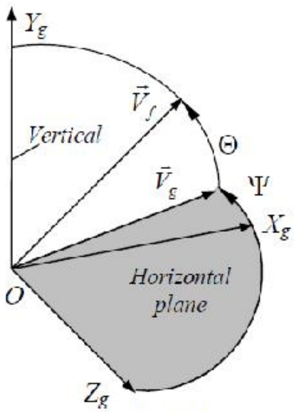


Fig. 1.17

Flight path angles, illustrated in Fig. 1.17, are used in the air navigation as the kinematic parameters, which characterize the motion of aircraft’s center of gravity.

Herewith angle Ψ is defined as *track angle*, that is, the angle between axis OX_g of the normal coordinate system (direction of geographical meridian) and direction of ground speed vector $\vec{V}_g$.

Flight path inclination angle Θ is the angle between vector of earth-fixed flight speed $\vec{V}_f$ and horizontal plane X_gOZ_g. Positive directions of these angles are shown in Fig. 1.17.

Note that definition of the track angle Ψ, used in navigation, corresponds to the definition of the course angle, but positive values of the track angle as the heading are measured clockwise from the direction fixed in the horizon plane.

Aircraft attitude (in the body-fixed coordinate system *OXYZ*) is determined by the pitch ν| and roll ↑ angles relatively to the horizon plane (in the normal coordinate system $OX_gY_gZ_g$).

Pitch angle ν| is an angle between aircraft longitudinal axis and the horizontal plane OX_gZ_g. The range of measurement for the pitch angle is Z90° ™ν|™90°. The positive (right) direction of pitch means that aircraft longitudinal axis is located higher than horizontal plane OX_gZ_g.

Roll angle ↑ is an angle between vertical axis of the aircraft and the vertical plane, which passes through longitudinal axis of the aircraft. The range of measurement for roll angle is Z90° ™↑™90°. The positive (right) direction of roll means that aircraft lateral axis is located lower than horizontal plane OX_gZ_g.

Let us introduce two additional interconnected flight parameters of aircraft: acceleration and overload.

Absolute acceleration of the aircraft center of mass $\vec{w}_{abs}$ is defined by the ratio of the resultant force, affects the aircraft, and mass according to the second Newton’s law, i.e.:

$$\vec{w}_{abs} \mathrm{X} \vec{F} / m. \tag{1.23}$$

Vector direction of the absolute acceleration $\vec{w}_{abs}$ coincides with the vector direction of the resultant force $\vec{F}$. Considering aircraft motion relative to the Earth's surface the vector of absolute acceleration can be represented as the sum of the relative $\vec{w}_{rel}$, transferring $\vec{w}_{tr}$ and Coriolis $\vec{w}_{cor}$ accelerations:

$$\vec{w}_{abs} = \vec{w}_{rel} + \vec{w}_{tr} + \vec{w}_{cor}.$$

Relative acceleration $\vec{w}_{rel}$ can be defined as the time derivative of earth-fixed flight speed vector $\vec{V}_f$. Transferring acceleration $\vec{w}_{tr}$ is an acceleration of aerospace point near to the Earth, where aircraft is located and which is determined by the Earth diurnal rotation. The vector of this acceleration is always directed as the perpendicular to the Earth's rotation axis, and its value is calculated as:

$$w_{tr} \mathrm{X} \vartheta_0^2 l,$$

where l is the distance to the Earth's rotation axis.

Value and direction of Coriolis acceleration vector $\vec{w}_{cor}$ are determined by dependence:

$$\vec{w}_{\mathrm{cor}} \mathrm{X} 2\vec{\vartheta}_0 | \vec{V}_f.$$

Sometimes it is easier to represent the resultant of all external forces, acting on the aircraft, as the sum of the gravitational forces $\vec{Q}$ and resultant $\vec{R}$. Resultant $\vec{R}$ is the resultant of all non-gravitational forces (engine thrust, aerodynamic force, etc). In this case instead of equation (1.23) we get

$$\vec{w}_{abs} \mathrm{X} \frac{\vec{Q}}{m} \Gamma \frac{\vec{R}}{m} \mathrm{X} \vec{g}_{\mathrm{g}} \Gamma \vec{a}.$$

Hence

$$\vec{a} \mathrm{X} \vec{w}_{abs} \mathrm{Z} \vec{g}_{\mathrm{g}},$$

where $\vec{g}_{\mathrm{g}}$ is vector of the gravity field intensity in the point of aircraft position, $\vec{a}$ is vector of so-called imaginary acceleration or acceleration from the resultant $\vec{R}$. Sometimes vector $\vec{a}$ is called as vector of the specific resultant, which accounts for the unit of aircraft mass. Definition of the 'aircraft overload' is practically identical to this definition.

Overload $\vec{n}$ is the ratio of the resultant $\vec{R}$, applied to the object, and the product of the object mass m and standard gravity acceleration g, i.e.

$$\vec{n} \mathrm{X} \frac{\vec{R}}{mg}.$$

Considering that

$$\frac{\vec{R}}{m} \mathrm{X} \vec{a},$$

we get

$$\vec{n} \mathrm{X} \frac{\vec{a}}{g}.$$

Hence it is seen that overload is unitless vector value, which has quantitative distinctions from acceleration vector under the action of non-gravitational forces by the constant multiplier $(g)^{\mathrm{Z}1}$, which has dimension, which is inverse to the acceleration dimension.

Overload vector direction $\vec{n}$ corresponds to the direction of so-called apparent (imaginary) vertical. Imaginary vertical is a direction, which is fixed by human vestibular apparatus or by physical pendulum, which is used for determining the vertical direction on the moving object. In the conditions of vehicle motion the direction, which is fixed by the physical pendulum, is defined by vector direction of gravity field intensity, as well as by the vector components of absolute acceleration of vehicle.

1. What groups can the civil avionics be conditionally divided into?
2. What systems can be referred to the class of autonomous navigation systems? What systems can be referred to the class of radio navigation systems?
3. Name the main navigation methods.
4. What is the base of the positioning navigation and dead reckoning method?
5. What Earth's shapes are practically used in the accurate navigation reckoning?
6. What relations are used for conversion from the geodetic coordinate system to the spherical one?
7. What direction is called true or celestial vertical?
8. What parameters are used for characterizing the Earth magnetic field?
9. What is standard atmosphere?
10. Name the main parameters of standard atmosphere.
11. Give examples of main formulas, which set the dependence between altitude and atmospheric (static) pressure.
12. What coordinate systems are frequently used in the airborne systems?
13. What is the feature of the great-circle coordinate system?
14. What is the great-circle course?
15. What celestial coordinate systems are usually used in the celestial navigation systems?
16. Give the example of equations of the navigation and celestial coordinates' relation.
17. What directions are more often chosen as the heading reference? What headings are herewith determined?
18. What altitude nomenclature is used in aviation?
19. How are the aircraft speeds differed relatively to the chosen reference system?
20. What angles define the aircraft angular attitude in the space and relatively to the horizon plane?

CHAPTER 2. AEROMETRIC NAVIGATION SYSTEMS

2.1. Air data computer systems

2.1.1. General information of air data computer systems

Aerometric devices provide autonomous continuous measurement of aerometric parameters for provisioning the prescribed modes of piloting and navigation. Construction principles of mechanical aerometric gauges (altimeters, airspeed and Mach number indicators, vertical speed indicators) are given in detail in other disciplines. However, it is considered more rationally to obtain a large number of aerometric parameters within a single system, which is able to provide both operation of indicators and presentation of information in a variety of airborne systems.

Examples of such aerometric systems are air data computer systems (ADC) and flight environment data system.

The air data computer system is designed for presenting the primary flight information on indicators in the cockpit and in other airborne systems. Before introducing ADC system, airplanes used separate sensors of aerometric parameters, annunciators and indicators. This was done for individual consumers independently from others. In this regard, there could be a redundancy of mismatched information, duplication of equipment mass, unnecessary material costs, and complications of equipment maintenance technology in general.

The air data computer system combines all sensors and indicators into a single ideology, eliminating the duplication and inconsistency of information. The air data computer system is the autonomous system consisting of sensors of primary aerodynamic parameters, computer and indicators. It presents information about the values of the primary aerometric parameters.

These parameters include: Mach number M, true airspeed V and indicated airspeed V_i, barometric altitude (absolute H_{abs} and relative and H_{rel}), outside temperature T, deviation of Mach number ΔM, of altitude ΔH and speed ΔV from the prescribed values.

The principle of operation of such systems is the aerometric method of determining the aircraft motion parameters. The aerometric method is based on dependencies that determine the standard atmosphere (see paragraph 1.2.4), as well as on certain equations of constraints between aerometric parameters. In this case, functional relationships of the aerodynamics between measured and calculated parameters serve as the equation of constraints.

The primary measured parameters are the static (atmospheric) pressure P_{st}, pitot pressure P_p, and temperature T_s of stagnated (fully decelerated) airflow. The dynamic pressure P_d, characterizing the impact pressure, is obtained by subtracting the static pressure P_{st} from the pitot pressure P_p.

In practice, the difference $(P_p - P_{st})$ coming from pitot-static tube differs from the impact pressure due to the tube inaccurate manufacturing, so the dynamic pressure calculation involves a correction factor ξ_{corr}, that may vary within 1.02 ... 0.98:

$$P_d = \xi_{corr}(P_p - P_{st}).$$

To calculate the aerometric parameters, the gradable formulas of separate aerometric devices are used.

A dimensionless characteristic of the aircraft flight speed is Mach number. To calculate Mach number for subsonic flight speed, the well-known gradable formula of Mach number indicator is used.

$$M = \sqrt{\frac{2}{k-1}\left[\left(\frac{P_d}{P_{st}}+1\right)^{\frac{k-1}{k}}-1\right]} = f_M\left(\frac{P_d}{P_{st}}\right), \qquad (2.1)$$

where $k = 1.4$ is the adiabatic constant.

If there is information about Mach number, the formula (2.1) for calculating the true airspeed V is converted to the following form:

$$V = aM,$$

where a is the speed of sound at a given height, which simplifies calculating operations, that is not to use the gradable formula in calculating the true airspeed.

The sound speed $a = \sqrt{kRT}$ depends on the air temperature T (R = 287,053J/kg K is the specific gas constant of air, k is the adiabatic constant). Since in the flight the temperature cannot be measured directly because of the gauge aerodynamic heating, it is calculated using the ratio

$$T = \frac{T_s}{1+0.2M^2},$$

where T_s is the stagnated airflow temperature (temperature of fully decelerated airflow). The denominator of the formula reflects the factor of aerodynamic heating of air during its stagnation.

Thus, when calculating the true airspeed the following dependence is used:

$$V = M\sqrt{kRT} = K_R\sqrt{T_s}\, f_V(M), \qquad (2.2)$$

where $K_R = \sqrt{kR}$; $f_V = \frac{M}{\sqrt{1+0.2M^2}}$ is the function formed by Mach number calculation.

The indicated airspeed $V_i = f(P_d)$ is obtained as function of only dynamic pressure P_d.

$$V_i = a_0\sqrt{5\left[\left(\frac{P_d}{P_0}+1\right)^{\frac{2}{7}}-1\right]},$$

where a_0 is a value of the sound speed under normal conditions for standard atmosphere at sea level, P_0 is the mean value of atmospheric pressure, corresponding to normal conditions at sea level.

The absolute barometric height H_{abs} is calculated by hypsometric equation:

$$H_{abs} = \frac{T_0}{\beta}\left[1-\left(\frac{P_{st}}{P_0}\right)^{R\beta}\right], \qquad (2.2)$$

where T_0 = 288K (15°C) is the average temperature value at sea level; $\beta = -6{,}5 \cdot 10^{-3}$ deg/m is lapse rate (temperature gradient).
Also absolute barometric height in (2.2) can be determined by Laplace formula

$$H_{abs} = RT_{ave} \ln \frac{P_0}{P_{st}}.$$

The value of average temperature T_{ave} depends on the height and on the midlatitudes it is defined by dependencies:

$$T_{ave} = \frac{T_0 + T_H}{2} \text{ for } H \le 11\,000 \text{ m};$$

$$T_{ave} = T_{11} + \frac{T_0 - T_{11}}{2} \frac{H_{11}}{H} \text{ for } 11000 < H < 30\,000\text{m},$$

The relative barometric height H_{rel} is calculated as follows:

$$H_{rel} = H_{abs} - H_{abs.\,giv.},$$

where $H_{abs.\,giv.} = f\,(P_{giv})$ is the absolute barometric altitude of a given point on the Earth surface, having the pressure P_{giv}.

The first air data computer systems were analog both in our country and abroad. They included the most common analog system СВС-ПН-15, СВС-72, installed in many aircraft, including IL-62, An-22, Tu-154, Yak-42, An-124 and others.

In the first analog ADC systems, aerometric parameters were calculated using self-balancing potentiometric bridges. To provide nonlinear functional relationships, the closed loops of bridge calculation circuits, the functional potentiometers or compacted components were included.

In more advanced analog ADC systems, calculation involves scale amplifiers, transformers for division and multiplication, and gradable dependencies resolving is carried out in non-contact functional voltage converters, reproducing nonlinear functional relationships.

In 1980s years, digital ADC systems began to be intensively implemented in exploitation. Aerometric parameters are calculated in the digital ADS by algorithms that are implemented as software of the digital processor.

The input information to calculate aerometric parameters comes to ADC systems from the dynamic and pitot-static tubes and from sensor of stagnated airflow temperature; in some ADC systems, information from vane sensors (about angle of attack and slip angle) is used.

2.1.2. Primary information sensors of aerometric systems

Motion parameters are unique functions of static and pitot pressure. The pitot and static pressures come to devices from pitot-static tubes, which together with the pneumatic pipelines create a powering system of aerometric devices and systems.

Pressure tubes largely determine metrological and technological characteristics of aerometric systems such as accuracy, measurement ranges, reliability, and quality of system output information and ease of its use. In this regard, aviation science and technology experts pay special attention to pressure sensors.

A lot of companies and enterprises are engaged in means of pressure measurement in flight. The basic developers of these means are Ulyanovsk Instrumentation Design Bureau and Aeropribor-Voskhod (Russia); Rosemaunt Inc. (USA); Smiths (England); Badin-Crouset (France); Dornier (Germany). These firms develop and deliver high-precision pressure tubes to be used in civil and military airplanes, helicopters and other vehicles.

Pitot pressure tubes (PPT) are intended only to perceive the pitot pressure of the ram airflow. The pitot pressure P_p is the pressure per unit of body surface, the plane of which is perpendicular to the velocity vector of airflow. The pitot pressure P_p is the sum of the static P_{st} and dynamic P_d pressures.

Constructively, PPTs are made in the form of a hollow cylindrical body (Fig. 2.1). Figure shows that the pitot pressure P_p of ram airflow over the body will be only at point A.

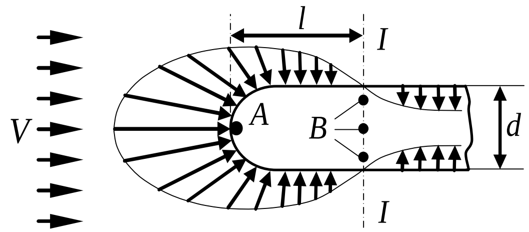


Fig. 2.1

Making a hole in the cylinder in the vicinity of point *A*, along its cavity there will be the pressure equal to the pitot:

$$P_p \,\mathrm{X} P_{st} \,\Gamma\, \zeta P_A \,\mathrm{X} P_{st} \,\Gamma\, P_d .$$

The dynamic pressure P_d numerically is equal to overpressure, i.e. the ram pressure q:

$$P_d \,\mathrm{X} P_d \,\mathrm{Z} P_{st} \,\mathrm{X} q \,\mathrm{X} \frac{\partial V^2}{2} .$$

As shown in Fig. 2.2 illustrating the streamline of pressure tube by airflow, the ram pressure q can be taken fully only at point *A*. Here d is the diameter of the hollow cylinder and l is its length.

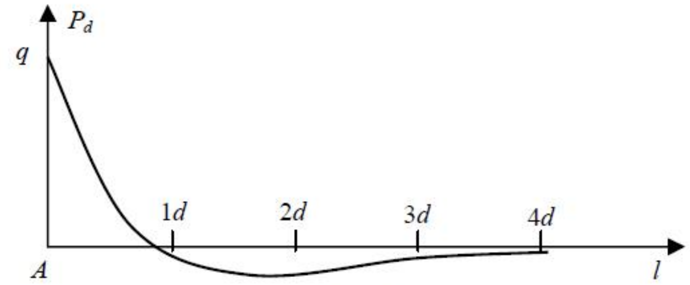


Fig. 2.2

Figure 2.3, a shows the most common constructive variant of the pitot pressure tube ППД-4, and Fig. 2.3, b presents the exterior of the pitot pressure tube ППД-11.

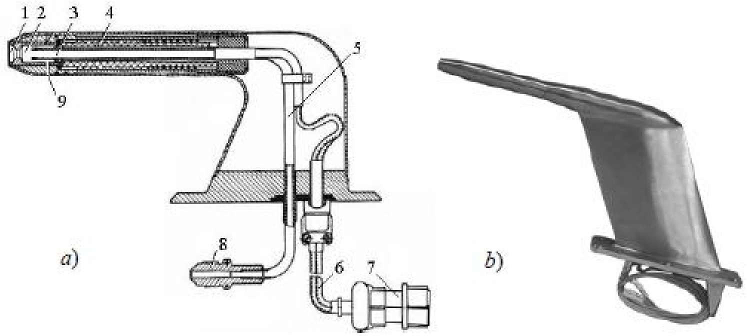


Fig. 2.3

The airflow coming (see Fig. 2.3, a) through the inlet 1 to the chamber 2 is stagnated. As a result, the pressure in the chamber equals the pitot pressure P_p, which is fed through the tube 3 and pneumatic pipeline 5 to the connecting pipe 8 connected to the pitot pressure line.

Drain holes 9 in the body are designed to drain water from the chamber 2. The pressure tube is heated by element 4 made from a nickel wire that is powered by the voltage coming through the wires 6 of the plug connector 7.

Obviously that relatively to the airflow the best location for PPT is provided when the plane of cross-section of pressure tube inlet is perpendicular to the velocity vector *V*. In this case the pressure tube error is only caused by losses of flow in the PPT channel cavity. This condition is equivalent since the tube longitudinal axis coincides with the direction of airflow. Even in this case, the pressure tube has error ΔP_p of about 2%.

In practice, the direction of PPT longitudinal axis and the airspeed vector direction differ in angles of attack α and slip angle β. This results in additional angular errors $\Delta P_p = f(\alpha)$ i $\Delta P_p = f(\beta)$. Another cause of the PPT angular error is the airflow wash in the tube installation point aboard. This error is the error of local attack angle difference from the true one (standardized by aircraft airworthiness standards within less than 3% throughout the aircraft speed range).

The most effective ways to reduce PPT angular errors are the following:

- PPT installation on a device that orients the pressure tube in the airflow during flight (used for measuring low flight speeds);

- optimization of PPT construction as an aerodynamic body to reduce angular errors.

Researches of PPT construction in wind tunnels show that the essential to the pitot pressure perception quality is the tube head shape and the ratio of internal and external diameters. It is proved that PPT head should not be rounded and of streamlined form, as for existing tubes, and the inlet diameter should be as large as possible.

The positive impact on PPT quality has the presence of stabilizing chamber and head inlet angle (Fig. 2.4, a, 2.4, b, respectively).

In terms of reducing angular errors, shielded PPTs are considered to be effective. The inlet tube in such PPT is placed in a hollow cylinder (Fig. 2.5) with much greater internal diameter.

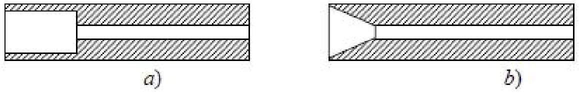


Fig. 2.4

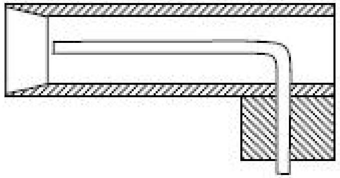


Fig. 2.5

The insensitivity to airflow wash of such tubes is explained by the fact that the airflow through the outer tube provides a constant flow direction inside the tube compared to the ram airflow. The presence of the external tube inlet confuser enhances this effect. The best is to locate the inlet tube at the end of the outer tube confuser.

Examples of this pressure tube type are the shielded tube 851FV made by Rosemaunt (USA) (Fig. 2.6, a) and the pitot pressure tube ППД-8В made by Ulyanovsk Design Bureau (Fig. 2.6, b).

Fig. 2.6

Due to selection of installation location aboard the aircraft, due to design techniques and calibration in wind tunnels, PPT errors are reduced to ± (0.005 ... 0.01) from the value of ram pressure.

For the perception of only static pressure of ram airflow, *static pressure tubes* (SPTs) are used. All SPTs, regardless of the specific construction, can be divided into two main groups: flowing and non-flowing tubes.

In the flowing type SPTs, the static pressure is taken from the inner (usually profiled) surface of the hollow cylindrical body. For non-flowing SPTs, the typical intake of pressure is made on the external streamline surface, which can also be profiled. The profiled surface refers to a regular change in the SPT internal or external diameters made as axisymmetric bodies of revolution. Both types of tubes have their own features and area of use.

When an object moves relative to the ambient air, the airflow is disturbed. Therefore, the static pressure should be measured at such a point on the object surface, where the flow is disturbed the least. The static pressure impacting on the object that moves in the air is considered the pressure that would exist at a given point of the air undisturbed by the object.

The diagrams of overpressure distribution on the surface of the hollow cylindrical body (see Fig. 2.1) show that in the section plane *I-I*, the pressure on the body surface is distorted the least. If along the perimeter of this hollow cylinder one makes inlet holes *B* in the section plane *I-I*, the pressure inside the body will be equal to the static pressure. The ratio of distance *l* from the holes to the cylindrical body nose and the cylinder diameter *d* is kept within l ⌉ $3d$.

In aviation, the role of hollow cylindrical tube at the subsonic speed can perform the aircraft fuselage, where the inlet holes (In) for static pressure are made (Fig. 2.7).

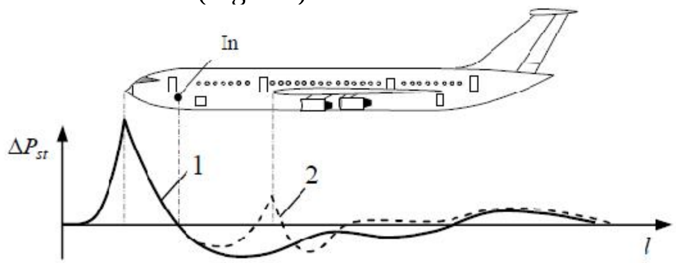


Fig. 2.7

Figure 2.7 shows the typical subsonic distribution along the fuselage line for the disturbed static overpressure ζP_{st} caused by the aircraft:

1 – taking into account only fuselage shape; 2 – taking into account the fuselage together with the wing and tail unit.

Used in practice the non-flowing SPTs taken the aircraft fuselage as the surface of static pressure tube, are made either in the form of inlet holes at certain points on the fuselage surface (Fig. 2.8, a), or as a special plates (plate tubes) (Fig. 2.8, b) with holes in these points. Together with the body, the plate simulates an instrument for the perception of static pressure.

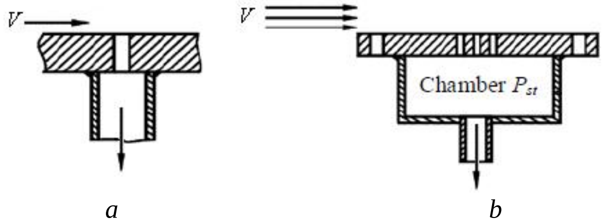


Fig. 2.8

On the fuselage, such places for installation of SPT are chosen, where there are the lowest levels of ζP_{st} (see Fig. 2.3). The tube plate is installed on the aircraft at the same level with the skin. The general view of the non-flowing plate SPT is shown in Fig. 2.9.

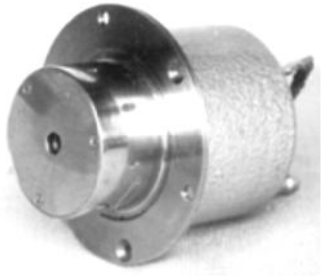


Fig. 2.9

In the flowing type SPTs, the static pressure is taken from the internal surface of the profiled tube. The structural diagram of the flowing type tube is shown in Fig. 2.10, a, and Fig. 2.10, b shows the exterior of static pressure flowing type tube ПДС-В1.

Structurally the tube consists of confuser (narrowing) and diffuser (expanding) parts. The confuser consists of two parts, separated by a cylindrical section, where the static pressure is taken.

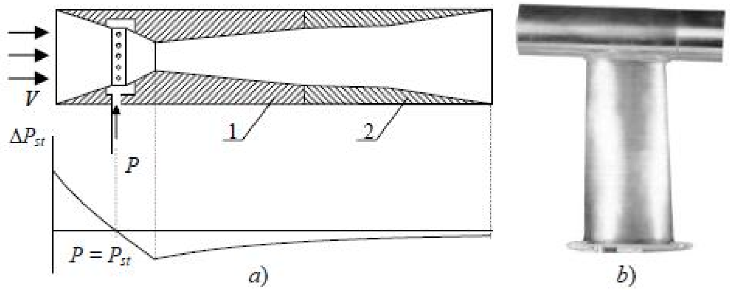


Fig. 2.10

As it can be seen from the graph presented in Fig. 2.10, a, at the tube inlet the increased pressure ($\zeta P_{st} > 0$) is created. As the flow narrows and accelerates, the static pressure in the flowing part decreases, reaching a minimum in the narrowest section. Then, through the flow extension in the diffuser, the pressure increases. Inlet holes for taking pressure are placed in the confuser at such a diameter, where $P = P_{st}$. Displacing the static pressure reception section, it is possible to change the coefficient of the pressure tube (to the right Z $\zeta P < 0$, to the left Z $\zeta P > 0$) and to perform the aerodynamic compensation of pressure measurement error.

The same problem can be solved by changing the configuration of the diffuser, which consists of two parts: the main part 1 and detachable tail piece 2. To correct the speed characteristics it is enough to install a tail piece on the tube.

The speed range of flowing tube is limited by the advent of critical mode when the flow rate in a narrow section reaches the speed of sound. In this case, the speed characteristic of the tube is significantly distorted. Therefore, the flowing SPTs are mainly used for low-speed aircraft and helicopters.

Non-flowing SPTs are often connected with PPT, thus creating combined tubes intended for simultaneous perception of pitot and static pressures in flight. These interconnected tubes are named as *pitot-static tubes* (PSTs).

When installing such a tube on the fuselage, the additional component of static pressure measurement error may appear that in accordance with the existing aircraft airworthiness standards should not exceed 5% of the ram airflow.

In this case, the measurement of static pressure with acceptable accuracy involves correction (compensation) of this error. The aerodynamic compensation is based on the artificial change of static pressure value in the vicinity of static pressure perception holes.


Fig. 2.11

Figure 2.11 shows the exterior of pitot-static tube ПВД-30, and Fig. 2.12 presents its structural scheme.

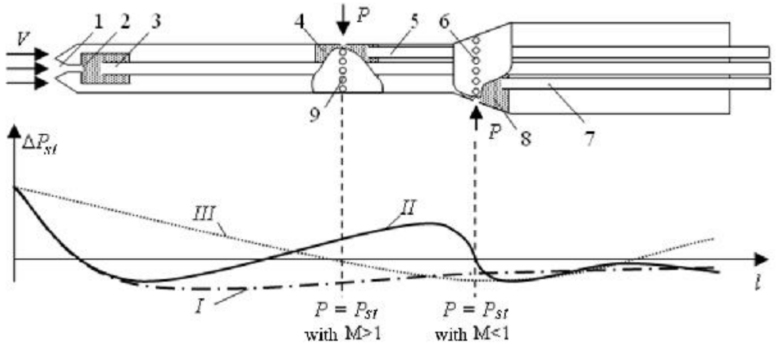


Fig. 2.12

During the aircraft flight, a part of the ram airflow is slowed down at the PST end piece and enters through the inlet 1 into the chamber 2. The kinetic energy of the airflow transforms into the potential energy, resulting in creation of full pressure in the chamber 2, which is fed to the pressure line through the pneumatic pipeline 3.

The static pressure of air surrounding the aircraft is perceived by holes 6 and 9, which are connected with two sealed chambers 4 and 8, output to connecting pipes.

At the subsonic speeds of ram airflow ($M<1$), flow is formed around the tube. At the PST nose, the static pressure is maximum and close to pitot. With removal from the tube nose that has a purely cylindrical shape (curve *I* in Fig. 2.12) it reduces and falls to a minimum ($\zeta P_{st} < 0$).

With the presence of correction circuit, the flow around the tube has the form of expanded cone, in which the pressure first decreases and then begins to increase (curve *II* in Fig. 2.12). The pressure growth continues to the compensation flute (correction cone). Here, because of the flow narrowing by compensation flute, the airflow is accelerated and its pressure decreases, reaching a minimum value at the maximal diameter of flute. The static pressure is taken from the static pressure subsonic chamber 8 through holes 6 which are made along the flute diameter, on a section, where $P = P_{st}$.

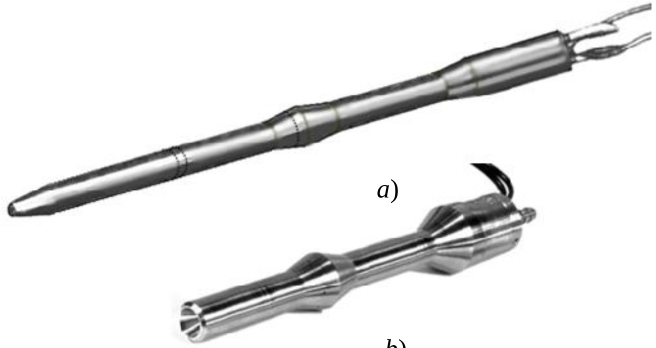
The supersonic airflow ($M > 1$) in front of the tube forms compression shock. The flow formed around the tube takes the form of *Laval nozzle*. In the section of the narrowing flow, the pressure decreases, and the speed increases, reaching values of sound speed in the narrowest section of Laval nozzle. Then, due to expanding the flow, the speed increases and at a certain section reaches the flow rate value that is incident on the tube. Here, the value ζP_{st} is close to zero (curve *III* in Fig. 2.12). In this section the pressure is taken through holes 9 and static pressure supersonic chamber 4.

From both tube chambers, the pressure is fed to the static pressure perception system through pneumatic pipelines 5 and 7. The static pressure perception system connected to this tube has a special device (pneumatic switch), which provides switching of static pressure lines depending on the flight speed.

In supersonic aircraft, due to the development of flow stall it is more difficult to find a place on the fuselage, which is consistently streamlined by the airflow. Therefore, such airplanes primarily use the tubes installed on the bar in front of the aircraft. Figure 2.13 shows the exterior of the tube ПВД-18.

The tube ПВД-18 has three separate static pressure chambers. The first two rows have holes located on the cylindrical section of the tube. The tube has two heating elements connected with each other in series. There is one pitot pressure connecting pipe and three static pressure ones.

PSTs are used in the same way in helicopters to avoid the impact of the flow from the main rotor on the perception of air pressure. Figure 2.13, b shows the exterior of pitot-static tube ПВД-К for helicopters.



a)

b)

Fig. 2.13

Developers work constantly both to improve the construction of tubes which are mass-produced, and to investigate the possibility of creating and implementing advanced PST. Among such developments one can distinguish static pressure spherical tube developed by Ulyanovsk Design Bureau (Fig. 2.14).

The main element of the tube is a sphere of porous nickel, inside which the static pressure perception chamber with holes of a certain diameter is located. The static pressure is vented from the chamber by pneumatic pipeline.

Fig. 2.14

The pressure coefficient of spherical SPT generally depends on the velocity V of streamlined flow due to the dependence of the tube total drag coefficient on Reynolds number:

$$Re = VD/\Delta,$$

where D is a diameter of spherical nozzle; Δ is a kinematic viscosity coefficient.

With the airflow, the most part of the pressure distributed along a spherical surface is lower than P_{st} ($\zeta P_{st} <0$). Since the main principle of operation of the spherical SPT is integration of pressure distributed along the spherical surface, then in the static pressure chamber, the negative pressure is formed that must be taken into account with sensor calibration.

When constructing the static pressure measurement systems using spherical tubes, while making the static pressure chamber in a porous sphere, it is enough to convert the pressure into an electrical signal and send it to a digital computer. To obtain the required static characteristic, the correction coefficients are entered in the computer memory. Their values are determined by individual calibration of the sensor.

The tube characteristics can be controlled by changing the location of the static pressure perception chamber inside the sphere, or changing the diameter of static pressure perception holes and using external perforated sphere protection.

The widespread introduction of spherical SPT is prevented by operational difficulties associated with sphere pores foul with particles in the air. In addition, not all problems of nozzle heating have been solved.

To measure local aerodynamic angles (angle of attack ℑ and slip angle ℘) the aerometric method is most commonly used. As the sensitive element of sensor that implements the aerometric method of measurement, a vane with the narrow symmetrical profile is used that is set in flight along the direction of ram airflow. Figure 2.15 shows the exterior of vane sensors of aerodynamic angles (aviation abbreviation - ДАУ): ДАУ-19-1 (Fig. 2.15, a) and ДАУ-72 (Fig. 2.15, b).

This vane sensor is installed on the aircraft fuselage where there is the minimum flow distortion. At the same time, vane should be extended outside the air boundary layer in the place where the sensor is installed.

To measure the angle of attack, the vane sensor is placed aboard the fuselage so that the rotation axis of vane is located perpendicular to the aircraft plane of symmetry as close to the aircraft center of mass as possible. To measure the aircraft slip angle, the sensor is usually placed at the bottom of fuselage, where the rotation axis of vane should be in the plane of symmetry or in a plane parallel to it.

a *b*

Fig. 2.15

For the vane sensor, the fundamental importance has the vane design because its characteristics influence all metrological parameters: static and dynamic errors; resistance to destabilizing climatic and mechanical factors (temperature, shock, vibration).

By aerodynamic characteristics, the best vane sensor design at subsonic speeds is a vane with trapezoid in a plan, with the right angle of leading edge to the flow (zero angle of sweep), with thin wedge-shaped profile and large elongation (Fig. 2.15, b). The vane extending is within 1.5 ... 2. To measure the aerodynamic angles at supersonic speeds, it is desirable to have an arrow-shaped vane (the sweep angle is about 50°) with the trapezoid in a plan and lamellar cross-section (Fig. 2.15, a).

The principle of operation of the vane sensor is illustrated by the kinematic scheme (Fig. 2.16).

In the absence of ram pressure, the mass of vane 13 is balanced relative to the axis of its rotation 12 by counterbalance mass 8. Due to this, the vane takes an arbitrary position in space. In flight, the vane takes the position strictly according to the airflow, rotating around the axis 12 with rigidly attached brushes 9 of the potentiometer 10, thereby fixing the vane rotation angle. To damp the vibration of sensor moving mass, a liquid damper 7 is used, whose rotor is connected to the gear 11 rigidly fixed on the axis 12. For normal operation in icing conditions, there is a tubular electric heater inside the vane. The circuit elements 1...6 are not involved in measurement mode and are used to form servo system during sensor testing without removing it from the board the aircraft.

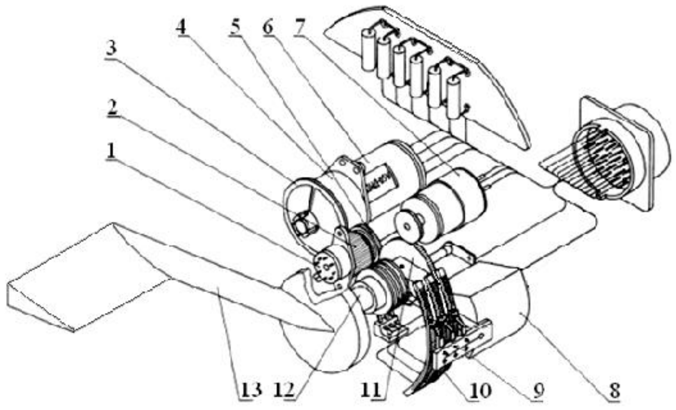


Fig. 2.16

In the measurement mode, the electromagnet 1 is electrically disconnected, the gear 5 is withdrawn from cohesion with the gears 3 and 11, and the axis 12 rotates independently on the circuit elements 1...6. In the test control mode, the signal is supplied to the electromagnet 1, the friction clutch 2 attracts the gear 5, entering it into cohesion with the gear 3 and 11. The circuit elements 1...6 and 10 together with a potentiometer that sets the vane angle of rotation create a potentiometric servo system, which verifies the sensor serviceability. The potentiometer, which sets the vane angle of rotation, is located on the remote control of the test control system in the aircraft cockpit.

The advantage of vane sensors is simplicity of design, and the shortcomings are weak damping and low accuracy (error in the range 1°...4°), especially at low flight speeds. Manufacturers are struggling against the problem of weak damping by improving liquid dampers, and to improve the accuracy they look for new variants of measuring the aerodynamic angles.

One of these options is the pneumatic sensor of aerodynamic angles. Figure 2.17 shows the exterior of the pneumatic sensor of 60TP type by Thomson firm.

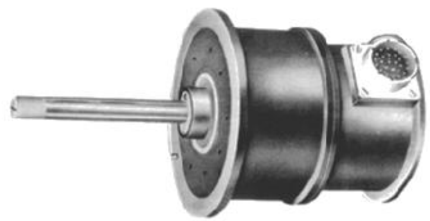


Fig.2.17

Like vane sensor, pneumatic one measures the local aerodynamic angles. The principal differences from the vane sensor are the enhanced dynamic performance, more correct form of sensing element, increased precision and high sensitivity to flow downwash at low flight speeds.

The basis of the aerodynamic angle pneumatic sensor operation is the dependence of the pressure distribution on the surface of sensing element in a form of symmetric body on the direction angle of airflow. The sensitive element is in the airflow outside the aircraft skin. During the flight, the sensing element is blown by the airflow varying in intensity and direction. Around the sensing element body, there is a pressure diagram. The law of flow depends on the sensing element shape and flow intensity.

Figure 2.18 illustrates the operation of pneumatic sensor of aerodynamic angles. On the surface of sensing element of gauge 1, made in the form of a cylinder, two pressure tubes Π_1 and Π_2 are installed at angle ⇐ With the help of pneumatic pipelines 6 and 5, laid in the cylinder body, pressure tubes Π_1 and Π_2 are connected to the respective chambers of pneumatic motor 2. When tubes are arranged symmetrically with respect to the airflow, pressures P_1 and P_2 measured by them will be equal (see the diagram of pressure distribution on the sensing element surface).

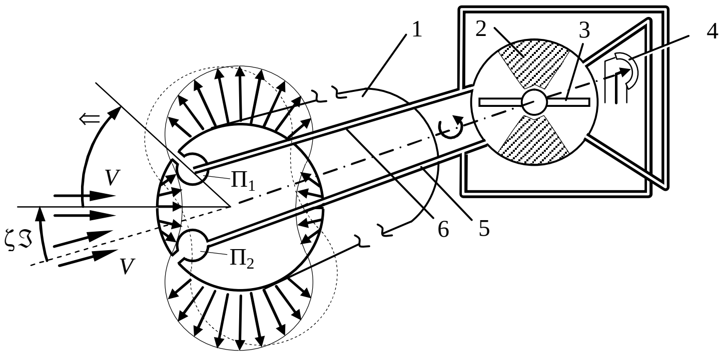


Fig. 2.18

In order to reduce errors of pneumatic sensor, combined probes are used. For example, ДАУ-ПФ sensor (Fig. 2.19), developed by Ulyanovsk Design Bureau together with scientists from Moscow Aviation Institute as combined probe, uses vane 1 and pneumatic 2 sensing elements. In this design, the vane sensing element, increasing the moment that turns the probes of the pneumatic sensing element in a symmetrical position relative the airflow, offloads the pneumatic motor. The feature of the device is that in addition to forming the torque, pneumatic motor acts as damping device.

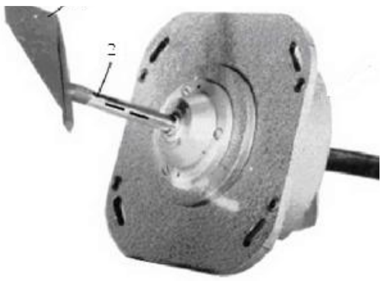


Fig. 2.19

The necessity to measure the aerodynamic angles at near-zero flight speeds leads to the development of electromechanical sensors distinguished by the electric servo drive amplifying the payload torque to overcome the friction torque of its moving parts.

For example, the compensation sensor ДАУ-К (Fig. 2.20) has a lightweight multi-blade pneumatic motor, which serves as an indicator. The amplified electrical signal comes from the indicator through the scheme elements to the electric motor, which turns the probe through the reduction gear in the equilibrium position. With reasonable accuracy this sensor starts to measure the aerodynamic angles ranging from 15...20 km/h of the flight speed.


Fig. 2.20

The principle in pneumatic sensor design is the overall refusal of rotary probes that is implemented in multifunctional sensors of aerometric parameters.

The multifunctionality is achieved by unifying functions of measuring the angle of attack, slip angle, static pressure, pitot pressure, and sometimes temperature of stagnated airflow in one design.

The example of unified gauges of aerodynamic parameters is helicopter gauge of angles of attack and slip Z ДУАС-В (Fig. 2.21), which is mounted on a remote rod. ДУАС-В design unifies two functions: measurement of angle of attack and slip, that is why two vanes are used. They are fixed on their axes at 90° to each other.

Fig. 2.21

The sensor ДУАС-ПВД unifies four functions in one design: measurement of angles of attack and slip, pitot pressure and static pressure (Fig. 2.22).

These instruments, as vane sensors, measure local aerodynamic angles in flight. However, they significantly increase the accuracy of perception of pressure and measurement of angles of attack and slip by placing the sensor outside the aircraft skin into the undisturbed ram airflow. In addition, there is a reducing of the total mass of instruments and the number of structural elements outside the aircraft skin.

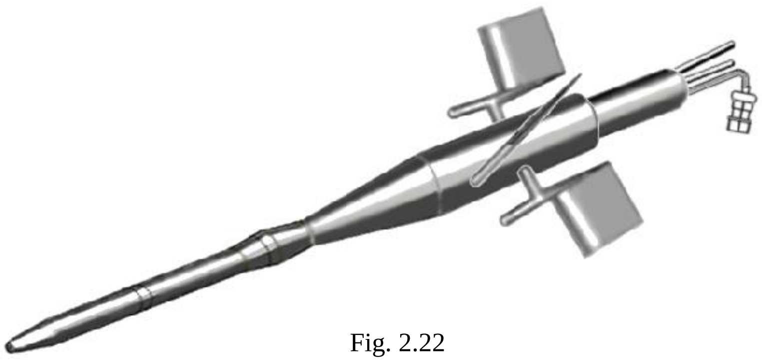


Fig. 2.22

In ДУАС-В and ДУАС-ПВД, there is the maximum approximation of local angles of attack and slip to their actual values. Within range $\alpha_L = \beta_L = \pm 15$ degrees, the maximum precision of perception of P_p and P_{st} pressures is also achieved.

Drawbacks of ДУАС-В and ДУАС-ПВД sensors can be the following: they distort the aircraft aerodynamics and obscure the field of view in front of the aircraft for pilots. Therefore, in the default mode, ДУАС-В and ДУАС-ПВД are installed mainly on helicopters, where other sensors cannot operate, and also installed on the supersonic aircraft.

A major disadvantage of ДУАС-ПВД is a narrow range of angles of attack and slip (± 15°), where perception of P_p and P_{st} is performed with acceptable accuracy. Beyond this range, the angular error of sensor increases catastrophically.

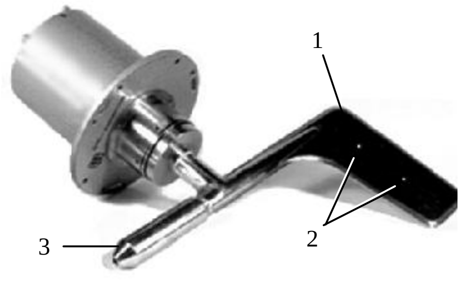


Fig. 2.23

To improve the accuracy of measuring P_p and P_{st} at large angles α and β, PST is used oriented by airflow vector (Fig. 2.23). This sensor is called aerodynamic parameters sensor. The sensor measures three aerodynamic parameters: pitot pressure P_p is perceived by the pitot pressure tube 3; static pressure P_{st} is perceived by holes 2, which are connected with two sealed chambers fixed by respective nozzles; angles α (or β) are measured using vane 1, which is set in flight along to the direction of ram airflow.

The sensor operates at speeds of 70...400 km/h and for angles of attack (slip) in the range ± 30° with accuracy ± 0.3°, and for the pitot pressure ± (0.02...0.03)q.

Unlike the pressure tubes rigidly fixed on the aircraft of PST and ДУАС-ПВД types, the aerodynamic parameters sensor is workable when flying with large angles of attack and slip. This property of sensor is used for installing aboard the high-maneuverable aircraft that can fly at angles equal to ± 90 degrees.

To avoid the shock stall when flying at high speeds, the vane has a triangular semi-wing shape (see Fig. 2.24), for which sweep angle is chosen depending on maximum Mach number. On the side surfaces of vane there are two groups of air intakes 3, symmetrically located relative to the vane leading edge. These intakes are used to improve the accuracy of perception of aerodynamic angle just as it does in ДАУ-ПФ with combined probe (see Fig. 2.19) by the pressure difference between two groups of intakes.

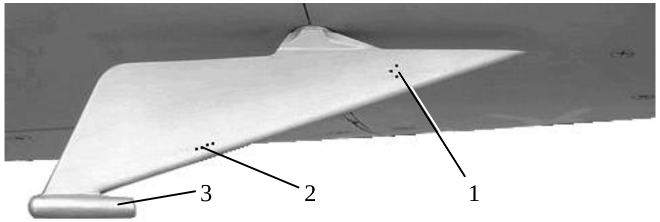


Fig. 2.24

Two groups of intakes 2, also symmetrically arranged relatively the vane leading edge, are used to perceive the static pressure in flight. Due to sweep angle and vane side surface configurations, one can receive the static pressure taking into account the compensation of atmospheric pressure distortions by the aircraft design in the place of sensor installation. The pitot pressure is perceived by the tube 3.

The multifunctional air pressure transceiver ППВД-40-Б (Fig. 2.25) developed by 'Aeropribor-Voskhod' Design Bureau provides perception of pitot and static pressure; of two pressures that are functionally dependent on the angle of attack; and of temperature of the stagnated airflow. Perceived pressures (including the static pressure taken by two independent chambers) is measured using the built-in high-accuracy pressure sensors, then they are transformed by the processor device into digital signals and as a serial code are transmitted to digital consumers of aerometric parameters. At the same time, due to lack of transmission pneumatic pipelines, the dynamic lag is drastically reduced and correspondingly the accuracy of pressure measurement is increased. To provide the air pressure to the backup devices, the transceiver ППВД-40-Б has pneumatic outputs of the pitot pressure and other statics.

Fig. 2.25

The pressures that are functionally dependent on the angle of attack are taken from holes symmetrically located relative to the streamlined body of the tube. The angle of attack is calculated by the difference between these two pressures. Thus, except of moving mechanical parts in sensor, the reliability of sensor operation increases along with design simplification.

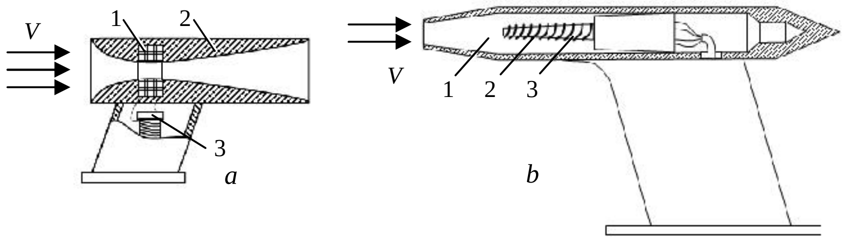


Fig. 2.26

In the widely spread flow-type temperature probe of type П-5 (Fig. 2.26, a), the heat sensitive element 1 is placed in the body, whose internal channel is shaped by the profile of Laval nozzle 2. At the narrowest section of the profile, the flow rate while accelerating reaches its maximum critical value, at which the ratio of temperature perceived by the sensor to the stagnation temperature is the constant value equaled to 0.978.

Thus, the sensor 1, installed in the narrowest section of the profile, measures the temperature $T_{sensor} = 0.978T_{stagnation}$. From the temperature sensor through the adjucting resistor 3, information about the stagnation temperature comes to consumers. The probe heat sensitive element is made from nickel wire.

The temperature of stagnated airflow is measured in the stabilizing chamber of pitot pressure. The principle of temperature measurement is the same as, for example, in the temperature probe of П-69-2М type (Fig. 2.26, b).

The heat sensitive element of probe П-69-2М is a cylindrical insulation coil 3, which has two separate bifilar spirals 2 of platinum wire with diameter in 0.04 mm.

The kinetic energy of airflow in the stagnation chamber 1, which in this case is the probe of stagnated flow temperature, is converted into the thermal energy. The stagnation temperature is measured by the thermistor sensing element and output to consumers.

In addition to improving the sensors of aerodynamic angles, the methods for determining angles of attack and slip through the simulation of aircraft motion dynamics are developed.

By nature of the primary data use, airborne devices of determining true angles of attack can be divided into several groups:

- devices to determine the aerodynamic angles by analyzing overload effecting the aircraft;
- devices to determine the angles of attack and slip by modeling the dynamics of the aircraft motion relative to the center of mass;
- devices which use the kinematic equations and geometric relationships between the parameters of the aircraft motion in different coordinate systems in their algorithms.

Information systems of the first group simulate equations of forces and moments acting on the aircraft, whose motion is regarded as a motion of the variable mass solid body. The accuracy of determining the angle of attack in such system is determined by errors of prior assignment of aerodynamic coefficient values and their derivatives, and also by errors of measurement of aircraft mass in flight, ram pressure and acceleration. The systematic error of determining the angle of attack can reach 1...2°, while the measurement error of the current aircraft weight is crucial.

In the information system of the second group the output parameters are current flight speed, overloads n_x, n_y, n_z, angular velocities ω_x, and ω_z, pitch ϑ and bank γ angles. For example, considering the relationship

$$n_z = \frac{c_z^{\beta} qS}{G}\beta$$

(where c_z^{β} is derivative of lateral force coefficient by angle of attack; S is lifting area of wing; G is gravity force) the signal, proportional to the rate of change of slip angle, can be obtained by converting readings of the accelerometer installed in the direction of the aircraft lateral axis:

$$\dot{\beta} = \frac{G}{c_z^{\beta} qS}\dot{n}_z = K\dot{n}_z.$$

On the other hand, equations of aircraft motion dynamics, subject to certain simplifications, can give

$$\dot{\beta} = \omega_x \sin\alpha + \omega_y \cos\alpha$$

or

$$K\dot{n}_z = \omega_x \sin\alpha + \omega_y \cos\alpha. \qquad (2.3)$$

Using the known kinematic relations for the angular rate of bank

$$\gamma = \omega_x \cos\alpha - \omega_y \sin\alpha \qquad (2.4)$$

and solving equations (2.3), (2.4) together, it is possible to get the value of true angle of attack α.

Such system is operable at changes of angle of attack up to 80°, while the static error of the angle of attack is 1.5°. While reducing the upper limit of changing the angle of attack up to 45°, the error is reduced to 1°.

Means of determining the true angle of attack from the third group solve the equation of kinematic and geometric relations between the aircraft parameters measured in different coordinate systems: normal, body-fixed and wind axes. For example, one of methods of determining the true angle of attack uses the equation of connection between parameters of the aircraft orientation in normal and wind axes coordinates:

$$\sin\theta = \sin\vartheta\cos\alpha - \cos\vartheta\cos\gamma\sin\alpha\cos\beta - \cos\vartheta\sin\gamma\sin\beta \qquad (2.5)$$

where θ is the trajectory angle, which is usually calculated from the relation of vertical flight speed and true airspeed.

Using the signals of sensors of vertical speed and true airspeed, pitch, bank and slip angles, it is possible to calculate the value of true angle of attack, which satisfies the equation (2.5).

The best accuracy of determining the aerodynamic angles in flight is provided in the integrated processing of signal from airborne sensors of local angles of attack and slip, and signals received from one or more of the above considered information systems.

This approach is particularly implemented in the helicopter flight environment data system СИ ВПВ-52 and in СИ ВПВ-35 (both systems are developed by 'Aeropribor-Voskhod' Design Bureau). Figure 2.27, a shows the exterior of СИ ВПВ-52.

a

b

Fig. 2.27

The probe is a monolithic metallic body, which consists of two triangular prisms with convex faces shifted with respect to each other at angle 60°. Each face of the probe has two receiving holes that are connected with the pneumatic pipeline that ends with a nozzle. Thus, each probe has six pneumatic channels.

Receiving channels of pitot and static pressures and of two pressures that are functionally dependent on the angle of attack (slip) are selected in the system computer. There, according to information received from external systems (regarding the components of vector of absolute angular velocity relatively to body-fixed axes; the components of the overload vector along body-fixed axes; angles of bank, pitch, heading; the current weight of helicopter) the true aerodynamic angles and other aerometric parameters are calculated analytically.

For heating the probe, the tubular electric heater is used, built-in the sensor body. A part of the probe that is in flow has 35 mm in height and maximum longitudinal size of 10 mm. The weight of sensor is less than 0.1 kg.

Figure 2.27, b shows the exterior of air pressure transceiver, which is a part of measurement system СИ ВПВ-35. The probe sensitive element is in the form of triangular prism similar to the module of air pressure tube system СИ ВПВ-52. The tube is designed to measure, calculate and output information on altitude, speed and aerodynamic parameters in airborne automatic systems. Calculating the true angles of attack (slip) is made in the probe processor using external information about the components of the aircraft angular velocity relative to the body-fixed axes and about the value of normal overload.

To measure the pressure in the probe body, there are compact pressure sensors with resonating cylinder of generator pressure sensor. The calculated and measured aerometric parameters are transmitted via electrical network to system СИ ВПВ-35 and to other digital consumers.

However, the majority of aircraft consumers receive information from pressure tubes through the supply system of aneroid-membrane devices and systems.

The schemes of supply systems depend on the constructive features of aircraft and on the installed consumers of pitot and static pressures. Figure 2.28 shows the aircraft supply system of aneroid-membrane devices, consisting of two pitot pressure pipelines and of three static pressure pipelines, one of which is backup (the static pressure pipelines are shown in thickened lines). Figure 2.28 also shows the sensors of aerodynamic angles (angles of attack) of port side 13 and of starboard 10, and temperature sensors 11, 12.

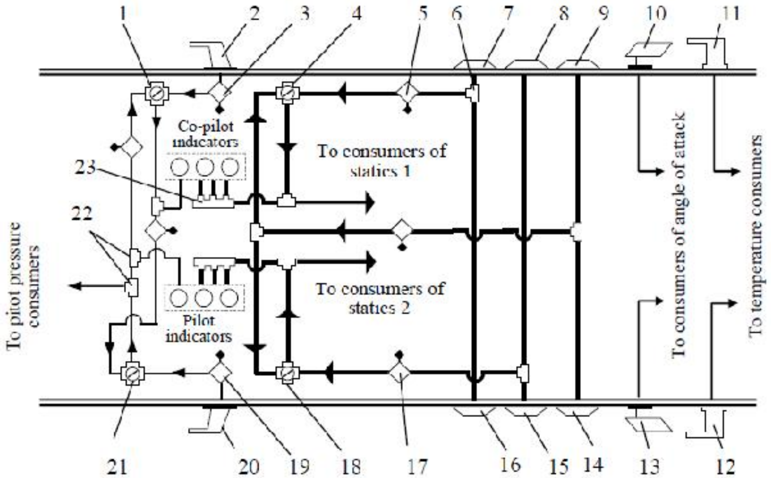


Fig. 2.28

The pitot pressure comes from two pitot pressure tubes 2 and 20 located on the aircraft fuselage skin on both sides. One of the tubes supplies the pilot-in-command indicators, air data computer system, other consumers of aerometric parameters and is a backup for the co-pilot indicators. The second tube 2 is the main for indicators of co-pilot, backup for the pilot-in-command and consumers of aerometric parameters.

Switching from one tube to another and vice versa is performed by pitot pressure valves 1 and 21.

The symmetrical arrangement of static pressure tubes 7Z9, 14Z16 (on the aircraft fuselage skin, three ones are on the port side and three ones are on the starboard) and their pairwise connection into three combined pipelines provide equalizing of the static pressure at the aircraft maneuvering. Switching from the main to the backup supply is performed by static pressure valves 4 and 18. The switching valves are controlled from the cockpit using the knobs located on the valve control panel, or are controlled directly by cables.

The pitot and static pressure pipelines are referred to pneumatic pipelines which connect the aerometric parameter probes with consumers. Pneumatic pipelines are metal or rubberized-fabric pipes. The inner diameter of static channel pneumatic pipeline must be at least 6 mm, and for the pitot pressure channel it must be not less than 4 mm.

Pneumatic pipelines are provided with water traps (3, 5, 17, 19) that protect pneumatic pipelines from accumulation of moisture. Water traps, whose exterior is shown in Fig. 2.29, are designed to collect and drain the condensate water formed in pipelines.

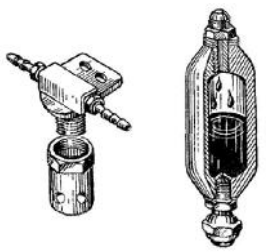


Fig. 2.29

Water traps are small capacities that are made of transparent material, with the nozzle at the top to connect pipeline and with drain plug that is screwed in the bottom part of water trap. In this case, the accumulation of water at the bottom when it freezes does not cause plugging the system.

Water traps are located near the pressure tubes and in the lowest places of pneumatic pipelines.

For branching of corresponding pressure when supplying it to instruments and systems, manifolds 23 and tees 22 (Fig. 2.28) are used. Exterior of the manifold is shown in Fig. 2.30, a, and Fig. 2.30, b shows the tee exterior.

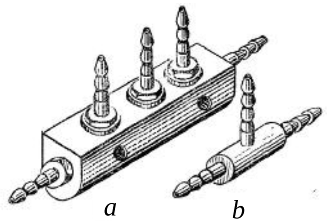


Fig. 2.30

In the steady state flight conditions, pneumatic pipeline parameters do not affect the measurement process. However, in the dynamic mode, a lag in pitot and static pressure lines affects the accuracy of measurement of altitude and speed parameters. For civil transport aircraft, lag coefficients of channels P_p and P_{st} are standardized. Thus, the lag of at the ground level for each static system after connecting all consumers must be less than 0.4 seconds with supplying sensors of autopilot systems and no more than one second when supplying flight and navigation instruments.

2.1.3. Analog air data computer systems

An example of the analog air data computer system is СВС-ПН-15-4. The system is an analog computing device to determine main aerometric parameters based on measurements of static P_{st} and dynamic P_d pressures and of temperature of the stagnated airflow T_s.

СВС-ПН-15 is installed on the subsonic aircraft and used to calculate signals of absolute H_{ab} and relative H_{rel} barometric altitudes, Mach number M, the equivalent (indicated) airspeed V_{IAS} and true airspeed V, and deviations from the given values of absolute barometric altitude, Mach number and indicated airspeed. Furthermore, there is a possibility to display the ground speed V_{gr}, measured by Doppler navigator of ДИСС type, and output a signal proportional to the relative height in the aircraft transponder СОМ-64, because as required by ICAO, all airplanes must transmit the height data to the dispatcher automatically.

Block diagram of СВС-ПН-15 is shown in Fig. 2.31. The system includes:

- computer of speed, Mach number and altitude ВСМВ-1-15;
- altitude indicator УВО-15 (AI) with supply and amplifier units БПУ-3 (SAU);
- IAS indicator (IASI) with БПУ-3 units;
- Mach number indicators (MI) УМ-1К with БПУ-3 units;
- potentiometric voltage converters (PVC);
- altitude controller КЗВ-0-15 (AC);
- ready signal units (RSU);
- IAS controller (IASC);
- Mach number corrector (MC);

- supply unit БП-27-2 (SU);
- filter Ф-115-1.

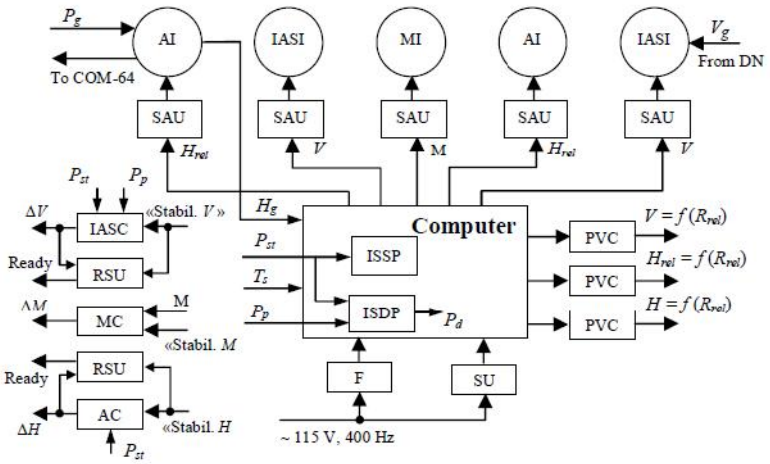


Fig. 2.31

The number of indicators, correction controllers and ready signal units is determined by a specific system configuration. In addition, the system may include the standardized computers of aerodynamic corrections, the forming laws of which depend on the type of aircraft.

The main element of the system is computer, which includes the aneroid-membrane sensors of static P_{st} and dynamic P_d pressures of inductive type – ISSP and ISDP, respectively. The dynamic pressure is formed directly in the aneroid capsule of ISDP by subtracting the static pressure P_{st} from the pitot pressure P_p. To reduce instrumental temperature errors of sensors, the temperature of 45 ± 5°C is maintained using the temperature control system.

The stagnated airflow temperature T_s is measured by gauge П-5 of the outside air thermometer ТНВ-15, which is not included in ADC system.

The signal H_g enters the computer from the pilot altitude indicator (it is formed based on the pressure *QFE* – the pressure at the runway threshold, which is set by indicator knob as P_g) and is used to determine the relative altitude H_{rel}.

Besides sensors, the computer includes functional voltage generator (FVG), amplifiers, transformers, switchgear elements and temperature control system.

Computer ВСМВ-1-15М is designed to solve the gradable formulas (1.1)Ž(1.4) and to output the signals proportional to H, H_{rel}, V as relative resistances R_{rel}. This signal conversion is carried out in PVC units, connected to the outputs of the computer.

To improve the accuracy of calculation, the functional dependences of calibration formulas are converted to logarithmic ones. In this case, the multiplication and division are replaced respectively by adding and subtracting logarithms of unknown quantities with subsequent potentiation of the result via FVG. With the same purpose, ISSP and ISDP sensors also have nonlinear dependences which are close to logarithmic.

Addition and subtraction operations are performed in computer using amplifiers. Nonlinear functions are implemented by diode FVG by their piecewise-linear approximation. Functional voltage generators consist of individual cells, each of which is designed for function approximation by a section.

In particular, when calculating Mach number, if M ™1, the following relationship is used similar to (2.1)

$$\frac{P_d}{P_{st}} \text{X} f_1(M), \qquad (2.5)$$

where $f_1(M) = (1 - 0.2M)^{3.5}$ J 1.

In the calculator, this expression is converted to:

$$\lg P_d \text{ J } \lg P_{st} = \lg f_1(M).$$

The functional diagram of Mach number calculation channel in computer ВСМВ-1-15М is shown in Fig. 2.32.

The static characteristics of ISSP and ISDP sensors are selected so that the output voltage dependences on the respective pressures are close to logarithmic in most of the working sections. With FVG1 and FVG2 converters, amplifiers Am1 and Am2 and bias voltages generated by Tr1 and Tr2 transformers, output signals of static and dynamic pressure sensors are adjusted to their exact coincidence with logarithmic dependences. As a result, the amplifier Am1 output forms the voltage proportional to lg P_{st}, and the amplifier Am2 gives the voltage proportional to lg P_d.

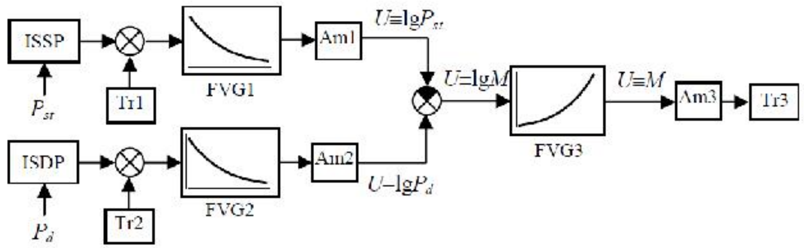


Fig. 2.32

These voltages are summed to form logarithmic dependence lg $f_1(M)$. With FVG the potentiating of logarithmic dependence is realized.

The voltage that is proportional to M-number comes to the transformer Tr3 through the amplifier Am3, where it is branched and comes to consumers.

The true airspeed V is calculated taking into account M-number determined in (2.5) by the following formula

$$V = f_2(M)\sqrt{T_s}\ ,$$

where $f_2(M) = \frac{c\ M}{\sqrt{1+0.2M^2}}$; $c = \sqrt{kR}$; $k = 1.4$ is the constant of adiabatic process; R = 287.053 J/kg K is the air specific constant.

The channel of calculating the true airspeed is built similarly to the above discussed Mach number computation channel.

The voltage formed in Mach number computation channel is proportional to lg $f_1(M)$ in the speed computation channel on the diode-functional FVG and amplifiers. This voltage is changed up to exact match with the logarithmic dependence lg $f_2(M)$. Potentiating this dependence using FVG, the function $f_2(M)$ is obtained. Introducing the function $\sqrt{T_s}$ is provided by the transformer multiplication circuit, at the output of which the voltage proportional to the true airspeed V is formed.

Just as in Mach number computation channel, the consumers receive the true airspeed signal through the multiplication transformer branching.

The absolute barometric height is calculated by standard atmosphere hypsometric formula:

$$H_{abs} = \frac{T_0}{\wp}\left[\left(\frac{P_{st}}{P_0}\right)^{-\frac{\wp R}{g_0}} - 1\right].$$

In calculating the absolute barometric height H_{abs}, the dependence lg P_{st} formed by ISSP sensor is used. This dependence, using functional transformers, is adjusted to logarithmic dependence

$$\lg f_H(H_{abs}) = f_P(\lg P_{st}).$$

Further, the discussed earlier potentiating procedure is performed, and the voltage proportional to the absolute barometric height is output to consumers.

To generate the signals proportional to H_{abs}, M, V in the form of relative resistances to consumers, PVC units are used which are connected to the computer outputs. Potentiometric voltage converter is electromechanical tracking system that works as compensation scheme with self-balancing. The input signal of PVC is the corresponding output signal of the computer, and the output signal is the output potentiometer resistance, whose brushes are rotated synchronously to the brushes of feedback potentiometer in the electromechanical tracking system.

The relative barometric height is formed in the computer as the difference $H_{rel} = H_{abs} - H_g$. The given height H_g is formed in the altitude indicator УВО-15 according to the information about the atmospheric pressure P_g (pressure at the runway threshold). This pressure is set by the pilot on the atmospheric pressure indicator of УВО type.

The aircraft automatic flight control system is the main consumer of information about deviation of current values of absolute barometric height, Mach number and indicator airspeed from the given ones. Information about the deviation ζM, ζH, ζV is generated by such units as M-number controller, altitude controller, IAS controller, respectively.

Calculated value of Mach number from BCMB-1-15M comes to M-number controller (MC), which produces the signal ζM, if the actual value of Mach number differs from the value the aircraft had at the time of switching autopilot in the mode of Mach number stabilization. M-number controller is essentially an electromechanical system that resets the current value of Mach number in tracking mode. When switching on the correction mode, the tracking system engine stops, and M-number controller forms the output signal to change Mach number relative to the value at which the tracking system was stopped.

The signal ζH is produced in the altitude controller (AC) according to information about the static pressure entering the device case. The altitude controller outputs the signals in the form of DC and AC voltages with the frequency of 400 Hz, which are proportional to the deviation from the given altitude. Structurally, altitude controller is electromechanical gauge to measure the deviation of the current barometric height from the given one. The given altitude is the height at which the aircraft was in time of activating the mode of barometric altitude stabilization (pressing button "Stabil. H_b" on the autopilot control panel). This moment of time is the beginning of measuring the barometric altitude.

The indicated airspeed controller (IASC) gives the signal ζV which is the deviation of current indicated airspeed from the given one. The given IAS is the speed that the aircraft had in time of activating the airspeed stabilization mode, particularly in the time of pressing button "Stabil. *V*" on the autopilot control panel. Structurally, IASC is electromechanical gauge to measure the indicated airspeed, which is a function of the dynamic pressure. To form the dynamic pressure $P_d \ \mathrm{X} (P_p \ \mathrm{Z} P_{st})$, IASC receives both static and pitot pressures.

The system СВС-ПН-15-4 has measurement range of altitude as 0...15000 m, speed Z 100...1200 km/h and Mach number Z 0.3...1.0. Instrumental errors of altitude measurement for the range 0...3000 m are ±10 m (±1% of the current altitude) and at altitudes from 3 to 13 km they are ±50 m. Instrument errors of true airspeed in the range from 150 to 400 km/h are ±25km/h; in the speed range of 400...900 km/h they are ±12 km/h and at the speed over 900 km/h the errors are ±24 km/h.

2.1.4. Digital air data computer systems

Digital ADC systems as well as analog ones are designed to calculate and output the altitude and speed parameters to airborne systems and to the crew. Depending on the modification these systems may also output the additional parameters, such as the ram pressure, atmospheric pressure and dynamic pressures as well as single signals at reaching the specific fixed values by some flight parameters.

In such systems the aerodynamic flight parameters can be calculated in aircraft special digital computers according to algorithms which operate with known calibration dependences converted to logarithmic ones in order to increase the accuracy of calculations. Nonlinear calibration dependences are stored in long-term memory device of computer.

Input aerodynamic and additional parameters in control display unit are converted into number codes. Number codes by poll commands of input converters come to the processor. The processor computes the aerodynamic parameters in accordance with the algorithm that is implemented in the form of program in machine codes of operation like addition, subtraction, etc.

After computing the information is transmitted to the output device that generates and presents code, analog and single signals to consumers.

The computed parameters are also presented to separate arrow indicators, as well as in electronic indication system.

To improve the accuracy of computing the aerodynamic parameters in some digital systems the special algorithms can be used, like algorithms of compensation of systematic component errors in the pressure tubes in the form of functions of angles of attack and slip, Mach number and flight altitude as an aerodynamic method of compensation of pressure tubes errors.

Nowadays at all modern aircraft the following digital ADC systems are used as the main airborne aids of measuring the aircraft altitude and speed parameters: СВС-2Ц-1 and its modifications; СВС-85, СВС-2Ц-У and their modifications; ВБЭ-СВС, СВС-2Ц-2 types.

As an example of the second generation digital ADC system can serve system СВС-85. It is designed to compute and indicate for the crew and in the airborne systems a number of aerometric parameters based on measurements of static P_{st} and pitot P_p pressures, temperature of stagnated airflow T_s, the local angle of attack $\Im_L$ and the entered in the system value of atmospheric pressure at runway threshold (pressure of the day) P_g. In addition, the system functioning requires discrete information on position of flaps and landing gear, on the serviceability of heating system pitot-static tubes and angle of attack sensors. Aerometric parameters that come from system CBC-85 include:

Z absolute barometric height H_{abs};
Z relative barometric height H_{rel};
Z indicated (instrument) airspeed V_{IAS};
Z true airspeed V;
Z vertical speed V_v;
Z Mach number;
Z maximum allowable indicated airspeed $V_{m.a.}$;
Z temperature of outside air $T_{\text{o.a.}}$;
Z temperature of stagnated airflow T_s;
Z dynamic pressure P_d;
Z pitot pressure P_p;
Z local angle of attack $\Im_L$;
Z true angle of attack $\Im$;
Z signals of barometric correction (pressure at the sea level and pressure at runway threshold P_g).

Sensors of primary information are frequency sensors of pitot and static pressures which are built into system computer, vane sensors of the angle of attack and temperature sensor of stagnated airflow of П-104 type. The static and pitot pressures are applied to the system from the aircraft pressure tubes. The system corrects the perception of the angle of attack, as well as perception of static and pitot pressures.

The system can be used on airplanes of 16 types, for each of them the correction coefficients for aerodynamic correction of altitude and speed parameters and for angle of attack are stored in the system memory device.

The system can switch to the mode of calculation without correction when applying special control signals or in case of absence of coefficients for this type of aircraft in the computer memory.

The main consumers of information from ADC system are shown in Fig. 2.33.

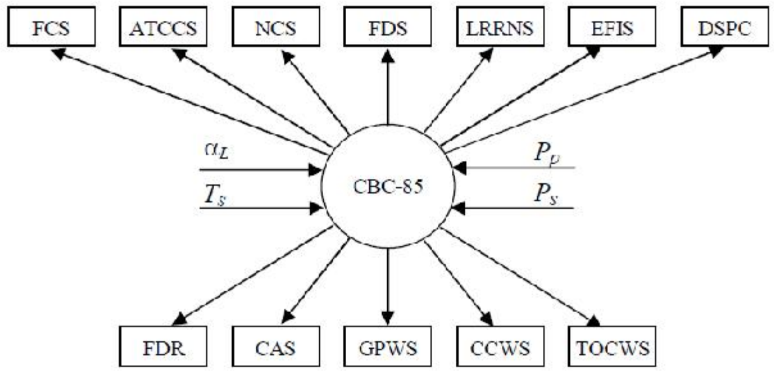


Fig. 2.33

Figure 2.33 contains commonly used aviation abbreviations: FCS – flight computer system; ATCCS Z auto throttle control computer system; NCS Z navigation computer system; FDS Z flight director system; LRRNS Z long range radio navigation system; EFIS Z electronic flight indication system; DSPC Z digital system of pressure commands; FDR Z flight data recorder; CAS Z collision avoidance system; GPWS Z ground proximity warning system; CCWS Z critical conditions warning system; TOCWS Z takeoff configuration warning system.

The basic of system CBC -85 is airborne specialized digital computer (ASDC) of *SISD* class (Single Instruction, Stream Single Data) with a single instruction stream and single stream of data that is the simplest and most common type of ASDC. The structure of the digital computer of *SISD* class is shown in Fig. 2.34.

For ASDC of *SISD* class, the following features are typical: only one stream of instructions, consistency and uniformity of information connections, fixed consequence of tasks solution. All instructions are processed one after the other, and each instruction initiates an operation with one data stream. Managing these computers is done by using a small number of signals and instructions.

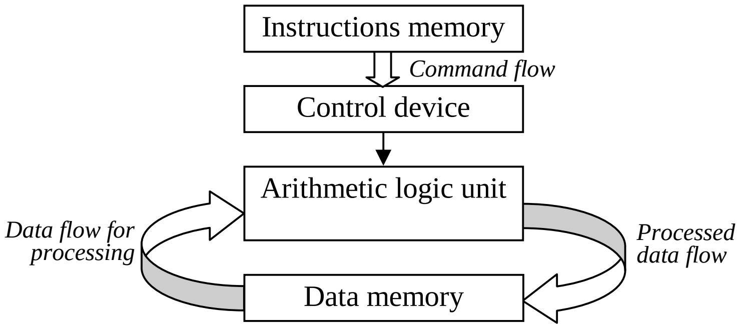


Fig. 2.34

Any ASDC includes the following functional units: central processing unit (CPU), memory storage device (memory), consisting of random access memory (RAM) and read only memory (ROM), input-output device (IOD), code transceiver (CT). The structure of ASDC CBC-85 also includes special RAM device (SRAM).

Block diagram of system CBC-85 based in ASDC is shown in Fig. 2.35.

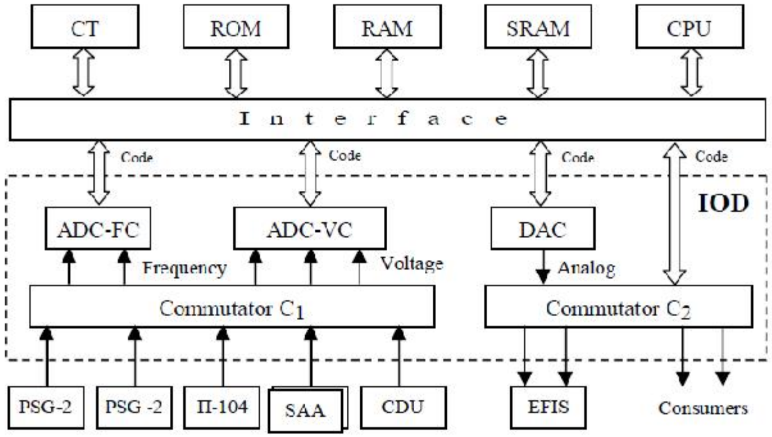


Fig. 2.35

Random access memory device provides the reception of discrete control and information signals (single instruction) and the transferring them in a form of parallel code to CPU; storage and selection of specific correction coefficients for different aircraft; ability to read the coefficients of pressure sensors by CPU.

Functional units are interconnected by interface system bus, which connects all the major components of computing system. The interface contains the address lines, data lines, lines of control, of synchronization, of testing, etc. Interface lines are grouped to perform one of the operations in the data transfer process. These groups are called interface buses.

The signals from the primary information sensors come to the input of IOD. Signals of pitot and static pressures come from the frequency pressure sensors of generator type PSG-2. These sensors are volumetric vibrating resonators, inner cavity of which receives the static and pitot pressures. The frequency of resonator vibration, i.e. pulse repetition frequency depends on air pressure inside the resonator.

To compensate the temperature errors the signals from sensors are issued on the temperature of their sensing elements, and to ensure the normal operation of the computer and to maintain constant temperature in it, the heating system and temperature control are used.

Pressure values P_{st} and P_p are calculated by CPU using code of sensors period, code of temperature and individual values of coefficients of every sensor, which are described by means of mathematical expressions and stored in ROM and SRAM.

Frequency sensor signals in analog-to-digital converter “frequency-code” (ADC-FC) are converted to a numeric code. In the ADC-FC the time-pulse conversion method is used like period-time-code. During the small cycle of the system operation (62.5 ms) in single-channel ADC-FC there is a serial conversion of signals of static pressure sensor, pitot pressure sensor and control frequency signal.

Analog signals from the output of sensor П-104 of the stagnated airflow temperature air and from synchro resolvers (SR) of two sensors of the angle of attack (SAA) are converted into numeric codes using analog-to-digital converters “voltage-code” (ADC-VC).

The main part of ADC-VC is 11-bit converter of unipolar constant voltage into binary code. The device ADC-VC uses the principle of serial computing. To increase the speed of operation the record of previous value of converted signal is used.

Values of local angles of attack come to the system CBC-85 from sensors of angles of attack on two aircraft sides (from port side and starboard). In the system there is an averaging of the values of the angle of attack from the right and left sensors. Averaging mode can be disabled by special command and with this the system switches to use the only one sensor value. In the system there is also the comparing of values of the angle of attack from the left and right sensors. In case of mismatch between values over the limit for the definite type of aircraft, there will be the message on malfunction. Comparison is blocked at the speed less than 140 km/h.

The true angle of attack in the system is calculated by the following expression:

$$\Im_{true}\ \mathrm{X} \frac{\Im_L}{K}\ \mathrm{Z} I_0,$$

where $\Im_L$ is the local angle of attack; K, I_0 are constant values, which are specific for each type of aircraft.

At the speed of aircraft less than 111 km/h, the value of true angle of attack at the system output equals zero at any value of the local angle of attack.

Signals of barometric correction come to the input of the system from the control display unit (CDU) of electronic flight indication system (EFIS). Two values of barometric correction QNH and QFE are transmitted into the system (QNH is the pressure at the height of mean sea level; QFE is the pressure at the runway threshold). Signals of barometric correction received at the input of the system can be of two types: analog or digital. Type of signals perceived by the system is defined by the type of contact of electrical connections.

Analog signals of barometric correction from SR of EFIS CDU are converted into numerical code using ADC-VC. Signals of digital barometric correction come to the input of the system in the form of bipolar serial code.

Signals to the input of ADC-FC and ADC-VC come through the commutator C_1 of IOD. Numerical coded signals come to CPU from the outputs of ADC-FC and ADC-VC.

Central processor unit is a computing kernel of the system and provides processing data that come to the memory in accordance with the given program recorded in machine code. Central processor unit contains a number of operational and arithmetic-logic devices, control device and local memory.

In CPU according to given algorithms all aerometric parameters are computed. With this CPU controls the operation of all external devices, retrieves the primary sensors information, generates the output code information and executes the subprogram of control.

Memory consists of RAM and ROM. ROM has a capacity of 6144 16-bit words and programmed when system is being manufactured. ROM device contains all the sequence of executable instructions by the CPU block and constant values necessary to calculate aerometric parameters. RAM device with capacity of 1024 16-bit words is used to store intermediate results of the calculations and ratios which are rewritten from ROM, sensors and SRAM unit.

Calculation algorithms are based on the method of piecewise polynomial approximation of non-linear calibration functions to be calculated similar to those in the previously discussed analog ADC systems. The essence of the method is that the interval, at which the function is given, is divided into a number of subintervals and within each subinterval the nonlinear function is replaced by a line segment. The flexibility of the method at providing the required accuracy is that the accuracy of calculations is achieved by increasing the number of subintervals and therefore the degree of the polynomial. For the same accuracy with the greater number of subintervals, the lower degree of the polynomial and higher speed of response are required. For each function which is calculated, there is a combination of the number of subintervals and degree of the polynomial. Polynomial coefficients are stored in ROM.

In the second and third generations of digital ADC systems for civil aviation aircraft, the concept of system unification for use on 16 types of aircraft is implemented. To correct coefficients of approximating polynomial for a specific type of aircraft, the additional options are used.

Thus, in accordance with the recommendations of standard ARINC-706 the aerodynamic error compensation is realized as a function of three parameters: height, speed and the angle of attack. Influence of slip angle on the error of the perception of static pressure is compensated by the circle connection of static pressure receivers located symmetrically on port side and starboard of the aircraft.

To correct the error of perception of static pressure in the function of Mach number the curve $f(M)$ is used:

$$f(M)\ \mathrm{X} \frac{\zeta P}{P_{st}}.$$

Specified function is being given by its values in the following points of Mach number: 0.1; 0.2; 0.4; 0.5; 0.6; 0.7; 0.8; 0.82; 0.84; 0.86; 0.88; 0.9; 0.92; 0.94. These values must be written in the SRAM unit for two flight modes of each aircraft type.

To correct the perception of static pressure in the function of Mach number and the angle of attack there is given a curve $f(M)\, \mathrm{X} \frac{\zeta P}{P_{st}}$ in the following points of Mach number: 0.2; 0.4; 0.7; 0.8; 0.82; 0.84; 0.86; 0.88; 0.9; 0.92. In these points the correction is calculated according to formula:

$$\frac{\zeta P}{P_{st}}\, \mathrm{X} A_2 \Im^2_{tr}\, \Gamma A_1 \Im_{tr}\, \Gamma A_0 ,$$

where A_2, A_1, A_0 are constant coefficients, $\Im_{tr}$ is true angle of attack.

For each type of aircraft in all ten indicated points of Mach number for two flight modes, the values A_2, A_1, A_0 should be recorded to the ADC system. One of the modes of correction is selected according to the state of single input command.

The calculated parameters are encoded in the form of numerical signals and in the form of voltages (after DAC) and they come through the commutator C_2 of IOD to consumers and for indication.

The integrated DAC is a converter with discreteness with 12 bits. Input signals of DAC unit are converted to constant voltage that varies in the range of 0.1 to 9.9 V.

Code transceiver (CT) forms and provides coded 32-bit words to buses. Formats of words contain information about the status and work ability of the system and its sensors; about switching off or malfunction of heating for air pressure probes; about features of correction of the angle of attack and the features of correction of error of static pressure perception, as well as on the excess of instrument flight speed.

The value of the maximum allowable indicated airspeed $V_{m.a.}$ is defined by curves of given values of $V_{m.a.}$ for altitudes ranging from 2048 feet to gaining 4096 feet. For each airplane, these values are given by a set of curves for nine heights. Selection of one of them is carried out by the obtaining the corresponding command. In the absence of control commands the graph with zero value is selected. In case of simultaneous receiving of multiple control commands, the graph with lower value of $V_{m.a.}$ is selected.

Equipment of built-in control in CBC-85 system ensures the control over the status of functional units of the system and the primary sensors in two modes: in the mode of the ground checking and in flight background mode with operation of main program.

As an example of digital ADC system of third generation there are systems, based on advanced microprocessor technology. Let us consider the ADC system "ВБЭ-СВС" (ADC system with electronic barometric altimeter).

The system ВБЭ-СВС is designed to measure, calculate and output the information about altitude and speed parameters to airborne systems, to indicate the barometric height and flight level, setting of flight level, and visual and aural signals on deviation from given flight level. The system provides permission of aircraft to fly in conditions of Reduced Vertical Separation Minimum (RVSM) and can be installed on 16 types of aircraft, in particular, in Be-103, Il-62, Il-76, Tu-154.

The system provides:

- measurement, calculation and output of information in a form of 32-bit binary serial code according to GOST 18977-79, PTM 1495-75 and ARINC 429 on such aerodynamic parameters as absolute height, relative altitude, vertical speed, instrument speed, true airspeed, temperature of stagnated airflow, pitot pressure;
- indication in meters and feet of relative barometric height adjusted taking into account the aerodynamic corrections by altitude and speed for the specific type of aircraft;
- manual input, indication in meters and feet, output of electrical signal of the given flight level;
- manual input and indication of atmospheric pressure at ground level in hectopascals hPa;
- manual input and alarm about setting the atmospheric pressure at ground level, which is equal to 1013.25 hPa;
- warning of flying at an altitude less than 1000 m;
- warning of deviation from the given flight level in range 60... 150 m;
- warning and output of electrical signal in case of deviation from the given flight level for more than 150 m;

- removing the visual warning in case of deviation from the given flight level for more than 150 m.

The exterior of the system is shown in Fig. 2.36. Structurally the system ВБЭ-СВС is a mono instrument (electronic barometric altimeter), which consists of sensor block for static and pitot pressures, of computer and of monochrome liquid crystal indicator.

Fig. 2.36

The block diagram of the system is shown in Fig. 2.37. The system ВБЭ-СВС, the same as system СВС-85, is an airborne special digital computer of class *SISD*, made on the basis of microprocessor set (CPU, ROM, RAM). The system ВБЭ-СВС consists of sensors for static and pitot pressures (pressure sensors Z PS) and indication unit (UI) with the controls.

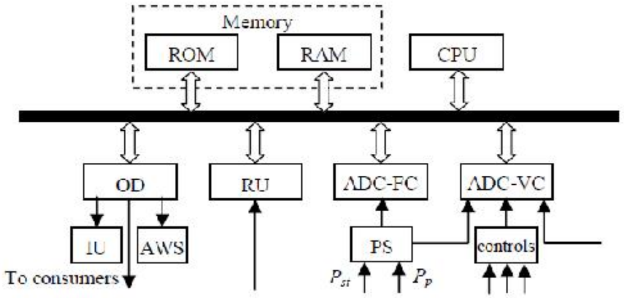


Fig. 2.37

The structure of the system largely repeats the structure of СВС-85. In this structure the same as in the СВС-85, the frequency sensors are used as sensors for pitot and static pressures. Frequency converter ADC-FC is designed to convert frequency sensor signals into binary code. ADC-FC converts the information by two channels from sensors P_{st} and P_p, as well as by the channel of check frequency.

Input signals of ADC-VC are the signals of receiver of stagnation temperature, signals of temperature channel of pressure sensors and signals of check voltage.

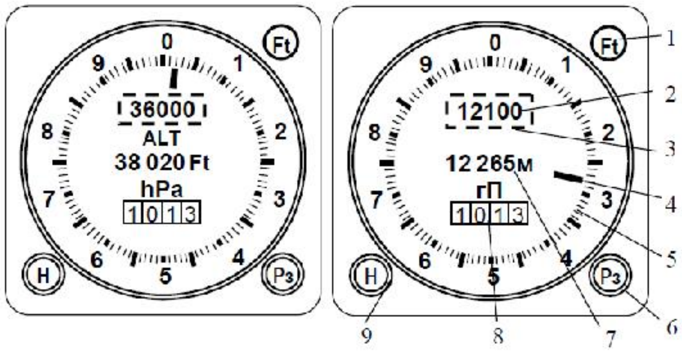
The memory device (memory) includes read-only memory (ROM) and random access memory (RAM), designed for storage of the program algorithm, constants and intermediate results in the calculation.

The receiving unit (RU) is designed for receiving one-time commands. The output device (OD) is for issuing information to consumers in the form of bipolar serial code, coded signals to the indication unit (IU) and electrical signals to the aural warning system (AWS). The exchange of information between CPU with OD is done in program mode.

The indication unit is designed to convert the coded information into visual one. The aural warning system converts the coded information into the signal of aircraft intercom.

The system operation is based on static and pitot pressures measurement (P_{st}, P_p) transmitted through pneumatic line from the receiver of pitot-static tube and of stagnation temperature coming from the receiver П-104 (which is not included in the system), and on calculating the height and speed parameters on the basis of this information. The static and pitot pressures come to frequency pressure sensors (PS) which output electrical signals that are proportional to the measured pressures.

Signals from sensors and control signals come to the computer. Control signals allow producing the electric signals of atmospheric pressure at ground level P_g and of given flight level H_l. In computer by signals P_{st}, P_p and signals P_g and H_l there is calculation of height and speed parameters, correction of signals of relative (H_{rel}) and absolute (H_{abs}) heights, generation of warning signals on deviation from the given flight level and other one-time signals. Signals which are proportional to height and speed parameters are supplied to interconnecting aircraft systems in a form of code. Values of P_g and H_l, H_{rel} (in meters or feet) are indicated on the liquid crystal display (Fig. 2.38, a – indication in feet, Fig. 2.38, *b* – indication in meters).



a *b*

Fig. 2.38

Signal H_{rel} comes to system indicator where is displayed on a five-digit counter *7* and simultaneously the values of three low-order digits of counter are indicated with help of moving arrow *4* and fixed scale *5*. The warning signals of deviation from given flight level coming to the system indicator, provide warning on deviation by means of a light frame *3* around the value H_l on five-digit counter *2* of the indicator.

If there is a deviation more than 150 m from given flight level, then on the indicating device there is a visual signal in a form of constantly visible frame *3* around the value of flight level. By pushing rack mechanism *9* (H_l) there is the suppression of the visual signal (frame disappears). If there is a deviation from given flight level ranging from 60 to 150 meters (from 200 to 500 feet) the frame *3* is in the flashing mode. If the aircraft does not deviate from the specified flight level for more than 60 meters (200 feet) then the indication H_l and frame *3* are absent. Upon entering zone of ⌇150 m from given flight level and exceeding zone of ⌇60 m the system generates the signal of aural warning.

Logic of generating the warning signals of deviation from given flight level is illustrated in Fig. 2.39.

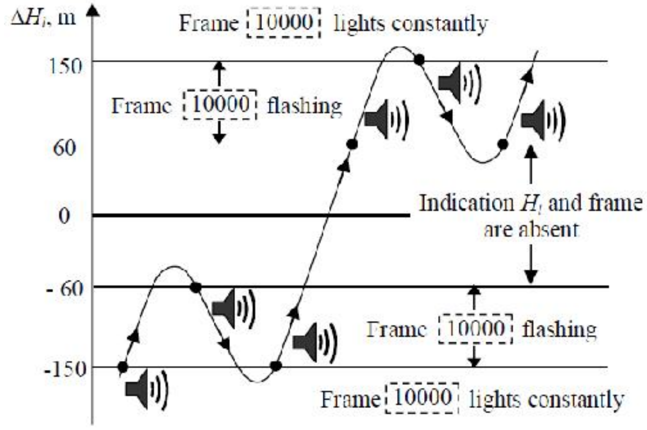


Fig. 2.39

Pressing and releasing the rack mechanism *6* (P_g) on the indicating unit *8* will set the fixed value of atmosphere pressure at ground level equal to 1013 hPa.

Pressing and releasing the button *1* (Ft) will change the background of indicator scale from green to yellow or from yellow to green. With this the mode of indication is also changing from meters to feet and vice versa.

If during the flight the relative barometric altitude is less than 1000 m, then at the counter of barometric height *7* instead of the high-order digit there will be visual signal “≡”. For negative barometric altitude the sign “Z” (minus) will be on the counter in the position before the first significant digit, while moving arrow *4* of indicator disappears.

The system ВБЭ-СВС is a part of height and speed equipment ББЭ-СВС-БСКА. Except two ВБЭ-СВС (blocks of pilot and co-pilot) the system includes two probes of stagnation temperature П-104, two blocks of communication and of БСКА control type, two alarm boards “CHECK H”. Information from the system comes to flight data recorder (FDR), to the traffic collision avoidance system TCAS 2000, to aircraft transponder СО-72М. Structure of ББЭ-СВС-БСКА is given in Fig. 2.40.

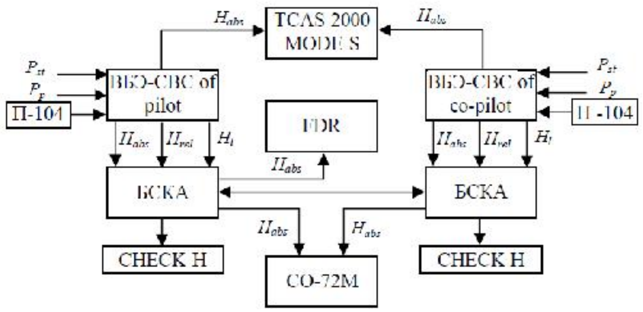


Fig. 2.40

Further development of ADC system is flight environment data system.

2.2. Information systems of height and speed parameters

Necessity to increase information content, accuracy and reliability of ADC systems, to enhance the depth of their control with failures warning leads to the creation of flight environment data system, so-called FEDS. They are a multi-channel information and measuring systems with built-in automatic control, designed to measure, calculate and produce information on the current values of height and speed parameters and information on the their deviations from the preset values, and indicate them for the crew and airborne consumers. These systems (if necessary) ensure the implementation of laws of aerodynamic errors compensation for pressure probes in a form of Mach number and the angle of attack functions. If there is radio altimeter in a system, it also gives information about the current true altitude of flight.

In addition, the system forms information on maximum allowable values of parameters and produces the warning signals to the crew about approaching to them. The system also provides entering a number of preset values of flight parameters and performance of reduced vertical separation minimum in 300 meters (1000 feet).

The basis of FEDS is ADC system that is used either as in non-reserved or in the reserved form. In some FEDS the information of ADC is integrated with information from other sensors and subsystems included in the system to improve its accuracy. This mainly concerns to the estimation of altitude and vertical speed, that is, those parameters which have information redundancy.

In the structure of the system like ИК ВСП-1-6 (Fig. 2.41) on the basis of the system type CBC1-72-1 the following components are used: ground proximity warning system (GPWS) with block of linear acceleration (BLA) and radio altimeter (RA); angle of attack and overload indicator (AAOI); computer of critical warning regimes (CCWS); flight level warning system (FLWS); block of formation and control (BFC).

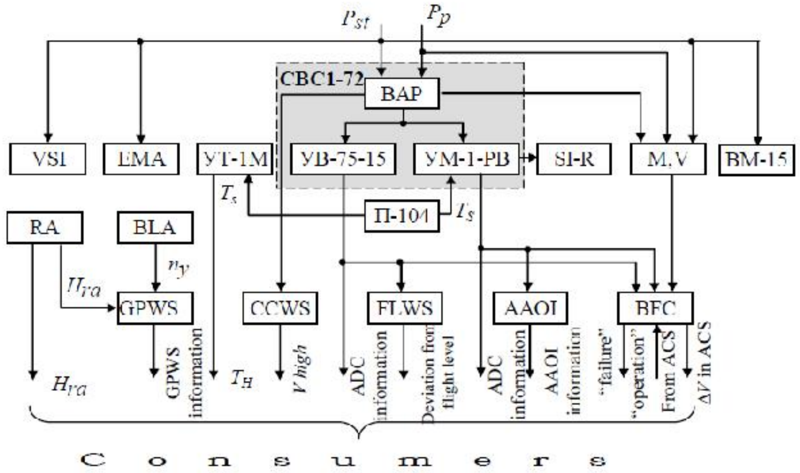


Fig. 2.41

In the base system of CBC1-72-1 type the aerodynamic flight parameters are formed in computers connected constructively with indicators. The signals which are proportional to the static and dynamic pressures and to the absolute altitude and indicator airspeed come to the computer from the block of air parameters (BAP). The signal that is proportional to the temperature of stagnated airflow comes from the stagnation temperature sensor П-104. If necessary, in computers the pressure and temperature at ground level and pressure of given datum can be manually entered.

The ground proximity warning system is designed to issue the warning signals on takeoff and landing stages by descending speed $V_y > V_{ycr}$ and in case of aircraft approach to the ground in dangerous height range. Dangerous height range is determined depending on barometrical radio inertial V_{ybri} or barometrical inertial V_{ybi} vertical speed. The speed V_{ybi} is received as a result of integration of barometric vertical speed V_y and vertical inertial speed calculated by integrating the vertical accelerations. The speed V_{ybri} is the result of integration of vertical velocity V_{yra} that is received by differentiating the signal of radio altimeter and vertical speed V_{ybi}.

The instrument OOAI is designed to measure, calculate and display the current values of the angle of attack and vertical overload and maximum allowable values of these parameters depending on the flight regime. Upon reaching the maximum allowable values of the angle of attack and overload the block OOAI outputs signals on warning devices.

The calculator CCWR issues the warning signals about the aircraft approaching to the maximum allowable value of indicator airspeed to other airborne systems.

The flight level warning system is designed to set the flight level H_{FL} and warning of crew by means of light and sound signals about the approach to the given flight level or about dangerous deviation from it.

The block of formation and control is intended for commutation and control of operable functioning of the system, and to identify and to output information on the failure of height and speed channels to the consumers, as well as for issuing the signal ζV to automatic flight control system (ACS).

Information on the current values of the height and speed parameters is indicated for the crew on the indicators of system СВС1-72-1 which are functioning as electromechanical computing devices (altitude indicator УВ-75-I5ПБ and Mach number and speed indicator УМ-1-ПБ). This information is duplicated by speed indicator, reserved SI-R, by combined speed and Mach number indicator (M,V), mechanical altimeter ВМ-I5ПБ, electromechanical altimeter EMA, vertical speed indicator VSI and indicator of the outside air temperature УТ-1М.

In other variant of the similar system (complex of ИК ВСП-1-1 type) in order to increase the reliability of information support three systems СВС-1-72-1 are used.

Complex ИК ВСП-1-1 is the information measuring system, divided into three independent subchannels of measurement, calculation and formation of signals on height and speed parameters covered by single checking system of correct functioning.

In each subchannel of measurement of height and speed parameters to improve the reliability of measurements and to ensure independence of outputs the quorum elements are used. Quorum element forms reliable output signal of three identical circuit elements, equal to the average value of the majority of the input signals which differ little in their value. Checking output signals is done on the majority logic principle ("voting by majority"). If one signal differs from the output signal of quorum element in a certain value, then the corresponding warning device of quorum element is triggered and outputs the signal to the logic circuit that removes signal of correct functioning from the corresponding subchannel. In case of failure of two subchannels there is removal of signal of correct functioning from the whole channel. In case of such failure the warning system outputs information about the failure to the crew.

Defective block of the system is determined by built-in control. Defective blocks may be replaced by normally functioning blocks, and then the FEDS should be checked again by built-in control.

System ИК ВСП-1-6 and its modifications are installed on aircraft An-72, An-74, Yak-42.

Considered variant of analog FEDS is essentially a set of separate subsystems (ADC, GPWS, CCWR, FLWS, AAOI types), each of them working in their own algorithm and solving the specific problem.

A higher degree of integration belongs to digital FEDS, e.g. flight environment data system ИК ВСП-10, which includes digital computer based on system СВС-2Ц; system of restrictive signals СОС 2-Ц; radio altimeter for small and medium altitudes РВ-21 "Импульс"; stagnation temperature sensor П-104; combination instrument ДА-200П; autonomous reserved instruments УСМ-2 and ВМ-30; air pressure probes ПВД-18-3М series 2 and ПВД-7.

When creating СВС-2Ц-2 the data fusion algorithms have been developed and put into practice for aerometric and inertial navigation systems. Algorithms provide the increase of dynamic accuracy of parameters measurement at unsteady flight modes and the improving of the accuracy of the vertical speed calculation.

Fig. 2.42

Structurally the system СВС-2Ц is a monoblock of computer В-2Ц-2 (Fig.

2.42). Digital computer of the system is a computer of *SISD* class. As part of a system the new precision pressure sensors of ДДГ-1 type are used. Airdata computer systems of this type are designed to work with EFIS.

The system of restrictive signals СОС 2-72 includes computer for restriction signals ВСО-1-1, block of formation of indication signals БФСИ-2-1, angle of attack and overload indicator УАП-5-13, sensor of aerodynamic angles ДАУ-72-1, sensor of aerodynamic angles ДАУ-72-4, linear acceleration sensor ДЛУ-26-02 and indicator of speed and Mach number УСМ-2.

The structure of the radio altimeter РВ-21Д includes two antennas А-061-4, transceiver А-35-1 and altitude indicator А-034-4.

System ИК ВСП-10 is a part of the flight and navigation system and is designed to calculate the current aerometric parameters of flight, true geometric height, current normal overload, as well as maximum and minimum allowable values of speed, normal overload, angle of attack, and to output data for indication and to other airborne systems. Structure diagram of IS HSP-10 is shown in Fig. 2.43.

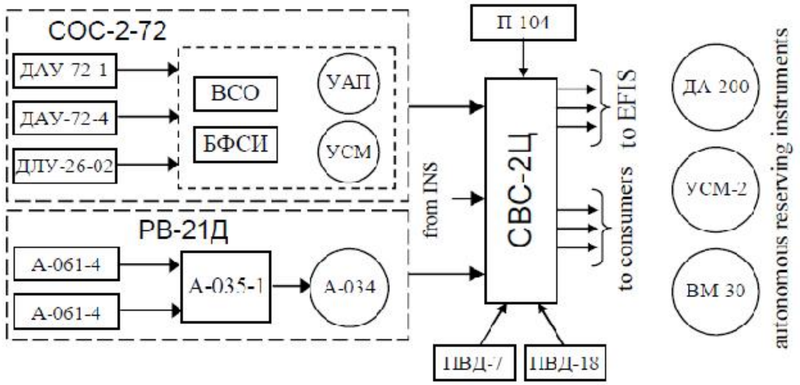


Fig. 2.43

Complex forms and outputs the signals of the approach to the critical values of angle of attack, normal overload, Mach number, instrument airspeed for indication and warning.

The system also provides:

- creation and output of deviation signals of current values of geometric height (H_g), barometric altitude and Mach number fixed in the moment of switching on the stabilization modes for AFCS;

- creation and output of one-time signals in the airborne systems in case of reaching a fixed value H_g;

- creation and output of the signals of normal operation of blocks, systems and communication lines in the airborne systems;

- indication on reserving instruments of parameters H_{rel}, V_{IAS}, V_y, H_g, n_y, and α.

The basis of algorithms of aerometric parameters calculation is the method of piecewise polynomial approximation of calculated nonlinear calibrated functions similar to those discussed earlier in analog and digital ADC systems. Aerometric parameters are calculated using known calibrated dependencies for standard atmosphere. The main difference of computational procedures is the transformation of computing algorithms, special algorithm for vertical speed calculation and optimal filtering of signals in the calculation of output parameters.

To calculate the main parameters the method of piecewise linear approximation of these dependencies is used:

$$\lg P_{st} = f_{st}(P_{st}); \lg P_p = f_p(P_p); H_{abs} = f_H(\lg P_{st});$$
$$M = f_M(\lg P_p - \lg P_{st}); T_H = f_s(T_s); V_i = f_V(\lg P_p),$$

where f_{st}, f_p, f_H, f_M, f_t, f_V are special transformative and approximation functions; T_s is a temperature of stagnated airflow measured by temperature probe.

Functions in computer are divided into 16 intervals, in each of them the desired function is approximated by a polynomial (from the 2nd to the 5th order). The values of the function within the interval are determined by so-called Horner's method (algorithm for computing the polynomial values recorded as a sum of monomials, for a given value of variable).

The scheme of calculating the value of x at point $x = x_0$, for example, for polynomial of 5th order is the following:

$$f(x_0) = ((((a_5 x_0 + a_4)x_0 + a_3)x_0 + a_2)x_0 + a_1)x_0 + a_0,$$

where $a_0 \ldots a_5$ are coefficients of approximation of function on a given interval stored in the computer memory.

In addition to these dependencies for correction of error of static pressure probe ПВД-18, the computer ИК ВСП uses another function: $P_{st} = f(M, H_{abs})$. This function is determined experimentally and is stored in the computer memory.

The given above approach to computation of basic parameters allows:

- getting the high accuracy of parameters calculation using relatively simple polynomial of approximation;
- excluding the division operations in the calculation of the Mach number with the help of logarithms, fulfillment of which in digital computers requires time;
- using single method of calculation and due to this to simplifying the system of operations and computer program.

In СВС-2Ц-1М for determination of V_y there is numerical differentiation of current values of absolute barometric altitude. The double-point algorithm for the numerical differentiation is implemented for this in the system computer.

In numerical differentiation the expressions of infinitesimal increments of functions and of argument are replaced by finite differences

$$\frac{dy}{dx} \approx \frac{\Delta y}{\Delta x}.$$

Obviously, the smaller the argument increment is, the more accurate the numerical value of derivative will be.

For double-point methods at calculating the derivatives the values of functions in two points are used. The argument increment is set by three ways, shifting $\Delta x = h$ to the right, to the left and in both directions from the given point. Accordingly, there are three double-point methods of numerical differentiation:

$$\frac{dy}{dx} \approx \frac{\Delta y}{\Delta x} = \frac{y(x+\Delta x) - y(x)}{\Delta x} \text{ is the first method;}$$

$$\frac{dy}{dx} \approx \frac{\Delta y}{\Delta x} = \frac{y(x) - y(x-\Delta x)}{\Delta x} \text{ is the second method;}$$

$$\frac{dy}{dx} \approx \frac{\Delta y}{\Delta x} = \frac{y(x+\Delta x) - y(x-\Delta x)}{2\Delta x} \text{ is the third method.}$$

In СВС-2Ц-1М for determination of V_y, the current values of absolute barometric altitude are differentiated using the first method of numerical differentiation:

$$V_y \approx \frac{\Delta H_{abs}}{\Delta t} = \frac{H_{\text{abs}}(t+h) - H_{\text{abs}}(t)}{h},$$

where $H_{abs}(t+h), H_{abs}(t)$ are values of absolute barometric altitude, calculated at previous and current cycles of calculation, respectively; h is duration of calculation cycle.

However, error values of this method of calculation for vertical speed are too high, because output errors of determination H_{abs} can be significant (e.g. only through delay of pitot-static tube system they may reach up to 50 m at certain flight modes).

The error of calculating V_y can be sufficiently decreased if instead of differentiation H_{abs} the optimal filter of data fusion between H_{abs} and signal of vertical accelerometer of inertial system is used.

This method of calculating V_y is used in modifications of system ИК ВСП-10, where there is optimal Kalman filtering.

The equations of filter are formed on the basis of observation equations and state equations. The state equations are the following:

$$\dot{H} = V_y;$$

$$\dot{V}y = a_y,$$

where H is current flight height; V_y, a_y are vertical speed and vertical acceleration.

Observation equations are derived from mathematical models of measurements of barometric altitude and vertical acceleration, which are the following:

$$T_H \dot{H}^* + H^* = H + \Delta H + \xi_H(t),$$

$$a_y^* = a_y + \Delta a_y + \xi_{ay}(t),$$

where H^*, a_y^* are measured altitude (measurements of ADC system) and transferring translational acceleration measured in INS; ΔH is slowly varied error that is due to deviation of pressure at altitude H from its standard value; Δa_y is slowly varied error of INS accelerometer;

$\notin_H(t), \notin_{ay}(t)$ are white noise components of altitude and acceleration measurement; T_H is time constant of altitude measurement that is caused mainly by the static pressure lines.

The apparent acceleration $\dot{w}_y$ measured by INS accelerometer, is indicated taking into account the transferring translational acceleration a_y and a_y^C that considers Coriolis (rotary) acceleration and acceleration of gravity force $\dot{w}_y \, \mathrm{X} a_y \, \Gamma \, a_y^C$.

Values of Coriolis acceleration and acceleration of gravity force are calculated in INS by special algorithms. So, the value $a_y \, \mathrm{X} \dot{w}_y \, \mathrm{Z} a_y^C$ should be considered as transferring translation acceleration calculated according to INS algorithms.

Neglecting the slowly varied errors of altitude ζH and acceleration ζa_y measurements, it is possible to calculate the following by optimal Kalman filter:

$$\begin{aligned} &\dot{\hat{H}} \, \mathrm{X} \hat{V}_y \, \Gamma \, k_1(T_H \dot{H}^* \, \Gamma \, H^* \, \mathrm{Z} \hat{H}); \\ &\dot{\hat{V}}_y \, \mathrm{X} a_y^* \, \Gamma \, k_2(T_H \dot{H}^* \, \Gamma \, H^* \, \mathrm{Z} \hat{H}). \end{aligned} \qquad (2.7)$$

where $\hat{H}, \hat{V}_y$ are estimated values of altitude and vertical speed.

The correction coefficients k_1 and k_2 can be calculated using equation of covarianes, and also as constant, pre-calculated ratios:

$$k_2 \, \mathrm{X} \sqrt{\frac{i_{ay}}{i_H}}, \quad k_1 \, \mathrm{X} \sqrt{2k_2}. \qquad (2.8)$$

where i_{ay}, i_H are intensities of white noises $\notin_{ay}$ and $\notin_H$.

Analysis of equations (2.8) Z (2.9) shows that the filter completely eliminates dynamic lags with estimation of vertical speed and altitude. Change of parameters of error models which differs from the optimal ones on 60 ... 70%, does not lead to significant degradation of the filter features.

Unaccounted slowly varied errors of altitude measurement ζH are directly included in the errors of estimation of altitude and vertical speed and cannot be completely eliminated. The components of the estimation errors which are caused by quasi stationary errors of accelerometer ζa_y can be reduced by using the third-order filter.

Without taking into account the altitude measurement errors ζH and with introducing the additional ratio $\zeta \dot{a}_y \, \mathrm{X} 0$ into observation equation, the optimal filter is synthesized, similar to the above considered one, but is already of the third order:

$$\begin{aligned} &\dot{\hat{H}} \, \mathrm{X} \hat{V}_y \, \Gamma \, k_1(T_H \dot{H}^* \, \Gamma \, H^* \, \mathrm{Z} \hat{H}); \\ &\dot{\hat{V}}_y \, \mathrm{X} a_y^* \, \mathrm{Z} \zeta \hat{a}_y \, \Gamma \, k_2(T_H \dot{H}^* \, \Gamma \, H^* \, \mathrm{Z} \hat{H}); \\ &\zeta \dot{\hat{a}}_y \, \mathrm{X} k_3(T_H \dot{H}^* \, \Gamma \, H^* \, \mathrm{Z} \hat{H}). \end{aligned}$$

Using this approach, the estimation system becomes astatic relatively slowly varied error ζa_y of measurement of transferring translational acceleration.

Further development of this series of FEDS is the system based on triplex ADC system CBC-85. Using the triple reserving allows not only improving the reliability of information support, but also to some extent increasing the accuracy of calculating of height and speed parameters.

For regional aircraft produced by Antonov firm, the company “Aviacontrol” (created on the basis of Kharkiv instrument making design bureau), which is specialized in developing and manufacturing the systems and instruments for aviation, organized the serial production of the following:

Z flight environment data systems ИК ВСП-140-01, ИК ВСП-140-74 for aircraft AN-140-100, AN-74TK-300, AN-74TK-200;

Z flight environment data system ИК ВСП-148 for aircraft AN-148 and AN-158;

Z coded system of flight level warning ССВЭК-1 for aircraft AN-124.

Information systems ИК ВСП-140-01, ИК ВСП-140-74 and ИК ВСП-148 are multi-channel information and measuring systems covered by automatic control in flight, and are designed to measure, calculate, form and provide the crew and airborne consumers with the following information:

- current values of height and speed parameters, temperature, angle of attack and vertical overload;
- maximum allowable values of flight parameters;
- warning signals of approaching to critical modes of flight;
- signals on correct operation of the system.

Systems ensure the implementation of laws which realize the compensation of aerodynamic errors of air pressure probes in the form of functions of Mach number and of angle of attack. Systems enforce implementation of rules for reduced vertical separation minimum (RVSM) in 300 meters (1000 feet).

Principles of structure and algorithms of system operation are similar to the digital FEDS. The structure of ИК ВСП-140-74 is shown in Fig. 2.44.

Fig. 2.44

The system includes:

- unit of air parameters МВП-1-1 - 3 pcs.;
- warning block БС-1;
- coded system of flight level warning ССВЭК-1;
- coded indicator of angle of attack and overload УАПК-1-ПБ1;
- coded indicator of altitude УВК-1М-ПБ - 2 pcs.;
- coded indicator of altitude УВК-1Ф;
- coded indicator of temperature УТК-1.
- coded indicator of speed and Mach number УСИМК-1 - 2 pc.;
- sensor of aerodynamic angles ДАУ-72-1 - 2 pc.;
- block of linear acceleration sensor БДЛУ 1-5;
- sensor of stagnation temperature П-104.

Products which are a part of the system, have “Certificate of component compliance” issued by International Aviation Committee and State Aviation Administration of Ukraine.

Triple reserving of computer МВП-1-1 allows increasing the reliability of the information support of consumers with current values of height and speed parameters.

Representative of the height and speed parameters and aerodynamic parameters measurement system belonging to the class of digital FEDS is a modern system of СИ ВСП-35 manufactured by JSC “Aeroprybor-Voshod” (Fig. 2.45).

Fig. 2.45

The system consists of two double-channel blocks of computer, four blocks of detector-transducers of pressure and two double-channel probes of temperature of stagnated airflow П-104М.

The main advantages of the system are:

- reducing the number and size of the elements contacting with the airflow and decreasing the weight of aerometric equipment which is especially important for multiple reserving;
- increased reliability due to eliminating the sensors of aerodynamic angles with moving mechanical parts;
- high degree of resistance to failures due to information redundancy and special algorithms of failures identifying;

- excluding the pneumatic lines on board the aircraft due to constructive combination of pressure sensors with air pressure probes (they are both implemented in single unit), which significantly reduces the dynamic lag in pneumatic lines and increases the measurement accuracy.

The system is based on receiver-converter of air pressure, previously considered, that not only measures the air pressure, but also on the basis of external information analytically calculates the true values of angles of attack and of slip.

Another development of the company is system for measuring the height and speed parameters СИ ВПВ-52.

The system СИ ВПВ-52 is included in the airborne equipment of the helicopter and is designed to determine the complete set of aerial flight parameters of helicopter and for information support of airborne systems and the crew at all flight modes, including flights forwards, backwards, sideways, up, down, and at modes of near-zero speeds and hovering. The system consists of two functionally completed modules (MIVP (Fig. 2.46), placed on consoles in areas of relatively “pure” airflow and connected by the bus of data exchange. The module is designed as a streamlined ellipsoid body with diameter 130 mm and length which is not exceeding 500 mm.

Each module contains:

- two pitot-static tubes (see Fig. 2.27, a);

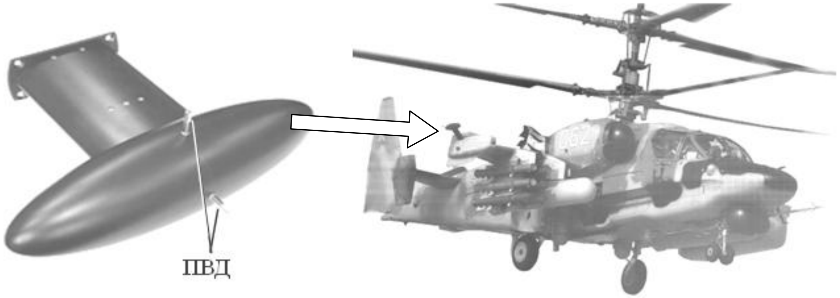


Fig. 2.46

- control unit of pitot-static tubes heating;
- pressure sensors block;
- outside air temperature sensors;
- computer of aerial parameters of flight;
- power supply.

2.3. Aerometric computing systems

Aerometric computing systems determine the coordinates of the aircraft by method of aerometric dead reckoning. This method of navigation is based on continuous calculating of the aircraft trajectory by values of magnitude and direction of ground speed vector and taking into account the coordinates of the initial point of motion.

The airplane relative to air mass (with no slip) moves with the true airspeed V in the direction of its longitudinal axis. At the same time it moves relative to the ground surface (in the selected navigation coordinate system) together with the air mass in the direction of the wind δ and with its velocity W.

In the aircraft navigation the wind direction δ is measured between the meridian and the wind vector. This wind is called navigation wind (where the wind is blowing *to*). The direction of meteorological wind differs from the navigation one in 180° (where the wind is blowing *from*).

As a result the aircraft flies relative to the ground surface by resultant force, built on the components of the airspeed vector and the wind velocity vector. Thus, when flying with crosswind the vectors of horizontal projection of true airspeed V_h, ground speed V_q and wind speed W form a triangle (Fig. 2.47), which is called the navigational speed triangle.

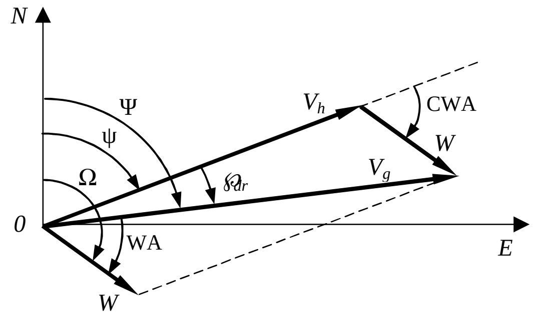


Fig. 2.47

Navigational speed triangle is usually built in a rectangular geotopic (normal earth-fixed) coordinate system. Mostly the orientation of the axes of horizontal rectangular coordinate systems is defined by the direction of geographical meridians and parallels. In the selected coordinate system the navigational speed triangle is characterized by:

- true (geographic) heading ψ;
- true airspeed (or rather its projection to the horizontal plane) V_h;
- track angle φ;
- ground speed V_g;
- drift angle β_{dr};
- wind speed W;
- wind direction δ

Orientation of wind velocity vector relative to ground speed vector is characterized by wind angle (WA) and relative to the projection of longitudinal axis of aircraft on the horizontal plane it is characterized by course wind angle (CWA).

Drift angle β_{dr} (angle between the horizontal projection of the airspeed vector V_h and direction of ground speed V_g) is counted from the vector of V_h to the track line rightwards with "+" sign and leftwards with "–" sign. At modern flight speeds the drift angle does not exceed 10...20°.

Wind angle is the angle between the direction of ground speed and direction of navigational wind. It is counted from the track line towards the wind direction clockwise from 0 to 360°.

Course wind angle is called the angle between the horizontal projection of the true airspeed vector V_h and the direction of navigational wind. It is counted clockwise from 0 to 360°.

Let us define the projections of ground speed on the axes of selected coordinate system. Projecting the speeds V_h and W on the axes ON, OE of geotopic coordinate system gives:

$$V_{g_N} = V_h \cos\psi + W\cos\delta,$$
$$V_{g_E} = V_h \sin\psi + W\sin\delta,$$

where V_{g_N} is projection of ground speed on the meridian direction; V_{g_E} is projection of ground speed on the parallel direction.

The true airspeed V and the true heading ψ are continuously measured by the sensors of speed and heading of aircraft. Speed W and direction of wind δ are determined according to weather station information or by one of the air navigation methods.

Calculation of the traveled linear path along the meridian S_N and parallel S_E is made by integration of components of ground speed V_{g_N} and V_{g_E}:

$$\begin{aligned} S_N &= \int_{t_0}^{t} V_{g_N}\,dt = \int_{t_0}^{t} (V_h\cos\psi + W\cos\delta)\,dt; \\ S_E &= \int_{t_0}^{t} V_{g_E}\,dt = \int_{t_0}^{t} (V_h\sin\psi + W\sin\delta)\,dt. \end{aligned} \tag{2.9}$$

These equations are called equations of course recording machine, or plotter. Considering the relationship between linear and geographic coordinates, and assuming the simplified (spherical) shape of the Earth, it is possible to calculate the geographic coordinates φ and λ (latitude and longitude) of the aircraft location:

$$\varphi = \varphi_0 + \frac{180}{\pi}\int_{t_0}^{t} \frac{V_h\cos\psi + W\cos\delta}{R_0}\,dt;$$

$$\lambda = \lambda_0 + \frac{180}{\pi}\int_{t_0}^{t} \frac{V_h\sin\psi + W\sin\delta}{R_0\cos\varphi}\,dt,$$

where φ_0, λ_0 are coordinates of initial waypoint; R_0 is the radius of the Earth; $R_0\cos\varphi$ is the radius of the given parallel circle; φ is the latitude of parallel.

Position of the aircraft is not always determined in geographical coordinate system. Existing aerometric computing systems solve this problem in the grid coordinate system, which is used as a rectangular system XOY that is rotated relatively the geographic coordinate system to the map angle ψ_m (Fig. 2.48).

Map angle (see Fig. 2.48) is the angle between the northern direction of the meridian and the axis OY of rectangular coordinate system. Map angle is counted from the meridian direction clockwise. Origin of the coordinate system XOY is connected with the initial waypoint or with any landmark.

Current aircraft coordinates X, Y under the assumption that the navigation problem is solved in the horizontal flight ($V_h \approx V$), are determined as follows:

$$X = X_0 + \int_{t_0}^{t} \left[V \sin(\psi - \psi_m) + W \sin(\delta - \psi_m) \right] dt;$$
$$Y = Y_0 + \int_{t_0}^{t} \left[V \cos(\psi - \psi_m) + W \cos(\delta - \psi_m) \right] dt; \qquad (2.10)$$

where X_0, Y_0 are coordinates of initial waypoint.

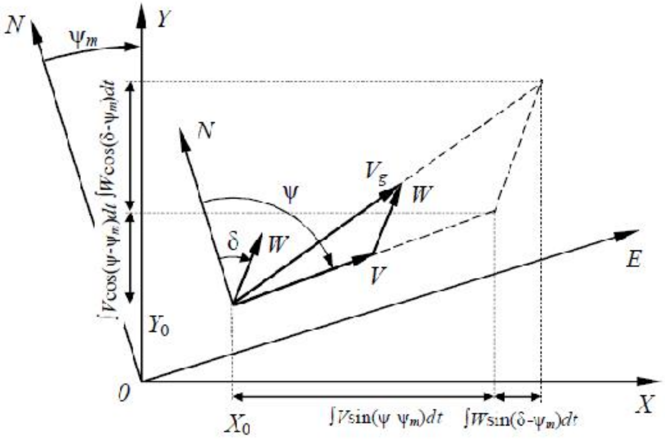


Fig. 2.48

The considered algorithms for aerometric dead reckoning are implemented in the navigational indicators of type NI-50, as well as in dead reckoning computers of type АНУ, НВУ and ЦНВУ.

The navigational indicator NI-50 is used to determine the coordinates of the aircraft in the grid rectangular coordinate system, as well as to determine the speed and wind direction, to control the moment of approaching to the given landmark, for dead reckoning, to approach the desired track line, etc.

The block diagram of navigation indicator is shown in Fig. 2.49.

The kit of navigational indicator includes the following basic devices: airspeed sensor (ASS), directional stabilizer (DS), wind selector (SW) and coordinate counter (CC). Temperature T_{st} of stagnated airflow required for calculating the true airspeed comes from the temperature sensor П-1. The navigational indicator is semiautomatic. One part of the initial data is entered the unit automatically, and the other part is entered manually.

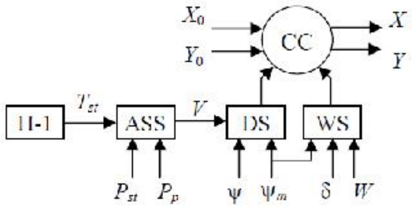


Fig. 2.49

Airspeed sensor is an electromechanical computing device that outputs the voltage proportional to V, the value of which is determined by the membrane mechanism that is supplied by the pitot P_p and static P_{st} pressures. The true airspeed is determined by the formula:

$$V = N\sqrt{T_{st}} \left(\frac{P_d}{P_{st}} \right)^m,$$

where $N = \sqrt{2R\frac{k}{k-1}}$ is constant value (R is specific air constant; k is adiabatic coefficient); T_{st} is temperature of stagnated airflow; P_d is dynamic pressure of airflow; P_{st} is static pressure; m is exponent.

Exponent is chosen from the condition:

$$\left(\frac{P_d}{P_{st}} \right)^m = \sqrt{1 - \left(\frac{P_d}{P_{st}} + 1 \right)^{\frac{1-k}{k}}}.$$

Solving the dependency (2.8) is ensured by means of converting the measured values T_{st}, P_p, P_{st} into electric signals and by electromechanical multiplying of the values proportional to $(1/P_{st})^m$, P_d^m, $\sqrt{T_{st}}$. Temperature T_{st} is measured by means of thermistor, the pressures P_p

and P_{st} are measured by means of pressure-gauge and aneroid capsules, deformation of which is converted into voltage.

Signal of true airspeed from airspeed sensor and the signal of the current magnetic heading from compass system come to computing device of processing system. The following values are entered this device manually from control panel of NI-50: map angle, wind speed and wind direction. At the time of flight through the given waypoint the pointers of coordinate counters “C” (North) and “B” (East) are set to zero.

Functional diagram of navigational indicator in the mode of dead reckoning in the grid rectangular coordinate system is shown in Fig. 2.50.

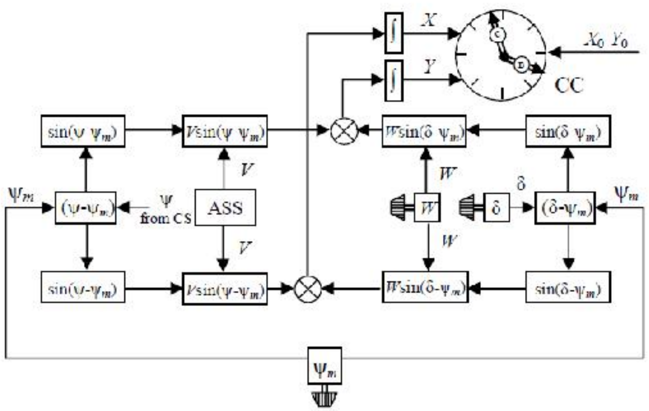


Fig. 2.50

All operations of sine-cosine transformation, multiplication and algebraic addition according to equations (2.9) are performed using potentiometric circuits.

The integration is carried out by integrating engines (DC motor), in which the speed of rotation is directly proportional to the supplied voltage.

Errors of navigation machines, as measuring devices of indirect measurement method are methodological and instrumental. Instrumental error of navigational indicator is 5 ... 7% of face value.

The main measurement-method error of dead reckoning computer based on the integration of ground speed is caused by the absence of accurate and continuous information on aircraft about wind speed and wind direction.

Error in setting the wind speed can reach 10...40 km/h and error in wind angle is 2...3⌠. In this regard, during the flight the correction according to the actual wind is periodically done. The frequency of input of wind correction depends on the average value of the wind speed, which is not taken into account by the automatic navigation machine.

In addition, there are measurement-method errors which are caused by errors of sensors of true airspeed and heading. During the flight the vector of true airspeed does not coincide with the longitudinal axis of aircraft. The presence of angles of attack and of slip, angle of trajectory inclination leads to errors of dead reckoning.

The most significant errors of dead reckoning are caused by slip angle and errors of airspeed sensor. The angles of attack and of trajectory inclination less affect the errors of dead reckoning.

For example, at constant angles $\alpha = \beta = \theta = 1$ ° the errors of dead reckoning from the slip angle are 1.75%, and from angles of attack and of trajectory inclination the errors are 0.03% of traveled distance. However, for maneuverable aircraft the angles θ can vary widely, resulting in the fact that errors of dead reckoning may be significant.

The errors of heading sensor can be divided into time-independent errors (deviation, dead zone errors, etc.) and periodic, or time-dependent errors. Therefore, the errors of dead reckoning which are caused by errors of heading sensor, are similar in nature.

Navigational indicator of NI-50 type is prototype of integrated aerometric-Doppler navigation systems. Aerometric method of dead reckoning is used as the basic one in navigating system БНК-62 of the aircraft IL-62. In this mode, the dead reckoning is performed according to the data from precise compass system, from ADC system and by wind parameters that were memorized or entered manually. Aerometric dead reckoning systems integrated with satellite navigation systems, have been widely used also on small unmanned aerial vehicles.

2.4. Features of aerometric navigation systems operation

In-flight system operation. At the stage of aircraft pre-flight inspection it is necessary to make sure that from the pitot and static tubes the covers and caps are removed and that they have no mechanical damage.

After switching the power supply, it is necessary to check the heating system of pitot-static tubes, to select and set the units of measurement of pressure (millimeter of mercury column or hPa) and of altitude (meters or feet), to input the pressure value in the system, which is given by the dispatcher. For Russian airfields and some aerodromes of former Soviet Union countries it is pressure *QFE*, in Ukraine and abroad it is pressure *QNH*.

Beforehand it is necessary to make sure that readings on indicators of the system correspond to the type and value of barometric pressure (0 or exceeding the datum of runway) and there is no signal of failure and malfunction. On the runway start it is necessary to switch on the heating of pitot-static tubes.

At climbing it is necessary to set *QNE* pressure at 760 mm of mercury (1013.25 hPa). Change of reference system is at transition altitude (TA). Since at the modern aircraft it is generally expected the installation of at least two major sets of sensors of height and speed parameters together with reserving piloting combined device of ППКР-СВС type, it is necessary to compare the readings of these devices, monitoring that the difference between them does not exceed the value specified by normative documents.

In flight it is necessary to compare periodically the altitude with the given flight level and readings of altimeters, airspeed indicators and vertical speed indicators. When piloting it is necessary to avoid exceeding beyond the limit values of airspeed, vertical overload and angle of attack.

At descending, the crew must request the pressure value at runway of landing airfield *QFE*, or *QNH* depending on the reference system adopted at the aerodrome. After receiving the pressure data and the landing clearance, it is necessary to input the type and value of the obtained pressure in the system. Change of reference system at descending is performed at transition level (TL).

After landing, it is necessary to turn off the heating of pitot-static tubes. After driving to hangar it is necessary to turn off the power supply of system.

Technical operation of systems. During pre-flight procedures before turning on the power supply it is necessary to make sure that the covers and caps are removed and that they have no mechanical damage.

Finding and troubleshooting are performed in accordance with the recommendations of the “Maintenance guidance”.

When performing assembling and disassembling it is not allowed applying great efforts in twisting-in the coupling nuts at fastening pipes and using the oil, causing an explosion of oxygen. Disconnected pipelines must be protected with plastic caps. Before blowing the pipes, it is necessary to disconnect all consumers of the system. Blowing the pipelines of pitot and static pressures is held at air pressure of 1.5...2 kilogram-force/cm^2.

When checking the pressurization of the system of pitot and static pressures and for estimation of accuracy characteristics at rarefication the readings of vertical speed must not exceed the values specified in the “Maintenance guidance”. The accuracy characteristics are estimated by comparing the readings of aircraft indicators with testing devices. Bothe pressure and rarefication are created smoothly, with estimation of smoothness of change in indicator values of the aerometric parameters.

When checking the probes of pitot and static pressures the attention is paid to the reliability of attachment, absence of external damage, cleanliness of input and drainage holes. Moisture chambers are checked for the absence of mechanical damage. In the presence of water in moisture chambers the water should be drained. Pipelines, rubber-canvas hoses at pipelines and fittings of devices must not have mechanical damage, cracks, cuts.

When checking the heating systems, the heating element must not be switched on over 5 minutes.

TEST QUESTIONS

21. Formulate the functionality of ADC systems. What is the principal difference of ADC systems from aerometric devices?
22. List the main aerometric parameters coming from the ADC system.
23. What values measured in ADC systems are primary?
24. Give an equation of relationship between primary measurable values which are implemented in ADC systems.
25. List the main types of air pressure probes.

26. What is the main feature of pressure probes in supersonic aircraft and helicopters?
27. What method is most commonly used to measure the local aerodynamic angles?
28. What is the basis of operation of the pneumatic sensor for aerodynamic angles?
29. How is the multi-functionality of sensors for aerodynamic parameters achieved?
30. Give examples of analog and digital ADC systems.
31. Why are the functional relationships of calibration formulas converted into logarithmic in ADC systems?
32. What types of pressure sensors are used in analog and digital ADC systems?
33. What are the main functional blocks included in computing device of digital ADC systems?
34. What method is the basis for the calculation algorithms of non-linear calibration functions in digital ADC systems?
35. What subsystems are included in the systems of ИК ВСП-1-6 and ИК ВСП-10?
36. What are algorithms of the complex data processing used in ИК ВСП-10?
37. What is the method of navigation used in aerometric computing systems?
38. What data are required to realize the aerometric method of dead reckoning in navigating system of aircraft IL-62?
39. What are altitudes at climbing and descending where there is a change of reference system of pressure?
40. What pressure must be set for the blowing of pipelines of pitot and static pressures?

Chapter 3. INERTIAL NAVIGATION SYSTEM

3.1. Tasks solved by inertial navigation systems and their classification

Inertial navigation systems (INS) are called systems that determine navigation parameters of object motion (ground, vertical speeds, and coordinates) using computations performed on accelerometer signals. Procedure of determination of ground and vertical speeds means preprocessing of accelerometer signals, with further integration. The orientation of accelerometers axis in space must be known in each moment of time. Fulfillment of this condition is provided either by setting accelerometers on a gyrostabilized platform, or by calculating their orientation from gyroscopic sensors.

By obtained ground speed vector it is possible to determine the coordinates of the object in the selected (great circle, geodetic, etc.) coordinate system using dead reckoning algorithms.

Beyond ground and vertical speeds and geographic or great circle coordinates of aircraft, this navigation system solves the problem of determination of roll, pitch and yaw angles and of accelerations and angular velocities. By definite algorithms it is possible to determine also trajectory angles, distance to the reference point with known coordinates, its azimuth and bearing, other navigation parameters.

Large amount of information, autonomy, and jamproofness provide leading role for INS in the structure of universal information system of the aircraft.

Often INS is classified depending on accelerometer location aboard the aircraft and on the role of computer in INS structure.

Depending on the accelerometers location there are platform and strapdown systems. In the first case the accelerometers are installed on the gyrostabilized platform, in the second case they are placed directly on the aircraft skin or in a special measurement unit, while the sensitivity axes of accelerometers do not change their orientation relatively aircraft axes.

Among platform INS, in turn, there is INS with free platform and INS with horizontal platform.

In INS with free platform the axes and accelerometers installed on the platform, are not rotated in inertial space.

INS with horizontal platform, in turn, is classified as INS with free in azimuth platform and INS with fixed in azimuth platform.

According to the computer role in determining the angular and linear coordinates there is geometric, analytic and semianalitic INS.

In geometric INS the basic element is gyrostabilizer that simulates the direction of inertial reference system axes. Platforms with accelerometers have sensitive axes which simulate the directions in the horizon plane and the direction of the local vertical. The role of computer is minimal. It is used to provide the correction of given position of the platform. Information about coordinates is taken from angular sensor of gyro stabilizer and platform.

Semianalytic systems include systems with horizontal platform. In these systems gyro platform with accelerometers reproduces the direction of normal (moving) coordinate system. From angular sensors of gyro stabilizer the information about roll, pitch, and aircraft heading is taken. INS computer solves the problem of determining the kinematic parameters of the aircraft center of mass and gives signals to correct the gyro stabilizer.

Analytical INS includes strapdown INS and INS with accelerometers on uncorrected or free gyro stabilizer.

Strapdown INS, in turn, can be classified by structure of primary data sensors, by algorithms of implementation of kinematic equations, in particular, by the chosen coordinate systems, in which the problem of inertial navigation is solved, and so on.

Computer of analytic INS does more computation compared with platform INS. In addition to determining the kinematic parameters of the aircraft center of mass, it analytically determines the angular orientation of normal moving coordinate system relatively inertial one and angular orientation of body-fixed moving coordinate system relatively normal one.

3.2. Fundamentals of inertial method for determining parameters of motion

The basis of the dead reckoning method is the basic laws of mechanics. Inertial dead reckoning can be done relative to the inertial reference system, created by reference (inertial) bodies which move uniformly and linearly in a space.

Inertial method of dead reckoning is based on physical and analytical modeling of the motion dynamics of the object under the action of resultant of external forces and moments which are applied to the object.

In the simplest case, the speed of motion and coordinates of aircraft can be determined by single and double integration of accelerations, measured by accelerometers taking into account the initial conditions.

Let's consider the basic equation of inertial method to determine the dynamics of the object motion which is written in vector form as

$$m\ddot{\mathbf{R}} = \mathbf{F},$$

where m is a mass of the object; $\mathbf{R}$ is a radius-vector (vector of position) of the center of mass of the object in inertial coordinate system; $\mathbf{F}$ is a resultant of external forces applied to the object. Force $\mathbf{F}$ can be written:

$$\mathbf{F} = m\sum_{i=0}^{k}\mathbf{g}_{0i}(\mathbf{R}_i) + \mathbf{F}_{ng}\,, \tag{3.1}$$

where $\mathbf{g}_{0i}(\mathbf{R}_i)$ is a vector of acceleration of gravity for i-th celestial body (if the flight takes place in aero space), which is a function of the radius vector $\mathbf{R}_i$; $\mathbf{F}_{ng}$ is a vector of non-gravitational external forces applied to the object. Dividing equation (3.1) by m, we obtain:

$$\ddot{\mathbf{R}} = \sum_{i=0}^{k}\mathbf{g}_{0i}(\mathbf{R}_i) + \mathbf{A}\,, \tag{3.2}$$

where $\mathbf{A} = \mathbf{F}_{ng}/m$ is an acceleration of the center of object mass, which is measured by accelerometer and is called the imaginary acceleration.

The differential equation (3.2) is the equation of inertial navigation in arbitrary inertial coordinate system for the general case. To determine the parameters of the spatial orientation of the object it is necessary to represent original equation (3.2) in the navigation coordinate system, which origin must be in some way connected to any specific celestial body, for example, the center of the Earth. Let's introduce an orthogonal right-handed coordinate system $O_O\xi_0\eta_0\zeta_0$ (Fig. 3.1), in which Newton's laws are valid and coordinate system $O\xi\eta\zeta$ with origin in the beginning of the center of mass of the Earth, the orientation axes coincide.

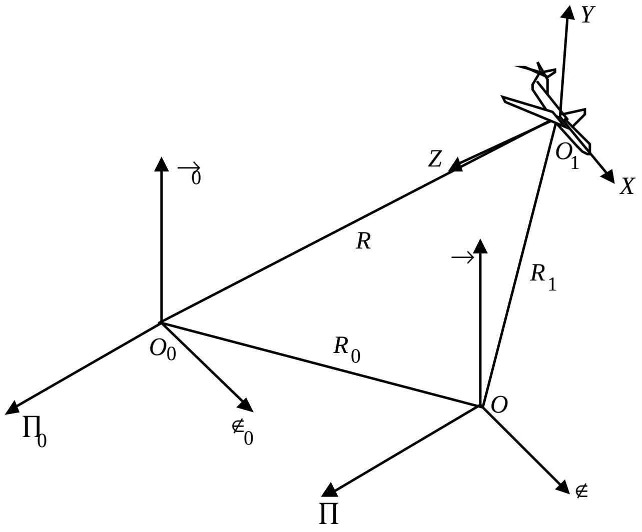


Fig. 3.1

Then the radius vector **R** of object in arbitrary inertial system is connected with the radius vector $\mathbf{R}_1$ in the coordinate system with the origin in the center of the Earth by ratio

$$\mathbf{R} \, X \mathbf{R}_1 \, \Gamma \, \mathbf{R}_0 \, , \tag{3.3}$$

where $\mathbf{R}_1$ is a radius vector of the point O_1 of center of object mass relative to the Earth's center of mass O; $\mathbf{R}_0$ is a radius vector of the center of mass of the Earth relative to the origin O_0 of inertial coordinate system.

Substituting expression (3.3) in equation (3.2), we obtain:

$$\ddot{\mathbf{R}}_1 \, X \mathbf{A} \, \Gamma \mathop{}\limits_{iX0}^{k} \mathbf{g}_{0i}(\mathbf{R}_i) \, Z \ddot{\mathbf{R}}_0 \, . \tag{3.4}$$

In the case when the motion of the object is near the Earth, i.e. the distance R_1 from the center of the Earth to the subject in many times smaller than the distance from the center of the Earth to the other celestial bodies ($R_1 << R_i$), then difference of accelerations of gravitational forces generated by *i*-th celestial body at the center of the Earth and at the center of object mass becomes neglible small compared with the acceleration of gravity $\mathbf{g}_0(\mathbf{R})$ of Earth's gravitational field. Taking into account these considerations and assuming that the origin O_0 of inertial coordinate system coincides with the center of mass of the Earth (R_0 = 0; R_1 = R), equation (3.4) becomes:

$$\ddot{\mathbf{R}} \, X \mathbf{A} \, \Gamma \, \mathbf{g}_0(\mathbf{R}) \, . \tag{3.5}$$

Vector **R** completely characterizes the current location of the object in a fixed inertial coordinate system and is determined in INS by double integration of the differential equation (3.5). However, the coordinates of the object are usually calculated in one of the navigation coordinate systems, connected with a rotating Earth, for example, in geographic (geodetic) coordinate system.

The absolute speed of the object in the inertial coordinate system in vector form can be written as

$$\dot{\mathbf{R}} = \mathbf{V}_r + \, t \, | \, \mathbf{R}, \tag{3.6}$$

where $\mathbf{V}_r$ is a vector of object relative velocity (velocity relative to the ground surface); t is a vector of angular velocity of the Earth's rotation; **R** is radius vector of object in inertial coordinate system relative to the center of the Earth; $t \, | \, \mathbf{R}$ is linear peripheral speed of the object, which is caused by the Earth's rotation.

Let's take the derivatives from the left and right parts of (3.6). As a result of differentiation we obtain the value of the absolute (full) acceleration

$$\ddot{\mathbf{R}} \, X \mathbf{w} \, X \dot{\mathbf{V}}_r \, \Gamma \, \mathbf{\Omega} \, | \, \dot{\mathbf{R}} \, . \tag{3.7}$$

Given the rotation vector $\mathbf{V}_r$ with absolute angular velocity $\mu_a = \mu + t$, caused by angular velocity μ, which occurs when flying around spherical surface of the Earth, which in turn rotates with angular velocity t.The derivative of vector of relative velocity $\mathbf{V}_r$, using the theorem of the derivative of vector in rotating coordinate system, can be represented in the following form:

$$\dot{\mathbf{V}}_\mathbf{r} \, X ƒ\dot{\mathbf{V}}_\mathbf{r} A_0 \, \Gamma \, \boldsymbol{\omega}_\mathbf{a} \, | \, \mathbf{V}_\mathbf{r} \, X ƒ\dot{\mathbf{V}}_\mathbf{r} A_0 \, \Gamma \, (\boldsymbol{\omega} \, \Gamma \, \mathbf{\Omega}) \, | \, \mathbf{V}_\mathbf{r} \, , \tag{3.8}$$

where $ƒ\dot{\mathbf{V}}_r A_0$ is the derivative of velocity in the earth coordinate system (acceleration relative to the Earth).

Substituting (3.6) and (3.8) to (3.7), we obtain

$$\begin{aligned} \mathbf{w} \, X ƒ\dot{\mathbf{V}}_\mathbf{r} A_0 \, \Gamma \, (\boldsymbol{\omega} \, \Gamma \, \mathbf{\Omega}) \, | \, \mathbf{V}_\mathbf{r} \, \Gamma \, \mathbf{\Omega} \, | \, \mathbf{V}_\mathbf{r} \, \Gamma \, \mathbf{\Omega} \, | \, (\mathbf{\Omega} \, | \, \mathbf{R}) \, X \\ X ƒ\dot{\mathbf{V}}_\mathbf{r} A_0 \, \Gamma \, (\boldsymbol{\omega} \, \Gamma \, 2\mathbf{\Omega}) \, | \, \mathbf{V}_\mathbf{r} \, \Gamma \, \mathbf{\Omega} \, | \, (\mathbf{\Omega} \, | \, \mathbf{R}). \end{aligned} \tag{3.9}$$

Vector sum in the right part is respectively called relative, Cariolis and transferring accelerations.

Taking into account (3.5), i.e., $\mathbf{w} = \ddot{\mathbf{R}} = \mathbf{A} + \mathbf{g}_0(\mathbf{R})$ the imaginary acceleration $\mathbf{A}$ of the center of object mass measured by accelerometer can be written using (3.9) as

$$\mathbf{A} = (\dot{\mathbf{V}}_\mathbf{r})_0 + (\boldsymbol{\omega} + 2\boldsymbol{\Omega}) \times \mathbf{V}_\mathbf{r} + \boldsymbol{\Omega} \times (\boldsymbol{\Omega} \times \mathbf{R}) - \mathbf{g}_0(\mathbf{R}). \qquad (3.10)$$

Typically, transferring acceleration caused by Earth's rotation, is summed up with the acceleration of the Earth's gravity force (Fig. 3.2) to form the gravity acceleration

$$\mathbf{g} = \mathbf{g}_0(\mathbf{R}) - \boldsymbol{\Omega} \times (\boldsymbol{\Omega} \times \mathbf{R}). \qquad (3.11)$$

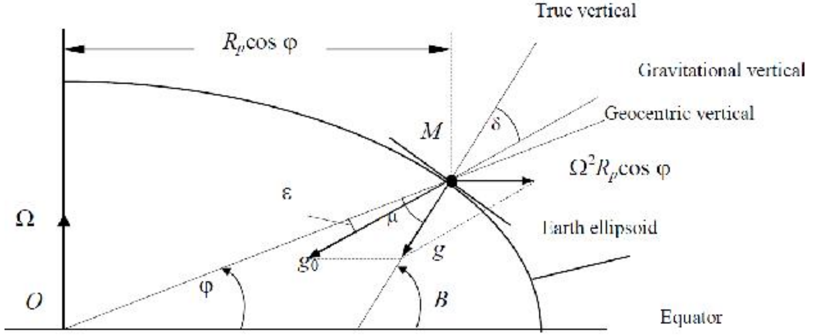


Fig. 3.2

Then taking into account (3.11) equation (3.10) for the imaginary acceleration of center of object mass, measured by accelerometer, takes the form

$$\mathbf{A} = (\dot{\mathbf{V}}_\mathbf{r})_0 + (\boldsymbol{\omega} + 2\boldsymbol{\Omega}) \times \mathbf{V}_\mathbf{r} - \mathbf{g}. \qquad (3.12)$$

It is seen from Fig. 3.2 that transferring acceleration in the point *M*, caused by the rotation of the Earth, is a centrifugal acceleration directed by normal from rotation axis of the Earth

$$f_c = \vartheta^2 R_p \cos\varphi,$$

where ϑ = 15.04107 deg/h is the angular velocity of the Earth; R_p is geocentric distance *OM* (radius of parallel); φ is geocentric latitude of point *M*.

The direction of the acceleration vector of gravity field $\mathbf{g}_0$ coincides with the direction of the gravitational vertical, which differs from the geocentric vertical (the direction to the center of the Earth) at angle

$$\delta = \varepsilon - \Omega,$$

where $\Omega \approx \dfrac{f_c}{g} = \dfrac{\vartheta^2 R_p \cos\varphi \sin\varphi}{g}$, and ε is an angle that describes the difference between geographical latitude B and geocentric latitude φ

$$\varepsilon = B - \varphi \approx 11.5' \sin 2\varphi.$$

Vector of gravity acceleration $\mathbf{g}$, as the resultant of centrifugal force f_c and the force of gravity field $\mathbf{g}_0$, sets the direction of true vertical, which is often chosen as the direction of the vertical axis of geotopic navigational coordinate systems used in inertial navigation systems.

As an example, let's choose the navigation coordinate system of the right-handed rectangular geotopic coordinate system OLRF (see Fig. 3.3). Axes OL, OF of this coordinate system lie in the horizon plane, but axis OR coincides with the local vertical.

Projecting the vector equation (3.12) on the axes of this coordinate system, we obtain readings of three orthogonal accelerometers oriented by axes OL, OR, OF:

$$\begin{aligned} a_L &= \dot{V}_L + V_R \omega_{F_\phi} - V_F \omega_{R_\phi} - g_L; \\ a_R &= \dot{V}_R + V_F \omega_{L_\phi} - V_L \omega_{F_\phi} - g_R; \qquad (3.13) \\ a_F &= \dot{V}_F + V_L \omega_{R_\phi} - V_R \omega_{L_\phi} - g_F, \end{aligned}$$

where $\omega_{F_\phi} = \omega_{F_V} + 2\vartheta_F; \omega_{R_\phi} = \omega_{R_V} + 2\vartheta_R; \omega_{L_\phi} = \omega_{L_V} + 2\vartheta_L$.

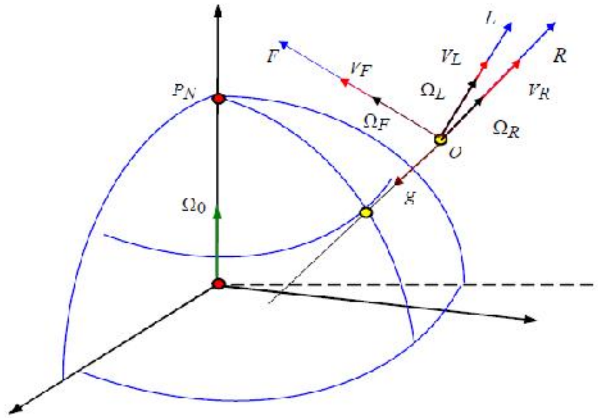


Fig. 3.3

Here $\in_{F_V}, \in_{R_V}, \in_{L_V}$ are projections of angular velocity of rotation of navigation coordinate system OLRF that occurs when flying above spherical surface of the Earth, and calculated by the speed of the aircraft, by the radii of curvature of the terrestrial spheroid and by direction of aircraft flight relative to terrestrial spheroid; $\vartheta_L, \vartheta_R, \vartheta_F$ are projections of the angular velocity of the Earth's rotation ϑ_0 on axes of navigation coordinate system OLRF which depend on latitude of aircraft location and on azimuth orientation of coordinate system OLRF.

In order to get results by integrating signals of accelerometers which measure imaginary acceleration of center of object mass, and to get the value of the Earth's velocity vector, it is necessary to subtract the components of Cariolis acceleration and the acceleration of gravity from readings of accelerometers (3.13). Then the Earth's velocity vector can be obtained by integrating equations

$$\begin{aligned} &\dot{V}_L \,\mathrm{X} a_L \,\mathrm{Z}(V_F \in_{R_\phi} \,\mathrm{Z} V_R \in_{F_\phi}) \,\Gamma\, g_L; \\ &\dot{V}_R \,\mathrm{X} a_R \,\mathrm{Z}(V_L \in_{F_\phi} \,\mathrm{Z} V_F \in_{L_\phi}) \,\Gamma\, g_R; \\ &\dot{V}_F \,\mathrm{X} a_F \,\mathrm{Z}(V_R \in_{L_\phi} \,\mathrm{Z} V_L \in_{R_\phi}) \,\Gamma\, g_F. \end{aligned} \tag{3.14}$$

By information about the components of the velocity vector of the Earth and the known coordinates of the starting point it is possible to solve the problem of dead reckoning of current coordinates of the aircraft. To determine geodetic (geographical) coordinates of the aircraft by dead reckoning it is necessary to take into account some geometric factors.

3.3. Platform inertial navigation systems

Platform inertial navigation systems provide the determination of flight and navigation parameters based on measurements of aircraft acceleration arising from the movement of bodies in gravitational field. Measurement of accelerations is done by using accelerometers located on the platform, which is kept in a certain position relative to inertial CS. Algorithms of determination of the flight and navigation information depend on the method of gyro platform orientation in particular INS.

Among existing platform INS the most widely used systems are horizontal platforms with great circle orientation and with free in azimuth orientation of their axes. Let us consider the principles of INS structure.

3.3.1. Principles of platform inertial navigation systems

The principle of action of platform INS is considered on the example of single-component INS. Let us make the following assumptions: aircraft moves only in one plane and on a constant distance *R* from the center of the Earth; strength vector of gravitational field is directed toward the center of the Earth. Coriolis accelerations caused by the Earth rotation and movement of aircraft relatively the Earth, are not taken into account.

Starting motion from the point O_1 (Fig. 3.4), aircraft moves to the point *O*, and the local vertical rotates at angle

$$\Im \,\mathrm{X} \frac{S}{R},$$

where *S* is the travelled distance.

The angular velocity of moving trihedral *XYZ*, with the axis *OX* to be tangent to the trajectory, and the axis *OZ* to be directed by local gravitational vertical, is determined by the ratio:

$$\dot{\Im} \,\mathrm{X} \in_y \mathrm{X} \frac{V}{R},$$

where *V* is flight speed of aircraft. Moreover,

$$\dot{S} \,\mathrm{X} V, \quad \dot{V} \,\mathrm{X} w,$$

or

$$S \,\mathrm{X} S_0 \,\Gamma {}^{t}_{0} V dt; \quad V \,\mathrm{X} V_0 \,\Gamma {}^{t}_{0} w dt,$$

where *w* is the absolute acceleration of the center of aircraft mass.

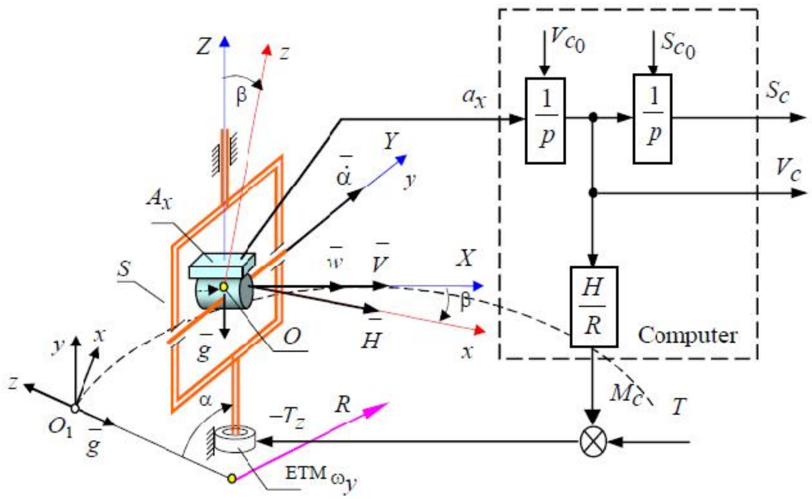


Fig. 3.4

Simulation of moving trihedral onboard is provided by gyroplatform, on which the accelerometer *Ax* is installed and which axes form the instrument trihedral *xyz*. Along axis *Ox* of instrument trihedral the sensitivity axis of accelerometer and gyro angular momentum vector $\bar{H}$ are directed. At the point O_1 of motion start the axes of instrument trihedral *xyz* with maximal accuracy are aligned with axes *XYZ* of moving trihedral. Also, initial values V_{c0} and S_{c0} are given for integrator inputs in computer. The values of these signals correspond to the initial value V_0 of aircraft speed and its coordinates S_0. Computer by the accelerometer signals continuously calculates the values of speed V_c and travelled distance S_c.

To keep gyroplatform always in horizon, it is necessary to create the precession rate of platform around axis *Oy*, i.e. to control the erection-torque motor (ETM) by speed $\in_y$ of gyro stabilizer in order to rotate the platform around axis *Oy* with a rate equal to angular velocity of aircraft motion relatively the Earth

$$\in_{yc} \mathrm{X} \Im \mathrm{X} \frac{V_c}{R}. \tag{3.15}$$

According to the precession rule the angular velocity of gyro $\in_y$ with the angular momentum *H* under the action of external torque T_z is determined by ratio:

$$\in_y \mathrm{XZ} \frac{T_z}{H}. \tag{3.16}$$

Comparing equations (3.15) and (3.16), it is possible to find the value of erection torque moment:

$$M_c \mathrm{XZ} T_z \mathrm{X} K V_c,$$

where

$$K \mathrm{X} \frac{H}{R}. \tag{3.17}$$

Expression (3.17) is frequently given in the form of $fK / H \,\mathrm{X}\, 1/ R$Ą and is called the condition of ballistic undisturbance. Fulfillment of this condition provides leveling of platform at aircraft motion with arbitrary acceleration.

Readings of accelerometer mounted on gyroplatform have the form:

$$a_x = w_x - g_x,$$

where w_x, g_x are correspondingly projections of absolute acceleration of the aircraft mass center and of acceleration of gravity force on sensitivity axes of accelerometer.

Since the gyroplatform always remains in the horizon plane, the influence of the gravity acceleration on accelerometer readings is eliminated, i.e. if $\wp = 0$ ($\wp$ is leveling error of platform), then g_x is also zero. In this case, there will be:

$$a_x = w_x = w. \qquad (3.18)$$

From equation (3.18) the necessity of continuous leveling of accelerometer sensitivity axis is seen. The leveling allows defining the problems solved by INS during the motion of aircraft:

- calculation of the kinematic parameters of movement of the aircraft center of mass by integrating (in this case they are speed V_c and traveled path S_c;

- continuous simulation of axes of moving trihedral using gyro stabilizer, i.e. simulation of vertical on board the aircraft (leveling of platform).

The algorithm of INS computer operation can be represented by such a set of equations:

$$\begin{aligned} V_c &= V_{c_0} + \int_0^t a_x dt; \\ S_c &= S_{c_0} + \int_0^t V_c dt; \\ M_c &= -T_{z_c} = \frac{H}{R} V_c. \end{aligned} \qquad (3.19)$$

Instead of the last equation of set (3.19), that is, equation for calculation of the motor erection torque of gyro stabilizer, it is possible to write equation for calculating the angular velocity of gyroplatform:

$$\in_c = \frac{T_{z_c}}{H} = \frac{V_c}{R}.$$

The considered algorithm of INS operation assumes ideal operation of gyrostabilizer and accelerometer with the erection torque motor system. However, in practice there is a difference $\dot{\wp}$ between the angular velocity of moving trihedral $\Im$ and precession rate of gyro $\in_p$ because of the presence of gyroscope drift $\in_d$ or of torque T (see Fig. 3.4), causing this drift:

$$\dot{\wp} = \in_p - \Im,$$

where $\in_p = \in_c + \in_d$.

For this reason, and also because of inaccuracy of initial alignment of vertical (initial deviation $\wp_0$ of gyroscope) the current error of leveling arises

$$\wp = \wp_0 + \int_0^t \dot{\wp} dt.$$

Uncompensated components of accelerometer errors can be characterized by component ζa_x. Because of errors of accelerometer, of gyro stabilizer and also because of initial inaccurate alignment of gyroplatform, the INS operation will be characterized by errors in speed ζV, and coordinates ζS, and by error of vertical simulation (error of leveling) $\wp$ Output accelerometer signal may be represented as:

$$a_x = w \cos \wp - g \sin \wp + \zeta a_x. \qquad (3.20)$$

At small angles $\wp$ the equation (3.20) can be written as

$$a_x = w - g\wp + \zeta a_x.$$

The considered principle of single-component INS operation can be represented by block diagram (Fig. 3.5).

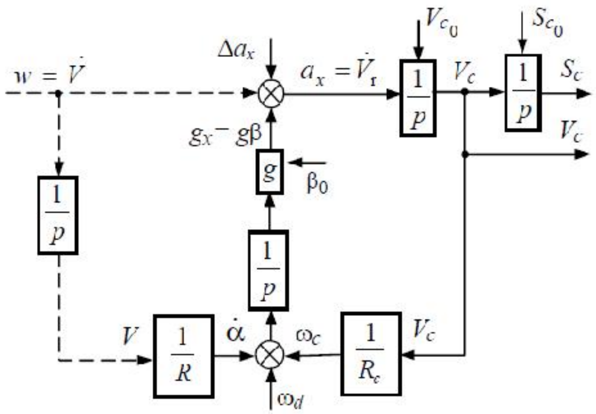


Fig. 3.5

Assuming the precession theory, the gyro platform will be represented by integrator link at block diagram. Input signals of the scheme are motion parameters of moving trihedral *XYZ* as an absolute acceleration *w* of center of aircraft mass as well as the angular velocity $\Im$ of rotation of local vertical. Other elements of the scheme correspond to elements of single-component INS (see Fig. 3.4).

By values of absolute acceleration $a_x \mathrm{X} \dot{V}_c$ measured by accelerometers the flight speed V_c and traveled path S_c are calculated. Calculated speed V_c is then used to calculate the required rate of gyroscope precession $\in_c$ that is equal to the angular velocity of aircraft motion relatively to the Earth. Signals ζa_x and $\in_d$ are the most significant sources of INS errors. Because of the current leveling error $\wp$ in the accelerometers readings the component $g_x \mathrm{X} g \wp$ appears, which in turn causes the error in calculation V_c. Leveling circuit (circuit of integral correction of horizontal position of platform) is a circuit with negative feedback. The presence of two integrator links indicates the structure instability of such circuit. With disturbance action, for example, in the form of errors a_x or $\in_d$ the circuit is excited to undamped oscillations with eigen frequency $\sqrt{g(R)^{Z1}}$.

In simplified statement the principle of platform INS operation is shown in Fig. 3.6. Let the platform with accelerometer is placed at point *A*. With motion relatively the Earth ($\Im$ is angular, *S* is linear displacement) the platform is kept in horizontal position with the help of gyroscope, which simulates the vertical and rotates the platform at angle $\uparrow = \Im$. Since the process of vertical simulation must be continuous, then instead of $\uparrow = \Im$ it is better to write:

$$\uparrow \mathrm{X} \Im\ \mathrm{X} \dot{S}/R,$$

where *R* is radius of the Earth; $\uparrow$ is rate of platform precession (corresponds to $\in_{y_c}$ in more detailed description of INS principle).

The value $\dot{s}$ is determined through the acceleration *a*, measured by accelerometer like $\dot{S}\,\mathrm{X}\,{}_{0}^{t}\, a\,dt,$, then $\uparrow \mathrm{X} \frac{1}{R}\,{}_{0}^{t}\, a\,dt.$ As a result the vertical is constructed, and the ground speed and travelled distance are determined as $V_g\ \mathrm{X} \dot{S}\ \mathrm{X}\,{}_{0}^{t}\, a\,dt,\ \ S\ \mathrm{X}\,{}_{0}^{t}\, V_g\,dt\ \mathrm{X}\,{}_{0}^{t}\,{}_{0}^{t}\, a\,dt\ \ dt\,.$

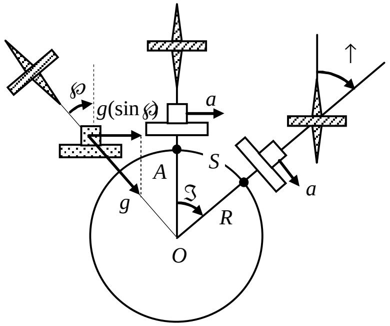


Fig. 3.6

The considered system has the properties of a pendulum. If at some point (Fig. 3.6) the normal to the platform deviates from the vertical at a small angle $\wp(\wp$is leveling error), then the accelerometer will measure the component of acceleration $a\,XZg\sin\wp$

Since the angle $\wp$ is small, then $\sin\wp|\,\wp$ and $a\,XZg\wp$and

$$\dot{\wp}X\frac{1}{R}{}_{0}^{t}\,a\,dt\,XZ\frac{g}{R}{}_{0}^{t}\,\wp dt. \quad (3.21)$$

By differentiating equation (3.21), there will be:

$$\ddot{\wp}\Gamma\frac{g}{R}\wp X0. \quad (3.22)$$

Equation (3.22) is the equation of mathematical pendulum relatively the true vertical in aircraft position, or so called Schuler's pendulum with oscillation period $T\,X2\Leftrightarrow\sqrt{R(g)^{Z1}}$.

Thus, the leveling circuit simulates so called Schuler's pendulum. If R = 0000006371000 m and g = 9.81 m/s^2, then the period of oscillations is 84.4 minutes. Schuler's pendulum is a pendulum with suspension length, equaled the radius of the Earth, and it is undisturbed from acceleration of suspension point. Another name for such a system is gyro-vertical with integral erection.

In the absence of sensor errors and errors of INS computer the input signal $\Im\,XwR^{Z1}/p$ on the bottom adder (see Fig. 3.5) is fully compensated by signal $\in_c Xa_xR_c^{Z1}/p$ if there is full equality $R_c\,XR$ which confirms the idea of ballistic undisturbance.

Figure 3.7 shows how the change in the length of the pendulum suspension affects the angle of deflection of the suspension under the action of the unit acceleration, which shifts the suspension point, but due to the inertia the mass center of the pendulum load does not change the position. Of course, only with the length of the suspension, equaled the distance to the center of the Earth, the pendulum remains undisturbed, that the angle of suspension deviation will be zero.

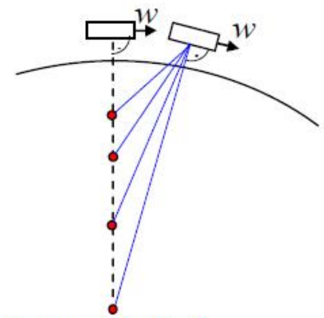


Fig. 3.7

Schuler's pendulum has the following property. Its displacement in space every second is equal to an arc, which is a shift of its suspension point along the surface of the Earth, that is, the pendulum always rotates around the suspension point at the same angle as the vertical does. This ideal sensor does not operate like a normal pendulum. It stays in equilibrium, despite of any movement of suspension point.

Structural analysis of the circuit (see Fig. 3.5) shows that constant error of accelerometer ζa_x causes the error $\wp$of vertical simulation and in the presence of constant platform drift $\in_d$, also because of gyroscope drift due to the diurnal rotation of the Earth, the vertical is reproduced without constant error, but there is a periodic error. Typical errors changes of vertical reproduction in the presence of a constant vertical accelerometer error and gyro platform drift are shown in Fig. 3.8.

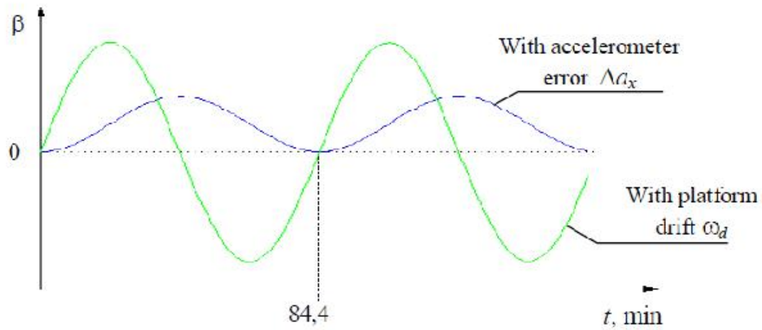


Fig. 3.8

Moreover, INS errors depend on the alignment error $\wp\theta$ of platform in horizon, on inaccurate initial values V_0 and S_0.

Thus, the following conclusion is made. Every time when readings of INS are affected by gravity acceleration g_x, this effect causes the error of INS. On the other hand, in INS algorithms this component plays very useful role, because thanks to this, INS errors remain limited.

3.3.2. Operating modes of inertial navigation systems

The considered algorithm of single-component INS practically is used with no changes in systems with horizontal, free in azimuth platform systems such as И-11. Unlike the single-component system, system И-11 uses three-axial gyrostabilized platform of indicated type with four frames (Fig. 3.9) on three-level float gyros of ГПА-20 type. In gyro stabilization systems of indicated type the gyroscopes are used as selectors of angular orientation of the platform. The given orientation of platform is provided by force servo systems.

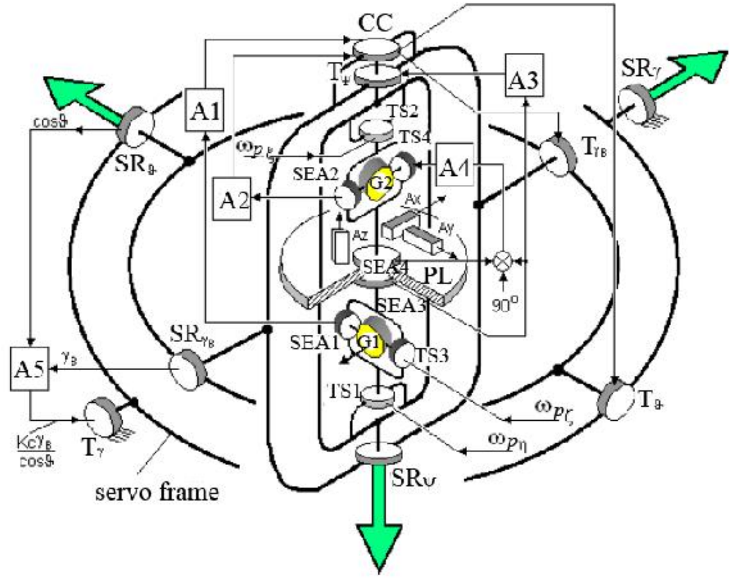


Fig. 3.9

The platform (PL) is oriented by the axes of gyroscope G1 and G2 by means of three servo systems. By heading the platform follows the axis of the outer frame of gyroscope G1 via servo system that includes sensor of error angle SEA3, amplifier A3 and gyro torquer $T_э$. The axis of the outer frame gyroscope G2 follows the axis of gyroscope G1 at outer frame using servo system, which consists of sensors SEA3 and SEA4, amplifier A4 and gyro torquer T4. According to the scheme (see Fig. 3.9) the signal at the output of amplifier A4 is:

$$(э_p \, Z э_{g1}) \, Γ \, (э_{g2} \, Z э_p) \, Γ \, 90^\circ \; X 90^\circ \, Γ \, (э_{g2} \, Z э_{g1}),$$

where $э_p$, $э_{g1}$, $э_{g2}$ are azimuths of platform, of the first and the second gyroscopes, respectively. Thus, the signal at the output of A4 equals the angle of deviation of external frames of gyroscopes G1 and G2 from orthogonal position, and the servo system provides orthogonality of planes of external gyroscope frames.

Two servo systems, or single two-dimensional servo system, include sensors of error angles SEA1 and SEA2, amplifiers A1 and A2, coordinate converter (CC) and gyro torquers Tν and T↑. They provide the parallelism of platform plane and axes of gyroscopes G1 and G2.

Since in three-axial platform the gyro torquers Tν and T↑ are rotated relatively the platform in variable angle э, then coordinate converter connected to the axis э, distributes signals for servo system of pitch and roll angles depending on the value of э.

The gyro stabilized platform is not tumbled because it has an extra servo frame that allows it to save axes of gimbal mount to be perpendicular with any aircraft maneuvers, i.e. with unlimited changes in roll and pitch. To ensure the constant amplification factor in control circuit the amplification factor of A5 is inversely proportional to cosν.

On the gyro stabilization platform there are three accelerometers Ax, Ay, Az of ДА-l type with axes of sensitivity which are oriented in three mutually perpendicular directions and create instrument CS.

Information about angular position is taken from the synchro resolvers (SR) with two levels of angles readout (coarse and precise): roll angle γ is taken from SRγ;

pitch angle ν is taken from SR_{ν}, and gyroscopic heading ψ is taken from SR_{ψ}.

By special erection torques which are created by torque sensors TS1, TS2, and by signals $\in_{p_{\notin}}$, $\in_{p_{\rightarrow}}$, the platform is stabilized relative to the local vertical, and with applying the signal $\in_{p_{\prod}}$on torque sensor TS3 the platform is properly orientated in horizontal plane relative to the local meridian.

High accuracy of INS is achieved by forming $\in_{p_{\notin}}, \in_{p_{\rightarrow}}, \in_{p_{\prod}}$, by the signals of apparent acceleration measured by accelerometers, with the excluding not only components of the gravity acceleration, but Coriolis acceleration too.

Two horizontal channels of INS are tuned on Schuler's pendulum period, which is undisturbed by accelerations of suspension point. For adjusting the length of pendulum suspension, which must be equal to the radius of the Earth, the ellipsoid (not spherical) model of the Earth is used.

The influence of the gravity acceleration is eliminated by continuous leveling of platform, and hence of sensitivity axes of accelerometers by dead reckoning data. In this case, the horizontal accelerometers fix only the relative accelerations and Coriolis accelerations. The latter ones are compensated analytically.

To solve navigation problems, two CS are used: navigation CS O∉→∏and CS O_1XYZ (Fig. 3.10):

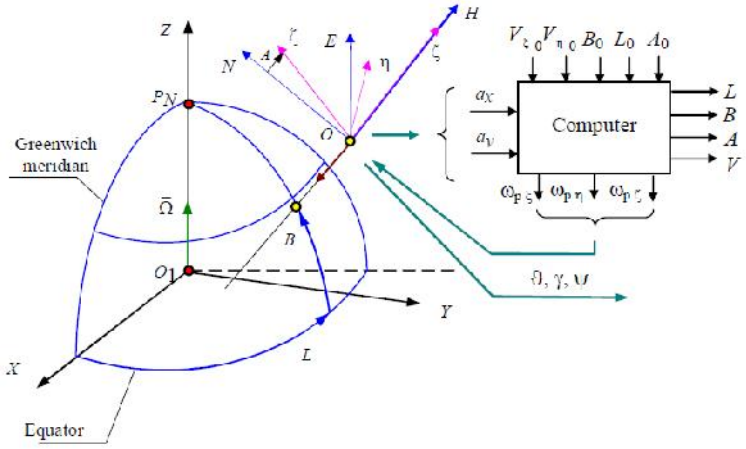


Fig. 3.10

The first is navigating the CS O∉→∏(moving trihedron). Centre of trihedron is point O, placed in the point of aircraft position. Axis O∏is directed vertically upwards by local vertical, axes O∉ and O→are in the horizontal plane and free in azimuth relative to movement of point O. The azimuth orientation of trihedron is determined by its azimuth angle *A*, that lies in the horizon plane and is measured clockwise from the northern direction *ON*. At the moment of start the axis O∉ is aligned with the longitudinal axis of aircraft and its azimuth angle is determined during the initial alignment of gyroplatform(so called gyrocompassing).

The second CS O_1XYZ is associated with the Earth. Axis O_1Z is directed by angular velocity vector of the Earth rotation t, axis O_1X is located at the intersection of Greenwich meridian plane with the plane of equator, axis O_1Y is directed in order to create right-handed Cartesian (rectangular) CS.

By the way, the instrument CS *Oxyz*, created by sensitivity axes of accelerometers Ax, Ay, Az (see Fig. 3.9) installed on the gyro stabilizer of platform, simulates the moving trihedron $O\notin\to\Pi$

Let us consider the operation of INS with the platform free in azimuth relative to the relative motion of the aircraft around the Earth, but coupled with the Earth in diurnal motion, that is, its angular velocity in azimuth equals the angular velocity of the Earth for local vertical. Azimuth gyro G1 of platform for such gyro stabilizer (see Fig. 3.9) is erected by torque sensor TS3 order to provide that its absolute vertical angular velocity $\in_{\Pi}$ will be $\in_{\Pi} X \vartheta_{\Pi} X \vartheta_0$ where B is geodetic latitude (see Fig. 3.10).

The navigation parameters are determined as follows. In the first stage, the derivatives of components of ground speed ($V_{\notin}$, $V_{\to}$) are found by subtracting the Coriolis components from the signals of apparent acceleration, measured by accelerometers

$$\dot V_{\notin} X a_{\notin} Z(\in_{\to} \Gamma 2\vartheta_{\to})\dot H \Gamma 2\vartheta_{\Pi} V_{\to};$$

$$\dot V_{\to} X a_{\to} \Gamma (\in_{\notin} \Gamma 2\vartheta_{\notin})\dot H Z 2\vartheta_{\Pi} V_{\notin},$$

where $a_{\notin} = a_x$, $a_{\to} = a_y$ are respectively signals of accelerometers Ax, Ay (see Fig. 3.9); $\dot H$ is flight height derivative; $\in_{\notin} E \in_{\to}$ are angular speed components of moving trihedron caused by flight over ellipsoidal surface of the Earth; $\vartheta_{\notin} E \vartheta_{\Pi} E \vartheta_{\to}$ are angular velocity components of the Earth rotation. Note that for the chosen method of azimuth erection, the angular velocity $\in_{\Pi}$ of rotation $O\notin\to\Pi$ around the Earth will be zero.

At the second stage, there is compensation of instrument errors in measured accelerations, and the procedure of integration is done (with initial values $V_{\notin 0}$, $V_{\to 0}$) to obtain components of ground speed. The corrections introduced at the stage of compensation of instrumental errors are obtained at the stage of pre-calibration of gyroplatform with mounted accelerometers.

Next, the components of the absolute angular velocity $\bar\in_p$ are determined. With this speed the gyroplatform must rotate relatively the inertial space. The components consist of the components $\bar\in_{\notin\to\Pi}$ caused by linear motion of aircraft relative to the Earth, and of components $\bar\vartheta_0$ caused by angular speed of the Earth rotation.

$$\bar\in_p X \bar\in_{\notin\to\Pi} \Gamma \bar\vartheta_0 .$$

In projections on the axes of navigation CS $O\notin\to\Pi$ we will get:

$$\in_{p_{\notin}} X \in_{\notin} \Gamma \vartheta_{\notin};$$

$$\in_{p_{\to}} X \in_{\to} \Gamma \vartheta_{\to};$$

$$\in_{p_{\Pi}} X \vartheta_{\Pi}$$

Projections of the angular velocity of the Earth rotation on the axes of navigation CS $O\notin\to\Pi$ are the following:

$$\vartheta_{\notin} \, \mathrm{XZ} \vartheta_0 \cos B \sin A;$$
$$\vartheta_{\rightarrow} \mathrm{X} \vartheta_0 \cos B \cos A; \qquad (3.23)$$
$$\vartheta_{\Pi} \mathrm{X} \vartheta_0 \sin B.$$

The projections of the angular velocity $\in_{\notin}, \in_{\rightarrow}$ of rotation of navigation CS $O\notin\rightarrow\Pi$ caused by the flight over ellipsoidal surface of the Earth, equal:

$$\in_{\notin} \mathrm{XZ}\frac{V_N}{R_m}\cos A \, \mathrm{Z}\frac{V_E}{R_p}\sin A; \quad \in_{\rightarrow}\mathrm{XZ}\frac{V_N}{R_m}\sin A \, \Gamma \frac{V_E}{R_p}\cos A,$$

where V_N, V_E are northern and eastern projections of ground speed (projections on the axes of CS *ONEH* (see Fig. 3.10); A is azimuth of moving trihedron, R_p, R_m are two radiuses of curvature of the Earth spheroid (ellipsoid of revolution).

Radiuses are usually calculated by the formulas:

$$R_m \, \mathrm{X} \frac{a \left(1 \, \mathrm{Z} e^2\right)}{\left(1 \, \mathrm{Z} e^2 \sin^2 B\right)^{\frac{3}{2}}} \Gamma H;$$

$$R_p \, \mathrm{X} \frac{a \cos B}{\sqrt{1 \, \mathrm{Z} e^2 \sin^2 B}} \Gamma H \cos B,$$

where e is eccentricity of the ellipsoid (e_2 = 6.73 10^{-3}), a is major semiaxis of ellipsoid (a = 6378388 meters); H is altitude of flight.

Northern and eastern projections of ground speed are taken by the components of ground speed $V_{\notin}$, $V_{\rightarrow}$ in navigation (instrumental) CS $O\notin\rightarrow\Pi$

$$V_N \, \mathrm{XZ} V_{\notin} \sin A \, \Gamma V_{\rightarrow} \cos A;$$
$$V_E \, \mathrm{X} V_{\notin} \cos A \, \Gamma V_{\rightarrow} \sin A.$$

The obtained values of the absolute angular velocity $\in_{p_{\notin}}$, $\in_{p_{\rightarrow}}$, $\in_{p_{\Pi}}$ of gyroplatform are compensated to avoid instrumental errors of gyroscopes, and then they are applied as control signals to torque sensors TS1, TS2, TS3 of gyroscopes G1 and G2 (see Fig. 3.9). By signals $\in_{p_{\rightarrow}}$, $\in_{p_{\notin}}$ two circuits of integral correction of platform horizontal position are realized with accelerometers. Signal $\in_{p_{\Pi}}$ provides free orientation of platform in azimuth with respect to the relative motion of aircraft around the Earth.

By the northern and eastern projections of the ground speed the coordinates of aircraft are calculated: geodetic latitude B and geodetic longitude L. Also there is determination of change in the platform azimuth A with known initial values of the coordinates L_0, B_0 and with defined in the initial alignment the azimuth angle A_0:

$$\dot{L} \, \mathrm{X} \frac{V_E}{R_p};$$
$$\dot{B} \, \mathrm{X} \frac{V_N}{R_m};$$
$$\dot{A} \, \mathrm{X} \frac{V_N}{R_m} \mathrm{tg} B.$$

This algorithm of dead reckoning is quite accurate, but it has a significant disadvantage. In the polar regions, where the latitude B approaches to $\{\Leftrightarrow 2$, the algorithm becomes computationally unstable. That is why for navigation systems that can be used in the polar regions, the all-latitude algorithm of dead reckoning is applied, free from this drawback, which is based on so-called Poisson equations

If in CS O_1XYZ to fix the unmoved vector $\bar{r}\,X(X,Y,Z)$, then the projection of this vector on the axes of CS $O\notin\rightarrow\Pi$ will be equal to

$$\begin{vmatrix}\notin\\ \rightarrow\\ \Pi\end{vmatrix} X\mathbf{B}(t)\begin{vmatrix}X\\ Y\\ Z\end{vmatrix}, \qquad (3.24)$$

where $\mathbf{B}(t)$ is the variable direction cosine matrix (orthogonal matrix of transformation from the system O_1XYZ to the system $O\notin\rightarrow\Pi$).

After differentiating (3.4) the velocity of vector end $\bar{r}$ relative to the system $O\notin\rightarrow\Pi$ will be:

$$\begin{vmatrix}\dot{\notin}\\ \dot{\rightarrow}\\ \dot{\Pi}\end{vmatrix} X\dot{\mathbf{B}}(t)\begin{vmatrix}X\\ Y\\ Z\end{vmatrix}. \qquad (3.25)$$

On the other hand, the speed of the vector end $\bar{r}$ relative to the CS $O\notin\rightarrow\Pi$ will be equal to the value $(Z\in|\bar{r})$:

$$\begin{vmatrix}\dot{\notin}\\ \dot{\rightarrow}\\ \dot{\Pi}\end{vmatrix} XZ\in|\bar{r}\,XZ\begin{vmatrix}\notin^0 & \rightarrow^0 & \Pi^0\\ \in_{\notin} & \in_{\rightarrow} & \in_{\Pi}\\ \notin & \rightarrow & \Pi\end{vmatrix} X$$

$$XZ\begin{vmatrix}0 & Z\in_{\Pi} & \in_{\rightarrow}\\ \in_{\Pi} & 0 & Z\in_{\notin}\\ Z\in_{\rightarrow} & \in_{\notin} & 0\end{vmatrix}\,|\begin{vmatrix}\notin\\ \rightarrow\\ \Pi\end{vmatrix} XZt^{*}\mathbf{B}(t)\begin{vmatrix}X\\ Y\\ Z\end{vmatrix}, \qquad (3.26)$$

where $t^{*} X\begin{vmatrix}0 & Z\in_{\Pi} & \in_{\rightarrow}\\ \in_{\Pi} & 0 & Z\in_{\notin}\\ Z\in_{\rightarrow} & \in_{\notin} & 0\end{vmatrix}$; $\notin^0, \Pi^0, \rightarrow^0$ are the unit vectors.

Comparing (3.25) with (3.26), Poisson equation are obtained:

$$\dot{\mathbf{B}}(t)\,XZt^{*}\mathbf{B}(t)\ .$$

Orthogonal direction cosine matrix $\mathbf{B}(t)$ of the transformation (3.24):

$$\begin{vmatrix}\notin\\ \rightarrow\\ \Pi\end{vmatrix} X\begin{vmatrix}b_{11} & b_{12} & b_{13}\\ b_{21} & b_{22} & b_{23}\\ b_{31} & b_{32} & b_{33}\end{vmatrix}\begin{vmatrix}X\\ Y\\ Z\end{vmatrix}$$

for the system $O\notin\rightarrow\Pi$ if Π^0 is directed by geodetic vertical, will have the form:

$$\mathbf{B}\,X\begin{array}{c|c|c} Z\sin L\cos A\,\Gamma\sin B\cos L\sin A & \cos L\cos A\,\Gamma\sin B\sin L\sin A & Z\cos B\sin A\\ \hline Z\sin L\sin A\,Z\sin B\cos L\cos A & \cos L\sin A\,Z\sin B\sin L\cos A & \cos B\cos A\\ \hline \cos L\cos B & \sin L\cos B & \sin B\end{array}\ .$$

For the selected method of platform azimuth correction, the angular velocity $\in_{\Pi}$ of rotation $O\notin\rightarrow\Pi$ relative to the Earth is equal to zero, and the skew-symmetric matrix t^{*} has the form:

$$\mathbf{\Omega}^{*} X\begin{vmatrix}0 & 0 & \in_{\rightarrow}\\ 0 & 0 & Z\in_{\notin}\\ Z\in_{\rightarrow} & \in_{\notin} & 0\end{vmatrix}.$$

Hence, determining the current position of the navigation trihedron $O\notin\rightarrow\Pi$ can be obtained by solving the differential equation

$$\dot{\mathbf{B}}(t)\,XZ\mathbf{\Omega}^{*}\mathbf{B}(t).$$

If the current value of the matrix $\mathbf{B}$ is known then the projections of the angular velocity (3.18) of the Earth rotation on the axis of the navigation CS $O\notin\rightarrow\Pi$ are obtained in the form:

$$\omega_{\xi} = \omega_{0} b_{13};$$
$$\omega_{\eta} = \omega_{0} b_{23};$$
$$\omega_{\zeta} = \omega_{0} b_{33}.$$

Without converting to the northern and eastern projections of ground speed, the components of angular velocity ω_{ξ}, ω_{η} of the rotation of the navigation CS O$\xi\eta\zeta$ arising during the flyby of ellipsoidal surface of the Earth, are determined in the form:

$$\omega_{\xi} = -\frac{V_{\xi}}{R_2} - \frac{V_{\eta}}{R_{add}};$$

$$\omega_{\eta} = \frac{V_{\eta}}{R_1} - \frac{V_{\xi}}{R_{add}},$$

where R_1, R_2 are basic ones and R_{add} is additional radii of curvature of the terrestrial ellipsoid:

$$\frac{1}{R_1} = \frac{1}{a}\left(1 - \frac{e^2 b_{33}^2}{2} + e^2 b_{13}^2 - \frac{H}{a}\right);$$

$$\frac{1}{R_2} = \frac{1}{a}\left(1 - \frac{e^2 b_{33}^2}{2} + e^2 b_{23}^2 - \frac{H}{a}\right);$$

$$\frac{1}{R_{add}} = \frac{1}{a}\left(e^2 b_{13} b_{23}\right).$$

These calculating formulas of the radii of curvature are used, for example, in the inertial system И-11. Other formulas of algorithm are not changed.

Using elements of the direction cosine matrix the following parameters are determined:

- geodetic latitude $B = \operatorname{arctg}\dfrac{b_{33}}{\sqrt{b_{13}^2 + b_{23}^2}}$ in the range $\pm 90°$;
- geodetic longitude $L = \operatorname{arctg}\dfrac{b_{32}}{b_{31}}$ in the range $\pm 180°$;
- azimuth platform $A = \operatorname{arctg}\dfrac{b_{13}}{b_{23}}$ in the range $0\ldots360°$;
- true heading $\psi = \psi_g - A$ in the range $0\ldots360°$;
- drift angle $\beta_{dr} = \varphi - \psi$ in the range $0\ldots360°$;
- ground speed $V_g = \sqrt{V_{\xi}^2 + V_{\eta}^2}$.

If there is additional information from ADC system about the true airspeed then the projections of wind speed on the axes of gyroplatform, magnitude of the wind speed and wind angle can be calculated.

Furthermore, from synchro resolvers SR_{ϑ}, SR_{ψ}, SR_{γ} of gyroplatform (see Fig. 3.9) information about the aircraft attitude is calculated: pitch angle ϑ, gyroscopic heading ψ_g and roll angle γ.

Inertial navigation systems require special pre-flight procedures, for example, for platform INS the task of preparing is initial alignment of gyroplatform in horizontal position, alignment of the measurement axes of accelerometers with axes of navigation CS and the entry of data about the aircraft coordinates and speed of motion.

3.3.3. Alignment of inertial navigation system

The process of INS alignment in horizontal position can be divided into stages: accelerated, coarse and precise alignments.

In the process of accelerated alignment in the horizon and in azimuth the electric caging of gyroplatform is performed in the horizontal channels of the gyroplatform body or by the signals of accelerometers. In the azimuth the gyroplatform is caged with the body, or by compass corrector, or by heading selector.

At the end of accelerated alignment, the caging circuit is disabled and further alignment is done as the process of correction of the gyroplatform attitude.

At the stage of coarse alignment for correction of leveling errors the signals from the accelerometers, bypassing the integrator come to the erection torque motors of gyro stabilizer ($ETM_{\omega\gamma}$ in Fig. 3.4). But turning off the integral erection, despite of improving the dynamic performance and stability of the leveling contour, anyway will degrade its static characteristics. As a result, there is the constant error of vertical simulation, which is proportional to the angular rate of gyroplatform drift caused by the projection of angular velocity of the Earth rotation, and is inversely proportional to the gain factor of leveling circuit.

In the process of precise alignment on the erection torque motor of gyro is supplied by not only the amplified signal from the accelerometer, but also by the signal from this accelerometer, that optionally passes through an integrator

The absence of accelerometer errors means the absence of leveling errors. The similar conclusion has already been made during the analysis of block diagram in Fig. 3.5. Thus, the output signal of the integrator at the end of the transition process will eliminate the cause of leveling errors. This allows for gyroscopes, pre-balanced, or free from drifting, to form or to calculate the projection value of angular velocity of the Earth rotation for the corresponding axis of gyroplatform in the absence of information about the azimuth of the platform axis and in the absence of data about the latitude of alignment position. This method is widely used for initial alignment of operational integrators in INS.

Azimuth alignment of gyroplatform in case of using the compass signals compass (corrector or heading selector), as well as alignment by "INS body" has no difference from horizontal alignment. However, the valid signal is taken not from the accelerometer, but from the corresponding angular position sensor.

In the case when ramp heading is unknown or determined with insufficient accuracy, in some INS the autonomous azimuth alignment modes are used by using the methods of physical or computational gyrocompassing. The process of gyrocompassing means the fixing of gyroplatform axes with accelerometers to the direction of the gravitational field strength vector and to the vector of angular velocity of the Earth diurnal rotation.

For physical gyrocompassing the property of gyropendulum is used (gyropendulum is gyroscope on torsional suspension, that is widely used in geodesic works). Such gyropendulum is aligned with the direction of tangent to the local meridian. Let us suppose that in the given time moment the gyroopendulum takes the required position, that is, its major axis (the vector of angular momentum) is horizontal and directed to the north, and it coincides with the direction of horizontal component of angular velocity of the Earth rotation ϑ_{0h} (Fig. 3.11, a). In this case, the pendulum is placed vertically. However, through the diurnal rotation of the Earth the gyro axis changes its position relative to the ground surface (rises over the horizon and drifts to the east). As a result, the pendulum rigidly connected with gyromotor, deviates from the vertical at the angle $\wp$(Fig. 3.11, b). The weight force of pendulum will create the moment M_v relative to the axis of gyromotor suspension. Under the influence of this moment the gyro starts precessing around the axis *OZ* of the outer frame and simultaneously aligning with the horizon.

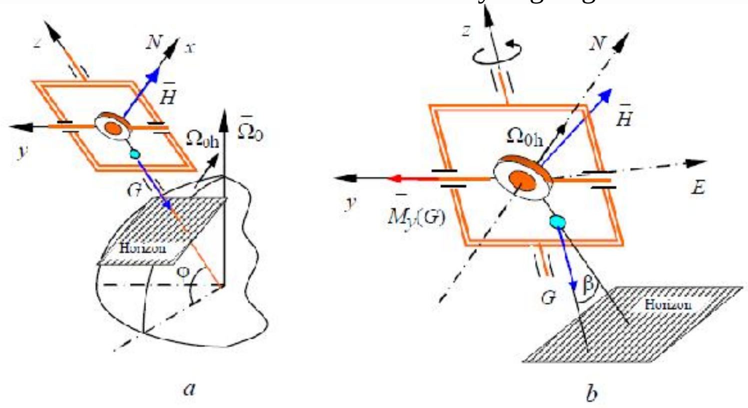


Fig. 3.11

Gyro precesses towards the elimination of drift of the gyro spin axis from the meridian plane. After the damping of all oscillations the spin axis returns both to the meridian plane and to the horizon plane. Because of small angular velocity of the Earth rotation $\vartheta_0 = 7.3\ 10^{Z5}s^{Z1}$, the process of physical gyrocompassing is long enough.

Implementation of the idea of INS physical gyrocompassing is realized by the corresponding scheme of gyro stabilizer erection (Fig. 3.12). The erection signals both in the horizontal and azimuth channels are taken from the corresponding accelerometers which measure the gravity field strength vector, that is, perform the role of the pendulum.

Leveling and gyrocompassing errors of corresponding axis of the instrument trihedron at the end of the transition process are determined by small angles, in this case they are angles $\Im$|and $\wp$

To improve the accuracy of gyro stabilizer alignment in the erection contour usually the signals corresponding to the calculated values of the projection of the angular velocity of the Earth rotation on the axes of moving trihedron $\vartheta_{Z\,c}$, $\vartheta_{Y\,c}$.

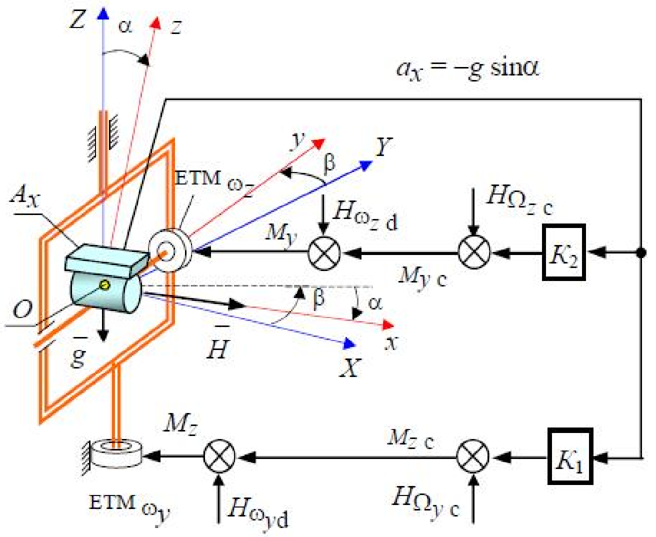


Fig. 3.12

In the sheme (see Fig. 3.11) these are signals $H_{\vartheta_{Zc}}$, $H_{\vartheta_{Yc}}$. Writing down the equations for the components of absolute angular velocity of gyro unit on the axes *OY* and *OZ* at small angles ℑ and ℘ there will be:

$$\in_y X Z\frac{K_1}{H} g\Im \Gamma \vartheta_{Yc} \Gamma \in_{Y\,d} X Z \dot{\wp}\vartheta_X \Gamma \vartheta_Y \Gamma \dot{\Im};$$
$$\in_z X Z\frac{K_2}{H} g\Im \Gamma \vartheta_{Zc} \Gamma \in_{Z\,d} X Z \Im\vartheta_X \Gamma \vartheta_Z \Gamma \dot{\wp} \qquad (3.27)$$

where ϑ_X, ϑ_Y are projections of the angular velocity of the Earth rotation on the axes of moving trihedron; $\in_{Yd}$, $\in_{Zd}$ are components of gyro drift.

At the end of the transition process, there will be:

$$\dot{\Im} X 0; \quad \Im X \Im_{st}; \quad \dot{\wp} X 0; \quad \wp X \wp_{st}, \qquad (3.28)$$

where $\sim_{st}$, $_{st}$ are steady state errors of horizontal and azimuth orientation of gyroplatform.

Solving equations (3.27) and taking into account (3.28), there will be:

$$\Im_{st} X \frac{\vartheta_{Zc} Z \vartheta_Z \Gamma \in_{Z\,d}}{\vartheta_X \Gamma \frac{K_2}{H} g};$$

$$\wp_{st} X \frac{1}{\vartheta_X} \frac{K_1}{H} g \frac{\vartheta_{Zc} Z \vartheta_Z \Gamma \in_{Z\,d}}{\vartheta_X \Gamma \frac{K_2}{H} g} \Gamma \vartheta_Y Z \vartheta_{Yc} Z \in_{Y\,d}\ . \qquad (3.29)$$

If $t_{Zc} = t_0$ *and* $\vartheta_X\ \Phi\Phi \frac{K_2}{H} g$, *then instead of equation (3.29) there will be*

$$\Im_{st} X \frac{H}{K_2 g} \in_{Z\,d}; \quad \wp_{st} X \frac{\aleph \in_{Z\,d} Z \in_{Y\,d} \Gamma \vartheta_Y Z \vartheta_{Yc}}{\vartheta_X},$$

where $\aleph X K_1/K_2$.

Since in gyrocompass schemes $\aleph = 10^{Z2} \ldots 10^{Z3}$, then the constant error of gyrocompassing with high accuracy is determined by the ratio

$$\wp_{st} \rceil Z\frac{\in_{Y\,d}}{\vartheta_X} Z \frac{\vartheta_{Yc} Z \vartheta_Y}{\vartheta_X}. \qquad (3.30)$$

The first term of the right side of equation (3.30) puts the severe requirements for the gyroscope drift around horizontal axis, and the second term requires the precise determination of the component of the angular velocity of the Earth rotation around the corresponding axis of moving trihedron.

If taking into consideration that the value ϑ_X is given by:

$$\vartheta_X = \vartheta_0 \cos \leftrightarrow \cos A_x,$$

where ↔ is the latitude of the alignment location; A_x is azimuth of corresponding axis of trihedron, then it is obvious that the accuracy of azimuth alignment increases with the increasing of latitude of alignment location and with the approaching of azimuth orientation of sensitivity axis of accelerometer to the eastern or western direction.

To ensure high accuracy of gyrocompassing in rather high latitudes, it is necessary to fully compensate the systematic component of the angular rate of gyro drift $\in_{Y\,d}$. The procedure of determination and compensation of gyroscope drift done during the pre-flight procedures is provided by the contours of the balancing modes.

Balancing modes, in turn, require certain preliminary orientation of gyroplatform. These circumstances cause long duration of autonomous azimuth alignment. Therefore, the operation of determining the gyro drift is done at the stage of pre-flight procedures or during maintenance checking. During pre-flight procedures the autonomous azimuth alignment of gyroplatform is made with "memorized" signals of gyro drift.

It is known that the best accuracy of gyroplatform balancing is reached for the northern axis of the instrument trihedron, with the decreasing of balancing accuracy for eastern axis. However, the drift relative to the eastern axis significantly affects the accuracy of gyrocompassing.

To improve the accuracy of the azimuth alignment in some INS so called method of "double gyrocompassing" is used. Its idea is as follows.

At first the eastern axis of gyroplatform is rotated to the north and the coarse gyrocompassing and balancing of the eastern axis of gyroplatform are made (memorizing of gyro drift), then the gyroplatform is rotated in 90⌈ (northern axis is oriented to the north) and the precise gyrocompassing and balancing of the northern axis are made with already balanced eastern axis.

3.3.4. Typical platform inertial navigation systems

The considered algorithms of operation of inertial navigation systems are implemented, for example, in the inertial systems И-21 and И-11.

Figure 3.13 shows the complete set of system И-21. The system consists of power supply unit БП-40 with frame РМ-49 (1), monoblock МБ-5 with frame РМ-23 (2), control display unit УВИ (3) and remote control unit ПУР-2 (4).

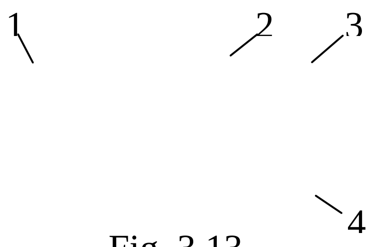


Fig. 3.13

Monoblock МБ-5 is the base of system И-21. It contains the gyroplatform, electronic devices and the digital computer that provides alignment by gyrocompassing thod, alignment in the specified heading, calculation of current navigation parameters, independent ground checking.

Mounting frame РМ-23 provides the mechanical coupling and coordination of monoblock axes with the aircraft construction axes, also provides the connection with airborne power sources and with airborne pneumatic system of electronics blowing and with gyro blocks.

Power supply unit БП-40 produces all kinds of power for sensitive elements, electronic circuits and heating system of monoblock.

The remote control unit ПУР-2 is used for setting the operating modes of the system and other auxiliary technological regimes. On unit ПУР -2 there are light indicators signaling the current state of the system.

Control display unit *УВИ* is the display, which is used for crew communications with the monoblock computer. *It* provides the possibility of entry in the computer of the fixed set of input information selectors, named buttons and the numeric dial pad.

The system И-11 consists of gyroplatform ПГ-1В-11, electronic unit БЭ-3, automatics unit БА-20, specialized power supply unit БСП-5, control panel ПК-36, digital computer system (ЦОК).

In turn, digital computer system consists of control display unit *УВИ, analog-digital connection block (АЦБС)* and specialized computing devices (СВУ). All electronic units of the system are mounted on shock-mounted platform П-21.

The basis for И-11 is free in azimuth gyro-stabilized platform ПГ-1В-11, kinematic diagram is shown in Fig. 3.9.

Figure 3.14 presents the simplified block diagram of the inertial system И-11.

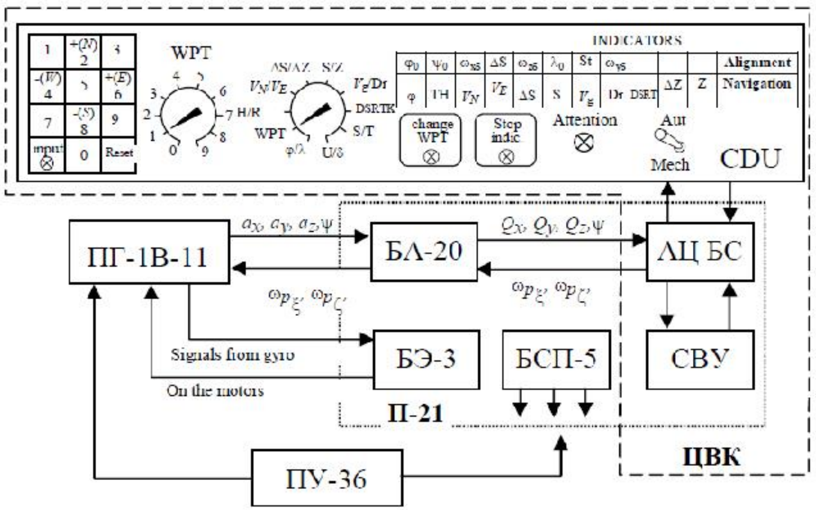


Fig. 3.14

Inertial system И-11 provides:

Z performing the en-route flight in accordance with a program entered the system before the flight or during it (up to nine waypoints of the route can be programmed);

Z continuous automatic calculation and indication of the current aircraft location in the geographic and great-circle coordinate systems, of track angle and true heading, of ground speed and drift angle, of desired track angle and cross-track error, of wind direction and speed, of time of flight and distance remaining to the next intermediate waypoint (WPT).

At operation process, the system И-11 obtains additional information about the true airspeed *V* (for calculation of wind parameters) and the absolute altitude H_{abs} (substitutes absent in system the unstable vertical channel) from ADC system.

Signals of accelerations a_x, a_y and a_z are measured by accelerometers (see Fig. 3.14). Through the automatics unit БА, they are converted into impulse signals Q_x, Q_y, Q_z and enter the *analog-digital connection block (АЦБС)*. From angle sensor located on the yaw axis of the platform, the signal э is also coming to the automatics unit БА. From the *analog-digital connection block (АЦБС) the* information comes to the specialized computing devices (СВУ). In its processor on the basis of the initial data and fixed programs, all tasks are solved to ensure the operation of system И-11, starting from the leveling of platform and finishing by the calculation of all navigation parameters.

In СВУ the values of absolute angular velocity $\in_{p_{\notin}}$, $\in_{p_{\rightarrow}}$, $\in_{p_{\Pi}}$ are calculated. The gyroplatform with accelerometers must rotate with such velocities relative to inertial space. They provide the contours of the integral correction of the horizontal position and through the units *АЦБС* and БА they come to the torque sensors of gyros. Gyroscopes precess, and from their angle sensors the signals come to the electronic unit БЭ-3, where they are amplified and sent to the motors of platform stabilization. The electronic unit provides also the caging and thermostabilization of gyros, platform and itself.

Using rotary switches and patchboard of control display unit the initial data are input in the system. This unit also performs the display function of the navigation parameters.

Control panel ПК-36 enables the system turning on, control of operation modes and methods of platform alignment and gives the opportunity to check the system performance.

The specialized power supply unit БСП-5 produces the power required to ensure the system performance.

Algorithms of И-11 operation are implemented in the digital computing complex of ЦВК type. According to the accelerometer signals, and data of ramp latitude and longitude, taking into account data of the intermediate waypoints, this complex produces the commands on the platform orientation and calculates the current navigation parameters. The digital computing complex also provides the implementation of preflight procedures of the system.

In system И-11 the following modes of operation are provided: "Standby", "Alignment", "Navigation", "Test" and "Attitude and heading reference system".

The mode "Standby" is intended to create the necessary temperature conditions of the system elements.

The mode "Alignment" is required for the platform orientation in the horizon plane and for the determination of true heading. In this mode at first the coarse electromechanical caging of platform axes are done along with respective axes of aircraft, and then the precise leveling of platform is done according to the accelerometer signals.

Autonomous determination of ramp true heading of the aircraft is implemented in ЦВК in gyrocompassing mode. With gyrocompassing the process of balancing is also carried out. Balancing is the computation and memorization of platform drift in the respective axes, caused largely by the gyro drifts. In algorithms of И-11 operation the memorized values of platform drift are taken into account which increases the accuracy of the system.

The precise determination of platform drifts is implemented in the mode of double gyrocompassing, but this mode is carried out not before each flight. All stages of mode "Alignment" are accompanied by indication of status factor on the control display unit.

The mode "Navigation" is a basic mode of system operation. In this mode, the system provides the dead reckoning and the calculation of all current navigation parameters.

The mode "Test" provides checking correct operation of the system as a whole. In case of failure, the command "System failure" is output and the fault location is remembered.

Moreover, the ground checking provides the verification of accuracy characteristics of the system. For this the system is aligned, and the flight simulates with certain flight speed entry in ЦВК. Computed navigation parameters ⇐ and ⇠ must not exceed the limited pre-calculated values. If the tested parameters exceed the limited values, then the command "Fault" is issued which means the unsatisfied accuracy of determining navigation parameters.

The mode "Attitude and heading reference system" is used in the case when the time for pre-flight procedures is less then the readiness time of the system, and from the system is not expected to output the current aircraft location and the dead reckoning. In case of computer failure the system automatically

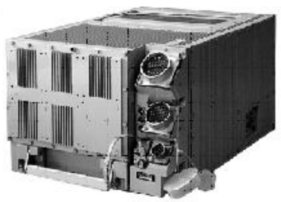


Fig. 3.15

switches to the mode "Attitude and heading reference system", which only provides the indication of signals about heading, pitch and roll of aircraft.

Modern platform INS, for example, ИНС-2000 (Fig. 3.15) by algorithms does not differ from the considered И-11. The differences are in more advanced element base, in particular, the element base of computer, the newest high-precision sensors of the primary information, advanced temperature control system.

ИНС-2000 is made in the form of monoblock consisting of gyrostabilized platform on the basis of dynamically tuned gyroscopes, service electronics, computer, block of modern interface. The system is deeply integrated in the navigation complex, so all devices of indication, control and testing are the integral devices of the complex.

The composition of INS may include the receiver of satellite navigation system (SNS), placed inside of monoblock. In this case, the system is equipped with the antenna device of SNS.

3.4. Strapdown inertial navigation systems

In strapdown inertial navigation systems (SINSs) the accelerometers are rigidly set directly on the aircraft body. The absence of horizontal platform requires the reproduction (modelling) of corresponding navigation coordinate system on the aircraft analytically, i.e. by mathematical modelling. Modelling of navigation coordinate system is based on signals from sensors of aircraft angular position. By projecting the signals of accelerometers into navigation coordinate system and separating the aircraft acceleration signals from their readings, in the SINS computer the direction of the vertical is determined analytically. Thus the problem of determining the angular orientation of the aircraft (angles of roll, pitch, grid heading) is solved.

The following calculations of current coordinates of the aircraft are reduced to integration of obtained accelerations, and to the solution of navigation problem of dead reckoning by known parameters of initial point and by continuous information about value and direction of the velocity vector in the selected navigation coordinate system. Herewith, the accuracy of dead reckoning is determined by the accuracy of the computer and, of course, by accuracy of primary navigation information sensors.

Among the potential benefits of SINS compared with platform INS there are the following ones:

- smaller size, weight and power consumption;
- significant simplification of the mechanical part of the system and, consequently, increasing of system reliability and reducing of its cost;
- the lack of limitations in angles of turn;
- reducing the time of initial alignment;
- universality of the system since transition to the determination of various navigation parameters is carried out algorithmically;
- simplification of solving reservation problem and control of system serviceability and its elements.

However, with the creation of SINS, principal difficulties appear, the main ones are:

- development of information sensors with a wide range of measurement and acceptable accuracy in terms of their rigid mounting on the aircraft body;
- development of computers with sufficient performance.

3.4.1 Algorithms of three-component SINS operation

The algorithm of SINS functioning includes a set of analytical dependences that allow continuously determining the current location coordinates, components of ground speed and angular position of the aircraft in the selected navigation coordinate system by measured values of apparent acceleration and absolute angular speed of the aircraft.

In the algorithms of three-component SINS operation, like in the algorithms of platform INS, the accuracy of dead reckoning of navigation parameters is achieved by eliminating components of gravity force acceleration and Coriolis acceleration from signals of the apparent acceleration, measured by accelerometers. But, unlike platform INS, the influence of these components is compensated only analytically.

The kinematic equations of inertial navigation are mainly determined by the selected coordinate system, i.e. by the navigational basis in which navigational parameters (coordinates and speed projections) are determined. In its turn, the choice of navigational basis depends on the type of aircraft, its features of trajectory motion, and the nature of missions.

For SINS of aircraft, that moves in atmosphere of the Earth the most frequently used coordinate system is rotating coordinate system with the base plane of local horizon and some orientation of horizontal axes in azimuth. The orientation of axes in azimuth means the possibility of their orientation, for example, by cardinal sides when two horizontal axes are directed to the east and north. Herewith, the positional information is determined by latitude ↔ longitude ⇐and height h, which are measured on Krasovsky ellipsoid or on GRS80 ellipsoid of international system WGS-84. Velocity is determined by projections on eastern V_E, northern V_N and vertical V_H axes, if the selected navigational system is the system with axes orientation by the cardinal sides or by projections on the axes of horizontal basis with other orientation. Orientation thus is determined by angles of roll, pitch and true heading.

Typical scheme of SINS construction is shown in Fig. 3.15. This variant implements the algorithm of the system that operates in the Earth rotating coordinate system.

SINS primary information sensors are angular speed sensors and accelerometers which are rigidly installed on the aircraft. Difficult conditions of information sensors operation lead to the arising of significant errors. Therefore, in SINS algorithms it is recommended to implement the analytical compensation of sensor errors (to carry out their in-flight calibration) before the signals will be used for calculating the orientation parameters and for determining the components of apparent acceleration along navigational axes.

For correction of primary information sensors readings the mathematical model of sensor is required, which usually takes into account: nonlinearity; orthogonality of sensors axes; drift; distortion of scale factor.

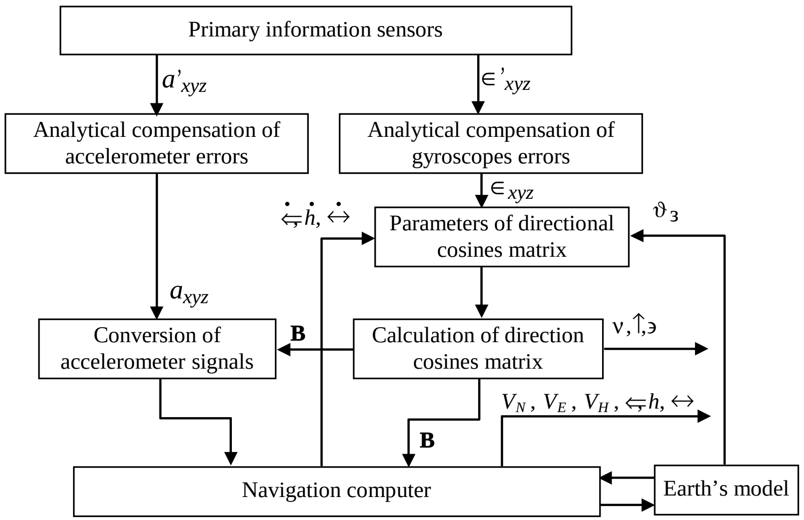


Fig. 3.15

The signals $\in_{x,y,z}$ from output of block of analytical error compensation are used to calculate the parameters of direction cosines matrix **B** which defines the relationship between two coordinate systems. Since the direction cosines matrix **B** is defined between axes of body-fixed coordinate system and axes of rotating navigation coordinate system, then it is necessary to involve calculated projections of angular speed vector of navigation coordinate system during the calculation of parameters of the matrix **B**. This is shown in the scheme by additional connections which take into account the angular speed that occurs during the overfly of the spherical Earth $(\dot{\leftrightharpoons}h, \dot{\leftrightarrow})$, and the angular speed of the Earth rotation (ϑ_0).

Conversion of apparent acceleration components $a_{x,y,z}$ from aircraft axes to axes of navigation coordinate system is performed by direction cosines matrix **B**. The navigation computer solves the problems, inherent to all platform systems, because at the input of this computer the projections of apparent acceleration on navigation coordinate system axes are formed, and the solution of this problem has nothing fundamentally new. At the output of SINS the radius vector of the aircraft location, velocity vector and the orientation angles of the aircraft are formed.

In the particular case, when the oriented by the cardinal sides horizontal trihedral is chosen as a navigational basis, at the output of the system the geographical (geodetic) coordinates of radius vector of location *B*, *L*, *H*, the projections of the relative motion velocity V_N, V_E, V_H, and the orientation angles of the aircraft at the geographical coordinate system (true heading э ,pitch ν and roll ↿) will be formed.

The volume of calculations in SINS is significant. This is mainly explained by the fact that airborne computer solves the problems related with the dynamics of the aircraft rotation, and with the dynamics of the aircraft translational motion. Translational speeds of aircraft are relatively small. For example, the speed of aircraft in flight towards the north in 1100 km/h corresponds to the rate of change of latitude only in 10 deg/h.

Thus, integration to obtain the speed and location can be accurately done using very simple methods of numerical integration at a low repetition frequency, in the typical case at 10…20 Hz.

Angular speeds of aircraft are typically greater in several orders than translational speeds. In particular, for maneuverable aircraft the angular speeds of rotation can be hundreds degrees per second. As a result, the integration of angular position in SINS is associated with severe requirements to airborne computer.

To provide the high accuracy of inertial navigation, it is necessary to have the errors of integration of angular position to be limited by several fractions of angular minute. So it is necessary to apply algorithms of integration of higher order at the typical repetition frequencies of 50…100 Hz.

Let us present the variant of SINS algorithms for the case when the oriented by cardinal sides horizontal trihedral is chosen as a navigational basis.

The navigational trihedral is taken the trihedral *NHE* connected with the ground surface.

The following direction of *NHE* axes (Fig. 3.16) is selected: *OH* coincides with the vertical; *ON* is tangent to the meridian; *OE* creates right-handed system.

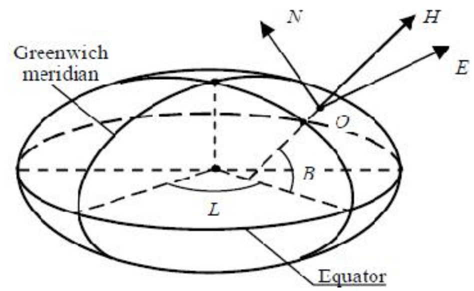


Fig. 3.16

In SINS algorithms usually the dynamic and kinematic equations are separated.

Dynamic equations realize three-component scheme of SINS where geodetic coordinates L, B, H are determined by the integration of equations:

$$\dot{L} = \frac{V_E}{(R_2 + H)\cos B};$$

$$\dot{B} = \frac{V_N}{R_1 + H};$$

$$\dot{H} = V_H,$$

where V_N , V_E are northern and eastern projections of ground speed, that are projections on *NHE* axes (see Fig. 3.16); R_1, R_2 are two radii of curvature of the terrestrial spheroid (ellipsoid of revolution); R_1 is radius of curvature of the ellipsoid meridional intersection (by plane *HN*); R_2 is radius of curvature of the ellipsoid intersection by plane *HE* (by plane of the first vertical);

$$R_1 = \frac{a(1 - e^2)}{(1 - e^2 \sin^2 B)^{\frac{3}{2}}}; \quad R_2 = \frac{a}{\sqrt{1 - e^2 \sin^2 B}},$$

where a is ellipsoid major semiaxis (a = 6378388 m); e is ellipsoid eccentricity (e^2 = 6,73 10^{-3}); H is flight altitude.

Here the same simplifications as in the platform INS can be used. In particular, functions $\frac{1}{R_1 \Gamma H}$ and $\frac{1}{R_2 \Gamma H}$ with the accuracy to terms of 10^{-2} order, can be represented as follows:

$$\frac{1}{R_1 \Gamma H} \rceil \frac{1}{a}\ 1 Z e^2\ Z\frac{H}{a} Z \frac{3}{2} e^2 \sin^2 B\ ;$$
$$\frac{1}{R_2 \Gamma H} \rceil \frac{1}{a}\ 1 Z \frac{H}{a} Z \frac{1}{2} e^2 \sin^2 B\ . \tag{3.26}$$

It should be noted that the use of simplifications (3.26) can lead to errors, compared with errors of high quality gyroscopic sensors used in SINS. So other more accurate forms of function representations can be used.

The components of the aircraft ground speed V_L, V_N, V_H are obtained as a result of integration of accelerometer signals projections, excluding the components of Coriolis acceleration and gravity force acceleration (see formula 3.14).

$$\dot{V}_E\ X a_E\ Z(V_N \in_{H_\phi}\ Z V_H \in_{N_\phi}) \Gamma g_E;$$
$$\dot{V}_H\ X a_H\ Z(V_E \in_{N_\phi}\ Z V_N \in_{E_\phi}) \Gamma g_H;$$
$$\dot{V}_N\ X a_N\ Z(V_H \in_{E_\phi}\ Z V_E \in_{H_\phi}) \Gamma g_N,$$

where $a_{E,H,N}$ are the projections of the aircraft apparent acceleration, measured by accelerometers on the navigational trihedral axes; $g_{E,H,N}$ are the projections of the gravity force acceleration vector, which take into account the acceleration of gravity force and the acceleration caused by inertia centrifugal force connected with the rotation of the Earth; components in brackets are projections of Coriolis acceleration on the navigational trihedral axes; $\in_{E_\phi}, \in_{H_\phi}, \in_{N_\phi}$ are projections of the angular velocity of the navigational trihedral relatively to the inertial space which take into account the projections of the angular velocity of the Earth rotation $\vartheta_E, \vartheta_H, \vartheta_N$ and the components of relative angular velocity of the navigational trihedral which are caused by the movement of the aircraft relatively to the Earth $\in_{E_V}, \in_{H_V}, \in_{N_V}$:

$$\in_{N_\phi} X \in_{N_V} \Gamma 2\vartheta_N;\quad \in_{H_\phi} X \in_{H_V} \Gamma 2\vartheta_H;\quad \in_{E_\phi} X \in_{E_V} \Gamma 2\vartheta_E.$$

In turn, the components of the relative angular velocity of the navigational trihedral and the Earth‘s rotational speed are determined by:

$$\in_{E_V}\ X Z \frac{V_N}{R_1 \Gamma H}\ X Z \dot{B};$$
$$\in_{HV}\ X \frac{V_E}{(R_2 \Gamma H)} \mathrm{tg} B\ X \dot{L} \sin B;$$
$$\in_{N_V}\ X \frac{V_E}{(R_2 \Gamma H)}\ X \dot{L} \cos B;$$
$$\vartheta_N\ X \vartheta_0 \cos B;\quad \vartheta_H\ X \vartheta_0 \sin B;\quad \vartheta_E\ X 0,$$

where ϑ_0 is angular velocity of the Earth rotation (ϑ_0 =7,27 10^{Z5} rad/s).

Deterministic mathematical model of the gravity force acceleration exists only for the normal component of the gravity force field, which corresponds to the earth ellipsoid with uniform distribution of mass in the volume of this figure. The gradient of this field at any point belonging to the surface of the ellipsoid is directed along the normal to it and is situated in the meridional intersection plane. As the point of the aircraft location does not belong to the ground surface, the vector of gravity normal field gradient $\bar{g}$ at this point will not be directed along the normal line omitted from this point to the terrestrial ellipsoid surface (axis *OH*). At the same time, this vector will be located in the meridian plane of the point *O*, i.e. in the plane of *NOH*. Then, using the potentional function of the normal gravity field of the terrestrial spheroid, with the accuracy of 10^{-2} order, the projections of gravitational field components $\bar{g}$ are as follows:

$$g_E \,\mathrm{X}\, 0\ ; \quad g_N \,\mathrm{X}\, 0;$$

$$g_H \,\mathrm{X}\mathrm{Z} g \left(1 \Gamma 5{,}2884\ 10^{\mathrm{Z}3} \sin^2 B\right) \left(1 \mathrm{Z} \frac{2H}{a}\left(1 \mathrm{Z} e \sin^2 B\right)\right),$$

where g = 9.78049 m/s^2 is gravitational acceleration on the equator.

There are other more accurate forms of given components expressions.

With the solution of kinematic equations the projections $a_{E,H,N}$ of aircraft apparent accelerations on the axes of navigational trihedral *NHE* are calculated by the readings of accelerometers from body-fixed coordinate system *XYZ* using the direction cosine matrix **B**

$$\begin{matrix} a_N \\ a_H \\ a_E \end{matrix} \ \mathrm{X}\mathbf{B} \ \begin{matrix} a_{x_{\mathrm{aircr}}} \\ a_{y_{\mathrm{aircr}}} \\ a_{z_{\mathrm{aircr}}} \end{matrix} \ .$$

The direction cosine matrix is as follows:

$$\mathbf{B}\,\mathrm{X} \begin{array}{c|c|c} \cos\text{э}\cos\nu & \sin\text{э}\sin\uparrow \mathrm{Z}\cos\text{э}\sin\nu\cos\uparrow & \sin\text{э}\cos\uparrow\Gamma\sin\uparrow\sin\nu\cos\text{э} \\ \hline \sin\nu & \cos\nu\cos\uparrow & \mathrm{Z}\cos\nu\sin\uparrow \\ \hline \mathrm{Z}\sin\text{э}\cos\nu & \cos\text{э}\sin\uparrow\Gamma\sin\text{э}\sin\nu\cos\uparrow & \cos\text{э}\cos\uparrow\mathrm{Z}\sin\text{э}\sin\nu\sin\uparrow \end{array},$$

where $\uparrow, \nu, \text{э}$ are angles of roll, pitch and yaw. The angle of yaw differs from the geographical heading э_g by the sign, $\text{э}_g = - \text{э}$.

The direction cosine matrix **B** can be obtained by different ways. Here are some examples.

The direction cosine matrix **B** can be obtained from the solution of the generalized Poisson equation by the information about the angular velocity of the aircraft relatively to the inertial space μ_{aircr} and about the angular velocity of navigational coordinate system relatively to the inertial space μ_{NHE}, which takes into account angular velocity of the Earth‘s rotation and angular velocity due to aircraft overfly of the spherical Earth

$$\dot{\mathbf{B}}\,\mathrm{X}\mathbf{B}\mu_{\mathrm{aircr}}\ \mathrm{Z}\mu_{NHE}\mathbf{B},$$

where

$$\mu_{\mathrm{aircr}}\ \mathrm{X} \begin{matrix} 0 & \mathrm{Z}\in_{z_{\mathrm{aircr}}} & \in_{y_{\mathrm{aircr}}} \\ \in_{z_{\mathrm{aircr}}} & 0 & \mathrm{Z}\in_{x_{\mathrm{aircr}}} \\ \mathrm{Z}\in_{y_{\mathrm{aircr}}} & \in_{x_{\mathrm{aircr}}} & 0 \end{matrix}\ ;$$

$$\mu_{NHE}\ \mathrm{X} \begin{matrix} 0 & \mathrm{Z}(\in_{E_V}\Gamma\vartheta_E) & (\in_{H_V}\Gamma\vartheta_H) \\ (\in_{E_V}\Gamma\vartheta_E) & 0 & \mathrm{Z}(\in_{N_V}\Gamma\vartheta_N) \\ \mathrm{Z}(\in_{H_V}\Gamma\vartheta_H) & (\in_{N_V}\Gamma\vartheta_N) & 0 \end{matrix}\ ;$$

$\in_{x\mathrm{aircr}}$, $\in_{y\mathrm{aircr}}$, $\in_{z\mathrm{aircr}}$ are the angular velocities of the aircraft relatively to the body-fixed axes measured by the angular velocity sensors; $\vartheta_E, \vartheta_H, \vartheta_N$ and $\in_{E_V}, \in_{H_V}, \in_{N_V}$ are determined above.

By elements of the matrix **B** the aircraft orientation angles: roll $\uparrow$, pitch ν, yaw (heading) ψ are determined:

$$\uparrow \text{X}\arcsin \frac{\text{Z}b_{23}}{\sqrt{1\,\text{Z}b_{21}^2}}\;;\nu\ \text{X}\arcsin(b_{21})\;;\text{э}\ \text{X}\arcsin\frac{\text{Z}b_{31}}{\sqrt{1\,\text{Z}b_{21}^2}}\ .$$

Here b_{ij} are elements of the matrix **B** (i is row number; j is column number).

Another algorithm of obtaining the direction cosine matrix suggests its formation directly by the angles $\uparrow$, ν, э. The angular orientation parameters $\uparrow$, ν, э (angular velocities $\dot{\text{э}}$, $\dot{\nu}$, $\dot{\uparrow}$) can be obtained by projecting the absolute angular velocity vector on the axes of body-fixed coordinate system.

Three dimensional direction cosine matrices are quite convenient for calculations in airborne computer. However, the formation of the matrix **B** by using the trigonometric functions requires considerable computing costs.

For determination of the aircraft orientation instead direction cosine matrix it is possible to use the parameters of Rodrigo-Hamilton transform in quaternions form. The advantage of the quaternions method is that it allows describing the conversion from one coordinate system to another with just only four numbers instead of 9 direction cosines.

The advantage of this method of construction of orientation matrix includes guaranteed orthogonality of orientation matrix. Furthermore, practice shows that the calculation using Rodrigo-Hamilton parameters gives the smallest computational costs compared with the other methods, provided with the same accuracy characteristics.

By the elements of matrix **B** according to (3.27) the aircraft orientation angles, roll $\uparrow$, pitch ν, yaw (heading) ψ are determined.

After finding the matrix **B** the system of equations for navigational calculations is closed.

The algorithm of navigational calculations in the case of forming the direction cosine matrix directly by the angles $\uparrow$, ν, э can be represented:

$$\in_{y_\phi}\ \text{X}\in_{y_{\text{aircr}}}\ \text{Z}\in_{y_{NHE}}\,;$$
$$\in_{x_\phi}\ \text{X}\in_{x_{\text{aircr}}}\ \text{Z}\in_{x_{NHE}}\,;$$
$$\in_{z_\phi}\ \text{X}\in_{z_{\text{aircr}}}\ \text{Z}\in_{z_{NHE}}\,.$$

$$\dot{\text{э}}\ \text{X}\int\!\in_{y_\phi}\cos\uparrow \text{Z}\in_{z_\phi}\sin\uparrow\text{Ā}\sec\nu\,;$$
$$\dot{\uparrow}\text{X}\in_{x_\phi}\ \Gamma\,\text{tg}\nu\int\!\in_{z_\phi}\sin\uparrow\text{Z}\in_{y_\phi}\cos\uparrow\text{Ā};$$
$$\dot{\nu}\ \text{X}\in_{y_\phi}\sin\uparrow\Gamma\in_{z\phi}\cos\uparrow;$$
$$\text{э}_{\text{г}}\ \text{XZэ}\,.$$

$$\mathbf{B}\,\text{X}\begin{matrix} \cos\text{э}\cos\nu & \sin\text{э}\sin\uparrow\text{Z}\cos\text{э}\sin\nu\cos\uparrow & \sin\text{э}\cos\uparrow\Gamma\sin\text{э}\cos\nu\sin\uparrow \\ \sin\nu & \cos\nu\cos\uparrow & \text{Z}\cos\nu\sin\uparrow \\ \text{Z}\sin\text{э}\cos\nu & \cos\text{э}\sin\uparrow\Gamma\sin\text{э}\sin\nu\cos\uparrow & \cos\text{э}\cos\uparrow\text{Z}\sin\text{э}\sin\nu\sin\uparrow \end{matrix}$$

.

$$\begin{matrix} a_N \\ a_H \\ a_E \end{matrix} \quad \mathrm{X}\mathbf{B} \quad \begin{matrix} a_{x_{\text{aircr}}} \\ a_{y_{\text{aircr}}} \\ a_{z_{\text{aircr}}} \end{matrix} \quad \bullet$$

$$\begin{aligned} &\dot{V}_E \ \mathrm{X} a_E \ \mathrm{Z} V_N (\in_{H_V} \ \Gamma 2\vartheta_H) \Gamma V_H (\in_{N_V} \ \Gamma 2\vartheta_N); \\ &\dot{V}_H \ \mathrm{X} a_H \ \mathrm{Z} V_E (\in_{N_V} \ \Gamma 2\vartheta_N) \Gamma V_N \in_{E_V} \ \Gamma g_H; \qquad \bullet \\ &\dot{V}_N \ \mathrm{X} a_N \ \mathrm{Z} V_H \in_{E_V} \ \Gamma V_E (\in_{H_V} \ \Gamma 2\vartheta_H). \end{aligned}$$

$$\dot{L}\,\mathrm{X}\frac{V_E}{(R_2 \,\Gamma H)\cos B};$$

$$\dot{B}\,\mathrm{X}\frac{V_N}{R_1 \,\Gamma H};$$

$$\dot{H}\ \mathrm{X} V_H.$$

$$\mu_{E_V} \ \mathrm{XZ}\dot{B};$$

$$\mu_{HV} \ \mathrm{X}\dot{L}\sin B;$$

$$\mu_{N_V} \ \mathrm{X}\dot{L}\cos B;$$

$$\vartheta_N \ \mathrm{X}\vartheta_0 \cos B;$$

$$\vartheta_H \ \mathrm{X}\vartheta_0 \sin B.$$

$$\begin{matrix} \in_{x_{NHE}} \\ \in_{y_{NHE}} \\ \in_{z_{NHE}} \end{matrix} \quad \mathrm{X}\mathbf{B}^{\mathrm{T}} \quad \begin{matrix} \in_{N_V} \ \Gamma\vartheta_N \\ \in_{H_V} \ \Gamma\vartheta_H \\ \in_{E_V} \ \Gamma\vartheta_E \end{matrix} \quad .$$

$$\frac{1}{(R_1 \,\Gamma H)} \rceil \frac{1}{a} \ 1\mathrm{Z}e^2 \,\mathrm{Z}\frac{H}{a}\mathrm{Z}\frac{3}{2}e^2 \sin^2 B \ ;$$

$$\frac{1}{(R_2 \,\Gamma H)} \rceil \frac{1}{a} \ 1\mathrm{Z}\frac{H}{a}\mathrm{Z}\frac{1}{2}e^2 \sin^2 B \ ;$$

$$g_H \ \mathrm{XZ}g \int 1\Gamma 5,2884 \ 10^{\mathrm{Z}3} \sin^2 B \big] \ 1\mathrm{Z}\frac{2H}{a} \int 1\mathrm{Z}e \ \sin^2 B \big] \ .$$

Modes of SINS alignment and calibration of primary information sensors are realized by separate algorithms.

3.4.2 Examples of strapdown INS

As examples of existing strapdown inertial navigation systems it is possible to consider strapdown laser navigation system БИНС-85 (analog of Litton-92) or systems ЛИНС-100РС, БИМС-Т, whose technical parameters are consistent with foreign analogues LTN-90 of Litton company and H-421 of Honeywell company (USA). These navigation systems are installed on aircraft modifications of IL-96; IL-76; TU-204; TU-214; YaK-130; AN70; AN-124; AN-148 and others. They are the major systems in flight and navigation complex.

Fig. 3.17

Figure 3.17 shows the exterior of strapdown inertial navigation system - ЛИНС-100РС, which is designated for use on mobile terrestrial, marine objects and airplanes for solution of orientation and navigation problems. ЛИНС-РС is based on ring laser gyroscopes and pendulum accelerometers. The structure of ЛИНС-100РС includes receiving device of satellite navigation system (SNS), placed inside a monoblock. ЛИНС-РС is equipped with antenna device of SNS and mounting frame to ensure accurate reference to object building axes.

Complete sets of SINS are the main sensors of flight and navigational parameters and spatial position parameters of aircraft. As the primary information sensors, the laser gyroscopes and non-thermostatic “dry” quartz accelerometers of compensation type are often used. The main feature of these accelerometers is realization of pendulum suspension from quartz glass which is characterized by high stability of elastic properties and practical absence of hysteresis.

In SINS the laser gyros and accelerometers are rigidly connected with the airplane, and their axes are oriented along its construction axes.

By information from laser gyros and accelerometers in the computer the definite navigational trihedral is simulated. On the axes of this trihedral the accelerations, measured by accelerometers are converted. Then the computer processes information according to algorithms of the system and of the system operation mode. The system has modes “Off”, “Alignment”, “Navigation”, “Attitude and heading reference system”, “Test”. The designation of these modes and calculated parameters are analogical to platform system И-11.

In SINS such parameters as pitch and roll of the aircraft are calculated but not read from synchro resolvers of gyrostabilized platform like in И -11.

3.5. PECULIARITIES OF INERTIAL NAVIGATION SYSTEMS OPERATION

Flight operation of system. Full-featured inertial navigation systems are part of the flight and navigation complex (FNC) and their flight operation is determined by regulations of navigation complex operation which have all the actions of the crew for FNC navigational preparation to the flight and FNC operation during the flight. Therefore here the operation questions only of inertial attitude and heading reference systems (AHRS) are considered.

AHRS is turned on after switching on the power supply of aircraft electrical power system. With receiving power supply AHRS starts the alignment procedure. The alignment process even for modern platform INS can last 15 minutes and more. For strapdown INS this process takes less than 30 seconds and is defined only by time of fibre-optical gyros readiness. During the process of INS alignment the aircraft must remain stationary but is allowed to be loaded.

After the alignment, roll, pitch and yaw (heading) will be inducted on the indicators of flight and navigation information. If the FNC includes navigation computer system (NCS) then heading selected in NCS will be shown. If NCS is turned off then the gyromagnetic heading will be shown on the indicators of flight and navigation information.

As on aircraft at least two INS are expected to be installed, then after the alignment it is necessary to control the deviation of attitude and heading reference system parameters which must not exceed the values, specified by technical specifications for the system, there must be no notifications of failures and deviations of attitude and heading reference system parameters.

At the taxiing it is necessary to make sure, that during the turning, the indications of roll and pitch do not change, and the heading values correspond to the selected one in NCS or to gyromagnetic heading.

At the preliminary start a line of the horizon on all indicators must occupy the horizontal position and must coincide with the aircraft figure. At the same time, there must be no notifications of failures and deviation of attitude and heading reference system parameters.

After the landing and parking before engine stopping it is necessary to turn off the system as prescribed in the maintenance instructions.

Technical operation of systems. INS and particularly strapdown INS with automatic balancing do not need specific maintenance but almost all INS have the mode of azimuth erection performed by magnetic (inductive) sensors.

During the installation and un-installation of these devices it is necessary to pay attention to the fact that the fixing screws of magnetic sensor must be made of non-magnetic metals.

Calibration of magnetic sensors is carried out during the stage of deviation works in the terms defined by the exploitation instruction of this device. Deviation works are also performed with replacing the inductive sensor.

Deviation works require the performance of the following operations:

- towing the aircraft on compass rose;
- eliminating the installation error of the magnetic (inductive) sensor;
- eliminating the semicircular and quadrantal deviations;
- determining the residual error.

To eliminate the installation error of the magnetic sensor the aircraft turns to the magnetic heading of 0 degrees with the tolerance of 5 degrees on the compass rose. With the help of a theodolite the installation error at this heading is determined.

In the flight complexes of modern aircrafts the elimination of semicircular and quadrantal deviations is done in the calibration mode. With turning on this mode, the aircraft turns with the interval of 45 ± 5º in the heading. When all data on eight headings come to the system, then it calculates errors and outputs these data to the calibration module.

The testing turns of the aircraft with the interval of 45 ± 5º the residual error is determined which should not exceed 1º.

TEST QUESTIONS

1. Information of what flight and navigation parameters can be determined in INS?
2. How can we classify INS depending on the methods of the accelerometers installation?
3. How is the value of the derivative of ground speed vector by the accelerometers information determined?

4. Why does INS gyroplatform always stay in the plane of horizon?
5. Why is the leveling contour adjusted to the oscillation period of Schuler pendulum?
6. How can we separate the relative acceleration from readings of horizontal accelerometers in platform INS?
7. What is the shape of the Earth to be usually used in INS kinematic equations?
8. What are the stages of the INS platform alignment in the horizontal position?
9. What are the methods of gyroplatform azimuth alignment?
10. How can the accuracy of azimuth alignment be increased?
11. How can we determine the accelerations in the navigation coordinate system in SINS with the absence of gyroplatform?
12. What components is it necessary to exclude from signals of apparent acceleration measured by accelerometers? How is the impact of these components compensated in SINS unlike to platform INS?
13. Why is direction cosine matrix computed in SINS algorithms?
14. What parameters (projections of angular velocities vectors) are usually used when calculating the parameters of the direction cosine matrix in SINS algorithms?
15. How can we obtain the direction cosine matrix in SINS algorithms?
16. How is the residual error of magnetic (inductive) sensor determined?

Chapter 4. Astronomical navigation systems

Modern astronomical navigation systems allow us to determine the flight course and the airplane coordinates. Astronomical systems have several advantages over others. The main is autonomy that is independency on ground equipment. Therefore, they can be used on routes of any length and in flights to any point of the world. Errors of determination of flight direction of the aircraft and its coordinates are practically independent on the duration of flight.

Astronomical navigation systems are the only compass devices which can determine the heading at high-latitude areas. While the magnetic and gyroscopic compass devices in these regions have very significant errors, using astronomical facilities in high-latitude areas the aircraft heading can be determined with accuracy up to unit of arc min.

Limited use of astronomical navigation systems on aircraft is caused by the presence of certain shortcomings. In particular, they cannot be used below the clouds and in the clouds, because of the lack of visibility of celestial bodies. Some difficulty is caused by identifying navigational stars in the daytime.

In the near future, these shortcomings can be overcome because the creation of star tracker allows using celestial tools regardless of weather conditions and time of day. They will not have interferences. Therefore, for a given celestial situation the promising astronomical navigation tools may be among the basic, especially when flying in Arctic and Antarctic.

Automatic celestial systems are called star trackers. Their main elements are sextant, which automatically provide range finding to celestial bodies, and computing device.

All star trackers require information about the position of vertical. At first, this data is obtained from the gyro vertical with positional pendulum erection torque system. Due to the widespread use of inertial navigation systems and airborne computers the structure of astro-inertial systems has been changed significantly. Based on INS and star trackers (automatic sextants) the high precision astro-inertial navigation systems have been created.

Thus, astronomical navigation tools can be divided into two groups: astrocompasses and celestial systems (star trackers).

4.1. Astrocompasses

4.1.1. Functionality and classification of astrocompasses

Astrocompasses are used to determine the true heading of aircraft. They are self-contained devices and are not exposed to both natural and artificial interferences.

According to principle of action, aircraft astrocompasses which provide direction finding to the celestial bodies, are divided into optical and radio direction-finding devices. The first can be non-automatic when direction-finding of bodies is carried out manually and automatic ones, with photo tracking systems. Radio direction-finding automatic astrocompasses use own radio emission of the Sun and of planets. Their use unlike optical astrocompasses does not depend on weather conditions.

By the type of CS adopted in the implementation of direction-finding devices there are astrocompasses of horizontal and of equatorial CS (horizontal and equatorial astrocompasses). The direction-finding plane of the horizontal ones is vertical, and the direction-finding plane of equatorial astrocompasses is the plane of declination circle of the celestial body. The most widely used astrocompasses are automatic optical solar horizontal astrocompasses of ДАК-ДБ type. As the emergency and auxiliary tools of heading measurement the non-automatic equatorial astrocompasses of АК type are used.

4.2. Horizontal astrocompass

Each astrocompass consists of the following parts:

- direction finder designated to determine the direction to the center of celestial body;
- computing device that calculates the azimuth of the celestial body, and also the true or great circle heading;
- selector of aircraft coordinates and of celestial bodies;
- clockwork mechanism to take into account the Earth rotation;
- indicator.

In the horizontal astrocompass (Fig.4.1) the true heading ψ is given by formula:

$$\psi = A - \beta, \qquad (4.1)$$

where $\wp$ is heading angle of celestial body (the angle between the horizontal projections of the longitudinal axis and of the direction to the body); A is azimuth of celestial body (the angle between the northern direction of meridian and horizontal projection of the direction to the body).

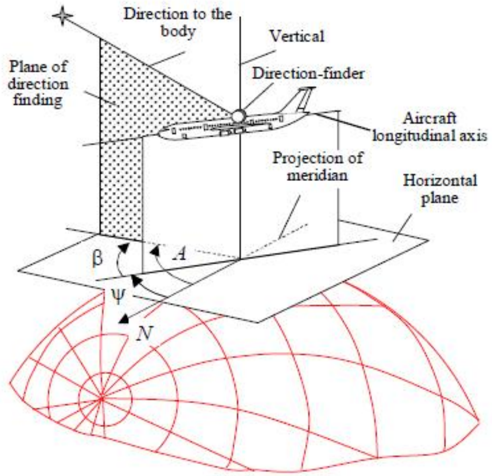


Fig. 4.1

In equation (4.1) the heading angle $\wp$ of body is measured by direction finder of astrocompass, and the azimuth A of the body is determined by two methods: by analytical calculation in microprocessor astrocompass (Fig. 4.2) and by spatial modeling of celestial sphere.

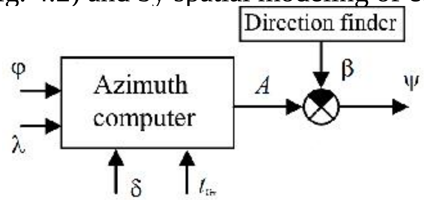


Fig. 4.2

Analytical determination of azimuth is based (see p. 1.3.3) on formulas of spherical trigonometry:

$$\cos h \sin A \, X \cos \Omega \sin t; \qquad (4.2)$$

$$\cos h \cos A \, X \cos \leftrightarrow \sin \Omega Z \sin \leftrightarrow \cos \Omega \cos t. \qquad (4.3)$$

Dividing equation (4.2) by (4.3), the formula to calculate the azimuth is obtained as:

$$A \mathrm{X} \operatorname{arctg} \frac{\cos\Omega \sin t}{\cos\leftrightarrow \sin\Omega \mathrm{Z} \sin\leftrightarrow \cos\Omega \cos t}.$$

Here $\leftrightarrow$δ, t Xt_{gr} Γ⇐ must be known, then the azimuth A of body is calculated as a function of the aircraft coordinates (latitude φ and longitude λ) and of the body coordinates (declination δ and Greenwich hour angle t_{gr}).

The values φ, λ, δ and initial angle t_{gr} are entered manually or automatically, and changes of the local hour angle t due to change in the angle t_{gr} are taken into account by clockwork mechanism.

In non-automatic astrocompasses the azimuth of body is calculated with the help of spatial mechanisms - spherants which simulate the spherical triangles discussed in p. 1.3.3 (see Fig. 1.13).

The simplified kinematic diagram of spherant is shown in Fig. 4.3, where for clarity the spherant elements are connected with the terrestrial sphere.

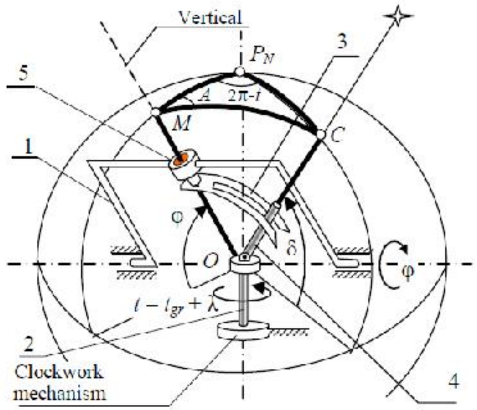


Fig. 4.3

Spherant main elements are:

Z latitude clamp *1*, rotating around the instrument axis "east-west";

Z hour axis *2* (instrument celestial axis);

Z altitude arc *3*, which rotates around the axis *OM* that simulates the local vertical and is used to measure the azimuth;

Z declination pin *4* that simulates the direction to the body, intersects with the altitude arc *3* and jointed with angle sensor 5.

For the convenience of clockwork implementation the meridian plane of the object is fixed, so changes both t_{gr} and λ are taken into account by declination pin rotation *4* (instrument plane OP_NC of body declination circle) correspondently to change in the local hour angle t

$$t = t_{gr} + \lambda.$$

By entering values φ, δ and t, the simulated triangle will be similar to one shown in Fig. 1.13, a. With the help of angle sensor *5* the calculated azimuth A of body can be measured. Stator of sensor *5* is connected with latitude damp *1* and rotor is connected with the altitude arc *3*.

To ensure the continuous alignment of the direction finding plane with the direction to the body that is prerequisite for accurate reference of aircraft true heading it is necessary not only to provide automatic tracking of the body, but also to compensate all evolutions and movement of the aircraft which violate the conditions of direction finding.

The main aircraft evolutions need to be compensated to ensure the accurate reference of the heading are:

Z changes in the aircraft attitude (roll and pitch) Z$\in_{\wp}$;

Z aircraft movement around the center of the Earth with angular velocity $\in_a$ that leads to changes in the geographic coordinates of the aircraft and to changes in horizontal coordinates of the body;

Z diurnal rotation of the Earth with angular velocity $\in_0$, which leads to changes in hour angle and to changes in horizontal coordinates of the body.

It is necessary to provide such compensation movement of direction finding plane $\in$, that must be equal to the disturbance motion in value and opposite in the direction

$$\mathrm{Z}\in = \in_0 + \in_a + \in_{\wp}$$

During en-route flight, the main changes in parameters of the aircraft angular orientation are due to roll changes. That is why all astrocompasses have compensators of roll component $\in_{\wp}$ which are based on sensors of the vertical, in particular, it may be liquid pendulum mechanism or gyro vertical.

Compensation of rotational motions $\in_0$ and $\in_a$ is slightly difficult and depends on changes in latitude and longitude of airplane, and on the speed of the Earth rotation, therefore, to determine the true heading in azimuth computer it is necessary continuously to introduce the current coordinates of the aircraft.

When flying by rhumb line it requires the integration of astrocompass with navigation system of dead reckoning in the geographic coordinate system. But for flying by the great circle route such use of astrocompass is significantly simplified. It should be noted that the aircraft route usually consists of great circle sections, or legs.

In horizontal astrocompasses which provide the flight by great circle route, the compensating rotation is done relative the vertical axis. The angular velocity of compensating rotation can be decomposed into two components: horizontal one and along with the celestial axis parallel to the Earth rotation axis.

The component along the celestial axis is $\in_N = Z \in_0$. Thus, the direction finding plane of horizontal astrocompass must be rotated around the celestial axis with angular velocity equal to the Earth rotation speed, but in the opposite direction. This is done by clockwork mechanism.

The horizontal component is equal to the angle of rotation ε relative to the center of the Earth (Fig. 4.4), that is, the distance travelled by airplane on the great circle route and expressed in arc. Thus, to ensure the flight by great circle route it is necessary to rotate additionally the direction finding plane relative to the horizontal axis that is perpendicular to the ground speed vector, and is proportional to the travelled path. Thus, it means to deviate it back (to the tail) at an angle

$$\varepsilon = S_{gc}/R_0,$$

where S_{gc} is the path travelled by the great circle route.

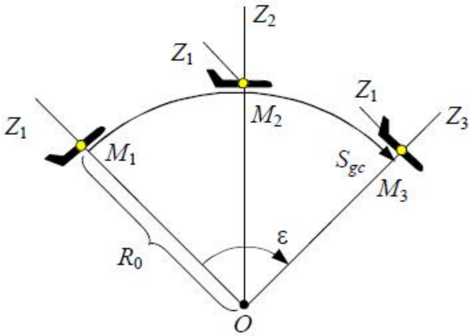


Fig. 4.4

As explained in Fig. 4.4, the direction finding plane in the points M_2-M_3 of airplane location will be parallel to the plane OZ_1 of initial point M_1. Because of the remoteness of the celestial body the direction to it from any point on the Earth is almost parallel, so the planes passing through the axes M_2Z or M_3Z_1 and the celestial body will be parallel to the body vertical in the initial point M_1. Therefore, the angle between the great circle plane and the direction finding plane is measured by the heading angle sensor and is equal to the heading angle of body β_0 at the initial waypoint (IWPT) at point M_1.

With keeping the constant value of true great circle heading, calculated by the great circle direction in the initial waypoint the aircraft will fly by the great circle route. The declination angle of the direction finding plane on the tail of the airplane proportional to the travelled path is calculated by track controller.

Track controller continually receives the signal from the clockwork mechanism and performs the dead reckoning

$$S = \int_0^t V_g\,dt .$$

where V_g is ground speed, which is set by the pilot on track controller.

If ground speed is unknown, then the track controller is set by true airspeed.

The tilting of the direction finding plane is carried out from 0⌈ to 10⌈, i.e. up to 1100 km, and then all settings are repeated again. For flying by great circle route it is necessary to enter the geographical coordinates of IWPT ($\leftrightarrow_{IWPT}$, $\Leftarrow_{IWPT}$) in the astrocompass computer together with equatorial coordinates of the Sun, keeping them further unchanged. After reaching IWPT the aircraft by astrocompass data is directed to true great circle heading. On the track controller the current ground speed is set, and the index of the travelled path is reset to zero.

4.1.3. Features of horizontal astrocompass construction

Widely used compasses are horizontal astrocompasses of ДАК-ДБ type. They determine the true heading and can also be used for flights by great circle route. In the daytime they carry out automatic direction finding of the Sun. At night the astrocompasses can work together with periscopic sextant, measuring the relative heading angle of bodies by their visual direction finding by operator.

Block diagram of astrocompass is presented in Fig. 4.5.

The set consists of sensor of heading angles, unit of amplifiers, computer, track controller and heading indicator. At the diagram (Fig. 4.5) there is direction finding head (DFH), amplifier (AM), tachometer generator (TG) and motor (M) which form a tracking system that automatically aligns the direction finding plane with the vertical of celestial body. The difference between the actual ℘and measured ℘heading angles of celestial body tends to zero.

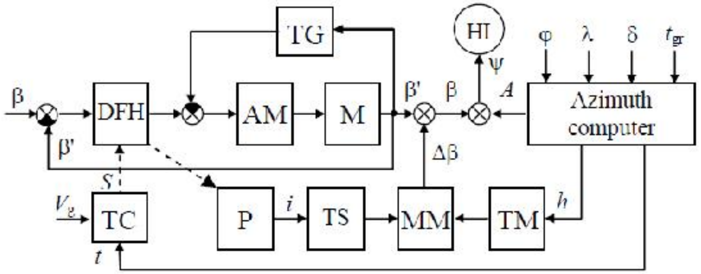


Fig. 4.5

The azimuth computer (its operation is based on mechanical modeling of parallactic triangle with help of spherant) outputs the azimuth *A* of celestial body using the given angles ↔⇆t_{gr} and Ω To avoid errors in measuring the heading angles at the aircraft banking the pendulum mechanism (P) determines the tilt angle *i* of the direction finding plane relative to the vertical of the body, which is used to form the bank correction Δβ.

Bank correction is calculated by the tilt angle *i* obtained from multiplying mechanism (MM), and by the signal of altitude angle *h* (coming from the azimuth computer). These signals are converted on potentiometers of the tilt sensor (TS) of direction finding head and on tangent mechanism (TM) into sin *i* and tg *h* and then are multiplied with the help of multiplying mechanism forming the correction

ζ ℘Xsin *i* tg*h*.

The result of addition of angles ℘Ɛ and ζ ℘is heading angle of body β, in which the error of the tilt of direction finding head is compensated. The heading indicator (HI) reproduces the measured heading

э = *A* Z ℘

The direction finding head with photo elements can be controlled by track controller (TC), which provides the measurement of great circle heading. To fly by great circle route in the track controller the ground speed is entered, and from the computer the time signals are supplied. At the output of TC the signal proportional to travelled path *S* is formed. Direction finding head tilts backward (with the help of potentiometric remote transmission) at the angle that is proportional to the travelled path (Fig. 4.5).

Electromechanical diagram of photo electric system of Sun direction finding is shown in Fig. 4.6.

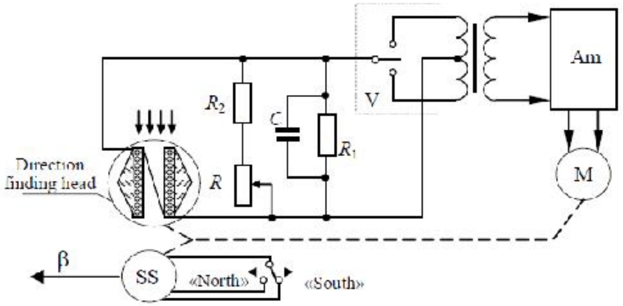


Fig. 4.6

Photo electric system of Sun direction finding includes the direction finding head (DFH) with two differentially connected photo elements, the vibrator (V) for conversion DC to AC, the amplifier (Am) and the electrical motor (M).

The motor rotates direction finding head at heading angle of the Sun (angle ℘), at the same time it rotates the rotor of the synchro sensor (SS). The signal taken from synchro sensor is proportional to ℘The rheostat (*R*) allows adjusting the sensitivity of the system respective to change in the brightness of the Sun and eliminating self-oscillations of the head.

Heading angle of the Sun ℘is measured by synchro. Differential circuit (R_1C) is used to improve the quality of the photo tracking system.

Cross section of direction finding head is shown in Fig. 4.7. Direction finding head consists of two photo elements *1* which have differential connection and are attached to cylindrical holder *4* with conical mirror *3*. Sensitive layers of photo elements are directed to different directions and covered with cones *2*, scattering the light. Shutters *5* with light filters *6* provide omni-directional view of the upper hemisphere. By the deviation of the direction finding plane from the vertical of the Sun, the beams fall on one of the conical mirror *3*, are reflected from it and pass through the lens *2* to the photo element *1*. Shutters *5* with light filters *6* limit the luminous flux incident on the photo elements.

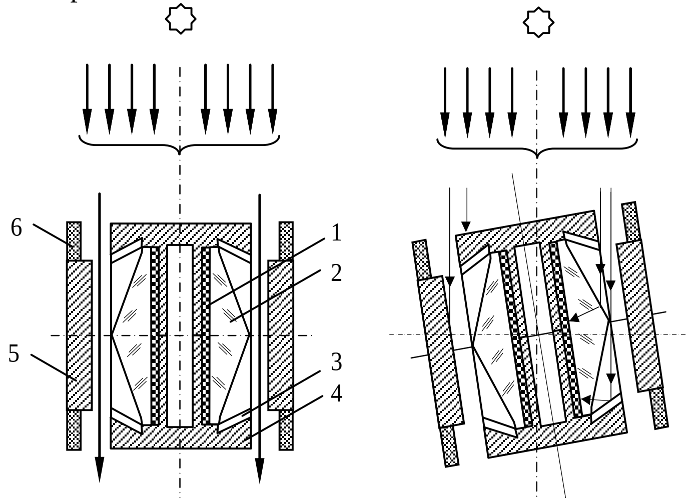


Fig. 4.7

Such construction of direction finding head provides direction finding of the Sun at any initial position relative to heading angle sensor. Exterior of the heading angle sensor and of front panel of astrocompass ДАК-ДБ computer is shown in Fig. 4.8.

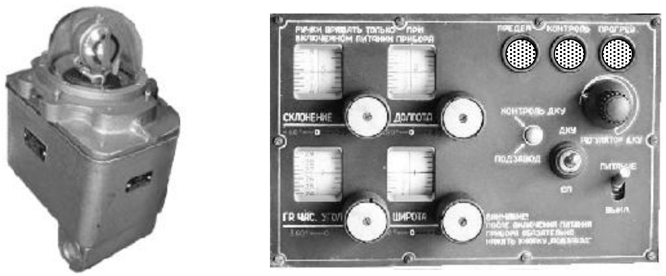


Fig. 4.8

For flights in the southern hemisphere the all-latitudinal astrocompass of ДАК-ДБ-5В type is used. Since the direct provision of all-latitudinal computing of azimuth is not possible for kinematic reasons, then the following method of solving this problem is used.

With using astrocompass in the southern hemisphere, the true values λ and t_{gr} and fictitious values φ' and δ' are entered in the azimuth computer, which are equal in magnitude, but opposite in sign to the true values φ and δ. This is explained in Fig. 4.9, where *M* and *C* denote the true position of the object and the body geographical position, and *M*' and *C*' are fictitious ones. As a result, the computer determines the angle *A*' equal to (π – *A*). The signal conversion *A*' into *A* is performed by switching the windings of SS (see Fig. 4.6) by the switch "North – South".

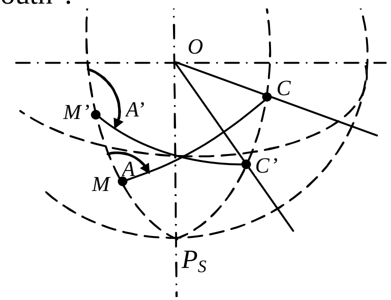


Fig 4.9

4.1.4. Equatorial astrocompasses

In the equatorial compasses there is no complex computing device and they do not need to enter the coordinates of the body declination. Astrocompass is done in the form of spatial model of the celestial sphere, which determines the direction that coincides with the geographical meridian of the given point. To determine the true heading it is necessary to know the latitude φ and longitude ⇐of the aircraft location and Greenwich hour angle t_{gr} of the body.

Among equatorial astrocompasses, the automatic astrocompasses of ДАК-Б type and non-automatic ones of АК-53П, АК-59П types are practically used. Devices of the last two types are used more widely as backup means of heading determination, requiring no power supply (excluding electric heating).

Unlike horizontal astrocompasses in which the direction finding plane coincides with the vertical of celestial body, in the equatorial astrocompasses the plane of direction finding coincides with the declination circle of the body. Spatial model of equatorial astrocompass is shown in Fig. 4.10.

The direction finder is fixed at hour axis $P_N\,P_S$ that simulates the celestial axis Z the axis of rotation of the Earth, so that the direction finding plane coincides with this axis. The hour axis is tilted relative to the plane of the azimuthal circle at the latitude angle φ. Since the hour axis simulates the celestial axis, the projection of this axis on the azimuthal circle located in the plane of the horizon, must coincide with the meridian line that connects south (*S*) with north (*N*), with this zero index of the azimuthal scale is in the plane of the observer meridian.

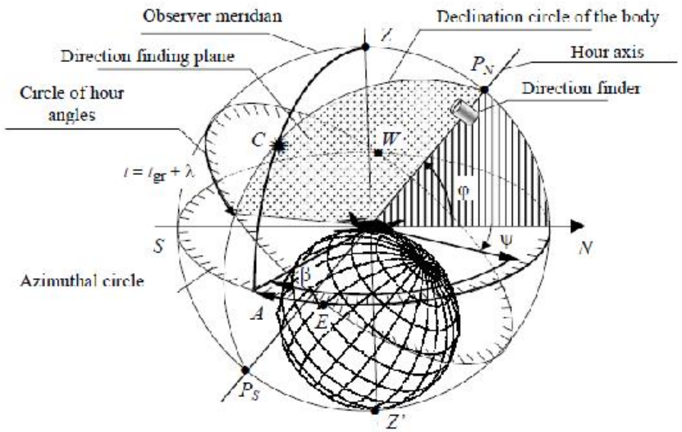


Fig 4.10

Points of east (*E*) and west (*W*) are taken remote in 90⌈ from south point (*S*) respectively counterclockwise and clockwise, when viewing from zenith (*Z*). Vertical line *ZZ*' connecting zenith and nadir is vertical axis of the equatorial astrocompass, and the plane *NESW*, in which azimuthal circle is lying becomes the plane of mathematical horizon.

Perpendicular to the hour axis, which is rotated by the clockwork mechanism, is the plane of the circle of hour angle. By the initial alignment of the clockwork mechanism, Greenwich hour angle t_{gr} and the current local longitude ⇐are set. Due to this, the direction finding plane rotates around hour axis at the body hour angle ($t = t_{gr} +$ ⇔).

By rotating the direction finder around vertical axis *ZZ*', the direction finding plane is aligned with the body *C* (with the center of body) and by the scale of the azimuthal circle the true heading is evaluated

$$э = A\ Z\ ℘$$

Therefore the true heading is defined as the angle between the simulated (instrumental) meridian line and the longitudinal axis of aircraft.

As an example of equatorial astrocompasses let us consider the design of non-automatic all-latitudional compass АК-59П. General view of the compass is represented in Fig. 4.11.

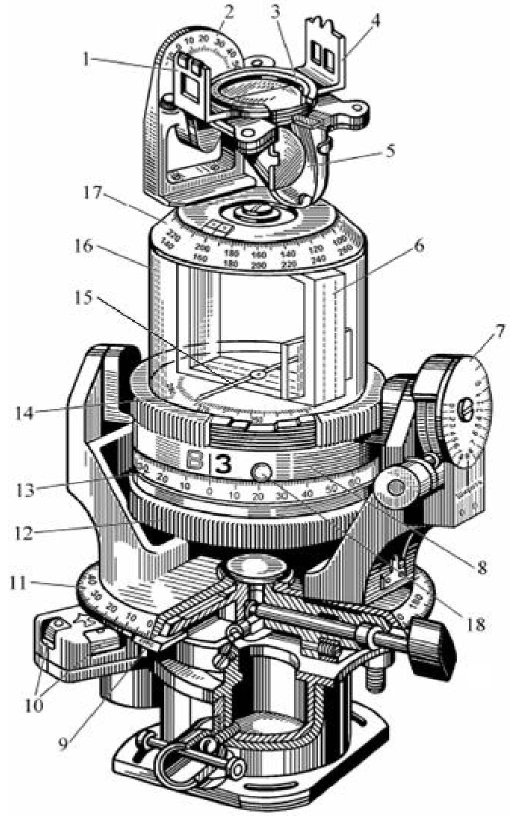


Fig 4.11

Compass АК-59П is designed to determine the true heading of the aircraft in the northern and southern hemispheres by the Sun, by the plane of polarization of the scattered in atmosphere sunlight, by the Moon, by the stars and planets.

The error in determining the true heading by solar and stellar sighting systems does not exceed ±2° at altitudes of bodies from 0° to 60°, in the range of declination of the Sun ±23.5°, of stars ±64°. The error in determining the true heading by the polarization sighting system is less than ±2°.

Astrocompass has sighting device *6* for direction finding of the Sun and the upper sighting system, the basis *17* of which has the scale of hour angles of the body. The scale consists of the devices *1*, *4* for direction finding the stars, the Moon and planets and of the polarization sighting devices *3, 5* for the direction finding the Sun in polarized light.

Block of upper sighting systems has no connection with the clockwork mechanism, so Greenwich hour angles are set manually at scale *17*. Block tilts relative to the equatorial plane at the angle δ of the point celestial bodies that are under direction finding. The angle δ is fixed by the scale *2*.

Polarization sighting system (analyzer and prism) are used for the direction finding the Sun in the twilight at the altitude of body $h < 0$ (to 7°). The polarization system also enables the determining heading in the case when the Sun is covered with clouds, but when in the plane of declination circle of the Sun there is the gap between the clouds.

The analyzer consists of three fields (Fig. 4.12): fields F_1 and F_2 with the planes of polarization at angle 90° relative to each other and of field F_3 with the plane of polarization at angle of 45° relative to the first two fields.

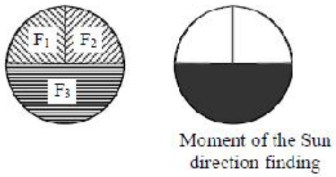


Fig 4.12

By rotation of the analyzer (around vertical axis), the illumination of its individual parts changes. Observation of the analyzer is carried out through the prism. At a time when the brightnesses of the fields F_1 and F_2 are the same, and the field F_3 is dark, the line of symmetry is aligned with the vertical of the Sun.

In solar sighting device (see Fig.4.10) the flow of light beams focuses on opal translucent screen *6* with two parallel marks. The axis of rotation of the direction finding plane of solar sighting device *6* is tilted relative to the azimuthal circle *11* at geographical latitude (evaluation of the angles is done by scale *7*).

Solar sighting device is rotated relative to the scale *14* at Greenwich hour angle using the clockwork mechanism with speed of 360° per solar day (rotation is done by ring *12*). To control the compensation of hour angle there is second hand *15*.

Solar sighting device is placed in a cylindrical transparent case *16* and rotates in the ring *8* at angle of longitude, measured by scale *13*.

Stellar sighting system has no connection with the clockwork mechanism, so Greenwich hour angle is set manually by the scale *17*.

The whole sighting system can rotate around the vertical axis perpendicular to the azimuthal circle *11*. The horizontal position of the circle is set by special adjusting screws and is controlled by level *10*. Searching movements with the direction finding of bodies are made around vertical axis. The evaluation of heading is made against the index *9* (labeled "Heading") by the scale *11*.

The height of the solar sighting device case is chosen so that at maximal angles of the Sun declination (±23°27ʼ) to provide its direction finding. When using astrocompass АК-59П in the southern hemisphere it is necessary to change the direction of rotation of the axis of clockwork mechanism.

Errors of determining the heading by polarization sighting device do not exceed ±3°, and with the help of other sighting systems they do not exceed ±2°.

Sighting system of *automatic* equatorial solar astrocompass ДАК-Б is similar to the considered above. Astrocompass has the scale and electric actuators for remote entering the latitude and longitude of aircraft in sighting system, as well as initial Greenwich hour angle of the Sun. In the selector there is a clockwork to take into account changes in hour angle.

The search and direction finding of the Sun are carried out automatically by the photo tracking system. But since the direction finding head has limited field of view, then for the initial capture of the Sun it is necessary to ensure the azimuthal rotation of the sighting system by pressing the button.

Sighting head is mounted at the pitch frame, which is controlled by the tracking system by signals of electrolytic pendulum. However, such a system of error compensation of astrocompass because of the changes in pitch is effective only in the absence of longitudinal accelerations.

Astrocompass ДАК-Б is used in northern latitudes in the range of latitudes from 45 to 90°. At changes of the Sun elevation in the range from 1° to 68.5° the maximal error increases from ±2° to ±4°.

4.2. Celestial navigation systems

The main purpose of celestial navigation systems is to determine the coordinates of the aircraft location. In addition, they usually measure heading, performing the same function as astrocompass.

Celestial navigation systems are used primarily for airplanes, flight routes of which exclude the possibility of using other means of location determination.

4.2.1. Methods and theoretical basics of celestial navigation

Methods of celestial navigation are based on relation of navigation and astronomical coordinate systems, analyzed in p. 1.3.3. All methods that have found practical application, are reduced to different methods of modeling spherical triangles *ZP'C* (*MP*$_N$*C'*) (see Fig. 1.13).

The method of two altitude angles is the most practically used and it corresponds to equations (1.22)

$$\begin{aligned} &\sin h_1 \mathrm{X} \sin \leftrightarrow \sin \Omega_1 \Gamma \cos \leftrightarrow \cos \Omega_1 \cos(S_{gr} \Gamma \Leftarrow Z \Im_1); \\ &\sin h_2 \mathrm{X} \sin \leftrightarrow \sin \Omega_2 \Gamma \cos \leftrightarrow \cos \Omega_2 \cos(S_{gr} \Gamma \Leftarrow Z \Im_2), \end{aligned} \qquad (4.4)$$

where indexes «1» and «2» correspond to numbers of celestial bodies. Using equations (4.4) it is possible to find values of $\leftrightarrow$ and $\Leftarrow$.

However, the solution of the equations set is not unambiguous, which is clearly illustrated by geometrical interpretation of method (see Fig. 4.13). It is based on the fact that the observer on the spherical Earth is at any point of the circle with "center" in geographical body position, and measures one and the same altitude *h*. Spherical distance from the geographical body position to the circle is equal to zenith distance

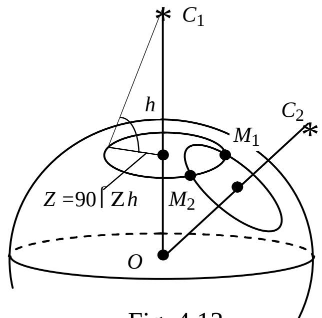


Fig. 4.13

$$Z = 90\lceil - h.$$

Measured values of two altitude angles correspond to two lines of position on the Earth - circles, which generally intersect at two points M_1 and M_2 of possible observer locations.

The ambiguity of the coordinate determination is excluded if the approximate location of object is known. If the altitude angle measurement is made relative to the geodetic vertical (which generally is done in practice), the method of two altitudes determines the geodetic coordinates of points M_1 (B_1, L_1) or M_2 (B_2, L_2). However, because of ease of using spherical triangles, in this case instead of geodetic latitude *B*, the geocentric latitude φ is used. Thus the obtained results are easily converted into geodetic coordinate system (with the specified measurement of h) by formal replacement of φ for B.

The accuracy of the method of two altitudes essentially depends on the difference $\Delta A = A_1 - A_2$ of body azimuth. Accuracy is maximum at ΔA, equal to $\pi/2$ or $3\pi/2$, and minimal at ΔA, equal to 0 or π. This is clearly explained by the graphic definition of the possible area of aircraft position in the presence of altitude measurement errors $\{\Delta h$ (Fig.4.14) for cases where the azimuth difference is close to $\pi/2$ (Fig.4.14, *a*) and when it is much less (Fig.4.14, *b*).

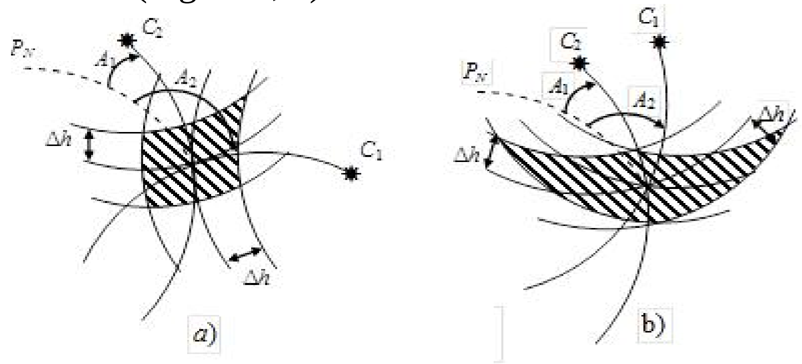


Fig. 4.14

In the second case, the dispersion of possible location points increases significantly, and with ΔA equal to 0 or π it is even possible to have non-overlapping circles, that is lack of sustainable solution.

Since instead of globes at airplanes the geographic maps are used, then in navigation practice the graphic-analytical method is widely used. It is based on the replacement of circles of equal altitudes in their points of intersection by segments of straight lines that is possible because of large radii of circles of equal altitudes. These lines (curves *1* and *2* in Fig. 4.15) are tangent to the circles of equal altitudes. The corresponding constructions of curves are explained in Fig. 4.15. The essence of the graphic-analytical variant of method of two altitudes, is the following. At time moments t_1 and t_2 using sextant the altitudes h_1 and h_2 of celestial bodies are measured. In the neighborhood of possible location the point M_0 with coordinates φ_0, λ_0 is put. For the point from handbooks, the altitudes $h_{1_0}(t_1)$ and $h_{2_0}(t_2)$ and azimuths $A_{1_0}(t_1)$ and $A_{2_0}(t_2)$ of celestial bodies are determined[1].

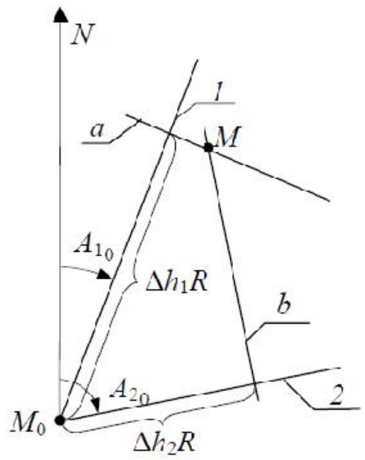


Fig. 4.15

[1] Using «Aviation astronomical yearbook», «Tables of star altitudes and azimuths», «Tables of altitudes and azimuths of the Sun, Moon and planets».

After that, lines *1* and *2* are plotted at angles A_{1_0} and A_{2_0} which are directions to geographical body positions. Then the altitude differences are determined and converted into linear values (1® is 1.85 km):

$$\Delta h_1 = h_1 - h_{1_0} \quad \text{and} \quad \Delta h_2 = h_2 - h_{2_0}.$$

Through points on lines *1* and *2*, separated by distance $\Delta h_1 R$ and $\Delta h_2 R$ (R is the Earth radius) from M_0 the linearized lines of position are plotted – straight lines *a* and *b* which are perpendicular to azimuth lines *1* and *2*.

At the positive values of Δh_1 and Δh_2 the segments $\Delta h_1 R$ and $\Delta h_2 R$ are drawn from M_0 by the azimuth lines to the celestial body, as it is shown in Fig. 4.15, and at the negative values the direction will be from the celestial body.

The intersection point M of lines *a* and *b* is true position of object by measuring two altitudes of celestial bodies. Using geometric constructions the corrections $\zeta\leftrightarrow$ and $\zeta\Leftarrow$ are found, so the coordinates are $\leftrightarrow = \leftrightarrow_0 + \zeta\leftrightarrow$, $\Leftarrow = \Leftarrow_0 + \zeta\Leftarrow$

It is impossible to measure two altitudes simultaneously in flight, so it is necessary to take into account corrections of position change M for time Δt between altitude measurements. So, it is necessary to shift the first line of position *a* on the value of the path projection travelled for Δt to the direction of line *1*.

As it has been shown, the accuracy of graphic-analytical method significantly depends on the difference of azimuths. Sextant measurement errors are also significant. However, substitutions of circles sections by lines may lead to significant errors only at small zenith distances.

In celestial navigation systems implementation there are some interesting methods like celestial orientation by one body for example by method of altitude and azimuth. Aircraft coordinates are determined as intersection of circle of equal altitude and curve of equal azimuth. However, the curve of equal azimuth A = const cannot be constructed accurately enough, because in the direct measurement of azimuth it is necessary to know aircraft true heading, which in contrast to the body altitude is measured by airborne compass system with significant errors. Because of the low accuracy of this method, as well as other possible methods for determining the coordinates by one body, which use, for example, information about the body altitude and rate of its change (line of position $\dot{h}$ Xconst), are not widely used.

When solving astronavigational problems analytically, the input parameters are usually measured in the horizontal coordinate system, including altitudes and heading angle. Star trackers using these input values are called horizontal. Like in the horizontal astrocompasses, they use celestial body verticals as direction finding planes.

Another group of celestial navigation methods, called geometric is based on direct simulation of relative position of equatorial and horizontal systems of astronomical coordinates using star tracker block like it is done in the equatorial astrocompasses. Star trackers which implement such methods are called equatorial because their direction finding planes are circles of celestial body declination. The advantage of such systems is the possibility of direct measurement of location coordinates and heading without computing devices. However, such systems have much more complex kinematics.

4.2.2. Principles of star tracker construction

Let us consider the principle of star tracker construction on the example of horizontal star-solar tracker (SST) of БЦ-63 type. It is designated to determine the aircraft geographic coordinates (by method of circles of equal altitudes) with direction finding of two stars by automatic sextants, which measure altitude and heading angles of stars. In the daytime flight after automatic or manual entering the aircraft coordinates, БЦ-63 is used as horizontal astrocompass to measure the true heading by the Sun.

БЦ-63 uses following solution of equations by method of two altitudes:

$$\begin{aligned}
&\cos h_1 \cos A_1 \,\mathrm{X} \cos B \sin \Omega_1 \,\mathrm{Z} \sin B \cos \Omega_1 \cos(S_{gr}\, \Gamma\, L\, \mathrm{Z} \Im_1);\\
&\cos h_2 \cos A_2 \,\mathrm{X} \cos B \sin \Omega_2 \,\mathrm{Z} \sin B \cos \Omega_2 \cos(S_{gr}\, \Gamma\, L\, \mathrm{Z} \Im_2);\\
&\cos h_1 \sin A_1 \,\mathrm{XZ} \cos \Omega_1 \sin(S_{gr}\, \Gamma\, L\, \mathrm{Z} \Im_1);\\
&\cos h_2 \sin A_2 \,\mathrm{XZ} \cos \Omega_2 \sin(S_{gr}\, \Gamma\, L\, \mathrm{Z} \Im_2)
\end{aligned} \qquad (4.5)$$

Two last equations in (4.5) refer to (1.21) and the first ones are derived from so-called formula of five elements of spherical trigonometry.

Equations (4.5) are solved in БЦ-63 using computing devices. As a result of solving with known values S_{gr} and equatorial coordinates of celestial bodies $\Im_1, \Im_2, \Omega_1, \Omega_2$ and measured altitudes h_1, h_2 we receive azimuths A_1 A_2 of celestial bodies and geodetic coordinates B, L of aircraft. The geodetic coordinates – coordinates on ellipsoid ($L = \Leftarrow$ and $B = \leftrightarrow + \Uparrow$, where $\Uparrow$ is amendment for ellipsoid oblateness) are taken because true astronomical measurements are based on true (geodetic) vertical.

One of the obtained values of azimuth is used to determine the true heading

$$э = A\ \mathrm{Z}\ \wp \mathrm{E}$$

where э is true heading of aircraft; $\wp$ is relative bearing (heading angle between the horizontal projections of the longitudinal axis of aircraft and the direction to the celestial body).

Geodetic coordinates B, L are converted into great circle ones by formulas of spherical trigonometry. Also true track angle $\wp_{gc}$ of great circle route is calculated as an angle between the geographic meridian and great circle parallel, required to convert the true heading э into great circle $э_{gc}$ which is counted from great circle parallel.

Simplified block diagram of БЦ-63 is shown in Fig. 4.16.

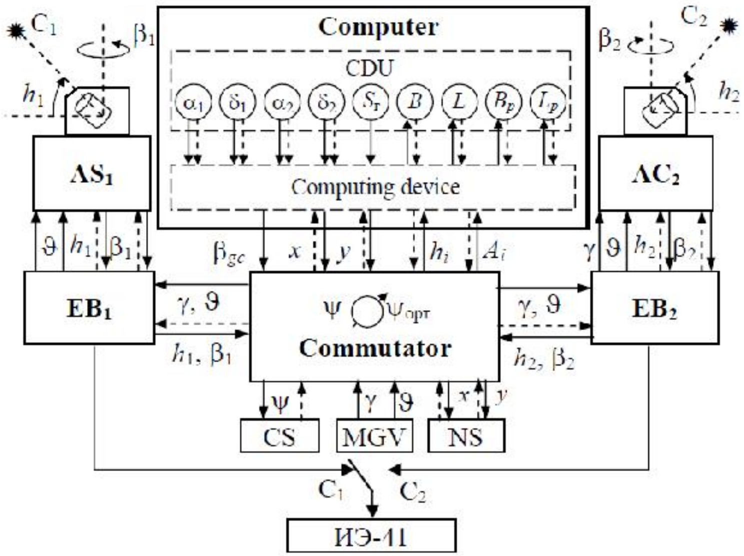


Fig. 4.16

Its main blocks are electromechanical computing device that solves equation (4.5) and performs the coordinate conversion, and two automatic sextants (AS) with electronic blocks (EB) which measure altitudes h_i and heading angles $\wp$ of two bodies (in the daytime they measure altitude and heading angle of the Sun). Commutator is designed to calculate the heading angles of celestial bodies, great circle and true headings, as well as for electrical connection of БЦ-63 blocks.

Horizontal stabilization of platform of sextant optical system is provided by roll $\uparrow$ and pitch ν signals from vertical simulator, e.g. master gyro vertical. Star tracker solves the problem of determination of aircraft position and heading, working sequentially in celestial body guidance mode and tracking mode.

For prior guidance of sextant on certain celestial bodies, and to exclude ambiguity of solution of equations (4.5), the adjusting mode is provided. Adjustment of star tracker on the selected bodies is made by solving the inverse problem. From the control display unit (CDU) the following parameters are entered in the computing device: the equatorial coordinates of celestial bodies $\Im_{i0}$, δ_{i0} and Greenwich sidereal time S_{gc0} (further changes are taken into account by the clockwork mechanism); aircraft coordinates B_0, L_0 and heading; coordinates of the great circle pole B_p, L_p. In flight adjustment of star-solar tracker the great circle coordinates x, y, coming from navigation system are used and converted into B_0, L_0.

Computer outputs the values h_{i_0}, A_{i_0} and $\wp_{gc_0}$. Knowing initial heading э or $э_{gc}$ which is entered in the commutator block by readings of compass system (CS) and controlled by the star tracker heading, in the commutator block by equations (2.6) the heading angles of celestial bodies $\wp_0$ are calculated. The obtained values h_{i_0}, $\wp_0$ are applied to automatic sextants. Sextants are rotated in the direction of the selected bodies. This mode is shown in Fig. 4.16 with dotted lines.

To capture the celestial body, considering the field of view of optical system (its half value is equal to the 1.5⌠) the aircraft coordinates and the heading must be known with accuracy up to 1⌠.

Operator, using special indicator ИЭ-41, controls star tracker operation in guidance mode. When the image on the screen indicates that celestial bodies are in the field of view of optical systems (telescopes), the operator switches sextants in the tracking mode. Tracking systems align the telescope optical axes with the direction to the celestial body, resulting in correction of h_i and q values.

The calculated values of altitude h_i are compared with altitudes h_{AS} measured by automatic sextants.

Knowing the difference $\Delta h_i = h_{AS} - h_i$ and calculating the azimuth A_i allows us to determine the latitude and longitude corrections ΔB, ΔL and thereby correcting the coordinates of airplane

$$B = B_0 + \Delta B; \quad L = L_0 + \Delta L.$$

Relationship between corrections ΔB, ΔL and measured deviations of altitudes Δh_1, Δh_2 is determined by

$$\begin{aligned} \Delta h_1 &= \Delta B \cos A_1 - \Delta L \cos B \sin A_1 \\ \Delta h_2 &= \Delta B \cos A_2 - \Delta L \cos B \sin A_2 \end{aligned} . \quad (4.6)$$

Solving equations (4.6), it is possible to find the increments of latitude and longitude:

$$\Delta B = \frac{\sin A_2 \Delta h_1 - \sin A_1 \Delta h_2}{\sin(A_2 - A_1)}$$
$$\Delta L = \frac{\cos A_1 \Delta h_2 - \cos A_2 \Delta h_1}{\cos B \sin(A_2 - A_1)} .$$

Thus, by measured angles h_i the coordinates B, L and celestial body azimuths A_i are calculated. Like in the horizontal astrocompass one of the obtained azimuth values A is used to determine the true heading by formula:

$$\psi = A_i - q.$$

Coordinates B, L are converted to great circle ones x, y. Great circle coordinate system in БЦ-63 is set by the geodetic coordinates of its «northern» pole – great circle pole. Great circle equator usually passes through the initial and final waypoints.

Formulas of conversion from B, L to great circle coordinates
x, y can be obtained from spherical triangle MP_NP_{gc} on the globe surface (Fig. 4.17, *a*), where M is aircraft position, P_N is the North pole of the Earth, P_{gc} is great circle pole, C is initial waypoint:

$$\sin x = \sin B_p \sin B + \cos B_p \cos B \cos(L_p - L);$$
$$\mathrm{tg}\, y = \cos B_p \,\mathrm{tg} B \,\mathrm{cosec}(L_p - L) - \sin B_p \,\mathrm{ctg}(L_p - L),$$

where B_p, L_p are geodetic latitude and longitude of great circle pole.

To determine the great circle heading, the track angle of great circle parallel q_{gc} as angle between geographical meridian and great circle parallel (q, q_1) is calculated continuously in the star tracker

$$\mathrm{tg}\, q_{gc} = \cos B \,\mathrm{tg} B_p \,\mathrm{cosec}(L_p - L) - \sin B \,\mathrm{ctg}(L_p - L).$$

By calculated values of ψ and q_{gc} the great circle heading is determined (Fig. 4.17, *b*)

$$\psi_{gc} = \psi - q_{gc} .$$

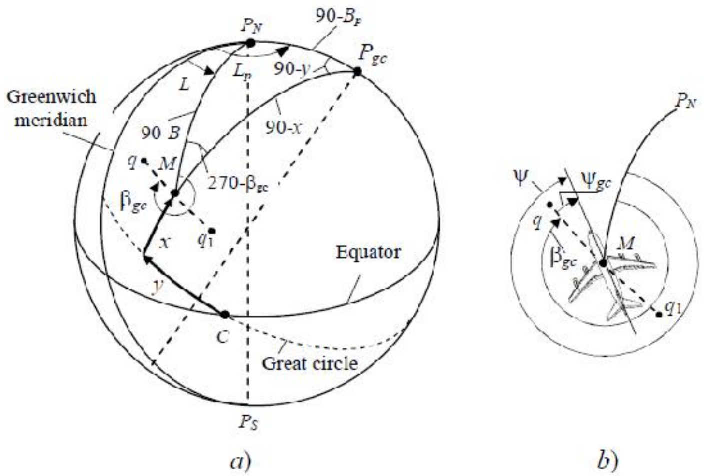


Fig. 4.17

Heading signals can be used for compass system correction, and *x*, *y* signals are used to correct navigation system.

If БЦ-63 is working in mode of single celestial body, to determine heading the periodic manual input of coordinates *B*, *L* into computer or automatic input of *x*, *y* coordinates from navigational system is required.

4.3 Errors of astronomical navigation aids

Like any information systems the astronomical navigation systems have methodical and instrumental errors. Measurement method errors are caused by the following reasons: errors of measurement of the altitude and heading angle of celestial body arising, in particular, through the optical distortion; input errors in the coordinates of the aircraft location and of the body coordinates; time dead reckoning errors; banking of direction finders, etc.

Instrumental errors depend on the constructive shortcomings of the device, and their total value is limited by the device tolerance.

The main sources of measurement method errors in measuring the altitude angle of celestial body are astronomical refraction, parallax of the body, semidiameter of celestial body.

Astronomic refraction is the refraction of light beam in the atmosphere of the Earth, as a result of that the visible direction to celestial body rises above the horizon. The main reason of the refractive phenomenon is a change of air density with height increase.

The density of air in the lower layers of atmosphere is greater than in the upper layers that is why the beam coming from the body *S*, is distorted (Fig. 4.18).

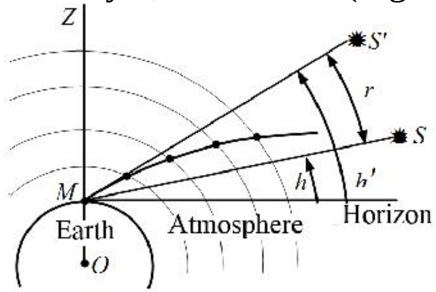


Fig. 4.18

The observer, located at point *M* sees the body at the point in the direction of the tangent to the curve of light beam path, i.e. in the direction *MS*’. The angle *S*’*MS* between the visible direction to the body *MS*’ and the true direction *MS*’, for which the body would be observed in the absence of atmosphere, is called the astronomical refraction (*r*).

The trajectory of light beam is located in the plane of body vertical, so the refraction changes only the altitude angle. The higher the altitude angle *h* is, the less the refraction will be. When the celestial body is on the horizon, the refraction is maximal and reaches 35'. Refraction depends on the density of air ∂. With increasing altitude the air density decreases, and therefore the refraction becomes smaller, thus $r = f(h, \partial)$.

In aviation astronomical yearbook the coordinates of celestial bodies are given relative to the center of the Earth, that is assumed that the center of the celestial sphere coincides with the center of the Earth, but astronomical measurements are performed on the ground surfaces, and this causes the phenomenon of so-called parallax. *Parallax of celestial body* is called the angle *p* (Fig. 4.19) between the direction from any point of the ground surface to the body and the direction from the center of the Earth to the same body.

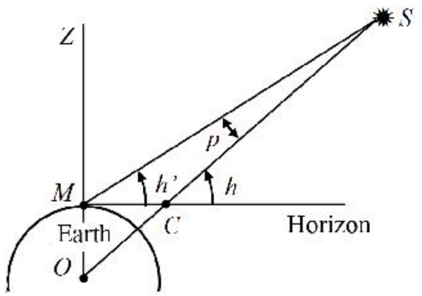


Fig. 4.19

Let us suppose that the observer is at point *M*, and the body is at point *S*. The following reference designations are used: *h* is geocentric altitude angle, *h*' is apparent altitude angle, $R = OM$ is the radius of the Earth, $D = OS$ is distance from the center of the Earth to the body. The angle *h* is the external angle of triangle *SMC*, then

$$h = h' + p,$$

where *p* is parallax of the body.

From the triangle *SOM* the following expression is obtained:

$$\sin p / R = \sin(90 + h') / D,$$

then

$$\sin p = R/D \cdot \cos h'$$

The largest parallax is formed in case of the body position on the horizon. The horizontal parallax of the Moon is 53...62'; 0.55' for Venus; 0.4' for Mars; 8,8'' for the Sun; parallax of stars is negligible small.

Semidiameter of celestial bodies is an error that occurs when there is direction finding of bodies with the disc shapes. Coordinates of all bodies given in handbooks are coordinates of the body centers, and in the measurement of altitude angle sometimes are not the center of the body, but its upper or lower edge of the visible disk is under direction finding.

The errors from optical distortions are related to the *methodical errors* of astronomical navigation aids. The errors of optical distortions occur when the sunlight passes through the Earth's atmosphere and aircraft astrodome. They also include interference from light reflected from clouds, etc.

Methodical errors of astronomical measurements and correspondingly of navigation parameters also arise due to difference of coordinates of location and of the body, which are entered the computer, from the true ones in the values Δφ, Δλ, Δt_{gr0} and Δδ.

For example, the error of azimuth calculation in horizontal astrocompass may be associated with episodic entry of φ and λ. The azimuth error due to inaccuracy of calculation of initial angle t_{gr0} (because of errors of navigational watches) is insignificant. The same is related to the error Δ*A* due to neglecting of changes in the Sun declination (δ) during the flight. Total azimuth error Δ*A* is defined as

$$\Delta A = \Delta\varphi \sin(\mathrm{tg}h) + \Delta\lambda(\sin\varphi - \cos\varphi\ \mathrm{tg}h \cos A).$$

Thus, error Δ*A* depends on the values of *A* and *h*, that is, on the location and time, in particular, it increases with altitude angle *h*. On the other hand in the afternoon time when the Sun azimuth *A* is close to 0 or π, the error Δ*A* is minimal.

To reduce Δ*A* (Δφ, Δλ) the intervals of input φ and λ should be reduced, and this complicates the navigator work. This problem does not exist if astronomical systems are part of the flight and navigation complex that continuously gives information about coordinates φ and λ in astronavigation system.

The methodical error of great circle heading measurement by horizontal astrocompass occurs because of deviation of rotation axis of direction finding head from the calculated position. The reasons for deviation of the rotation axis can be the aircraft banking, rotation of aircraft symmetry plane from the great circle route plane (e.g. at drift angle), lateral deviation of aircraft from the great circle route, errors of ground speed entry to the track controller. Measurement error of great circle heading increases with increasing the traveled distance.

The equatorial compasses have the methodical errors as for horizontal compasses. The errors are caused by change of zenith distance of the body as a result of atmospheric refraction, by inaccuracies in coordinates of aircraft location and of geographic location of the body, by tilting of direction finding plane of bodies.

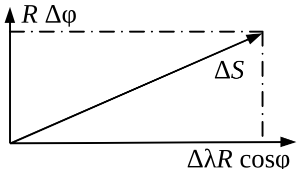


Fig. 4.20

For star trackers the methodical error of dead reckoning Δ*S* depends on the choice of celestial bodies and is determined (Fig. 4.20) by geometrical sum of errors in determining the latitude and longitude:

$$\Delta S = \sqrt{(R\Delta\varphi)^2 + (\Delta\lambda R\cos\varphi)^2}.$$

Using the ratio

$$\begin{aligned}\Delta h_1 &= \Delta\varphi\cos A_1 + \Delta\lambda\cos\varphi\sin A_1;\\ \Delta h_2 &= \Delta\varphi\cos A_2 + \Delta\lambda\cos\varphi\sin A_2,\end{aligned} \qquad (4.7)$$

after simple transformations there will be the following:

$$\Delta S = \frac{R}{\sin(A_2 - A_1)}\sqrt{\Delta h_1^2 + \Delta h_2^2 - 2\Delta h_1\Delta h_2\cos(A_2 - A_1)}. \qquad (4.8)$$

Thus, the errors in determining the aircraft location strongly depend on the difference of azimuth of navigation bodies. The smallest error will be for azimuth difference close to 90°. In practice, it is necessary to choose navigation bodies with azimuth difference in the range 30°... 90°.

To clarify how the methodical errors of dead reckoning ΔS depend on errors of keeping the vertical by automatic sextant of star tracker, let us consider Fig.4.21. It shows the horizontal plane and the plane of sextant stabilization, which is declined from the horizon line at angle α.

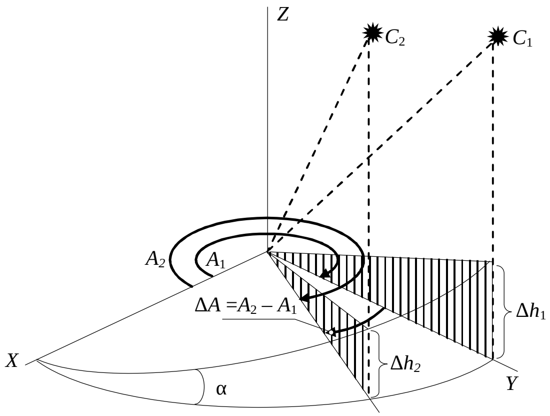


Fig. 4.21

This angle corresponds to errors in determining the altitude angles of bodies:

$$\Delta h_1 = \alpha \text{ and } \Delta h_2 = \alpha \cos(A_2 - A_1).$$

Then the error in determining the location coordinates ΔS taking into account (4.6), (4.7) is defined by the expression:

$$\Delta S = R\Im.$$

Thus, the error ΔS of vertical keeping errors is independent on the azimuth difference. However, the requirements for vertical keeping must be quite severe since the error of vertical keeping in one angular minute causes the error in determining the aircraft location in one nautical mile (1.85 km).

Banking methodical error of astronomical navigation aids occurs when the direction finding plane tilts through changing angles of pitch ν and roll ↑. The expression for banking methodical error $\Delta\wp$ of astronomical aids is taken directly from the general formula of banking error of compass devices:

$$\zeta\wp = \mathrm{arctg}\frac{(\cos\wp\sin\nu\,\sin\uparrow - \mathrm{tg}\,h\cos\nu\,\sin\uparrow + \sin\wp\cos\uparrow)}{\cos\wp\cos\nu + \mathrm{tg}\,h\sin\nu} - \wp$$

Compensation of banking methodical error can be made either by stabilization of direction finding device from the vertical, such as master gyrovertical, or by the entry of amendments to the computing device.

Instrumental errors of astronomical navigation tools consist of errors of direction finder, errors of the vertical, computing device errors, indicator errors, errors which occur due to refraction of astrodome. These errors depend not only on the parameters of separate parts and components, but also on external conditions: overloads and vibrations, temperature, humidity, supply voltage. In addition, theses errors change over time due to aging of materials, changes in the properties of oils and others. It is impossible to estimate the value of all errors, that is why usually the most significant and the most probable errors are calculated.

Errors of direction finder arise from the presence of the dead zone, static and dynamic errors of photo tracking device. These errors appear both in the plane of body vertical and in the transverse plane Z plane of heading angle.

Refraction errors of astrodome are caused by different geometric shape of astrodome Z spherical or flat. Astrodome of spherical shape is deformed under the influence of airflow, irregular heating of surface and pressure difference between the atmosphere and the air in the cabin of aircraft. Deformation of astrodome causes the refraction, that is, the deflection of light beam coming from the celestial body.

For flat astrodome there are deformations caused by pressure difference between the atmospheric and cabin airs and by temperature difference on the outer and inner surfaces of the glass. The latter causes unequal expansion of external and internal surfaces of glass, resulting in convex astrodome.

Refraction of astrodome can be decomposed into two components - one is in the plane of body vertical (leads to errors in the measurement of altitude angle), the other is in the plane of the heading angle.

4.4 Features of astronomical navigation aids operation

Operation of horizontal astrocompass of ДАК-ДБ series. Functional test before the flight in daytime conditions and the astrocompass switching on in flight are done in the following order. At first, Greenwich time is determined by the formula:

$$T_{gr} = T_d - (N + 1),$$

where T_d is daylight saving time (readings of navigator watches); N is number of time zone.

Then Greenwich hour angle t_{gr0} and declination δ of the Sun are found by determined value T_{gr} for the observation date from the aviation astronomical yearbook. After powering on and running the clockwork mechanism of astrocompass, the object coordinates φ, λ and body coordinates δ, t_{gr0} are entered in the azimuth calculator. When using astrocompass in the southern hemisphere it is necessary to enter values (-δ) and (-φ) instead δ and φ, and the switch "North - South" must be in the position "South".

To determine the true heading on the scale of track controller, zero values of traveled path and ground speed are set using the corresponding knobs. Observations of the Sun without overshooting and oscillations are adjusted by the rheostat of sensitivity adjustment in photo tracking system.

When using astrocompass to fly by great circle route, the coordinates φ_0, λ_0 of initial waypoint and value δ and t_{gr} are entered the computer in advance. In the moment of initial WPM passing the ground speed is set for track controller. Further in the flight by astrocompass index the computed heading is kept. The tilt angle of the direction finder head axis can be adjusted in accordance with the actual traveled path *S* by the knob "Path" on the control display unit in the check points of great circle route. After the flight of great circle route segment of

1100 km, the rotation of this knob sets zero value of S, and the direction finder head axis takes the vertical position. The coordinates φ, λ of point $S = 0$ are entered the computer, and the heading by astrocompass indicator is specified for new values of heading angles $\wp_\theta$.

At night astrocompass can be used together with periscope sextant for occasional determination of the true heading ψ. With the help of sextant the direction finding of the Moon, planets or stars is done visually, and the heading angle of the body is measured. The astrocompass computer where coordinates of object φ, λ and of the body are entered, determines the azimuth of celestial body. If the true heading э is known from some other compass device, then the detection and direction finding of stars are simplified in sextant. For this the computer is supplied with the coordinates of object and star. The drift sight system of sextant is rotated in azimuth until the readings of astrocompass indicator become equal to э. As a result, the direction finding plane of sextant will be close to the body vertical, and to make the star in the field of view of sextant, it is enough to specify the guidance by its altitude angle.

To control the photo tracking system in flight there is a button on the front panel of azimuth computer. By pressing it, the circuit of photo elements is de-energized. Through another contact group, the signal on one of the amplifier stage is supplied, resulting in the rotation of direction finder head by motor. After releasing the button the initial indications of heading must be restored (in case of constant heading). If there is periscope sextant, then to check astrocompass serviceability it is recommended to perform the direction finding of the Sun via sextant. If sextant and astrocompass are serviceable, then the obtained value of heading must not differ more than in 1-2°.

To check astrocompass there is checkout equipment which allows us to check the instrumental errors, the gain coefficients of amplifiers, the precision of clockwork mechanism in ground conditions.

Features of astronomical navigation systems operation. In flight, to adjust the star tracker the great circle coordinates x, y can be used from the navigation system of dead reckoning numerous ways. These coordinates in star tracker computer are converted in B_0, L_0.

The operation of photo tracking system is controlled by operator on special indicator. When the form of images on the screen indicates that the celestial bodies are in the field of view of optical systems (telescopes), the operator puts the sextant in tracking mode. Photo tracking system combines the optical axes of telescopes with the directions to the bodies, resulting in specifying the values h_i and $\wp_i$. The operating mode of star tracker is realized. By measured angles h_i the coordinates B, L and azimuths A_i of bodies are calculated. Coordinates B, L are converted into the great circle ones x, y, and the angle $\wp_\omega$ is found. The signals x, y are used to correct the navigation system. By measured angles $\wp_i$ and by calculated values A_i and $\wp_\omega$, the great circle or true heading is determined. The heading signals can be used to correct the compass system.

For star tracker operation by single celestial body, the occasional manual input of coordinates B, L in computer or the automatic input of coordinates x, y from the navigation system are required for determination of heading (i.e. angles A_i and $\zeta\omega$).

Errors of star trackers strongly depend on the difference of body azimuths. For this reason, in star tracker the range of azimuth difference $|\zeta A|$ is limited by value $90 \pm 60°$.

Errors of horizontal stabilization of sextant also lead to the errors of coordinates determination. And for horizontal star trackers of star-solar type with telescopes stabilized in the horizontal plane by signals of the pendulum erected gyroverticals, this error is the main cause of dead reckoning errors. High precision of star trackers is achievable only when using signals of inertial gyroverticals, undisturbed by accelerations of aircraft motion.

TEST QUESTIONS

1. What groups can the astronavigation systems be divided into?
2. What formula determines the true heading in horizontal compass?
3. What method is used to determine the azimuth of body in horizontal compass?
4. What is the main functionality of spherant? How is the change in local hour angle taken into account in spherant?
5. How is it necessary to deflect the direction finding plane for providing the great circle route flight?
6. List the components of horizontal astrocompass ДАК-ДБ.
7. Which device is used in ДАК-ДБ to exclude errors in measurements of heading angles with aircraft banking?
8. What data are entered the azimuth computer of ДАК-ДБ in case of using astrocompass in the southern hemisphere?
9. How is direction finding plane oriented in the equatorial astrocompass unlike horizontal ones?
10. How is the true heading determined in the equatorial astrocompass?
11. What is functionality of polarizing sighting system of astrocompass?
12. What method of astronavigation is the most widely used in astronavigation systems?
13. How is the ambiguity of the coordinates determination in astronavigation systems excluded?
14. What is the influence of azimuth difference on the accuracy of two altitudes method?
15. What parameters are entered the computer БЦ-63 for preliminary guidance of sextant to the selected stars?

16. How is the operation of star tracker controlled in the guidance mode?

Chapter 5. Radio altimeters

5.1 Functionality and principles of radio altimeter operation

Aerometric method of measuring the altitude of aircraft flight is based on the method of barometric leveling. This method of measuring the altitude depends on several factors, and neglecting them leads to significant errors in altitude determination. There are, for example, errors of setting the initial pressure and temperature, neglecting of non-linearities of temperature changes in the inversion layers of atmosphere, etc. Also, the relief of locality can be not always taken into account accurately, for example, in the aerodrome area to calculate the true flight height, especially in cloudiness conditions.

These errors can lead to relative error in measuring the absolute altitude in the range of 10...20%, and sometimes more.

Therefore there were attempts to build an altimeter on any other principle. In particular, to realize the idea of measuring the height is based on the sound echometer.

In echometer the strong sound signal is emitted, which after reaching the ground surface, is reflected from it and comes backward to the aircraft. On the aircraft the time interval between the moment of emission of main signal and the time of arrival of the reflected signal is measured. Knowing the speed of sound propagation and the measured time interval it is possible to determine the distance traveled by sound signal. Obviously, this distance equals the doubled flight altitude of the aircraft.

However, the height measured by echometer will also have errors. These errors arise because of instability of the speed of sound propagation and because of different reflectivity of the ground surface. Noises of aircraft engines also introduce errors in measuring the absolute altitude. Much better results are reached by using radio signal since the speed of radio waves propagation is constant and does not depend on the propagation medium. This idea is the basis of radio altimeter (RA) construction.

Radio altimeters are designed to measure the absolute altitude of aircraft flight. They belong to the class of autonomous radionavigation systems, because they do not need additional ground equipment to implement the measurement channel.

In radio altimeters the radar principle for determining the height by the reflected signal is used. Radio altimeter transmitter generates radio waves which by means of transmitting antenna are directed to the ground surface. The reflected signal is taken by the receiving antenna and further comes to the receiver. In computing device of receiver the signal is produced proportional to the travel time of radio waves to the ground surface and backwards, that is, proportional to the absolute altitude. Depending on the maximal measured height the radio altimeters are divided into RA of small and large heights.

Radio altimeters of small heights are designated to measure the flight altitude of aircraft in the range from 0 to 1500 m. They are used mainly for aircraft control on the stages of landing approach and landing and during flights at low altitudes. In RA of this type so-called frequency method of measurements of height is widely used, the main advantage of which is small limits of measured height and high accuracy.

In frequency method the RA transmitter continuously with constant power emits continuous frequency-modulated signal (frequency of signal is rapidly changed with time between the highest and lowest values). The receiver in each moment of time receives two signals. One signal comes directly from the transmitter to the receiver; another signal is reflected and comes from the transmitter to the ground and from the ground to the receiver. It is obvious that the frequencies of two received signals are different, and the difference of frequencies depends on the time interval between receiving the main signal and the reflected one. By measuring the frequency difference, it is possible to calculate the time delay of the reflected signal which is proportional to the absolute altitude. Measuring the frequency difference can be made with very high accuracy which allows measuring the absolute altitude of the aircraft with the accuracy necessary for low-altitude flight, that is, within accuracy of 2...3 m.

However, the frequency altimeters cannot provide measuring of high altitudes, because it needs to increase power of transmitter to get the reflected signal suitable for processing. This causes significant increase in the weight and dimensions of the altimeter.

Therefore, to measure high altitudes (up to 8...10 km) the pulsed radiation is used. Radio altimeters of high altitudes are used as navigation aids for aerial photography of terrain and for other purposes.

The impulse method of height measurement includes the transmitter radiation by pulses with constant frequency. Impulses are generated through small intervals. In the pauses between the pulse radiation the energy is accumulated in the transmitter and then is emitted in a pulse. Thus, the pulse energy is in many times higher than the average transmitter power. Measuring height in this case is done as a delay measurement of the reflected pulse coming from the ground compared to the arrival of direct pulse in the receiver.

Accuracy of pulse altimeter is less than of frequency altimeter, but significantly exceeds the accuracy of barometric one.

Radio altimeters unlike barometric devices of measuring altitude provide the possibility to determine the absolute altitude of the aircraft quite accurately directly without complex calculations and without entering additional information. Using radio altimeter it is also possible to determine whether aircraft flies over land, over the sea or the mountains.

5.2. Radio altimeter of low altitudes

5.2.1. Frequency method of measuring altitude

The frequency method of altitude measurement is based on the frequency modulation of the emitted oscillations. Figure 5.1 shows simplified block diagram of RA of low altitudes with frequency method of altitude measurement. Figure 5.2 shows time diagrams at different points of the circuit.

Frequency modulator produces low-frequency modulating voltage U_{mod} that controls the oscillations frequency of high frequency generator (HFG). The modulation by symmetric linear-segment law is widely used (Fig. 5.2, *a*). It is also possible to use modulation by sinusoidal law and by asymmetric saw tooth law (used in RA A-034). Frequency modulation law does not provide significant effect on the altimeters operation.

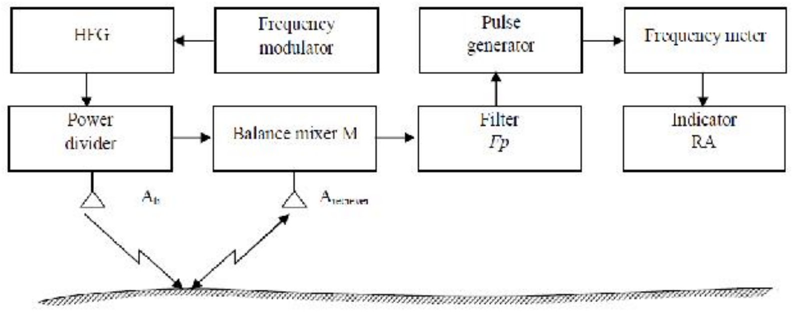


Fig. 5.1

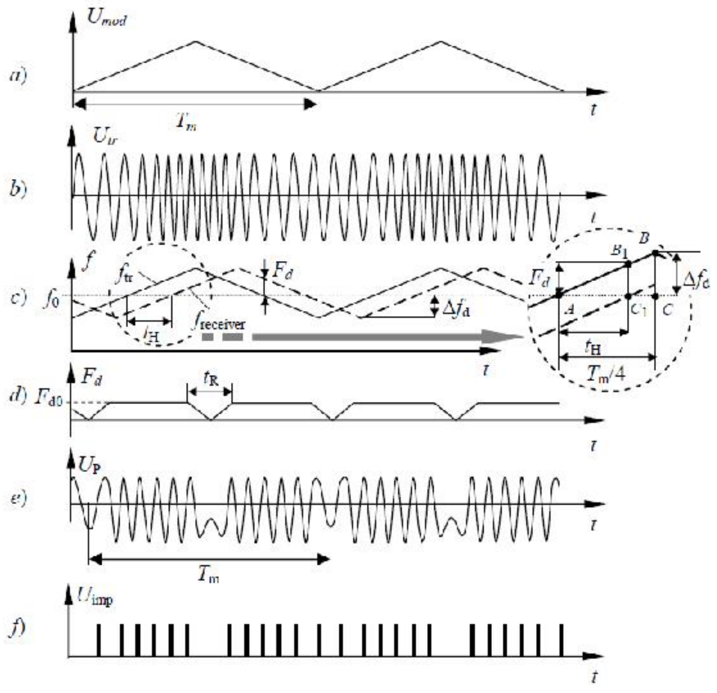


Fig. 5.2

Under the influence of modulated voltage U_{mod} of frequency modulator, HFG produces continuous oscillation U_{tr} (transmitter) (Fig.5.2, *b*), the frequency f_{tr} of which changes relative to the average value f_0 on the value of deviation frequency $\pm \Delta f_d$ with modulation period T_m.

Signals of HFG via power divider (Fig. 5.1) enter the transmitting antenna (A_{tr}) and are radiated in the direction to the ground. The signal reflected from the ground surface is received by receiving antenna ($A_{receiver}$) and comes to the balance mixer (M), to where simultaneously the part of the power oscillations as heterodyne transmitter signal is branched and applied.

Two separate antennas (transmitting and receiving ones) provide uncoupling of the receiver and transmitter of RA as radar with continuous emission. In RA the horn antennas are commonly used, forming symmetrical, matching in space diagrams with width to 50°, directed vertically downward. Diagrams are selected relatively wide to ensure the constant contact with the ground under aircraft maneuvers.

Reflected signal exactly repeats the probing signal in the form, but with a time delay:

$$t_H \text{ X} \frac{2H}{c},$$

where H is flight height; c is speed of light.

The dependence of frequency of the received signal $f_{receiver}$ on time is shown in Fig. 5.2, *c*. It varies by the same line-segment law, as the frequency of radiated signal f_{tr} but with delay in time t_H:

$$f_{receiver}(t) = f_{tr}(t \text{ Z } t_H).$$

At the current time moment t the same signal is received that was emitted before at the time ($t \text{ Z } t_H$).

At the output of the mixer M (Fig. 5.1) with the help of filter *Fp* the frequency difference is separated which is equal to the absolute value of the difference of the instantaneous frequencies of the emitted and received oscillations:

$$F_d = |f_{tr}(t) \text{ Z } f_{receiver}(t)|.$$

Frequency difference separated in the detector of low frequency of the filter F_d serves as a measure of the absolute flight altitude.

In Fig. 5.2, *c* $F_d(t)$ is determined as the vertical distance between the segment lines $f_{tr}(t)$ and $f_{receiver}(t)$.

Frequency difference is determined by examining the triangles *ABC* and AB_1C_1 (see enlarged zoom in Fig. 5.2, *c*):

$$AC_1/AC = B_1C_1/BC, \quad Fp = B_1C_1 = AC_1 \cdot BC/AC.$$

So, the frequency difference is:

$$F_d = |f_{tr}(t) \text{ Z } f_{receiver}(t)| = [\Delta f_d/(T_m/4)]t_H,$$

and considering that $t_H = 2H/c$ it is possible to obtain:

$$F_d = [8\Delta f_d/(cT_m)]H,$$

that is, F_d is proportional to the absolute altitude H.

The greater the average value of the frequency f_0 is and the greater the value of the frequency deviation is, the greater the sensitivity of radio altimeter will be.

The average value of the frequency equals f_0 = 400...600 MHz and frequency deviation is ± 0.5% from the average value at the range of measured altitudes from 0 to 1500 m.

At landing of the aircraft the sensor switches to the low range (0...150 m), and the frequency deviation increases 10 times more, up to ± 5% from the average frequency.

Figure 5.2, *d* shows the dependence $F_d(t)$ on time, Figure 5.2, *e* shows the voltage of frequency difference $U_d(t)$.

At intervals of duration *t*, called reversal zones (Fig. 5.2, *d*), $F_d(t)$ decreases from F_{d0} to zero and increases to F_{d0} again. The frequency F_{d0} is called the main one, or rangefinder, or maximal difference frequency. Cycles of changes $F_d(t)$ are repeated with half-period of modulation $T_m/2$.

In practice, the modulation period T_m is selected high enough in order to fulfill the relation T_m>>t for any altitudes. Then reversal zones will be relatively small, and it is possible to assume that the average difference frequency during the period T_m is equal to F_{d0}: $F_{d.av.} \text{ β } F_{d0}$. In this case the problem of altitude measurement is reduced to measure $F_{d.av}$ of the voltage of difference frequency $U_d(t)$.

At the output of the filter of difference frequency F_d (Fig. 5.1) the oscillations $U_d(t)$ using generator of pulses are converted into sequence of pulses (Fig. 5.2, *f*). Standard impulse is formed each time with passing through zero of voltage $U_d(t)$ from minus to plus.

Frequency of pulse passing is determined by frequency meter. The average frequency of pulse passing is equal to mean-square frequency of voltage $U_d(t)$ in the absence of obstacles.

Counter of frequency meter produces the voltage proportional to the number of pulses which are received, and inertial link in the frequency meter averages this voltage.

Indicator of RA (Fig. 5.1) shows the measured altitude taking into account the scale factor.

Depending on the bandwidth of the amplifier of converted signal in filter F_d, the frequency radio altimeters are divided into narrowband and broadband ones.

Broadband RA has the band amplifier, the bandwidth is $\zeta F_{a.av.}$ ΨΨζF_s, where ζF_s is spectrum width of converted signal. Extreme frequencies F_{min} and F_{max} of bandwidth define the range of measured altitudes.

Narrowband radio altimeters are widely used as precision measuring instruments of small heights, particularly, in the low-altitude flight systems or at automatic landing of aircraft.

In narrowband RA the bandwidth of processing channel of converted signal is selected close to the width of the spectrum of the converted signal ζF_s. The servo system is used that allows combining the average frequency of converted signal with frequency f_0 of channel tuning. These radio altimeters have high accuracy even at low signal-to-noise ratio at the input of RA, due to both the decrease in noise power at the input of frequency meter at narrowing of bandwidth of processing channel and the reduction of systematic error that occurs because of difference of middle frequencies of signal and noise.

Block diagram of narrowband radio altimeter is shown in Fig. 5.3). It operates by turn in the measurement mode and in the control mode with calibration of scale factor.

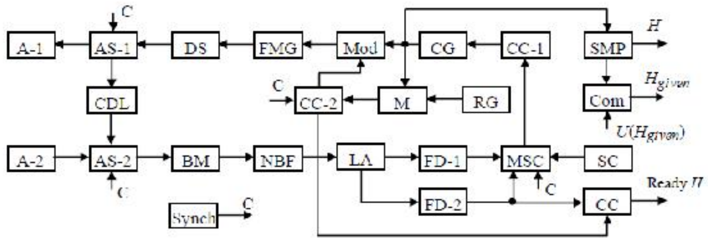


Fig. 5.3

Measurement mode is implemented in the channel consisting of controlled generator (CG); frequency-modulated generator (FMG); directional splitter (DS); transmitting and receiving antennas A-1 and A-2; balance mixer (BM); narrowband filter (NBF); limiting amplifier (LA); frequency discriminator (FD-1) and control circuit of modulating frequency of oscillations (CC-1).

According to block diagram, radio altimeter is closed-loop servo system with sensitive element FD-1 of constant tuning frequency f_0. At frequency f_0 the NBF is tuned, bandwidth of which ζF_{NBF} is close to the width of the spectrum of the converted signal ζF_s ($\zeta F_{\mathrm{NBF}}.\rceil\ \zeta F_s$).

Frequency discriminator produces the voltage proportional to the deviation of the average frequency ζF_{av} of converted signal spectrum from the tuning frequency f_0. This voltage is integrated in the CC-1 and is used to control the frequency of modulating voltage generator. Frequency of modulating voltage F_m changes toward the reducing the error:

$$\zeta F = \zeta F_{av}\ \mathrm{Z}\, f_0.$$

For preliminary rough adjustment of frequencies ζF_{av} and f_0 it is necessary to have the search circuit (SC) by which the frequency F_m gradually changes until spectrum of converted signal gets in bandwidth of NBF and the mode selection circuit (MSC) switches RA in tracking mode by frequency of converted signal. The last is usually chosen equal to 25 kHz. In the circuit the period of oscillation modulating is measured from the output of reference generator (RG). Sensor of modulation period meter (SMP) outputs the signal H which is proportional to flight altitude.

Checking mode is turned on several times per second using synchronizer (Synch). Control channel includes the main devices of basic channel and calibrated delay line (CDL), which is connected to the channel via antenna switches AS-1 and AS-2 and frequency discriminator FD-2. This part of the circuit operates in the same way as the basic one. The voltage from FD-2 is used to control the level of signal.

In the checking mode the constancy of scale factor is checked. Modulation frequency from CG (value of frequency is determined by delay in CDL and is constant in the checking mode) is applied to the BM, where oscillations from RG are also coming.

Frequency difference separated by mixer contains information about the value and sign of the frequency deviation at the output of CG from a given value in CDL and is used in the control circuit (CC-2) to change the amplitude of modulated oscillation, and, hence, the deviation of signal frequency. Control of amplitude is done in modulator (Mod).

Checkout circuit (CC) at presence of the converted signal and constancy (within established ranges) of scale factor generates readiness signal (Ready H). This signal indicates serviceability of RA. Comparison circuit (Com) is used for receiving the signal H_{given} at the height reducing to the given value.

5. 2.2 Radio altimeter PB-5

Radio altimeter PB-5 is designated to measure the absolute flight altitude in the range of 0...750 m and to alarm about aircraft reaching the given height, the value of which is preset on the indicator. Radio altimeter is a set of electronic equipment with antennas.

In the crew cabin on the flight deck only indicator is present that immediately shows the altitude and usually it functions as warning system about dangerous height with sound and light alarm.

General view of the set of radio altimeter PB-5M is shown in Fig. 5.4

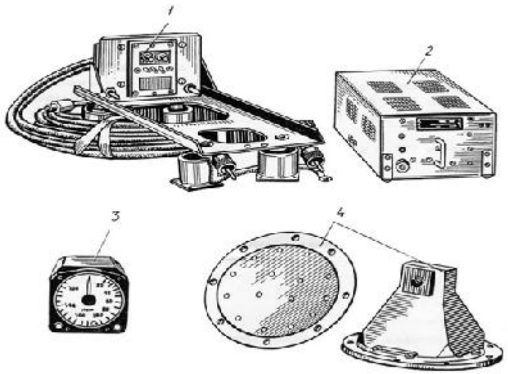


Fig. 5.4

The set of PB-5 includes frame *1* with high-frequency cables, transceiver *2*, altitude indicator *3*, antennas *4*.

Technical data of PB-5M:

- range of measured altitudes (1st range is 0...120 m; 2nd range is 100...1200 m);

- measurement error is 5% of the measured altitude;

- carrier frequency of transmitter is 444 ± 2 MHz;

- modulation frequency is 124 ± 3 Hz.

- radiated power is not less than 0.15 W.

The altitude indicator of PB-5 is device УВ-5 mounted on the flight deck. On the flange of the device the knob "SET ALTITUDE" is placed with built-in yellow signal lamp and there is "TEST" button with the built-in red signal lamp. Turning the knob "SET ALTITUDE" provides setting the warning of given altitude, whose value is evaluated by triangular yellow index that moves on indicator scale. When the aircraft reaches the given altitude, the yellow lamp blinking begins and in pilot headphone simultaneously for 3...9 seconds the audio signal beeps with the frequency of 400 Hz. By pressing button "TEST" on serviceable altimeter the indicator is set to the check altitude of 15 ± 1.5 m; in the absence of the button the radio altimeter shows the absolute flight altitude (or H = 0 at the ground).

Examples of the low-altitude radio altimeters are also altimeters of type A-037 (Fig. 5.5, *a*) and A-053 (Fig. 5.5, *b*).

Radio altimeter A-037 has no error; it is designed for light aircraft and helicopters, but can be installed on all types of aircraft and be used instead of the radio altimeter PB-5. Altimeter A-053 is designated for general aviation, aircraft and helicopters of major airlines.

Radio altimeters A-037, A-053 are airborne radar station with continuous radiation of frequency-modulated radio waves. Radio altimeters are equipped with one or more indicators. They have small dimensions and weights, high reliability of output information, fulfill the requirements of DO-160 (standard for environmental testing of avionics equipment), TSO-S87 (standard of radio altimeter certification) and airworthiness requirements of aircraft.

a)

b)

Fig. 5.5

Altimeter may be included to set of ground proximity warning system (such as TAWS or EGPWS) and to be an important part of them.

5.3 Radio altimeters of high altitudes

Increasing the flight altitude, the power of the reflected radio signal drastically falls, and it is difficult to identify valid signal from noise. The intensity of the valid signal can be increased by increasing the transmitter power, but the required power increases in proportion to the 4th power of height increase. For example, to increase the range of continuous radiation radio altimeter from 1500 to 15 000 m, the power of radio transmitter would have to be increased by 10 000 times.

Therefore, in the radio altimeters of high altitudes, the pulsed method of measuring distance from the aircraft to the ground is used. Pulses of constant frequencies at small regular intervals are emitted through the transmitting antenna. In pauses between impulses there is an accumulation of energy, which is then emitted again in the pulse. The ratio between the instantaneous power P_{imp} emitted in space in a pulse, and the average power P_{av} of the transmitter is equal to:

$$\frac{P_{imp}}{P_{av}} X \frac{T}{\tau_0},$$

where T is period between impulses; τ_0 is the pulse duration.

If, for example, $T = 1$sec, and $\tau_0 = 1$ microsecond, then $T/\tau_0 = 1000$. Consequently, the instantaneous power in the pulse will exceed 1000 times the average power of transmitter.

Measurement of altitude of H_{RA} is realized as measurement of delay time ζt of the arrival of the pulse reflected from the ground in comparison with the arrival in the receiver of direct impulse:

$$H_{RA} X 0.5 c \zeta t,$$

where $c = 299\ 762$ km/sec is velocity of radio waves propagation in the atmosphere.

5.3.1 Principle of high altitude radio altimeter operation

Block diagram of high altitude radio altimeter of PB-25 type is shown in Fig. 5.6.

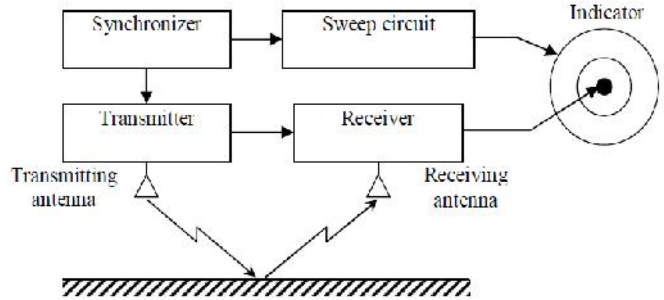


Fig. 5.6

Transmitter of radio altimeter PB-25 generates high frequency pulses of duration $T = 0.5$ microseconds. The repetition frequency is set by synchronizer. The synchronizer also controls the sweep circuit of indicator tube. The probing pulses are emitted by the transmitting antenna, reach the ground surface and are reflected from it, then they are accepted by the receiving antenna. The amplified and converted signals from the receiver output are sent to the indicator.

Indicator tube (IT) is used as the altitude indicator with a circular sweeping and radial deflection of the beam (Fig. 5.7). The synchronizing generator (SG) is realized according to the scheme with quartz stabilization of frequency and produces sinusoidal oscillations with frequency varying depending on the operation mode of radio altimeter (rough or accurate measurements, for example, 14989 or 149895 Hz. The same voltage is used to form the pulse in the modulator (M), that controls the high frequency generator (HFG) of the transmitter (TR), and to obtain two quadrature sinusoidal voltages required for the operation of ramp generator (RG), which operates in the standby mode.

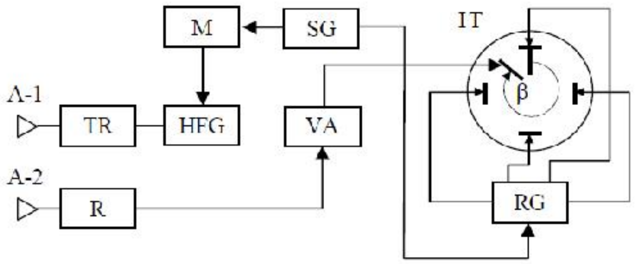


Fig. 5.7

The reflected signal from the receiver (R) is amplified by video amplifier (VA) and applied to the central deflecting electrode of IT.

For altitude reading, the scale in metres is used. It is deposited on a transparent organic glass and superimposed on the screen of indicator tube.

On the display screen of altitude indicator the reflected pulse is seen in the form of radial splash. Moreover, on the display screen the probing signal will be viewed in the form of amplitude marks.

If the movement of the beam by circle from zero mark of the scale starts at the time of arrival of the direct (probing) pulse, and at time of arrival of the reflected pulse the electron beam receives the radial splash, then the angular position β of this splash (see Fig. 5.7) will be proportional to the measured altitude:

$$\beta = \omega t = (2\omega/c) H_{RA} = S H_{RA},$$

where $S = (2\omega/c)$ is the device sensitivity; ω is angular velocity of the electron beam sweeping.

Sensitivity S and accordingly the accuracy of the reading can be raised by increasing the sweeping speed ω. However, at too high speed, the beam can make several turns until the moment of reflected pulse arrival, and there will be uncertainty of the indications associated with ignorance of the number of revolutions made by the beam.

Obtaining unambiguous results can be reached by switching the ranges: for low speed of sweeping the rough evaluation of altitude is done, and for high speed there is precise evaluation.

The division value of radio altimeter is 200 m in rough scale (Mx10) (see Figure 5.8) and is 20 m in precise scale (Mx1).

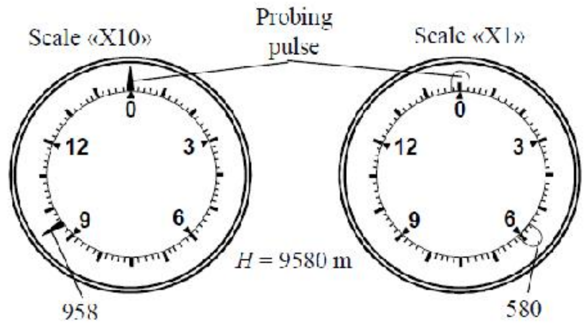


Fig. 5.8

Thus, when measuring the altitude with the limit precision to half scale the evaluation may be made to the nearest 10 or 100 m respectively to the scale of measurements.

The reading of altitude is performed in the following sequence. On the coarse scale M(10) the altitude is read and rounded to the nearest smaller value multiple to 1500 m (Fig. 5.8). Then on the precise scale M(1) the altitude is determined, not included in the first measurement. The amount of values obtained on both scales will constitute the geometric altitude.

Main technical characteristics of radio altimeter PB-25 are:

- range of measured altitudes is from 100 to 17 000 m;
- error of altitude measurement is 15 m ⌇ 0.25% from measured altitude on the precise scale, and 150 m ⌇ 0.25% from measured altitude on the coarse scale;
- transmitter power is not more than 0.2 W;
- carrier frequency of transmitter is 440 ± 1 MHz;
- pulse repetition frequency is 149895 ± 25 Hz on the precise scale, and 14989 ± 20 Hz on the coarse scale;
- duration of probing pulse is 0.5 ⇑s;
- transmission bandwidth of receiver is not less than 5 kHz;
- intermediate frequency is 30 MHz;
- consumption power is not more than 140 W.

The high altitude radio altimeters are usually used as auxiliary navigation tool, for example, for aerophotography and other purposes, but for the problems of piloting at high altitudes they are not applied. That is why, the set of some high altitude radio altimeters does not even contain altitude indicator. Information about the current absolute altitude comes directly to consumer equipment. In addition, instead of two antennas only one receiving-transmitting antenna is used. Figure 5.9 shows a set of impulse high altitude radio altimeter A-075-05.

Fig. 5.9

5.4. Errors of radio altimeters

The first group of errors is formed by the methodical errors caused by the discrete nature of the altitude evaluation, by change of the measurement conditions (the influence of roll and pitch of aircraft on the accuracy), as well as by other factors.

The second group of errors is caused by fluctuations of signal through the process of scattering of radio waves, by noises of the external and internal origin, by randomness of the received signal.

The third group is related to dynamic errors, in particular, the delay in the measurement of altitude.

The fourth group consists of instrumental errors due to the passage of signals through the antenna, transceivers and measuring channels of radio altimeter and of the errors due to circuit design and technological solutions of radio altimeter blocks.

Methodical errors of radio altimeters. In RA of low altitudes for measuring the frequency, the method of "zeros count" is used that is the count of number of pulses generated from the converted signal at a certain time interval.

Let us suppose that the altitude gradually increases. The number of pulses will vary discretely, increasing on one, then another, etc., leading to discrete samples of the difference frequency and accordingly to flight altitude.

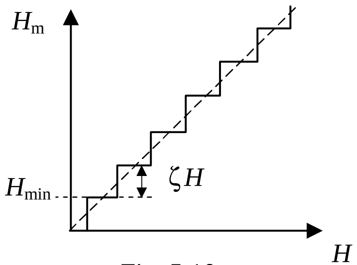


Fig. 5.10

The error resulting from the discrete nature of the altitude evaluation is called discrete or steady one, since during the flight at constant altitude, this error has constant value. The dependence of the measured altitude on the actual one in the presence of discrete error leads to limitation of the minimal altitude H_{min} (Fig. 5.10).

The minimum number of pulses that can be defined on the interval of the modulation period T_m is equal to one, i.e. the minimum of the measured difference frequency $F_{d\min}$ is equal to the modulation frequency F_m, which corresponds to the minimal measured altitude. For symmetrical laws of frequency modulation it is:

$$H_{\min} \mathrm{X} \frac{c}{4\zeta f_d F_m} F_{d\min} \rceil 37.5 \zeta f_d^{-1},$$

where ζf_d is deviation of frequency, Mhz; c is the speed of light.

If ζf_d is equal to 50 Mhz, then $H_{\min} = 0.75$ m. To reduce $H_{\min}$ it is necessary to increase the deviation of frequency ζf_d.

In practice of RA building the effective methods of dealing with discrete uncertainty are developed, which are mainly based on smoothing of the discrete structure of spectrum.

Errors of measurement conditions arise in the presence of significant angles of roll or pitch of the aircraft through the narrow directivity of the radio altimeter antennas, because in this case the slant range to the ground surface D is measured, not the the geometric height of the flight H_g (Fig. 5.11).

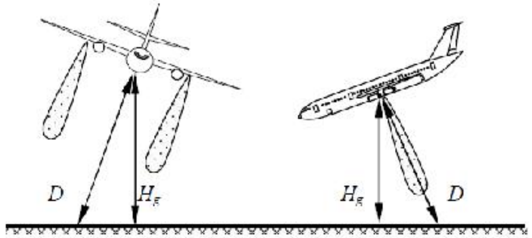


Fig. 5.11

Methodical errors of high altitude RA also arise because of the instability of radio waves propagation speed in case of changes in radiation conditions. This causes the changes in the speed of radio waves propagation, which differs from the calculated speed (299 762 km/sec).

Fluctuation error of radio altimeters. Fluctuation error of height measurement occurs through noise-like structure of the converted signal spectrum. With sufficiently large signal-to-noise ratio, the standard root-square errors $\exists_F$, $\exists_H$ of measurement of average difference frequency and of height due to the random nature of the converted signal are determined by the formula:

$$\exists_F \ \mathrm{X} K\sqrt{\frac{\zeta F_{ef}}{T}}, \quad \exists_H \ \mathrm{X} m_H \exists_F ,$$

where ΔF_{ef} is effective width of the spectrum of the converted signal; T is time of averaging when measuring; K = 0.1...0.7 is coefficient depending on the method of frequency measurement; m_H is scale factor.

For ΔF_{ef} = 2290 Hz, if T = 0.5 s, K = 0.3 and m_H = 0.0074, the fluctuation error will be $\exists_H$ = 0.15 m. To reduce fluctuation error, it is advisable to increase the averaging time T, however, this increases the inertia of radio altimeter as a measuring device, which leads to the growth of dynamic errors under aircraft maneuvering. Therefore, there is an optimal value of T, which usually lies in the range of 0.1...1 sec.

Dynamic error of radio altimeters. Since the measuring devices of radio altimeters have the dynamic characteristics of the corresponding order, when measuring the altitude of highly rugged relief, there is a time lag in measuring the altitude, and the dynamic error occurs.

In the presence of vertical speed of aircraft at low altitudes, there may be an error due to the influence of Doppler phenomena on the frequency method. This error is referred to as dynamic one. To eliminate this error, it is necessary to choose the RA parameters so that the vertical speed could not exceed a certain critical value. In practice, this condition is fulfilled with reserve, and the requirement is provided:

$$F_{D\max} \leq 0.1 F_{d0\min},$$

where $F_{d0\min}$ is the difference frequency, which corresponds to the minimum height $H_{\min}$, at which the flight with vertical speed $V_{y\max}$ is possible; $F_{D\max}$ is Doppler frequency, which corresponds to $V_{y\max}$.

By selection of RA parameters the dynamic error because of influence of Doppler phenomena can be reduced to 1...2% from $H_{\min}$.

Instrumental errors of radio altimeters. Like any measuring instrument, the radio altimeter has its instrumental errors. They are caused by non-ideality of characteristics of the individual blocks, as well as by the conditions of their placement. The most common causes of instrumental errors are parasitic amplitude modulation of the radiated signal, the influence of vibration, causing low-frequency modulation noise of the transmitter, the penetration of direct signal from the transmitter to the receiver via the incomplete isolation of the receiving and transmitting antennas, the deviation of modulation parameters, the errors of frequency meter.

Radical way to deal with the noises introduced by the direct signal is to improve the noise characteristics of the generator and to improve isolation of receiving and transmitting antennas by their rational installation. The implementation of the last recommendation requires the proper installation of antennas in different places of aircraft within a few meters from each other. At low altitudes there will be "hypotenuse error" (Fig. 5.12, a) caused by the difference of the average path length of reflected signal and doubled height. There may be also an error resulting from multiple reflection of signals from the ground and from the aircraft surface (Fig. 5.12, *b*).

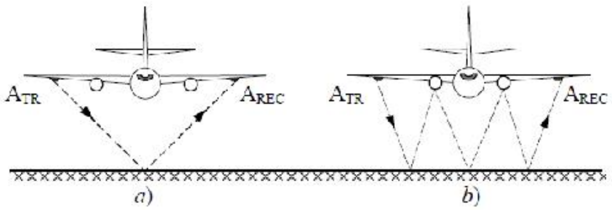


Fig. 5.12

As additional way of noise reduction, the filters are used which cut out the low frequency part of the spectrum of the converted signal, where the main amount of energy of parasitic noises is concentrated.

One of the reasons of instrumental error is unsatisfactory tuning and regulation of radio altimeters. Therefore, while developing equipment, the special methods to reduce such instrumental errors are used.

The errors in altimeter readings can be caused by the distortion of sweeping shape and by the eccentricity of sweeping circles relative the scale, by the inaccurate installation of probing pulse to zero of the altimeter scale, which is mostly caused by the shape and amplitude of probing pulse, and also by the unstable operation of synchronizer. As a result, the probing and the reflected pulses observed on the display screen of radio altimeter will travel (so called "trembling").

Errors of impulse RA also consist of errors from radio interference and errors from the instability of angular speed of sweep.

Errors of high altitude RA may occur because of incorrect adjusting of synchronizer, resulting in the violation of relationship between the scale and the sweep duration, caused by the period of sinusoidal oscillations of synchronizer.

Comparative analysis of RA errors caused by various factors, shows that the greatest influence on the resulting error has the fluctuation errors and the instrumental ones. In modern RA at the minimal altitudes, the maximal error does not exceed the value $\Delta H_0 = \pm(0.3...0.9)$ m. With altitude increase the measurement error raises.

5.5. Features of radio altimeter operation

Flight operation of radio altimeters. After switching on the radio altimeter it is necessary to verify that the readings of altitude indicator and of decision height indicator do not exceed 2 m.

Before takeoff it is necessary to ensure that the decision height indicator is set at the value corresponding to the value of the safety altitude of the aerodrome.

During climbing and descending it is necessary to test the signal of decision height. At the altitude less than

1500 m it is necessary to compare the readings of pressure altimeter and radio altimeter and to check the readings of decision height indicator during descending.

When flying at low altitudes above thick layer of ice (snow) the radio altimeter may measure the height with significant error, since the height is measured from the lower edge of the ice (snow) cover. Height to the top edge of the ice (snow) cover is measured only when flying above moist or polluted ice or snow. Therefore, when flying at altitudes less than 50 m above thick layer of ice (snow) it is necessary to take precautions.

When flying at low altitudes above forests, depending on the composition and density of forest the radio altimeter may measure the height to the top edge of tree crowns (dense forest) or to the ground surface (sparse forest). That is why when flying above forests at altitudes less than 50 m, it is necessary to take precautions.

When flying above mountainous terrain, where abrupt changes in altitude may be beyond the range of measured heights, the radio altimeter is not recommended for use.

At the angles of roll and pitch greater than 20°, the error of flight altitude measurement increases because of the influence of slant range.

At the angles of roll and pitch more than 40°, the radio altimeter is not recommended for use.

When flying above the operating range of heights or with defective RA, on the front of altitude indicator the warning flag of red color appears, and the indicator pointer is in the dark sector of high altitudes.

When checking RA in the mode "Test" that can be carried out at any height, the required testing time is determined by the time of indicator arrow passing from dark sector to the altitude of 15 m. After the checking this mark, the warning flag must disappear.

When aircraft flies from top to bottom and passes the dangerous height that is defined the altitude indicator, then on the front panel the lamp lights up. Its brightness is adjusted by rotation of the lamp knob.

When disconnecting the supply voltage from RA on the front panel of altitude indicator the warning flag appears, and the indicator arrow can be in any point of the scale.

As previously noted, high altitude radio altimeter has no indicator, and information on the flight altitude is used in other complexes and systems, therefore, its usage is largely confined to timely switching on and off the power supply in compliance with safety measures similar to the measures of safety in the operation of low altitudes RA.

In the operation of high altitudes RA, in which there is the altitude indication, it is necessary to compare readings of RA and pressure altimeter periodically.

Maintenance of radio altimeters. Before starting the checking of RA serviceability on the ground, it is necessary to ensure that there are no persons or foreign objects under the antenna of radio altimeter.

After switching on the radio altimeter via the time required for removing the signal "RA failure", the altitude indicator works off the height depending on the placement of antennas relative to the object center of gravity, on the loading of the shock absorbers of landing gears, on the tilt angle of antennas. Therefore, it is necessary to remember that the indicator arrow will be within the tolerance at zero altitude when the object is in the landing configuration, when antennas are located near the centre of gravity, as well as when there is small weakening of shock absorbers of landing gears.

If antennas are placed on front part of the object relative to the center of gravity, then the indicator arrow will show the negative altitude, and when placing antennas in the back of the object, then the readings will be positive.

If the index of dangerous height is set in the range of measured heights starting from 5 m, then with arrow passing through index of dangerous height, on the front panel of altitude indicator the signal lamp will light up and in the pilot headphones there will be audio signal.

It is necessary to ensure that the systems connected with radio altimeter (display system, ground proximity warning system and other aircraft systems that are functionally associated with RA) are serviceable. Also it is required to ensure that these systems are issued one-time signals and signal of dangerous height.

TEST QUESTIONS

1. What is the principle of height determination used in radio altimeter?
2. What method of altitude measurement is the most common in low altitude radio altimeter?
3. What method of altitude measurement is the most common in high altitude radio altimeter?
4. What principles is the frequency method of altitude measurement based on?
5. What frequency separated by the filter-detector of low frequency is a measure of the absolute altitude at low heights radio altimeter?
6. Which parameters of the transmitter frequency of low altitude radio altimeter does its sensitivity depend on?
7. What parameters of transmitter frequency of low altitude radio altimeter change when switching the altimeter to small range?
8. What frequencies in low altitudes radio altimeter determine the range of measured heights?
9. Give examples of low altitude radio altimeters.
10. How is it possible to increase the range of radio altimeter of continuous radiation? What does it lead to?

11. What measured parameter is a measure of absolute flight altitude in impulse high altitude radio altimeters?

12. What is the range of measured heights of radio altimeter PB-25?

13. Why do the errors of measurement conditions arise in low altitude radio altimeters?

14. What error arises due to the discrete nature of the altitude evaluation in low altitude radio altimeters?

15. How is it possible to improve the decoupling of the receiving and transmitting antennas of low altitude radio altimeter?

Chapter 6. Doppler Navigators

6.1. Doppler method of measurement of ground speed and drift angle

Radio method of measurement of aircraft ground speed is based on the effect of Doppler frequency shift, the essence of which is the change of frequency of received oscillation with the relative motion of the transmitter and the receiver.

If the oscillation source is moving relative to the medium, then the distance between the peaks of waves (wavelength) depends on the speed and direction of movement (Fig. 6.1). If the source is followng the emmited wave, then the wavelength is decreased (the frequency is increased); if the source is moving from this wave, then the wavelength is increased (frequency is reduced). Frequency shift is increased with increasing the velocity of the receiver relative to the transmitter.

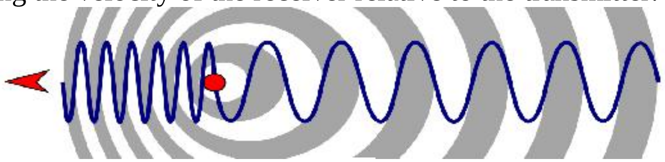


Fig. 6.1

The same effect occurs if the transmitter and receiver are unmoved relative to each other and located on the moving object, and oscillations are received after reflection from unmoving object, for example, ground surface.

Let us consider Doppler phenomena with continuous radiation of non-modulated radio signal.

Let us suppose that aboard aircraft performing the horizontal flight with constant ground speed V_g = const, the receiver and transmitter are installed. The latter emits the signal of the form:

$$e_{TR}(t) = E_{m\,TR}\cos(f_{TR}\; t),$$

where $e_{TR}(t)$ is instantaneous value of the electric field strength generated by the transmitter at the output of transmitting antenna; $E_{m\,TR}$ is the amplitude of the field strength generated by the transmitter; f_{TR} is the oscillation frequency of the transmitter.

Oscillations arriving at the receiver input are delayed relatively to the emitted ones by time $\tau = 2R(t)/c$, where $R(t)$ is current variable distance between the aircraft and elementary reflector; c is speed of light. Then the signal received by the receiver can be written as:

$$e_{REC}(t) = E_{m\,TR}\cos[f_{TR}(t - \tau)].$$

The instantaneous value of frequency of the received signal, which is determined as the time derivative of the total phase

$$\varphi(t) = f_{TR}(t - \tau) = f_{TR} \cdot t - 2R(t)/c',$$

equals

$$f_{REC} = f_{TR} - \frac{2 f_{TR}}{c}\frac{dR}{dt}.$$

Thus, the received oscillations vary in frequency from the frequency of emitted oscillation by the amount

$$F_D = f_{REC} - f_{TR} = -\frac{2 f_{TR}}{c}\frac{dR(t)}{dt},$$

that is called Doppler frequency shift (Doppler frequency).

If the speed of approach (or removal) changes over time, Doppler frequency $F_D(t)$ also becomes a function of time.

Derivative $\frac{dR(t)}{dt}$ is projection of aircraft velocity vector on the direction of the antenna beam, i.e. radial velocity W_R of aircraft relatively the reflector $-\frac{dR(t)}{dt} = W_R$. Taking this into account, let us represent the expression for Doppler frequency shift as:

$$F_D = f_{REC} - f_{TR} = \frac{2 f_{TR}}{c} W_R = \frac{2}{\lambda} W_R,$$

where λ is wavelength of oscillations emitted by the transmitter.

Doppler navigator (DN) uses exactly this effect. During the flight the fragment of ground surface that is irradiated by DN antennas, moves relative to the airplane with speed equaled to the ground speed of aircraft. If the surface is not perfectly smooth, then at each irradiated section, there is at least one point - elementary reflector that creates the reflection towards DN. Absolutely flat surface creates mirror reflection with which DN is not working.

Frequency of oscillations received by the DN receiver and reflected from the ground is different from the frequency of radio waves emitted by the transmitter. By the difference between these frequencies (by Doppler frequency F_D) it is possible to determine the radial velocity W_R of aircraft, and with the known orientation of the antenna beam relative to the aircraft, the ground speed of aircraft V_g and drift angle φ_{dr} can be found.

6.2. Measurement of ground speed vector by single beam meter

In horizontal flight the oblique irradiation of the ground surface is used at angle B_0 (Fig. 6.2, *a*) to provide sufficiently large projection of the aircraft velocity vector on the antenna beam direction and at the same time to keep an acceptable power of reflection towards the DN receiver.

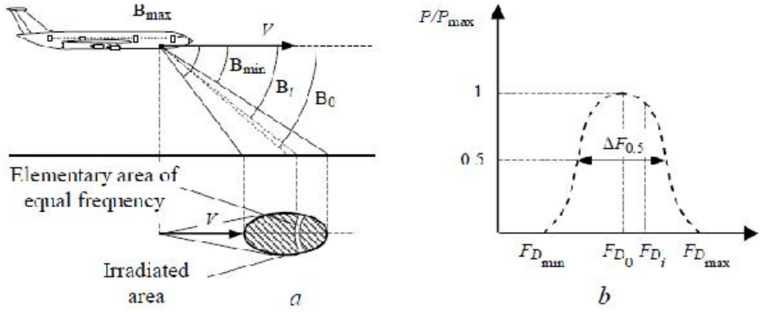


Fig. 6.2

DN antenna beam has a finite width, and therefore the reflection of radio waves occurs from a fairly large area of the ground surface. This section contains a set of elementary reflectors, independent on each other and randomly located within the area that is irradiated.

Since within the width of the beam the irradiation of ground surface section is done at different angles in the vertical plane, then the reflected elementary signals have different Doppler frequency shifts relative to the frequency of emitted signal

$$F_{D_i} = 2V\cos\delta_i/\lambda,$$

where *i* is number of elementary reflector, δ_i is the angle at which it is irradiated, λ is wavelength of signal emitted by the transmitter.

To determine the frequency spectrum of the reflected signal, let us cut out elementary strip from the irradiated area. All points of strip are located in directions which constitute angle δ_i with velocity vector *V*. At the oblique irradiation the borders of strips on the ground surface have the form of hyperbola.

All points of such elementary strip create the reflected signal with frequency $f_i = f_{TR} + F_{D_i}$. Having considered that each of *N* elementary strips corresponds to Doppler frequency shift it is possible to represent spectrum of the reflected signal by sequence of frequencies

$$f_{TR} + \frac{2V}{\lambda}\cos\delta_i.$$

Thus, the total Doppler signal is the sum of a number of elementary signals with random initial phases and amplitudes and with regular change of amplitude and frequency.

The power distribution of sums of elementary signals with the same Doppler frequencies averaged over the set of random reflectors, is called Doppler spectrum of the signal reflected from the ground surface. The form of the envelope of Doppler spectrum is shown in Fig. 6.2, *b*.

Average frequency $F_{D\,av}$ of Doppler spectrum is called the frequency that divides the power spectrum in half so that the total powers of spectral components with frequencies $F_D \,\Phi F_{D\,av}$ and $F_D \,\Psi F_{D\,av}$ are equal to each other.

If the ground surface reflection coefficient is constant within the width of DN antenna, then $F_{D\,av}$ coincides with the frequency F_{D_0} of Doppler signal of elementary reflector irradiated at angle B_0, and the form of spectrum envelope is determined by the form of the antenna directional diagram (ADD) of sensor in the vertical plane.

The maximum power in this case (Fig. 6.2, *b*) has the signal at average frequency of spectrum corresponding to the direction B_0 (axis of ADD), and spectrum width at the level of half-power is:

$$\zeta F_{0.5} \,\text{X} \frac{2V}{\Leftarrow} \cos\ \delta_0 \,\text{Z} \frac{\delta_A}{2} \ \text{Z}\cos\ \delta_0 \,\Gamma \frac{\delta_A}{2} \quad \text{X} \frac{4V}{\Leftarrow} \sin\delta_0 \sin\frac{\delta_A}{2},$$

where B_A is width of ADD in the vertical plane.

At sufficiently narrow ADD used in DN, it is possible to assume that $\sin(B_A/2) = B_A/2$, in this case

$$\zeta F_{0.5} \,\text{X} \frac{2V}{\Leftarrow} \delta_A \sin\delta_0 .$$

The relative width of Doppler spectrum is determined by the expression

$$\frac{\zeta F_{0.5}}{F_{D\ av}} \,\text{X} \delta_A \text{tg} \delta_0 ,$$

and depends on ADD width and beam tilt angle in the vertical plane. With the increase of inclination angle of the beam the width of Doppler spectrum increases.

In modern DN $\delta_A = (4...5\lceil)$, $\delta_0 = (65...70\lceil)$. Then at the horizontal flight of aircraft with angle of ADD $\wp_{dr} = 0$ we obtain

$$\frac{\zeta F_{0.5}}{F_{D\ av}} \,\text{X} 0.1\ldots0.5.$$

Thus, Doppler signal is narrowband.

Increasing ground speed, the spectrum of Doppler signal enlarges and moves (see Fig. 6.3) to the high-frequency area.

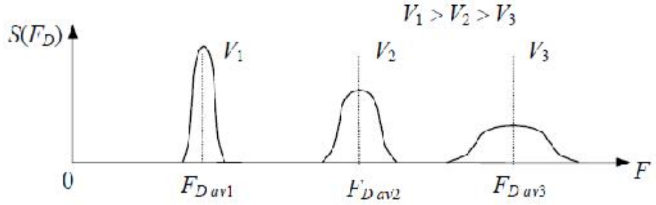


Fig.6.3

To measure the ground speed of aircraft it is necessary to determine the average frequency of Doppler spectrum $F_{D\,av} \rceil F_{D0}$:

$$F_{D_0} \,\text{X} \frac{2}{\Leftarrow} W_R . \qquad (6.1)$$

Radial speed W_R can be connected with the ground speed vector V_g and with drift angle of aircraft $\wp_{dr}$.

Figure 6.4 shows a diagram explaining the measurement of the ground speed and drift angle in the assumption that the flight is in the horizontal plane and there is no slip. Here V is vector of true airspeed (V_h is its horizontal projection); W_w is vector of wind speed; $\wp_{dr}$ is drift angle; W_R is radial velocity of aircraft (projection of ground speed vector V_g on the direction of the antenna beam).

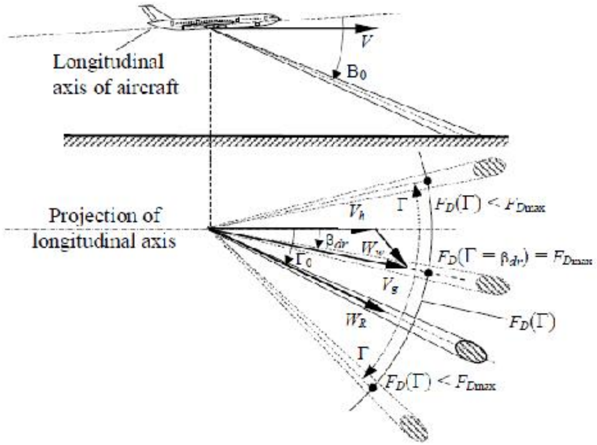


Fig.6.4

Assume that aboard aircraft there is an antenna with a narrow ADD, that is, electromagnetic energy is reflected from the limited area of ground surface. The antenna is rotated in the horizontal plane at angle Γ_0 relative to the longitudinal axis of the aircraft and at angle B_0 in the vertical plane. Angles Γ_0 and B_0 are called adjusting angles.

Since the vector W_R is the projection of the ground speed vector V_g on the direction of ADD axis, then

$$W_R \, X V_g \cos\delta_0 \cos(\iota_0 \, Z \wp_{dr}) \,. \qquad (6.2)$$

Substituting (6.2) into (6.1), the basic equation of DN is obtained:

$$F_{D_0} \, X \frac{2V_g}{\Leftarrow} \cos\delta_0 \cos(\iota_0 \, Z \wp_{dr}) \,. \qquad (6.3)$$

Doppler frequency includes information both about ground speed and drift angle, that is in equation (6.3) there are two unknown quantities ($\wp_{dr}$ and V_g), therefore it is necessary to use special methods to solve this equation.

For the single-beam DN it is the use of rotary antenna. When using rotary antenna, Doppler frequency F_{D0} will vary depending on the variable angle Γ, that is rotation angle of the antenna in the horizontal plane relative to the longitudinal axis of the aircraft.

When aligning the irradiation direction in the horizontal plane with the ground speed vector V_q, the angle ($\Gamma \, Z \, \wp_{dr}$) becomes equal to zero, and increasing the frequency reaches the maximum:

$$F_{D\max} \, X \frac{2V_g}{\Leftarrow} \cos\delta_0 . \qquad (6.4)$$

If λ and δ_0 are known, the ground speed

$$V_g \, X \frac{\Leftarrow}{2\cos\delta_0} F_{D\max}$$

can be determined by the direct measuring of $F_{D\max}$ by frequency meter. Drift angle $\wp_{dr}$ is equal to the angle between the axis of the airplane and the axis of the antenna at the time of its alignment with ground speed vector, i.e. at $F_D = F_{D\max}$.

Single-beam system does not find practical application due to the small slope of the curve $F_D(\Gamma)$ at maximum area (see Fig. 6.4), resulting in reduced sensitivity to changes in angle Γ with small misalignments in directions of V_g and ADD axis in the horizontal plane. This leads to low accuracy of determination of the drift angle as function $F_D(\Gamma) = F_{D\max}$. To understand the impact of errors on determining the frequency $F_{D\max}$, let us assume that at the moment of measurement $F_{D\max}$ the angle ($\Gamma \, Z \, \wp_{dr}$) = $\zeta\Gamma \,|\, 0$. At the same time Doppler shift differs from F_D in the value:

$$\zeta F_D \, X F_{D\max} \, Z F_D \, X \frac{2V_g}{\Leftarrow}\cos\delta_0 \, Z \frac{2V_g}{\Leftarrow}\cos\delta_0 \cos\zeta\iota \;\; X$$

$$X \frac{2V_g}{\Leftarrow}\cos\delta_0 (1 Z \cos\zeta\iota) \rceil \, F_{D\max} \frac{\zeta\iota^2}{2} .$$

Hence it is possible to determine the measurement error of drift angle $\wp_{dr}$, caused by inaccurate alignment of ADD axis with vector V_g due to the error of determination $F_{D\max}$:

$$\zeta \wp_{dr} \, X \zeta\iota \rceil \sqrt{\frac{2\zeta F_D}{F_{D\max}}} \,.$$

At the relative error $\Delta F_D/F_{D\max} = 0.01$ the error in measuring the drift angle is $\Delta \wp_{dr} = 0.14$ radian or approximately 8°, and in terms of aeronautical measurement it is unacceptable.

Error in measurement of $F_{D\max}$ causes also corresponding error of ground speed measurement, the value of which can be found directly from the expression

$$\frac{\zeta V_g}{V_g} \mathrm{X} \frac{\zeta F_D}{F_{\mathrm{D}\max}}.$$

Another important reason of the error of single-beam measuring devices is roll of aircraft. Suppose that because of the roll the actual value of adjusting angle B differs from the calculation angle B_0 in the value ΔB_0. Differentiating expression (6.4) by the parameter B_0, it is possible to write:

$$\frac{dF_{D\max}}{d\delta_0} \mathrm{XZ} \frac{2V_g}{\Leftarrow} \sin\delta_0 .$$

In the finite differences it will be:

$$\zeta F_{D\max} \ \mathrm{XZ} \frac{2V_g}{\Leftarrow} (\sin\delta_0)\zeta\delta_0 .$$

Hence, taking into account formula (6.4) it is obtained

$$\frac{\zeta V_g}{V_g} \mathrm{X} \frac{\zeta F_D}{F_{D\max}} \mathrm{XZ} \frac{\sin\delta_0}{\cos\delta_0} \zeta\delta_0 \ \mathrm{XZ}\zeta\delta_0 \mathrm{tg}\delta_0 .$$

In real systems, the angle of irradiation B_0 is selected about 70°. In this case, the relative error of ground speed measurement is 0.05 for each degree of error ζB_0 for determining the irradiation angle B_0.

Reduction of error caused by roll can be achieved by stabilization of the antenna in the horizontal plane or by the insertion of corrections to roll in the computing device when processing data. However, this leads to the significant complication of sensor, but it does not eliminate the disadvantages of single-beam method, which should also include requirements for high stability of frequency of irradiated oscillations.

The most radical way to improve the accuracy of measurement of ground speed and drift angle is the use of multi-beam measuring devices.

6.3. Measurement of ground speed and drift angle using multi-beam systems

Multi-beam DN for measuring the drift angle uses the method of comparison. It does not require the moving antennas and provides automatic and simultaneous measuring of ground speed and of drift angle with acceptable accuracy.

As follows from (6.3) - DN basic equation, Doppler frequency contains information both about ground speed V_g and drift angle $\wp_{dr}$, that is there are two unknown quantities. To determine two unknown quantities it is necessary to have at least two equations. With unmoving antennas in DN this can be achieved using two beams with different adjusting angles Γ_{01} and Γ_{02}. Then for two beams, two values of Doppler frequency are obtained and there will be corresponding set of two equations with two unknowns:

$$\begin{aligned} F_{D_1} \ \mathrm{X} \frac{2V_g}{\Leftarrow} \cos\delta_0 \cos(\iota_{01} \ \mathrm{Z} \wp_{dr}); \\ F_{D_2} \ \mathrm{X} \frac{2V_g}{\Leftarrow} \cos\delta_0 \cos(\iota_{02} \ \mathrm{Z} \wp_{dr}). \end{aligned} \tag{6.5}$$

In the unilateral two-beam system the error of drift angle determination is about 30 times less than in single-beam system. However, the error of ground speed measurement remains the same as in single-beam system.

Accuracy of ground speed measurement increases considerably when using bilateral systems with two beams directed forward and back. But in such a system the drift angle is determined with the error of single-beam system.

Moreover, equations (6.5) are valid only for level flight with no vertical component of the velocity vector. In the presence of roll or pitch the accuracy of two-beam DN is reduced.

These disadvantages are reduced sufficiently when using DN with three or four antennas, directional diagrams of which are located relative to the aircraft in the shape of letters «Y» or «X».

Measurement of ground speed and drift angle is carried out by comparing Doppler frequencies of two beams. In the absence of drift the ground speed vector coincides with the longitudinal axis of the aircraft, so that Doppler frequencies of both beams are the same and their difference is equal to zero. In the presence of drift, Doppler frequencies are not the same, because ground speed vector deviates from the longitudinal axis of the aircraft on the value of the drift angle.

Three-beam antenna system is shown schematically in Fig. 6.5 in two dimensions: horizontal and vertical, that passes through the beams *1-3*.

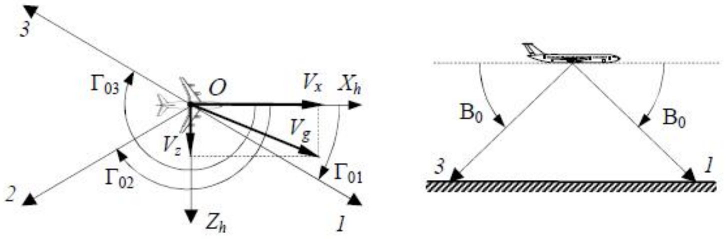


Fig.6.5

As it is shown in Fig. 6.5, three-beam antenna DN system uses symmetrical arrangement of beams in which:

$$B_{01} = B_{02} = B_{03} = B_0 ;$$

$$\Gamma_{01}=\Gamma_0, \quad \Gamma_{02} = 180^\circ - \Gamma_0, \quad \Gamma_{03} = 180^\circ + \Gamma_0;$$

$$|\cos\Gamma_{02}| = |\cos\Gamma_{03}| = \cos\Gamma_0 .$$

In three-beam system the expressions for Doppler frequencies for each beam (channel) are as follows:

$$F_{D_1} = \frac{2V_g}{\lambda}\cos B_0 \cos(\Gamma_0 - \psi_{dr});$$

$$F_{D_2} = \frac{2V_g}{\lambda}\cos B_0 \cos(\Gamma_0 + \psi_{dr});$$

$$F_{D_3} = \frac{2V_g}{\lambda}\cos B_0 \cos(\Gamma_0 - \psi_{dr}).$$

The sum and difference of frequencies of pair channels are determined by the following expressions:

$$\begin{aligned} F_{D_1} + F_{D_2} &= \frac{4V_g}{\lambda}\cos B_0 \cos\Gamma_0 \cos\psi_{dr} = \frac{4V_x}{\lambda}\cos B_0 \cos\Gamma_0; \\ F_{D_3} - F_{D_2} &= \frac{4V_g}{\lambda}\cos B_0 \sin\Gamma_0 \sin\psi_{dr} = \frac{4V_z}{\lambda}\cos B_0 \sin\Gamma_0, \end{aligned} \quad (6.6)$$

where V_x, V_z are projections of ground speed vector V_g on the axes of horizontal body-fixed coordinate system OX_h, OZ_h (see Fig. 6.5).

From (6.6) it is possible to obtain the calculation algorithm of navigation parameters:

$$V_x = \frac{\lambda}{4}\frac{F_{D_1} + F_{D_2}}{\cos B_0 \cos\Gamma_0}; \quad V_z = \frac{\lambda}{4}\frac{F_{D_3} - F_{D_2}}{\cos B_0 \sin\Gamma_0};$$

$$\psi_{dr} = \operatorname{arctg}\frac{V_z}{V_x} = \operatorname{arctg}\frac{F_{D_3} - F_{D_2}}{F_{D_1} + F_{D_2}}\operatorname{ctg}\Gamma_0; \quad (6.7)$$

$$V_g = \frac{V_x}{\cos\psi_{dr}} = \frac{\lambda}{4}\frac{F_{D_1} + F_{D_2}}{\cos B_0 \cos\Gamma_0}\sec\psi_{dr}.$$

Four-beam system with designating the angles B_0, Γ_0 for level flight is shown in Fig. 6.6.

To determine three components of the velocity vector of the aircraft, in fact, it is enough to have three beams. Therefore, information obtained by the fourth beam is redundant and navigation algorithms for calculating parameters are described by the same formulas - formulas (6.7).

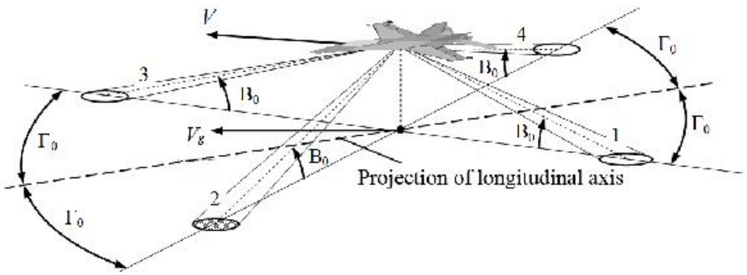


Fig.6.6

In four-beam DN there are the following relation between four Doppler frequencies:

$$F_{D_1} \Gamma F_{D_3} \mathrm{X} F_{D_2} \Gamma F_{D_4} . \qquad (6.8)$$

This relation is true for any evolutions of aircraft and can serve as a criterion of correct operation of all channels in the sensor. From (6.8) it is seen that one of Doppler frequencies can be expressed with the help of three others.

Determination of navigation parameters by formulas (6.7) corresponds to coherent mode of DN, i.e. separate processing of information for each beam. In four-beam DN the pairwise processing of beam data is also possible (auto coherent mode). In this case differences of Doppler frequencies of beams lying in one plane are measured in DN:

$$F_{D_{13}} \mathrm{X} F_{D_1} \mathrm{Z} F_{D_3}; \quad F_{D_{24}} \mathrm{X} F_{D_4} \mathrm{Z} F_{D_2} ,$$

by which the ground speed and drift angle are determined:

$$\wp_{dr} \mathrm{X} \mathrm{arctg} \frac{F_{D_{24}} \mathrm{Z} F_{D_{13}}}{F_{D_{13}} \Gamma F_{D_{24}}} \mathrm{ctg} \Gamma_0 ;$$

$$V_g \mathrm{X} \frac{\Leftarrow F_{D_{13}} \Gamma F_{D_{24}}}{8 \cos B_0 \cos \Gamma_0} \sec \wp_{dr} .$$

The accuracy of calculating the ground speed thus is slightly improving. The advantage of DN with pairwise processing of information is reducing of channels of signal processing from four to two, which leads to the reduction of mass and size characteristics of the system.

In multi-beam DN the measurement errors of navigation parameters arising from changing the roll and pitch angles are reduced significantly. The need for at least three beams and three frequencies is considered on the example of the influence of pitch ν. Figure 6.6 by solid lines shows the direction of the beam at $\nu = 0$, and by the dashed lines - at $\nu > 0$. The pitch leads to the decrease of angle B_0 in the first beam and to the increase of angle B_0 in the second and third beams. Thus, the sum and difference of frequencies in (6.7) vary significantly less than for each frequency separately.

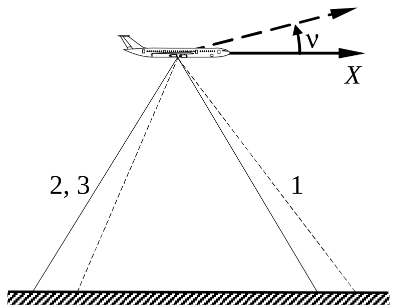


Fig. 6.7

6.4. Construction principle and block diagram of Doppler navigators

Construction of Doppler sensors is substantially conditioned by the mode of radiation. There are systems of continuous radiation with frequency modulation and without, and systems of pulse radiation with low off-duty ratio (quasi-continuous) and with large off-duty ratio.

The main advantage of DN with continuous unmodulated radiation is the focus of Doppler signal spectrum within a single, very narrow, frequency band, providing the most complete use of signal energy. The second advantage is in relatively simple structure of transmitter, receiver and indicator.

Disadvantage of DN with continuous radiation is the difficulty of eliminating the transmitter signal that leaks to the receiver input. This signal reaches the receiver input both as a result of coupling between receiving and transmitting antennas that are usually located nearby, and because of reflection of direct signal from elements of aircraft construction.

As an example, let us consider the generalized block diagram of three-beam DN with radiation of unmodulated continuous oscillations and with direct conversion of reflected signals to low frequency (Fig. 6.8).

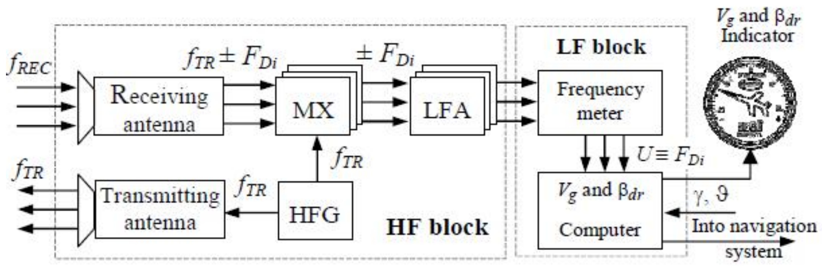


Fig. 6.8

DN main functional elements are high frequency generator (HFG); transmitting and receiving antennas; forming waveguide antenna system; balance mixer (MX); band-limited low frequency amplifier (LFA). In modern DN the listed blocks are structurally integrated into the high-frequency block (HF block).

Electromagnetic waves with required power of frequency f_{TR} generated by HFG of klystron or semiconductor type, passing through the slotted-guide bridge, are divided into three equal parts and pass to three horn radiators of transmitting antenna. Transmitting antenna generates three narrow beams. The beams are directed at specified angles down and sideways relative to the axis of antenna system that coincides with the longitudinal axis of the aircraft.

The signals reflected from the ground surface with frequencies $f_{REC_i} = f_{TR} \pm F_{D_i}$, $(i = 1,2,3)$ are taken in the receiving antenna by three narrow beams and then come into the three identical paths of receiving-measurement channels, which consist of the balance mixer (MX) and low frequency amplifier (LFA). Directional diagrams of receiving and transmitting antennas coincide and create the total ADD of DN antenna system.

To the input of each channel the signals from HFG also come, which play a role of reference signals in the balance mixer. At the output of the balance mixer the low-frequency oscillations of Doppler spectrum are separated and applied to LFA of each channel for amplification.

Bandwidth of LFA is selected based on the range of possible Doppler frequencies. Such transformation of frequency loses the sign of Doppler shift, but it is not essential for aircraft DN.

Doppler signals after amplification in HF block come into the low frequency block (LF block) to the input of frequency meter of average Doppler frequency.

To measure the average Doppler frequency it is possible to use counter of the number of times when the low-frequency voltage crosses zero level (zero-crossing counter); auto-correlator that measures the correlation time that is inversely proportional to the average frequency; frequency discriminator.

All three methods have similar errors, but in practice it is easy to implement zero-crossing counter, or the counter of pulses generated by the scheme of limiting and differentiating at the zero-crossing points of voltage on LFA output.

Voltages proportional to the value of Doppler frequencies, from the output of frequency meter enter the computer, which takes into account data on the roll γ and pitch ϑ of aircraft, as well as data on the angular orientation of DN beams. The computer calculates values of ground speed V_g and drift angle β_{dr}.

Measured Doppler frequencies depend to some degree on the nature of reflective surface, so simultaneously with the measurement of Doppler frequency shift, DN determines the correction to nature of the reflecting surface, which is taken into account when calculating the ground speed.

The obtained data on V_g and β_{dr} comes further in the navigation system to determine the coordinates of the aircraft by dead reckoning method and to the indicator of ground speed and drift angle.

Modern DN measures the ground speed in the range of 150...2500 km/h, the drift angle within $\pm$ 45°. They can operate at altitudes of 15...21 000 m. However, during the climb and descent, as well as maneuvers of aircraft with pitch and roll angles of more than 5...10° the accuracy is significantly reduced. Compensation of errors in measurement of ground speed and drift angle is carried out in different ways.

6.5. Doppler navigator ДИСС-013

Doppler navigator ДИСС-013, the set of which is shown in Fig. 6.9, is designed for continuous automatic measurement of ground speed and drift angle, and in the navigation systems it is used for dead reckoning.

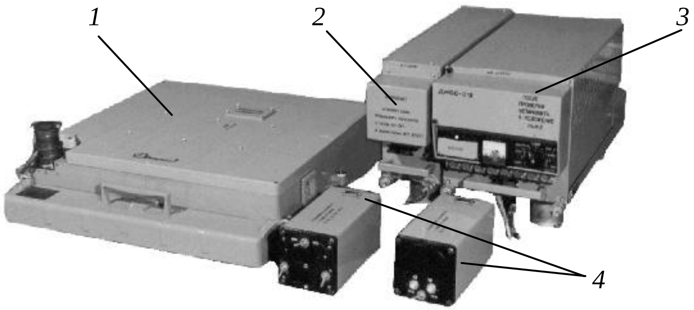


Fig.6.9

Structurally the functional blocks of ДИСС-013 (Fig. 6.9) are high-frequency block *1* with waveguide antenna system, interface block *2*, low-frequency block *3*, and display devices in the cockpit *4*.

Information about ground speed and drift angle comes to DN indicator, and in the navigation computer like АНУ or ЦНВУ and in the automatic flight control system of the aircraft. It is used for navigation for the given route of any type of the flyover surface independently on the optic visibility.

To take into account the nature of reflection of flyover surface, when flying over the sea the correction is entered the computer with the help of switch "land-sea" on the control display unit.

ДИСС-013 provides the following modes of operation: "Test", "Navigation", and "Memory." Turning the desired mode is carried out by mode selector.

"Navigation" mode is designed for automatic continuous measurement of ground speed and drift angle, for output their values on display and to consumers.

Mode "Test" is intended to check the functioning of device using built-in testing circuit. In this case Doppler frequency measuring circuit is connected with the special reference frequency generators which simulate Doppler frequencies.

Signals of reference frequency generators are applied to frequency meter instead of Doppler frequencies and on the indicator the reference values of the ground speed (710 ± 20 km/h) and of the drift angle (0 ± 30°) are displayed as calculated by DN schemes.

Activation of "Memory" mode is performed from the control display unit with turning off the radiation or automatically: if there is no reflected signals or low levels of them at the receiver input (for example, when flying over calm sea); if receiving or transmitting channels are not working; if there are angles of roll or pitch more 10 ± 2°. In this case the memorized at the moment of activation of "Memory" mode values of ground speed and drift angle are displayed.

Modifications of ДИСС-013 are installed on the aircraft An-22, An-26, An-30, Tu-134, Tu-154, Il-62 and on their modifications.

Operational and technical characteristics of ДИСС-013 are shown in Table 6.1.

Table 6.1

Type of characteristics	**Values**
Range of measurable ground speeds, km/h	180...1300
Range of measurable drift angles, deg	± 30
Operating range of altitudes, m	10...15 000 m
Ground speed measurement error (2∃), %	0,4%
Drift angle measurement error, ang.min	20
Mass, kg	27.5

Doppler navigator ДИСС-013, communication block and navigation computer form Doppler navigation system.

6.6. Helicopter Doppler navigator ДИСС-15

Helicopter Doppler navigators significantly differ from the aircraft ones in principles of their construction, primarily due to the different functionality and modes of helicopter and aircraft flights.

Motion of the helicopter in the space can be arbitrary. In some cases, such as in hovering mode the velocity vector can be equal to zero or near to zero and can change its direction. Therefore, the helicopter Doppler navigator should measure Doppler frequencies up to values close to zero, and determine their sign.

By design the helicopter DN can be divided into devices which measure:

- low velocities in the ranges of transition to hovering mode;
- high velocities in the range of navigation mode;
- low and high velocities.

Doppler navigator ДИСС-15 in combination with installed aboard compass system and gyrovertical is designed for independent measurement and display of components of velocity vector (during hovering of helicopter), ground speed and drift angle (during cruise flight), dead reckoning and indication of great circle or geographical coordinates of the helicopter location in both modes.

Structurally ДИСС-15 represents two large blocks Z high frequency block with waveguide antenna system (see Fig. 6.10) and low frequency block, and coordinate computer, control display unit. ДИСС-15 is combined with gyrovertical, compass system and airspeed sensor.

Fig. 6.10

Basic operational and technical characteristics of ДИСС-15 are represented in Table 6.2.

Table 6.2

Parameter	Values of continuous unmodulated oscillations
Frequency, MHz	13 325
Power, W	2...10
Working altitude range, m:	
in navigation mode over sea and land (sea choppiness > 1)	10...3 000
in hovering mode over land	2...1 000
in hovering mode over sea (sea choppiness > 1)	5...500
Measurement range of ground speed in navigation mode, km/h	50...399
Measurement range of longitudinal component of speed vector in hovering mode, km/h	Z25...+50
Measurement range of transverse component of speed vector in hovering mode, km/h	± 25
Measurement range of vertical component of speed vector in hovering mode, m/s	± 10
Measurement range of drift angle in navigation mode, deg	± 45
Errors in measurement of ground speed, km/h	0.5% ± 1.5
Errors in measurement of drift angle, ang.min.	25
Errors in measurement of longitudinal and transverse components of speed vector, km/h	± 1.5
Errors in measurement of vertical component of speed vector, m/s	± 0.4

Doppler navigator has three-beam receiving and transmitting antenna system. The antennas are arranged so that directional lobes of each antenna are directed downwards at some angle. Together they look like diverging from the helicopter beams by the type of trihedral pyramid. Its basis is the surface of land or water, vertex is helicopter itself, and edges are beams of narrow-band antennas of each channel. The orientation of the beams of antenna system in ДИСС-15 is shown in Fig. 6.11.

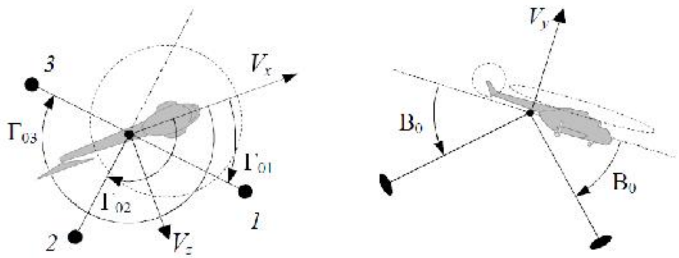


Fig. 6.11

In antenna systems ДИСС-15, the symmetrical arrangement of beams is used, in which

$$B_{01}\ \mathrm{X}B_{02}\ \mathrm{X}B_{03}\ \mathrm{X}B_0;$$

$$\Gamma_{01}\ \mathrm{X}\Gamma_0\ \mathrm{X}45\lceil,\quad \Gamma_{02}\ \mathrm{X}180\lceil \mathrm{Z}\Gamma_0,\quad \Gamma_{03}\ \mathrm{X}180\lceil\Gamma\Gamma_0;$$

$$\left|\cos\Gamma_{01}\right|\mathrm{X}\left|\cos\Gamma_{02}\right|\mathrm{X}\left|\cos\Gamma_{03}\right|\mathrm{X}\cos\Gamma_0\ \mathrm{X}\sin\Gamma_0.$$

For each of three beams of the antenna system, Doppler frequency is $F_{D_i}\ \mathrm{X}\frac{2}{\Leftarrow}W_{R_i}$ $(i\ \mathrm{X}1,2,3)$, where W_{R_i} is radial velocity (projection of the ground speed vector V_g of the helicopter on direction of i-th antenna beam). It can be expressed through the projections (V_x, V_y, V_z) of ground speed vector V_g on the axes of body-fixed coordinate system and through the orientation angles ι_0, B_0 of antenna coordinate system relative to the body-fixed coordinate system:

$$F_{D_1}\ \mathrm{X}\frac{2}{\Leftarrow}(V_x\cos\iota_0\cos\delta_0\ \Gamma V_z\cos\iota_0\cos\delta_0\ \mathrm{Z}V_y\sin\delta_0);$$

$$F_{D_2}\ \mathrm{X}\frac{2}{\Leftarrow}(\mathrm{Z}V_x\cos\iota_0\cos\delta_0\ \Gamma V_z\cos\iota_0\cos\delta_0\ \mathrm{Z}V_y\sin\delta_0);$$

$$F_{D_3}\ \mathrm{X}\frac{2}{\Leftarrow}(\mathrm{Z}V_x\cos\iota_0\cos\delta_0\ \mathrm{Z}V_z\cos\iota_0\cos\delta_0\ \mathrm{Z}V_y\sin\delta_0).$$

Reflected signals received by the receiving antenna, are applied to the device of separation of Doppler signal by the sign of the frequency shift.

The magnitudes and signs of the signals of Doppler frequency shift, which is taken by three antenna beams, are used to compute the components of the ground speed vector in the body-fixed coordinate system. The calculation is carried out according to the formula:

$$V_x\ \mathrm{X}\frac{\Leftarrow}{4}\frac{F_{D_1}\ \mathrm{Z}F_{D_2}}{\cos\iota_0\cos\delta_0};$$

$$V_y\ \mathrm{X}\frac{\Leftarrow}{4}\frac{F_{D_1}\ \Gamma F_{D_3}}{\mathrm{Z}\sin\delta_0};$$

$$V_z\ \mathrm{X}\frac{\Leftarrow}{4}\frac{F_{D_2}\ \mathrm{Z}F_{D_3}}{\cos\iota_0\cos\delta_0}.$$

Then the projections of the components of the velocity vector $V_{x_h}, V_{y_h}, V_{z_h}$ on the axes of horizontal (earth-fixed) coordinate system are calculated by the components of the velocity vector V_x, V_y, V_z using the direction cosines matrix **B**

$$\begin{matrix} V_{x_h} \\ V_{y_h} \\ V_{z_{hr}} \end{matrix}\ \mathrm{X}\mathbf{B}\ \begin{matrix} V_x \\ V_y \\ V_z \end{matrix}\ .$$

The direction cosines matrix can be obtained by signals of roll $\uparrow$ and pitch ν of helicopter, which are coming from the connected with ДИСС-15 gyrovertical in the following form:

$$\mathbf{B}\mathrm{X}\ \begin{array}{c|c|c} \cos\nu & \mathrm{Z}\sin\nu\cos\uparrow & \sin\uparrow\sin\nu \\ \hline \sin\nu & \cos\nu\cos\uparrow & \mathrm{Z}\cos\nu\sin\uparrow \\ \hline 0 & \cos\ni\sin\uparrow & \cos\uparrow \end{array}\ .$$

In ДИСС-15 computer, the simplified scalar expressions to calculate $V_{x_h}, V_{y_h}, V_{z_h}$ are used. Here by the horizontal components of the velocity vector the drift angle is determined as $\wp_{dr}\ \mathrm{X}\mathrm{arctg}(V_{z_h}/V_{x_h})$ and ground speed is as $V_g\ \mathrm{X}V_{z_h}/\cos\wp_{dr}$. The computed motion parameters of the helicopter are shown on the indicators.

In navigation mode by information about the longitudinal and transverse components of the velocity vector and about the difference between the current heading э, that comes from the connected with ДИСС-15 compass system, and map angle (MA) there are calculation of the distance traveled along the great circle route and of cross-track error of helicopter from the great circle route.

The introduction of MA sets the track angle of the great circle route, while the coordinates in great circle coordinate system are computed according to the formulas:

$$X\ \mathrm{X}\ {}_{t_1}^{t_2}[V_{x_h}\cos(\text{э}\ \mathrm{Z}MA)\,\mathrm{Z}V_{z_h}\sin(\text{э}\ \mathrm{Z}MA)]dt;$$

$$Z\ \mathrm{X}\ {}_{t_1}^{t_2}[V_{x_h}\sin(\text{э}\ \mathrm{Z}MA)\,\Gamma V_{z_h}\cos(\text{э}\ \mathrm{Z}MA)]dt.$$

The computed values of X and Z are displayed on the scales of coordinates indicator. At the same time the values of X and Z through the servo system enter the map indicator. Here the mechanic reduction gears convert the values of X and Z in the motion of sight cursor, which marks the location of the helicopter on the map.

When working by DN data at the coordinate computer the signal from the airspeed sensor comes, that replaces the signal V_{xh}. According to this data, there is a calculation of the path traveled along the great circle route.

At low flight speeds V_{x_h} ™50 km/h, ДИСС-15 switches to hovering mode, there is no indication $\wp_{dr}$ and V_g (drift angle indicator is dragged, and the ground speed indicator is closed by shutter). With increasing the longitudinal component of the velocity vector (V_{x_h} ™50 km/h) the navigation mode is activated and the indication of drift angle and ground speed is restored.

When flying over water the switch "land-sea" is set in position "sea" on the control panel. And in the computer the correction to the nature of the reflecting surface is introduced, which is taken into account when calculating $\wp_{dr}$ and V_g.

When reducing the amplitude of Doppler signal below the acceptable level in any of the three channels of Doppler navigator, the system switches to "Memory" mode, with the slowing down the indication on all indicators. The pilot receives the warning message Z display "Memory" lights.

6.7. Features of Doppler navigator operation

Flight operations of Doppler navigator. The operation of the system is performed in the order given in instructions to the crew for the certain modification of Doppler navigator.

Before taxiing out it is necessary to make sure that the switch "land-sea" on the control panel of Doppler navigator is set in the "land" position.

After takeoff, at altitude more than 300 m the mode selector of Doppler navigator on the control panel must be set to position "ON", within 3 minutes the panel "Memory" lights up at indicator of Doppler navigator. After 3 minutes after switching on the high voltage the system is ready for operation Z panel "Memory" goes out.

After 20 seconds after extinction of panel "Memory" Doppler navigator outputs the readiness signals to all connected systems. On the flight deck the panel "ADC dead reckoning" goes out. Further dead reckoning of aircraft location is performed by data of Doppler navigator.

When flying over water, the switch "land-sea" must be set to position "sea".

Under the maneuvers with roll more than 20⌠, the errors of the system increase. Therefore, after finishing the turn, the evaluation of ground speed and of drift angle is carried out not earlier than in 20...30 sec.

When flying over water with the sea choppiness of less than 1.5, over the sands, over mountain terrain, the information from Doppler navigator may disappear or be unstable.

In the absence of Doppler information, the system automatically switches to "Memory" mode. In this case, after 20 seconds on the flight desk the panel "ADC dead reckoning" lights up, and on the indicator of Doppler navigator the panel "Memory" lights up. In the "Memory" mode the system does not measure ground speed and drift angle. The instrument displays the values of these quantities calculated before switching to "Memory" mode.

When restoring Doppler information, the system leaves the "Memory" mode, the panel "Memory" goes out, and after 20 seconds the panel " ADC dead reckoning" goes out too.

When flying over smooth water surface the deviation and outliers can be observed in the readings of Doppler navigator without switching to "Memory" mode. In such cases, it is recommended the forced switching of Doppler navigator in "Memory" mode using the corresponding switch on the control panel (mode selector of Doppler navigator on the control panel must be set to position "Memory") or the switching off of the system. Reconnection of Doppler navigator to power is possible after aircraft approaching to favorable area of the flight from the point of view of the reflection surface.

Shut down of Doppler navigator before landing is carried out after the flight over outer marker.

Maintenance of Doppler navigator. All works for the assembly and disassembly of units and components of Doppler navigator are carried out only with fully de-energized equipment. External objects penetration in open plug connectors, pipe ventilation and blocks of DN, especially metal objects, is unacceptable. When working with HF block it is necessary to be especially careful. Damages of antenna mirrors are unacceptable. When Doppler navigator is working it is forbidden to be under antenna.

Special regulations and checks, elimination of complicated faults should be performed only in specialized areas, on special stands. When working with the system on the stands it is necessary to take appropriate measures of shielding of high-frequency energy radiation.

Places for working with Doppler navigator should be such that the radiation of high frequency energy in the adjacent rooms is not more than 0.1 μW/cm^2. To fulfill this condition the walls can be plastered by absorbing materials or shielded by metal grid.

All works on checking and adjustment of the system on aircraft are conducted in the sequence given out in the rules of technical operation for certain modification of Doppler navigator. During operation it is necessary to check: values of supply voltages; operation of the magnetron; accuracy of determining the ground speed and drift angle.

To check the accuracy of determining navigation parameters there is testing generator in electronic block. The test is carried out at low frequency.

Pre-flight checking of computational accuracy of ground speed and drift angle is performed in the following sequence:

Z to switch on automatic circuit breaker of Doppler navigator;

Z to set the mode selector of Doppler navigator on the control panel to position "ON". On the control panel the green lamp should light up, and after 2...3 minutes the red one should be lighted, signaling about the turning on the high voltage of transmitter;

Z to set the mode selector to position "Testing". With this, the red lamp should go out, signaling about removing the high voltage from the magnetron.

Z after 5 minutes to read the ground speed and drift angle from the indicator of Doppler navigator. The parameter values must be within the ranges: ground speed is in 850…890 km/h; the drift angle is in 0... ± 30'.

For the full checking of the technical condition of Doppler navigator on the front panel of the unit, LF has a controlling voltage meter. Using the switch, also located on the panel of the unit LF, checking the currents of the mixers; the voltage of the local oscillator; the output voltage of the amplifier of low frequency and high voltage on the reflection oscillator. After the inspections, the switch should be toggled to the "Off" position.

For a full review of the technical state of Doppler navigator on the front panel of LF block there is reference voltage gauge. Using the switch, also located on the panel of LF block, it is possible to check: the currents of mixers; the heterodyne voltage; the output voltage of LF amplifier; the high voltage on klystron. After the checking, the switch should be toggled to the position "Off".

In case of necessity to calibrate DN the set of checkout equipment is used. Calibration of system on the ground is carried out using the control unit of ПКД type from the set of checkout equipment, which allows testing all supply voltages, currents of the crystals in each channel, serviceability of magnetron. It provides the calibration of DN in two points: by ground speed and by drift angle. The control unit also allows checking the computer by switching it to testing mode.

TEST QUESTIONS

1. How does the frequency of wave change if the radiation source is moving to receiver?

2. What parameter is called Doppler frequency shift (Doppler frequency)?

3. Using radial velocity W_R, write down the expression for the Doppler frequency shift.

4. What parameters do we need to know to determine the ground speed and drift angle by Doppler frequency?

5. What are ranges of the bandwidth and the angle of inclination of the beam in the vertical plane in modern Doppler navigator?

6. How does the spectrum of Doppler signal change with increasing the ground speed?

7. What measures could be taken when using single-beam Doppler navigator and why are gauges of this type not applied in practice?

8. Why are not two-beam systems found the practical application?

9. What orientation of the beams is used in three-beam and four-beam Doppler navigators?

10. How many beams are sufficient to determine three components of aircraft velocity vector?

11. How is the measurement of the ground speed and drift angle carried out in three-beam Doppler navigator?

12. List the main functional elements of high frequency block of Doppler navigator?

13. How is the nature of the reflecting surface (land-sea) taken into account in the calculation of ground speed?

14. What operation modes are there in ДИСС-013?

15. What is the main feature of determining the parameters of the velocity vector in helicopter Doppler navigator?

16. What are two modes depending on the flight speed in helicopter Doppler navigator?

17. What coordinate system in helicopter DN is used for the initial calculation of the velocity vector components?

18. How is the pre-flight checking of ground speed and drift angle accuracy carried out in Doppler navigator?

Chapter 7. Overview-comparative method of navigation

One of the oldest methods of navigation – overview-comparative one – becomes very important in modern aircraft. Modern overview-comparative systems provide complete representation of full navigation data, interact with airborne digital computers, correct other sensors of navigation information and are essential information elements of system "aircraftZcrew."

Despite of the variety of technical implementation, the essence of overview-comparative navigation methods is in determination of object location by comparing the areas shown on the map or one that is in memory, with its actual view, observed using airborne observation devices (sights, viewfinders, television, radar, etc.) or visually. If the image of area on the map and its observed image are the same, then the location of object is considered to be recognized, and coordinates are determined.

Advantages of overview-comparative methods of navigation are:

Z high reliability and accuracy of measurements and the absence of accumulated errors;

Z possibility to measure in all regions of the Earth and near-Earth space;

Z high level of information redundancy of measurements;

Z wide possibilities to use manual (visual orientation) and automated measurement tools.

However overview-comparative methods have both disadvantages and limitations. Measuring is possible only when the ground surface or landmark are visible. The influence of obstacles Z clouds, fog, low illumination Z can significantly reduce the effectiveness of overview-comparative navigation. Also, for flying over terrain without landmarks (sea, deserts and so on) this type of navigation cannot be applied.

For successful implementation of overview-comparative method of navigation it is necessary to have sufficient provision by route maps or by required amount of prior information about landmarks planned for observation.

7.1. Classification of overview-comparative navigation systems

Overview-comparative navigation systems are classified according to the following criteria:

Z according to the physical nature of the measured signals there are optical, infrared, magnetometric, gravimetric, radar;

Z according to the degree of activity of measuring device there are passive (systems which use direct signals from landmarks) and active (systems obtaining the image of ground surface and landmarks through their radiation and receiving of reflected signals);

Z according to the degree of autonomy there are limited-autonomous systems (based on using natural, existing in nature landmarks) and non-autonomous systems (based on using artificial terrestrial or celestial landmarks);

Z according to the type of information, there are systems with point probing (so called probing of feature landmarks), systems with contour probing and with frame probing;

Z according to the level of automation there are visual, semiautomatic, automatic systems.

Z according to the amount of cartographic information there are systems with memory and without.

Systems without memory are systems in which landmarks are fixed during operation of the system (so-called correlation meters of speed). Such systems can only determine the rate of change of aircraft coordinates relative to the landmarks using any surface fields, including unstable over time (for example, clouds).

Systems with memory have prior (reference) information about landmarks that is accumulated before the operation. In systems with full prior information, the map is recorded with high accuracy in the memory block of navigation system for the whole area of flight. In systems with minimal prior information in the airborne memory device, the information about the coordinates of individual landmarks located along the desired track (point or extended types) is stored.

Systems with incomplete prior information provide the correcting of navigation systems both from point and extended landmarks and continuous correction at separate sections of route.

7.2. Principles of construction of overview-comparative navigation systems

Navigation content of overview-comparative measurement methods is defined by types of landmarks and their quantity. In the system there is comparison of physical parameters of landmark (area, geometrical shapes, radiation spectrum, etc.) stored in the memory of the system with the measured parameters and thus an identification of landmark is carried out.

In visual non-automatic systems the recognition task is solved by a pilot (operator). There is visual orientation by means of airborne observation devices (sights, viewfinders, television, radar and other) or it can be based on visual featured landmarks, i. e., whose characteristics are very different from neighboring similar objects (by special configuration or by the presence of the distinct elements of object).

During pre-flight procedure the typical landmarks are selected and in memory of airborne computer their coordinates are programmed. At flight after the landmark identification the navigation parameters of landmark location vector relative to the aircraft is determined and correction of aircraft coordinates is performed.

If in the same time several pre-defined landmarks are used, the information about them is stored in memory as parameters and coordinates of their relative position. The advantage of this approach is significant amount of navigational information, less dependence on the loss of planned landmarks and on the impact of interference. However, amounts of map information and time on its processing are increased, so the overview-comparative systems of this class can only be of automated type.

In general, a set of landmarks is considered as certain anomaly of physical field (optical, gravitational, terrain fields, etc.). The essence of this overview-comparative navigation is based on properties of anomalous fields as unique correspondence of field parameters distribution to specific area of the ground surface.

In determining the degree of similarity of the observed landmark or anomaly of physical field to those stored in memory, correlation function is used. If the number of reference landmarks is significant, then it is necessary to determine the maximal (or extreme) value of the correlation function that is, the greatest similarity of the observed areas of the ground surface to the mapped surface. Therefore, overview-comparative systems of this type are called correlation-extreme navigation systems (CENS).

7.3. Correction of aircraft current coordinates using overview-comparative systems with point probing

Overview-comparative point probing systems are used as additional tool to correct coordinates. To do this, in the airborne computer memory the coordinates of feature landmark must be programmed by which it is supposed to correct aircraft location.

The main task performed by the navigation system for correction by landmarks is the maximal automation of correction in order to release the operator from calculations and mapping works, to improve the accuracy and efficiency of orientation. Semi-automatic (with operator participation) correction process by landmark is carried out as follows.

During the flight (at estimated distance to the landmark) the crew obtains a warning signal and automatically the sighting device is actuated. On its screen the mark is formed (crossing) as possible position of landmark in coordinates of viewfinder.

The operator detects and identifies landmark, aligns the crossing of viewfinder with image of real landmark and at the moment of alignment pushes the correction button (referencing). All computing operations are performed by computer of navigation system and the operator is signaled on the implementation of correction.

Most suitable systems for such semi-automatic correction are surveillance airborne radar systems (RS) and electronic-optical viewfinders.

Use of angular electronic-optical viewfinders. Using of electronic-optical (television, infrared and others) viewfinders as a correctors compared with the radar station has the following feature: the viewfinders do not directly determine the distance to the landmark, they only measure the angular position (heading angle β_{LM} and elevation ε_{LM}). Therefore, for correction it is necessary either to use this viewfinder with rangefinder or to estimate the range analytically, such as by angle of elevation of landmark ε_{LM} and by absolute altitude H (Fig. 7.1).

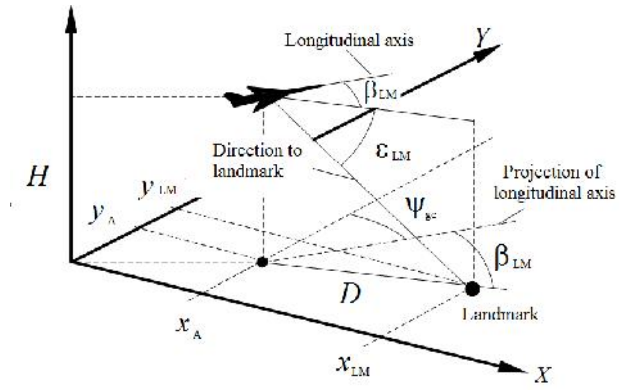


Fig. 7.1

Because of the small limit range of direction finding of ground landmarks by such viewfinders the calculation of the aircraft coordinates is carried out without regard spherical shape of the Earth:

$$x_A = x_{LM} - D\sin(\psi_{gc} + \beta_{LM});\quad y_A = y_{LM} - D\cos(\psi_{gc} + \beta_{LM}),$$

where D is distance to the landmark.

It is calculated by the angle of elevation and the absolute flight altitude by the formula:

$$D = H\ \mathrm{ctg}\,\varepsilon_{LM}.$$

The accuracy of determination of aircraft coordinates using optical and electronic-optical viewfinders is characterized by root mean square radial error, which can be obtained as:

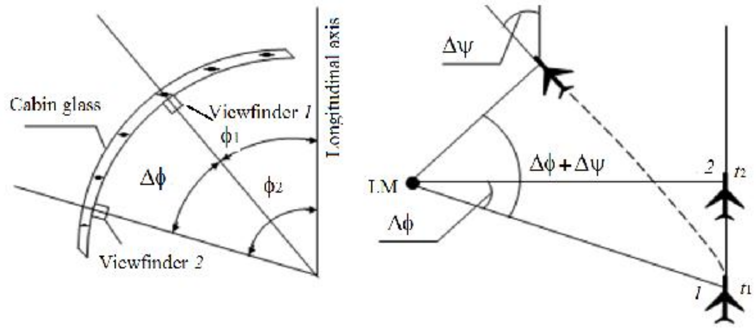
$$\sigma_r = \sqrt{\sigma_D^2 + D^2(\sigma_\beta^2 + \sigma_\psi^2)},$$

where σ_D is accuracy of range measurement; σ_β is accuracy of measurement of landmark heading angle; σ_ψ is accuracy of heading measurement.

It is necessary to take into account that due to the small range of direction finding of ground landmark, the excess of landmark above the reference level of absolute altitude at the time of correction significantly influences the accuracy of aircraft coordinate determination by analyzed viewfinders. This especially influences when flying at low altitudes. If flight mission involves such modes of flight, then during the preparation of system to operation it is necessary to choose a landmark with minimal excess.

Use of side-looking electronic-optical viewfinders. Another method of correcting of the aircraft calculated coordinates by landmark is to use side-looking viewfinders. Let us consider one of the algorithms of their use. In the simplest case, side-looking viewfinder is a transparent prism with two vertical bars installed at the side of the cabin. Due to the thickness of the prism these bars in their alignment by pilot eye create cross-section or plane of landmark direction finding.

On each side of the cabin there are two such viewfinders (Fig. 7.2, *a*), which form two planes of direction finding intersecting at angle $\zeta\leftarrow$. Heading angles of these planes ($\leftrightarrow_1$ and $\leftrightarrow_2$) are constant and determined by location of viewfinders.



a *b*

Fig. 7.2

With aircraft flying, the programmed landmark observed initially enters the cross-section of first viewfinder (Fig. 7.2, *b*), and then in the cross-section of the second one. In each of these time moments (t_1 and t_2) the pilot presses the fixing button. Solving the geometric problem of aircraft coordinates relative to the landmark and forming the corrections occur in the computer of navigation system. For this purpose, in each reference moment the angular position of aircraft ($\uparrow, \nu, \ni$) are fixed together with calculated coordinates (x, y) and absolute altitude H. Values $\leftrightarrow_1, \leftrightarrow_2$ and landmark coordinates are stored in the computer memory.

Formulas for calculating the corrections are very complicated (especially taking into account spatial maneuver), so the implementation of this method is appropriate only at presence of the computer. With the correction of aircraft location by side-looking viewfinder, the method of two curves of equal bearing is used. The accuracy of aircraft location determination is obtained by the approximate formula:

$$\exists_r \text{ X} \frac{\sqrt{f\exists_{\text{DF}} D_1 \text{Á}^2 \,\Gamma\, f\exists_{\text{DF}} D_2 \text{Á}^2 \,\Gamma\, fV_g \exists_t \text{Á}^2}}{\sin f\zeta\leftrightarrow \,|\, \zeta\ni |\text{A}},$$

where $\exists_{\text{DF}}$ is angular error of direction finding; $\exists_t$ is error at time moment of referencing; V_g is ground speed of aircraft; D_1, D_2 are distances to the landmark in the moment of the first and second direction finding; $\{|\ni|$ is change of aircraft heading between direction finding moments (sign + is related to the turn to the landmark); $\zeta\leftarrow$ is the angle between two planes of direction finding.

Assuming that $\exists_{\text{DF}}$ X1...2°, $\exists_t$ X0.2...0.4 seconds, at typical values D_1, D_1 = 2…3 km, V_g = 250 m/s and $\zeta\leftrightarrow$X30° $f\zeta\ni$ X0° $Á$ we will get $\exists_r$ X0.2...0.4 km.

In case of using sensor-correctors in structure of navigation systems together with the correction of calculated coordinates of aircraft location by information from these sensors it is possible to solve the problem of correction of the aircraft heading. The problem of heading correction can be performed by different systems and tools using a variety of methods. The simplest correction with the required degree of accuracy is performed using airborne viewfinder by single landmark.

Let us consider the essence of the heading correction by this method. Heading correction is performed after correction of computed coordinates of aircraft location. Due to this, the errors of determination of aircraft heading are reduced, which are caused by errors in determining the location coordinates. Further by known landmark coordinates (x_{LM}, y_{LM}) *and corrected coordinates of aircraft location* (x, y)*, the azimuth (bearing)* A *of landmark is determined (Fig. 7.3):*

$$A \, \mathrm{X} \, \mathrm{arctg} \, \frac{x_{\mathrm{LM}} \, \mathrm{Z} \, x}{y_{\mathrm{LM}} \, \mathrm{Z} \, y} .$$

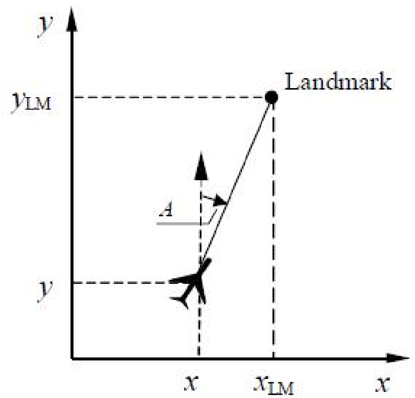


Fig. 7.3

Data about landmark azimuth and aircraft heading issued by the compass system allows obtaining the calculated value of heading angle of landmark:

$$\wp_{\mathrm{LM}} = A \, \mathrm{Z} \, \ni_{gc},$$

where $\ni_{gc}$ is great circle heading of aircraft before correction

Obtained value $\wp_{\mathrm{LM}}$ is used to form the longitudinal mark of viewfinder crossing (bearing label). The error of aircraft heading measurement $\zeta \ni_{gc}$ causes the error in the calculated value $\wp_{\mathrm{LM}}$. With alignment of crossing marks of viewfinder with the image of landmark, by the angle of additional turn of mark it is possible to determine the correction in heading of the aircraft. This method of heading correction is quite simple and reliable. In the same conditions the highest accuracy of heading correction can be achieved in case for the correction from landmark located at the maximal possible distance.

The disadvantage of electronic-optical systems is that they normally operate only with optical visibility of targets and with sufficient illumination. The radar systems have significant advantages in this sense.

Correction of location coordinates by radar data from single landmark. Radiolocation is the detection, determination of the location and of the properties of moving and stationary objects using radio waves reflected or emitted by these objects. Technical means of obtaining radar data are called radar station or radar. The combination of radar and technical auxiliary means, interrelated to each other and designated to solve tactical missions of radiolocation, is called the radar system (RS).

Correction begins with the formation of feature O$_c$, that determines the ability to perform correction in these conditions by the given radiolocation landmark, such as:

$$\varsigma_c \, \mathrm{X} \bullet D_{\min} \, \Phi D_{\mathrm{LM}} \, \Phi D_{\max} \quad | \wp_{\mathrm{LM}} | \Phi \wp_{\mathrm{limit}} ' , , \qquad (7.1)$$

where D_{LM} is current value of distance to radar landmark; $D_{\min}, D_{\max}$ are minimal and maximum distances, which determine the ability of coordinate correction by current landmark using RS; $\wp_{\mathrm{LM}}$ is current heading angle of landmark; $\wp_{\mathrm{limit}}$ is maximal limit value of heading angle of landmark, which provides execution of correction.

With fulfillment of condition (7.1) on the radar screen the crossing appears (Fig. 7.4). Position of the crossing on the radar screen corresponds to the calculated position of landmark relative to aircraft. Calculation of crossing position on the radar screen and further solving the problem of correction of calculated coordinates is as follows.

1. Programmed geographical coordinates of landmark $f \leftrightarrow_{\mathrm{LM}}, \Leftarrow_{\mathrm{LM}} \mathrm{A}$ are converted into corresponding great circle coordinates $f x_{\mathrm{LM}}, y_{\mathrm{LM}} \mathrm{A}$.

2. The guiding of electronic crossing on airborne radar screen for the image of landmark is performed using normal (horizontal) coordinate system of the aircraft. The initial position of crossing on the radar screen is determined by range $D_{+} f t_0 \mathrm{A}$ and heading angle $\wp f t_0 \mathrm{A}$, which are calculated by the formulas:

$$D_{+}(t_0) \, \mathrm{X} \sqrt{(x_0 \, \mathrm{Z} \, x_{\mathrm{LM}})^2 \, \Gamma \, (y_0 \, \mathrm{Z} \, y_{\mathrm{LM}})^2};$$

$$\wp(t_0) \, \mathrm{X} \, \mathrm{arctg} \frac{z_{h_+}(t_0)}{x_{h_+}(t_0)},$$

where x_0, y_0 are the determined great circle coordinates of aircraft at the moment of correction start; $x_{h_+}(t_0)$, $z_{h_+}(t_0)$ are the calculated *coordinates of landmark (crossing) in the normal coordinate system of the aircraft (Fig. 7.4).*

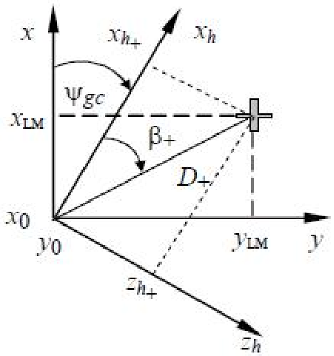


Fig. 7.4

Image of crossing moves on the radar screen, primarily as a result of the aircraft motion. Rate of change in great circle coordinates of crossing per a cycle of the computer operation for this reason is as follows:

$$\Omega x_\Gamma \,\mathrm{X}(V_g \;\; t_{\text{cycle}})\sin \ni_{gc};$$
$$\Omega y_\Gamma \,\mathrm{X}(V_g \;\; t_{\text{cycle}})\cos \ni_{gc}, \qquad (7.2)$$

where V_g is ground speed of aircraft; $\ni_{gc}$ is great circle heading of aircraft; t_{cycle} is duration of the calculation cycle.

Position of the crossing and its coordinates are changed by the operator (navigator) using knob.

Speed of moving of crossing on the screen can be determined, for example, using the following ratios:

$$\dot{\Omega}^*_{x_+} \,\mathrm{X} k_D \,\Im_D \,\mathrm{sign}\Im_D;$$
$$\dot{\Omega}^*_{y_+} \,\mathrm{X} k_\wp \Im_\wp D_+ \mathrm{sign}\Im_\wp \qquad (7.3)$$

where $k_D, k_\wp$ are certain coefficients (selected from experience); $\Im_D, \Im_\wp$ are angles of deviation of mechanism of crossing control by range and by heading angle.

Current position of crossing relative to aircraft with (7.2), (7.3) can be determined using the relations:

$$\zeta x_+ \,\mathrm{X}\, \zeta x_{+_0} \,\Gamma \mathop{}_{i\mathrm{X}1}^{n} \Omega x_{+_i} \,\Gamma \mathop{}_{i\mathrm{X}1}^{n} \dot{\Omega}^*_{x_+ i} t_{\text{cycle}};$$
$$\zeta y_+ \,\mathrm{X}\, \zeta y_{\Gamma_0} \,\Gamma \mathop{}_{i\mathrm{X}1}^{n} \Omega y_{\Gamma_i} \,\Gamma \mathop{}_{i\mathrm{X}1}^{n} \dot{\Omega}^*_{y_+ i} t_{\text{cycle}},$$

where $\zeta x_{+_0} \,\mathrm{X} x_0 \,\mathrm{Z} x_{LM}$ and $\zeta y_{\Gamma_0} \,\mathrm{X} y_0 \,\mathrm{Z} y_{LM}$ are initial relative coordinates of the crossing.

3. Directly the determined coordinates of aircraft location are corrected at the moment of alignment of the crossing and of image of landmark on the radar screen. Thus, calculated values $D_+ ftA$ and $\wp_+ ftA$ respectively are equal to actual values of range to landmark $D_{LM} ftA$ and of heading angle of landmark $\wp_{LM} ftA$. Thus, the following conditions are fulfilled, $\zeta x_+ \,\mathrm{X} \zeta x_{LM}$ and $\zeta y_+ \,\mathrm{X} \zeta y_{LM}$. Great circle coordinates of aircraft in this case are determined by following formulas:

$$x \;\; \mathrm{X} x_{LM} \,\Gamma \zeta x_{LM};$$
$$y \;\; \mathrm{X} y_{LM} \,\Gamma \zeta y_{LM}. \qquad (7.4)$$

This method of correction is semi-automatic and needs to perform some operations by the aircraft crew Z search for selected landmark of correction, alignment of crossing with the landmark image on the locator screen. These ones are its main disadvantages. Furthermore, the method does not ensure high accuracy of correction since accuracy of measurement of navigation parameters using radar is not very high.

Correction of coordinates by two landmarks. To improve the accuracy of calculating the corrected coordinates, the correction method by two programmed landmarks can be used. In this case (Fig. 7.5) aircraft location coordinates are obtained by solving:

$$(x_{LM_1} \,\mathrm{Z} x)^2 \,\Gamma (y_{LM_1} \,\mathrm{Z} y)^2 \,\mathrm{X} D_1^2;$$
$$(x_{LM_2} \,\mathrm{Z} x)^2 \,\Gamma (y_{LM_2} \,\mathrm{Z} y)^2 \,\mathrm{X} D_2^2.$$

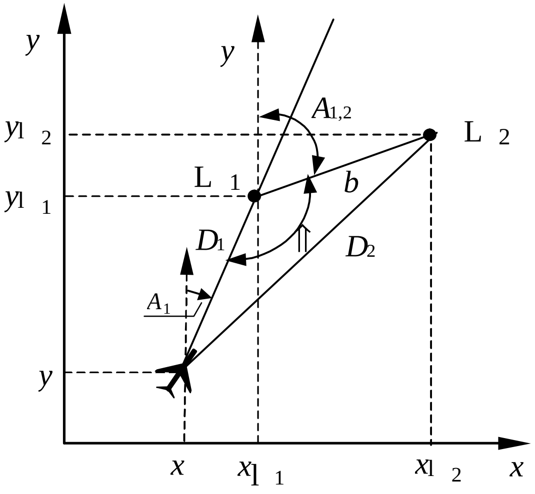


Fig. 7.5

Some complexes can use the modification of algorithm of calculated coordinates of aircraft location correction. The algorithm uses the information about two landmarks. When using this algorithm, the coordinates of aircraft are determined by two ranges D_1 and D_2 to the selected landmarks and by previous calculations of bearing (azimuth) of one of them (for example) A_1:

$$x = x_{l_1} - D_1 \sin A_1;$$

$$y = y_{l_1} - D_1 \cos A_1.$$

Bearing of the first landmark A_1 is calculated by the measured ranges and by the known distance between landmarks (see Fig. 7.5).

$$A_1 = A_{1,2} - (180^\circ - \psi); \quad \psi = \arccos \frac{D_1^2 + b^2 - D_2^2}{2D_1 b};$$

$$A_{1,2} = \operatorname{arctg} \frac{x_{LM_2} - x_{LM_1}}{y_{LM_2} - y_{LM_1}};$$

The scheme considered above is the scheme of correction of the calculated coordinates of aircraft location. It is simplified, here the motion of aircraft during the time between measurements of the range to the first and second landmarks is not taken into account.

The correction accuracy of the calculated coordinates of aircraft location using radar station is determined by accuracy of the range measurement (σ_D), accuracy of measurement of landmark heading angle (σ_φ) on the screen of radar station, and also by accuracy of measurement of the heading (σ_ψ). Modern airborne radar stations measure the range with the accuracy $\sigma_D \approx 0.1...0.5'$ km depending on the type of the radar station and used methods to compensate errors. The accuracy of the heading angle measurement is $\sigma_\varphi \approx 0.1...0.5'$ deg. The total error of determining the aircraft location by the above described method is determined by the following ratio:

$$\sigma_r = \sqrt{\sigma_D^2 + D^2(\sigma_\varphi^2 + \sigma_\psi^2)}.$$

The basic errors of measurement of the heading angles and ranges are connected with technique of the area reproducing on the screen of radar station and with reading this information by the operator. One of the major ways to improve the accuracy is to use the scanning of the microplan type where the image zoom is much larger than in the surveillance mode.

The accuracy of correction is significantly affected by technical implementation of radar station, its tactical and technical characteristics, the features of operational restrictions, and so on. Let us consider the principles of radar overview - comparative systems.

7.4. Construction principles of radar overview -comparative systems

7.4.1. Physical basis of radiolocation

Physical base of determination of objects location is that in homogeneous medium the radio waves propagate by straight lines and with constant speed $c \approx 3 \times 108$ m/s. This allows determining the direction of radio waves to the radiator and traveled path (range) $R = c\tau$, by measuring the propagation time τ between transmitter and receiver.

The problem of radio detection of object is to identify the signal emitted or re-radiated by this object against the background of various kinds of interferences. There are two types of radiolocation: passive and active. Passive radiolocation is based on receiving of own radiation of object. With active radiolocation the radar emits its own probing pulse and receives its reflection from the target. Depending on the parameters of the received signal the characteristics of the target (landmark) are determined. Active radiolocation, in its turn, may be of two types: with active response (on the object there must be radio transmitter that emits radio waves in response to the received signal) and with passive response (the interrogation signal is reflected from the object and is received by the radar as a response).

Active radiolocation is based on the phenomenon of reflection or scattering of radio waves, if on the way of their distribution there is an object with the electrical parameters which differ from the parameters of the medium. This object, irradiated with electromagnetic oscillations, becomes a source of reflected, i.e. secondary electromagnetic field. Power of secondary radiation depends on the intensity of the primary field around the object, on object parameters (size, shape, electrical properties), on position of the object relative the source of the probing signal, on polarization of the primary field, on wavelength λ.

The dependence of the power of secondary radiation on λ is particularly important because its nature determines the range of the radio waves, suitable for radio detection. If the linear dimensions l of object are such that

$$l \gg \lambda \qquad (7.5)$$

then the power of the secondary radiation is practically independent on λ. If $l << \lambda$, then the power of the secondary radiation is inversely proportional to λ^4. Thus, with increasing of λ the power of secondary radiation drops sharply, that leads to the corresponding decreasing in detection range. Therefore, in the radiolocation the radio waves are basically used with wavelength satisfying the ratio (7.5). Correspondingly, for radiolocation observations of targets like aircraft, vehicles and so on it is necessary to use the range of meter and shorter waves ($10...10^{-3}$ m).

Passive detection is based on the use of own radiation of object including thermal one. Any physical body, the temperature of which is above absolute zero emits electromagnetic waves, so there is a fundamental ability to detect any objects without previous irradiation. Maximum of thermal radiation of the ground surface and of many other objects lies in the area of infrared range. For detection the radio emission can be also used caused by the operation of various radio devices, which are available on the object, launching of rockets, etc.

Radio signal in radar station is usually a narrow-stripped process that can be expressed in the form of quasi harmonic oscillations

$$s(t) = S(t)\cos[\omega_0 t + \Phi(t)],$$

where $S(t)$ and $\Phi(t)$ are the envelope and phase of process which are slowly changing in comparison with the time functions and characterize the amplitude and angular modulation of the carrier signal oscillations.

The carrier circular frequency ω_0 determines the position of the signal spectrum on the frequency axis. The message can be in any of the parameters of radio signal: in amplitude, phase, frequency deviation from the carrier.

In the electrical circuits of the transmitter and receiver the radio signals of radar station operate in the form of currents and voltages. In the space the electromagnetic waves propagate, which are characterized by the vectors of strength of electric $\vec{E}$ and magnetic $\vec{H}$ fields and by Poynting vector (energy-flux vector) $\vec{\Pi}$ (Fig. 7.6), that determines the direction of radio waves propagation and its power per area unit.

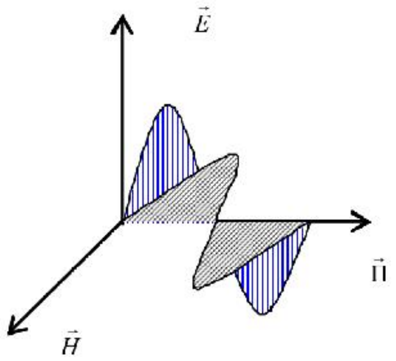


Fig. 7.6

In addition, the characteristics of radio waves are the polarization and frequency (or wavelength). Accepted by the receiving antenna the radio wave creates the radio signal, from which the valid component is separated. For radar station the parameters of received radio wave depend on the azimuth α and elevation angle β of object location and also on spatial delay $\tau = 2R/c$, where R is distance to the target; c is speed of radio wave propagation. By evaluating these parameters, it is possible to determine the location of landmark in space.

7.4.2. Characteristics of radar targets

The phenomenon of the secondary radiation, which is the basis of active radiolocation, is inherent to the waves of any nature. It occurs every time when wave meets an obstacle in the way of its propagation. The wave which is falling on an obstacle is called primary and the reflected or scattered one is secondary. The obstacle in this case is secondary passive radiator.

The nature of the secondary radiation depends on many factors, the main ones are electrical properties, geometry, motion and relative movement of elements of reflecting object, the ratio of the object size and length of the irradiation wave, the ratio of the object size and volume unit (the object is considered to be concentrated if it falls within the boundaries of volume unit, and volume distributed one if it takes a few permitted volumes), the law of modulation and polarization of irradiative electromagnetic wave.

Secondary radiation is divided into the specular reflection, diffuse scattering and resonant radiation. Specular reflection occurs during irradiation of smooth surfaces, dimensions of which are much greater than wavelength λ of the irradiative radio waves and dimensions of roughness do not exceed λ/16. In this case, the law of specular reflection is satisfied Z the angle of incidence equals the angle of reflection.

The property of the diffuse scattering has large surfaces with roughness of order of the wavelength of irradiative radio waves. Resonance radiation occurs when the size of the object is multiples of odd number of half-wave. Secondary radiation also depends on the size and configuration of reflective objects.

The main characteristics of radar landmarks are reflecting ability, statistical parameters and spectra of the fluctuations of the amplitude and phase front of the reflected signal.

7.4.3. Radar stations of ground surface surveillance

The radar stations of ground surface surveillance, which are designated for mapping the ground surface, for solving problems of aerial reconnaissance, etc., have high resolution that determines quality of radiolocation image, its details. This is accomplished by the significant increasing the size of antenna, located along the aircraft fuselage, that allows increasing the resolution in comparison with circular radar station with panoramic view, or by using the method of artificial aperture of the antenna, which allows approaching to the resolution of optical means of observation. So, the resolution does not depend on the range of observation and on the wavelength of the probing signal.

By type of observation the radar stations are classified by radar stations with side-looking observation and radar stations with omni-range observation.

Radar stations with side-looking observation, in turn, are classified by radar stations with sharp directed antenna, radar stations with synthetic aperture, radar stations with real aperture and panoramic view radar stations.

Radar stations with sharp directed antenna have beams, which are directed perpendicular to the axis of the aircraft. During the aircraft motion the irradiation of two strips occurs to the left and to the right of the track line, i.e. side-looking surveillance.

Radar station with synthetic aperture is coherent and its usual antenna becomes an "element" of some artificial grid during each radiation pulse. The distance between these elements is determined by the movement of aircraft. Radar station antenna moves on a straight-line trajectory, consistently taking position 1, 2, ..., *N*. In each location the antenna operates on the transmission and reception, that is, emits probing pulse and receives the signal reflected from the point target in the form of flat electromagnetic wave.

Radar station with real aperture solves the problem of range resolution by using radiation of short pulses. Azimuthal resolution is determined by the width of the antenna direction diagram (ADD) and range resolution is determined by duration of pulses.

Most of the panoramic view radar stations have rectangular antenna, larger side of which is oriented along the direction of the aircraft, and aperture is positioned so that the antenna beam is directed away from the platform. Antenna direction diagram in particular plane is inversely proportional to the size of the antenna in this plane.

Radar stations of circular observation have fan-like ADD in vertical plane. Since areas of the ground surface, which are on different distances, have different effective reflecting areas and give the reflected signals of varying intensity, then when the signals get to the control electrode of the cathode-ray tube with radial-circular sweep, these signals create an image similar to map of the area.

By the method of processing signals the radar station are classified by the radar stations with optical and digital processing.

By the range of operating frequency the radar stations are classified in accordance to international standards, adopted in IEEE.

7.4.4. Principle of operation of ground surface surveillance radar stations

The intensity of the reflection or scattering of radio waves (the intensity of the secondary field) depends on the degree of difference of electrical characteristics of the object and the medium, on the shape of the object, on the ratio of its size and wavelength, on polarization of waves.

Secondary resulting electromagnetic field consists of the reflection field that propagates toward the irradiating primary field and shadow field that propagates beyond the object (in the same direction as the primary field).

With the help of the receiving antenna and the receiver the part of the scattered signal can be received, converted and amplified for further detection. Thus, the simplest radar station may consist of transmitter that creates and generates radio signals, transmitting antenna that emits these radio signals, receiving antenna that receives reflected signals, radio receiver that amplifies and transforms the signals and finally output device that detects the reflected signals (Fig. 7.7).

As a rule, the amplitude (or power) of the received signal is low, and the signal has random nature. Low signal power can be explained by large distance to the object (target) and by the energy absorption of the signal on its propagation. In addition, the size of targets significantly affects the intensity of the reflected signal.

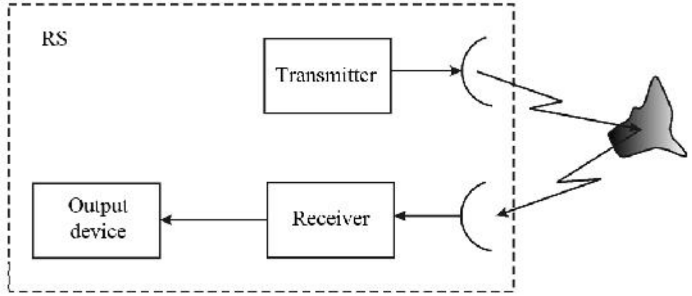


Fig. 7.7

The random nature of the signal is the result of the fluctuation of the reflected signal due to the random movement of elements of complex shape target during reflection of radio wave, random changes in the amplitude of the signal during propagation and other factors. As a result, the signal is accepted similar to noise and interferences in the receiving channel by type, intensity and nature of changes. So the first and primary goal of radar station is to identify valid radio signal, that is, decision making on the presence of valid signal. This statistical problem is solved by special detection device, in which optimal algorithm of detection of valid signal is used. The quality of the detection process is characterized by the probability of correct detection of valid signal, and by the probability of false alarm, when the interference (noise) signal is taken as valid.

There are serial and parallel variants of surveillance. In serial surveillance one and the same device analyzes the radar signal in each element, passing them in sequence, one after another. This single-channel system is maximally simple in structure, but needs plenty of time for observation in the whole surveillance area.

At the same time there are losses of signals energy of objects in the elements, which are not inspected at the moment. In parallel surveillance the multi-channel device is used, every channel of which processes the signals from one element. Channels work almost simultaneously, so the surveillance of the whole area takes approximately the same time, required for surveillance of single element.

However, the required number of channels can be so high, that the realization of such systems will be complicated. In pulsed radar station the surveillance can be carried out by using an electronic indicator with amplitude labels. Pulses with duration τ create labels, dimensions of which correspond to the distance $\zeta D\, X \varnothing / 2$ (here *c* is speed of light), which determines the resolution of radar station by range. This radar system of surveillance can be considered as parallel. The number of channels equals the number of target labels, which the operator can simultaneously observe on the line of scanning.

In the two-dimensional radar stations, in which one of the measured coordinate is range, two methods of surveillance are used: omnidirectional and side-looking (Fig. 7.8). During omnidirectional surveillance the antenna with beam, which is narrow in the horizontal plane, continuously rotates or performs swinging around the vertical axis. As the output device the indicator of omnidirectional surveillance is often used with the mark in brightness. The radial scanning line on the screen is reproduced synchronously and in phase with the antenna rotation. The combination of scanning lines creates a raster in the shape of circle or sector. When swinging the beam, except of sectoral, there is also rectangular scanning, at which the distance and direction to the target are reproduced in rectangular coordinate system.

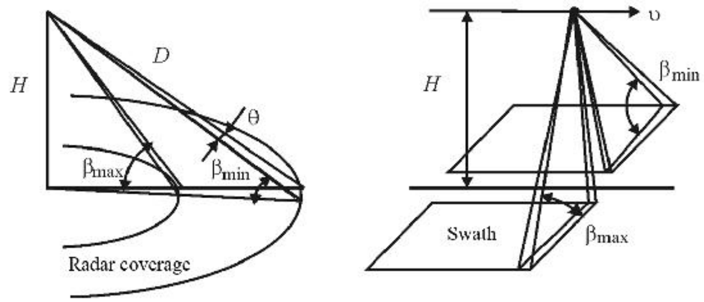


Fig. 7.8

Signals of point target, that is entered boundaries of antenna beam, create the label, the angular size of which $\zeta\Im$, excluding the light spot, equals to the beam width $\forall$. Thus, the dimensions of the radar coverage element are equal to $\zeta\Im = \forall$ by the angle and $\zeta D\, X \varnothing / 2$ by the range.

For maximum range D_{max} of observation the time of one probing of the most remote target is $2D_{max}/c$. If it is assumed that for its observations the energy of *n* periods of irradiation should be accumulated, then for surveillance of one angular element the time $n2D_{max}/c$ will be taken, and for the whole sector *A* the minimum amount of surveillance time will be $\varnothing_0\, X 2D_{max} f A\;\; n A / f c\;\; \forall A$ All points of surveillance area, regardless of their distance, are irradiated during the same time intervals.

Omnidirectional or sectoral surveillance is used both in stationary and mobile radar stations, if speed of carrier is relatively small. Side-looking surveillance is used only in airborne radar stations on fast moving platforms Z aircraft, helicopters, space vehicles. Fixed beam of antenna is oriented sideway of the flight direction and moves forward through its own movement of radar station (Fig. 7.9).

The radar coverage area has form of strip. The time required to observe the most remote target is $n2D_{max}/c$ and must be equal to the time $\forall D_{max} / \Delta$ during which the beam in its forward movement crosses the target at speed Δ. For other targets this time decreases inversely proportional to the distance.

In the two-dimensional radar stations (for example during the surveillance of the ground surface) fan-shaped beam is usually used, since different distances correspond to different elevation angles. The output devices of side-looking radar stations are special photo recorders, which record images of swath on the film.

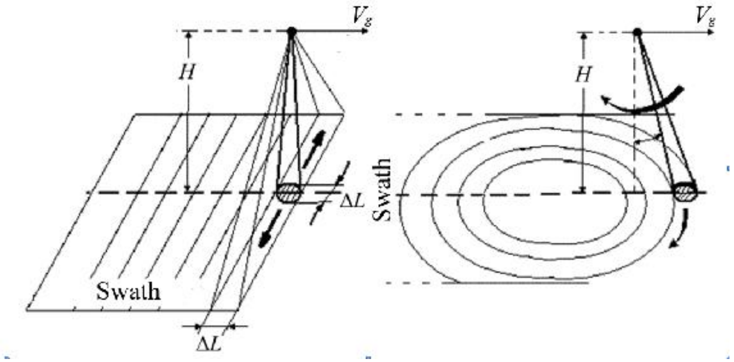


Fig. 7.9

Methods of spatial scanning with needle beam are used for surveillance of the surface under the aircraft, where the range resolution is very small. So, for surveillance of the swath along the direction of the movement of radar station the ordered method of surveillance is used (Fig. 7.9). Transverse lines are formed during the swinging of the antenna beam and are shifted as a result of the movement of the aircraft. Cycloid method of surveillance has certain advantages, when the beam rotates for cone generatrix, and as a result of translational movement the trace looks like a cycloid on the ground surface.

The most common radar stations of surveillance of the ground surface, which are used on civil airplanes, include meteo-navigation radio locators such as "Groza" and "Buran".

7.4.5 Meteo-navigation radar station "Buran A-140"

The station is designated to operate aboard the major aircraft and aircraft of local air-lines of civil aviation, in particular the aircraft An-148.

MNRS (Meteo-navigation radar station) solves the following problems: review of airspace in order to obtain and display on the indicator screen the information about meteorological formations, found in areas of intense rainfall, zones of turbulence, assessing the degree of danger at a distance, sufficient to bypass them at a safe distance; review of the vertical cross-section of meteo-formations in a given direction; review of terrestrial and aquatic surfaces to perform navigation by typical terrestrial and aquatic benchmarks; identification of mountain peaks, high rise buildings, aircrafts, standing in the radar coverage; determining the coordinates (slant range and heading angle) to the observed radiolocation objects (RLO).

To perform various tactical tasks in the radar station there are two main modes of operation: "GROUND" and "METEO".

To evaluate the meteorological situation and to select the optimal flight route in simple and complex weather conditions, the station "Buran A-140" in mode "METEO" analyzes the state of detected meteorological formations with the most informative characteristics.

In this case the following problems are solved: determination and display of the degree of danger of meteorological formations and their separate zones by the intensity of precipitation (radar reflection) in the form of zones of green, yellow and red colors; detection of zones of dangerous turbulence in meteorological formations and displaying them in violet color; getting the vertical cross-section of meteorological formations in the chosen direction in order to evaluate their vertical development (sub-mode "PROFILE").

The listed functions are extended by the mode of memorization of the image on the indicator screen "MEMORY". In this mode, images of objects are stored in two adjacent passing of the radar antenna and superimposed on each other. In this case, due to the relative motion of the aircraft and of meteorological formations, a trace is created that allows determining their relative displacement. The trace is marked with blue-green color.

In order to perform navigation by the feature ground and water landmarks, the radar in the mode "GROUND" forms the radar map in the coverage area with the allocation of four groups of landmarks such as water, point, extended ones and background of the flyover surface. Their set allows the pilot to recognize the terrestrial objects and determine the aircraft location.

The group of water landmarks includes rivers and lakes, as well as the water areas of seas and oceans. On the indicator screen, water landmarks are displayed in black color and are identified by the configuration of the coastline, shaped with background image of the flyover surface. The background of the flyover surface is created by unbuilt land areas, open spaces with herbaceous cover, forest arrays. The background is displayed in dark green color on the indicator screen.

The group of point-based landmarks includes railway bridges, metal supports of high-voltage transmission lines, ships, large constructions of industrial enterprises, as well as settlements at great distance. On the indicator screen, the point landmarks are displayed in violet color.

The group of extended landmarks includes large, medium-sized cities and towns, industrial centers, mountains and islands. On the indicator screen, extended landmarks are displayed in red color, and landmarks below the selection threshold are of bright green color.

Images of the groups of water, point and plane landmarks, as well as the background of the flyover surface are formed in the radar automatically. The pilot can manually select the objects of interest.

Calibration labels of range and heading angles are used to determine roughly the coordinates of feature ground and water landmarks, as well as meteo formations and dangerous areas. They are displayed on the indicator screen in blue.

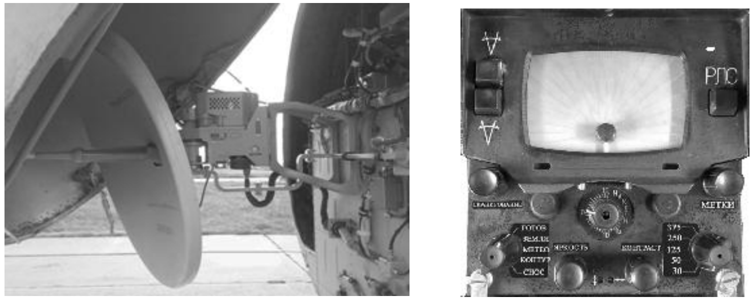
The electronic viewfinder is used for accurate measurement of slant range and heading angle of selected terrestrial and aquatic landmarks and boundaries of meteo formations and dangerous areas. They are displayed on the indicator screen in white. Electronic viewfinder can also be used to calculate the speed and direction of movement of the selected landmarks. The numerical values of the slant range and the heading angle of viewfinder are displayed on the indicator screen in white. The accuracy of measuring the coordinates by means of electronic viewfinder by slant range is of 0.4% of the established range and by angle is of 0.2°...0.3°.

Operation and structure of radar "Buran A-140"

The operation of "Buran A-140" is based on the radiation of the narrow beam of antenna direction diagram of the powerful radio frequency pulses, on reception, amplification, digital processing and accumulation of signals reflected from the ground surface, terrestial, aquatic objects, meteorological formations and the displaying of radar information on the screen.

Coverage view of space is carried out by the antenna directional diagram by scanning in the azimuthal plane at different tilt angles.

The antenna is of slotted-guide type Z antenna block (Fig. 7.10, *a*). The control of the operating modes of radar ("TEST", "GROUND", "METEO"), of sub-modes ("TURBULENCE", "PROFILE", "TILT", "AUTOMAT", "STABILIZATION") and of regulation is carried out with the help of controls located on the front panel (Fig. 7.10, *b*).



a *b*

Fig. 7.10

The functional scheme of the radar is shown in Fig. 7.11.

Radiation of microwave pulse. The synchronization of the radar transmission channel is carried out at frequency of 400 Hz from the power supply. The modulator 2БМ of transceiver generates high-voltage impulses which arrive at magnetron that generates microwave pulses. The circulator switches the antenna from the receiving to the transmitting channel. The ferrite rotary device of the polarization plane changes the polarization of high frequency oscillations. Depending on the polarization of the microwave oscillations of the reflector of the antenna block, either fan diagram or narrow beam is formed.

Reception of reflected microwave pulses. Reflected from radio contrasting obstacles, signals (microwave pulses) taken by the

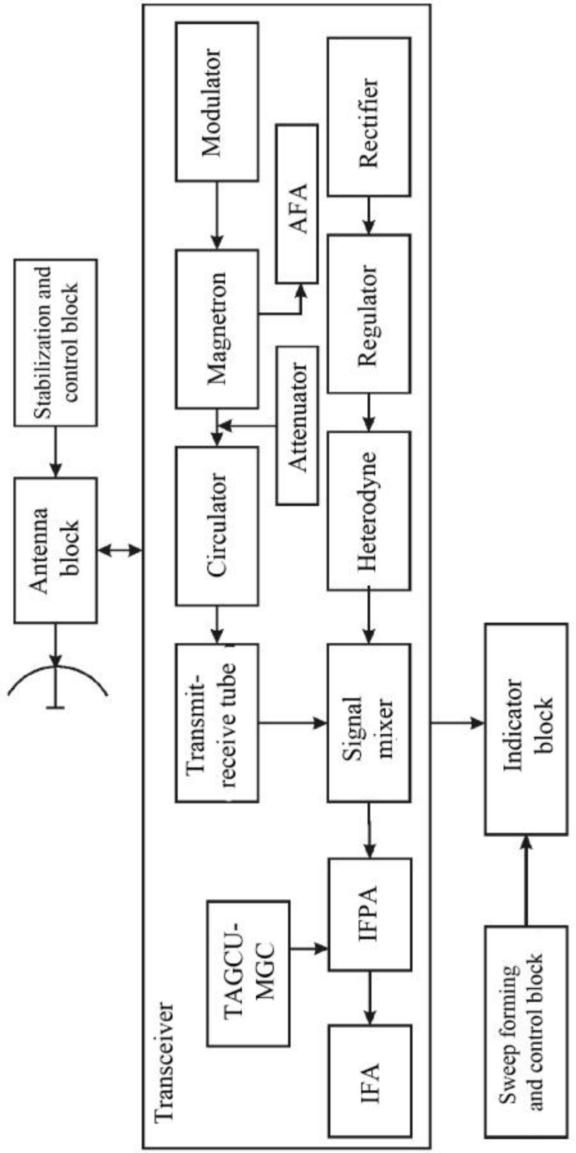


Fig. 7.11

antenna block of the radar, along the waveguide through the circulator, gate and transmit-receive tube come to the signal channel mixer. The mixer also receives the microwave signal from the heterodyne. After conversion from the mixer output, intermediate frequency pulses are fed to intermediate frequency pre-amplifier (IFPA), then into intermediate frequency amplifier (IFA), where the intermediate frequency signals are amplified and detected.

The temporal automatic gain control unit (TAGCU), which starts synchronously with the radiation impulse, regulates the gain of the receiver after the radiation of the microwave pulse, and also provides its closure for the duration of the powerful pulse of the magnetron. Manual gain control of the receiver (MGC) is carried out using variable resistor.

Automatic frequency adjustment (AFA). The circuit of AFA is used to maintain the constant difference in the frequencies of the magnetron and the heterodyne (intermediate frequency). A part of the microwave energy of the magnetron generator is fed to AFA mixer through the limit attenuator. The mixer also receives microwave signal from the heterodyne. After the transformation at the mixer output, impulses of intermediate frequency are formed which enter the AFA input, which produces the voltage proportional to the deviation of the intermediate frequency from its nominal value. This voltage, acting on the control electrode of the heterodyne, leads to the change in its frequency, reducing the deviation of the intermediate frequency from the nominal value.

Indication of signals. From the IFA output of transceiver, the video signal is fed to the input of the video amplifier of the indicator block. The characteristic of the video amplifier in the operation of radar in the mode "GROUND" is stepwise, in the mode "METEO" it is linear with indication of maximum amplitudes of signals in the direction of each heading angle. In the video amplifier there is amplification of video signal and mixing it with the calibration labels of the range produced by the sweeping unit. In addition, the illumination pulse comes to the video amplifier from the sweeping unit. This pulse provides backlighting of the direct sweep on the CRT screen. The amplified video signal, mixed with the calibration labels, enters CRT. With the help of sweeping unit the radial-sector scanning is created in the coordinates of the azimuth-range on the screen.

Synchronization of radar channels operation. The transceiver modulator generates the blanking pulse and start-pulse, which synchronize the operation of the indicator and receiving channels of the radar. Start-pulse is formed at the moment of microwave radiation pulse. From the output of the modulator, the start-pulse enters the sweep forming and control block.

The sweeping block produces saw-like sweep current and the backlighting pulse, the beginning of which coincides with the moment of arrival of the start-pulse, that is, with the moment of radiation of the microwave pulse. In addition, the sweeping block forms the calibration labels of the range, the first of which coincides with the moment of radiation, that is, with zero distance. Start-pulse switches on the power supply of the indicator block.

Basic technical data of RS "Buran A-140"

1. Maximum detection range, not less, (km):

- cumulonimbus cloud formations Z550;
- large cities Z 590;
- background of medium-rugged terrain Z 300...360;
- coastal lines of seas, large reservoirs Z 100...150;
- small towns, settlements Z 50...100;
- industrial objects, bridges, dams Z 40...80.

2. Basic radar parameters:

- readiness time for operation, not more than 3 minutes;
- time of view by azimuth in wide sector, not more than 6 s;
- time of view by azimuth in narrow sector, not more than 2 s;
- limit angle of the antenna by inclination angle, upward Z 15.75⌠; downward Z 16⌠;
- frequency of radiated microwave oscillations, 9345 ± 15 MHz;
- sensitivity of the receiving channel for RLO, not less, 100 dB/mW;
- pulse power of radiated microwave oscillations, not less, 3 kW;

3. Values of direct visibility range depending on the flight altitude is given in Table 7.1.

Table 7.1

Flight altitude, m	Direct visibility range, km
200	60
500	100
1000	150
3000	250
5000	300
7000	350
9000	400
11000	450

4. Radar power:

- on the network ~ 115 V 400 Hz, not more, 125 V A;
- on the network = 27 V, not more, 180 W;

5. Mass of radar, not more, 24.5 kg.

7.4.6. Advantages and disadvantages of radar system of the ground surface surveillance

Panoramic radar stations

The increase in the resolution of aircraft radar stations is limited primarily to the complexity of the placement of rotating antennas of circular view. On the other hand, the decrease in the wavelengths of radio waves and the transition, for example, from centimetric to millimeter waves, is not always promising, especially for long range radars. Millimeter waves, as known, do not allow realizing the main advantage of radar equipment Z the independence of obtaining information on the meteorological conditions. The peculiarity of the use of millimeter waves in radar is the following. With its propagation the radio emission attenuates in the atmosphere and in atmospheric formations. By the interaction of radiation with the environment there are processes of scattering, attenuation and depolarization of radiation, as well as amplitude and phase distortion of signals. Attenuation of radio emission in the atmosphere has general tendency to increase with increasing frequency and depends on weather conditions. However, on millimeter waves, the absorption intensity of radio waves is not so high as for the submillimeter range and caused by the presence of oxygen molecules and water vapor.

In general, millimeter waves are related to waves with varying range of action due to relatively large molecular absorption in water vapor and air oxygen, as well as through attenuation in different layers of the atmosphere

For evaluation of the possibilities of obtaining detailed radar images, the linear resolution by azimuth is of decisive importance. It is decreasing with increasing distance to the targets, which significantly impairs the efficiency of panoramic radars over long distances. Therefore, in solving problems associated with the detection and recognition of small-scale objects such as aerial reconnaissance, mapping, etc., panoramic radar systems are limited in use.

Side-looking radar stations

The radars of the side-looking view are considerably inferior in its capabilities to optical devices. Since, unlike the aerial photographing taken with the help of reflected sunlight, the radar antenna is itself the source of illumination for the observed area, then the image is formed due to reflected radiation of radar. It is possible to assume that the waves from the radar are distributed in a straightforward manner, therefore, the areas enclosed by hills or other large vertical objects are not illuminated. Therefore, they do not return back ultrahigh-frequency (microwave) impulses. Shaded areas on radar images are perceived as dark void. They do not resemble the lightly illuminated areas of ground surface by sunlight scattered in atmosphere which are in the shadows, for example, when photographing. The nature of the radar image depends on the wavelength and the polarization of the incident signal, as well as on the geometric characteristics and electrical properties of the displayed areas of the terrain.

When approaching the viewing area to the track line, the resolution by azimuth sharply worsens. This does not allow getting high-resolution radar images in the front viewing area.

The advantages of panoramic radars are the following. The coverage area is circle or sector with the radius corresponding to the radar range. The side-looking radars have the advantages of high angular resolution; the irradiation time is much larger than with the circular view; much larger radar range.

However, the main common disadvantage of all radar overview-comparative systems is the need to identify the landmarks with the help of the pilot-operator, which limits their functionality. Full automation of navigation for overview-comparative systems is possible only in CENS.

7.5 Correlation Extreme Navigation Systems

CENS is a system of processing the information, presented in the form of random functions (fields), intended for determination of coordinates. The basis of operation is the correlation connection between the realization of random functions, and the determination of the initial values (coordinates of the location or their derivatives) is carried out by searching for the extremum of the correlation function or any other statistical evaluation of the realization of random functions.

Navigation using CENS is carried out with the help of information derived from geophysical fields with random structure, parameters of which are closely related to certain areas of the ground surface.

Object control is carried out by determining its location in the process of comparing the current (taken in motion) distribution of the field with the reference distribution (mapped previously) of the same field, which is connected to the area with high accuracy.

Since the distribution of the current and reference fields along the route are random processes, the degree of their matching can be determined by the magnitude of the cross correlation function. The maximum (extremum) of this function will indicate that the current realization of the field coincides with certain section of the reference map of this field whose coordinates are known with high accuracy.

7.5.1. General structure of CENS

In the analysis of existing variants of CENS structure (Fig. 7.12), it is possible to distinguish the following common blocks: physical field sensor (sensors); cartographic block; correlator; automatic optimizer; georeferencing block; rough navigation system (RNS).

The physical field sensor provides information in the form of the current realization of the field (current field image $l_i(F)$), which can be different depending on the method of field probing (point probing, probing along the line of position, frame probing).

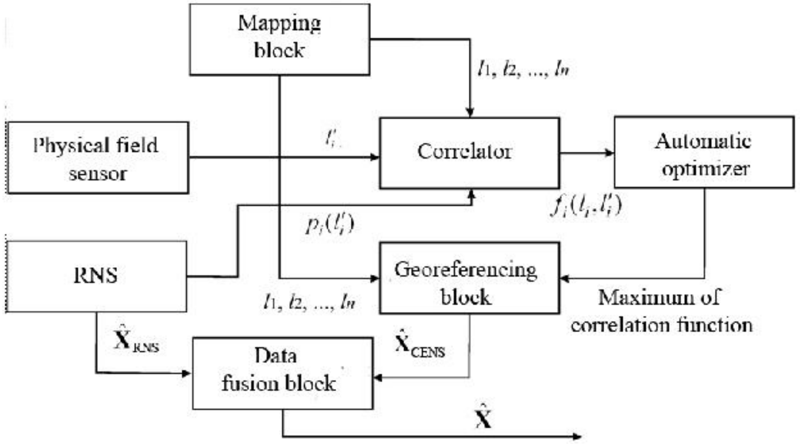


Fig. 7.12

For the first probing option, the field parameter is taken in the form of scalar value at each time moment, and it is possible to use both surface and spatial physical fields. For the second probing option, the sensor measures field parameters along the arbitrarily selected line instantaneously or over a short period of time. For the third option, the field parameters are measured from the ground surface during short scanning cycle.

The mapping block contains information about the reference realization of the field $l_1, l_2, ..., l_n$, which can be represented as regular or irregular grids, in the form of isolines or analytical model.

Correlator, depending on the type of CENS, can calculate the correlation function value $f_i(l_i, l_i')$ for each template stored in memory (so-called search CENS), or, in the case of GNS or prior knowledge of the flight path, can calculate the correlation function for only one template (so-called searchless CENS).

In the latter case, the presence of automatic optimizer is optional, since there is no need to find the extremum (maximum) of the correlation functions of the pairs of template and current field realizations.

The degree of coincidence of the template and current field realizations is the normalized correlation function. Correlation algorithms are selected based on the minimum of operations necessary to calculate the correlation function.

The simplest digital correlation algorithm is based on the use of paired functions which create the number of pairs of map elements with quantization levels i and j, which coincide with the shift ζx and ζz. The algorithm of summation of pair functions, that is the analog of the normalized correlation function, is realized as follows:

$$\partial_\phi(\zeta_x, \zeta_z) = N^{-1}\sum_{i=0}^{n-1} F_{ij}(\zeta_x, \zeta_z), \tag{7.6}$$

where n is the number of quantization levels, and N is the number of map elements. The algorithm for multiplying paired functions is described as:

$$\partial_\pi(\zeta_x, \zeta_z) = \sum_{i=0}^{n-1} F_{ij}(\zeta_x, \zeta_z)\left(\sum_{j=0}^{n-1} F_{ij}\right)^{-1}, \tag{7.7}$$

where $\sum_{j=0}^{n-1} F_{ij}$ is number of elements of the template image with the level of quantization equal to i.

For binary quantization ($n = 2$), there are four types of paired functions (Fig. 7.13) that make up the matrix

$$\begin{vmatrix} F_{00} & F_{01} \\ F_{10} & F_{11} \end{vmatrix},$$

where to simplify the recording is accepted $F_{ij}(\zeta_x, \zeta_y) = F_{ij}$.

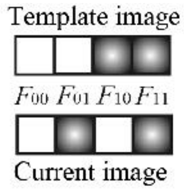


Fig. 7.13

Then algorithms (7.6) and (7.7) are written in the form:

$$\partial_\phi(\zeta x, \zeta z) = (F_{00} + F_{11})N^{-1};$$

$$\partial_\pi(\zeta x, \zeta z) = (F_{00}/N_0)(F_{11}/N_1),$$

where N_0 and N_1 are number of map elements containing respectively 0 and 1, and $N = N_0 + N_1$.

Let us consider an example of comparison of binary-quantized template and current maps containing $N = 10$ map elements and shifted to $\zeta x = 2$ (Fig. 7.14).

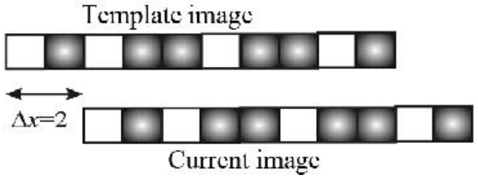


Fig. 7.14

With the image shift all paired functions are equal to 2, the numbers of matching elements 00 and 11 respectively are equal to $F_{00} = 1$ and $F_{11} = 3$.

Determining the value of paired functions are:

$$\partial_{\phi}\ X\frac{1\Gamma 3}{10}X0.4;\ \ \partial_{\pi}\ X\frac{1}{4}\ \ \frac{3}{6}X0.125.$$

By the next shift of the template realization relative to the current one, it is possible to reach the maximum value of the correlation functions equal to 1. Further image shift will result in a decrease of the values of the correlation functions.

Depending on the type of RNS used, CENS may differ in the degree of structural connectivity of systems and in algorithms of data fusion of navigation information. Since CENS itself cannot act as the sole and main source of navigation information for the vast majority of mobile objects, including unmanned aerial vehicles, then in future instead of term "GNS" the complex navigation system (CNS) will be used. Taking into account the CNS errors on the input of the correlator the prior probability of the object location comes in the form of the probability of coincidence of the current field realization with some template $p_i(l_i)$. Also, from the output of CNS to the data fusion block, the estimate of state vector $\hat{\mathbf{X}}_{\text{RNS}}$ is coming, which is specified using the previous navigation information $\hat{\mathbf{X}}_{\text{CENS}}$ from CENS georeferencing block.

7.5.2. Classification of physical fields

There are following classification features of the geophysical fields.

Physical origin

1. Relief field of the ground surface, which is characterized by mutual heights, that is the elevation regarding certain level, such as sea level. Surface elevation has geometric content that does not depend on objects which create these inequalities of ground surface. The measurement of the field parameter on the aircraft is done using radio altimeter or rangefinder with simultaneous using signals of barometric or inertial altimeters to set the reference profile of the terrain.

2. Optical field of ground surface, which is created by individual objects and their relative positions in the visible range of radiation. It is characterized by geometric and spectral characteristics of the image. Airborne measurement of the field parameters are carried by optical or television imaging devices.

3. Thermal field of ground surface is created by the electromagnetic radiation of individual elements (soil, water, vegetation, metal and concrete structures) in the infrared, the centimeter or millimeter wavelength range. It is characterized by temperature of these objects. This temperature is measured at a large distances by special devices (radiometers) which operate in corresponding wavelength ranges.

4. Radio reflection field (radar contrast field) is created by surface elements and is characterized by reflectance properties by areas in the radio wavelength range. Numeric values of reflectance coefficient are measured by airborne radar with scanning beam.

5. Magnetic field of the Earth (normal and abnormal in total) is caused by magnetic rocks in the Earth's core and its surface layer and is characterized by strength vector and direction of magnetic field strength lines. Airborne measurements of these fields are carried out by induction, ferromagnetic, quantum magnetometers.

6. Gravitational field of the Earth, created by the mass of the Earth and masses of natural and artificial formations on its surface, is characterized by the force of gravity. Airborne sensors of this field (gravimeters) are based on measurement of gravity force of reference masses installed aboard, to the ground surface in the given point.

In addition, there is a number of other fields (the field of natural gamma radiation, electrostatic, etc.) which are less stable in time or poor studied.

Spatial structure

1. Spatial fields are fields with parameters defined in each point of near-Earth space, e.g. magnetic and gravitation fields.

2. Surface fields have parameters defined only for the ground surface – relief field, thermal, optical, radio wave reflection coefficient fields.

Measurement of surface field can be done in the form of two-dimension or “volumetric” frames whereas spatial fields allow performing only point probing.

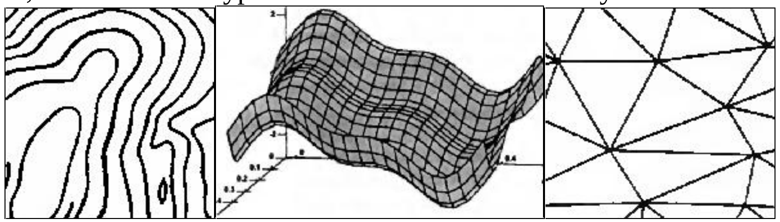
The most informative fields are surface fields because of the information content of any small-sized object on the ground surface in combination with increasing flight altitude due to so-called smoothing effect on height.

However, high informatively of spatial field is limited by such important disadvantages as weather condition, time of day of measurement, seasonal changes, sensitivity of airborne sensors and their noise immunity.

Form of representation on a map.

1. Fields are represented by continuous grid of points or in the form of analytical model. If the surface field is used, it is clear that that the surface can be described both in explicit and in parametric form. The main advantage of parametric representation is invariance of model to the geometric rotation.

2. Fields are represented by isolines (Fg. 7.16, *a*). Typically, these fields are represented as lines of equal values of the field, in general the curves are approximated by splines. As the basis function the polynomial, harmonic and hyperbolic function are commonly used.



a) *b)* *c)*

Fig. 7.15

3. Fields are represented by regular grid (*GRID*). The value of field is stored as two-dimensional array data, indexes match the coordinates of the surface (Fig. 7.16, *b*). Grid spacing is set according to the degree of geophysical data. Such representation of field has a number of significant advantages, namely: the information can be effectively compressed; data occupies less space, respectively, its processing requires less time, moreover, the possible use of complex nonlinear interpolation algorithms.

4. Fields are represented by irregular network (*TIN – Triangle Irregular Network*). This presentation is called triangular grid where the surfaces is represented in the form of triangle using triangulation algorithm, such as Delone triangulation. The features of this presentation is possibility to work with the geophysical data collected with variable sampling, interpolation of information at unexplored areas, support for on-line adding and updating mapping information. In addition, completeness of maps can be determined by the size of triangles (Fig. 7.16, *c*).

Principle of formation

According to the principle of formation, the fields are classified into natural and artificial. The natural ones include magnetic, gravitational, relief, optical, thermal and radio thermal fields. It is clear that the term "natural field" is rather arbitrary, because people and their activities affect the current realization of these fields, in particular magnetic field anomalies can be caused by the presence of large piece of metal structures, anomalies of radio thermal field may actually be different artificial interferences of radio systems.

The artificial fields are created by deliberate irradiation of ground surface and determined by the energy distribution of the reflected signal. They include radar, laser and ultrasonic long-range fields.

Type of measured signal

By type of measured signal the geophysical fields can be classified as fields with time, spatial, and spatial-time signals. The parameter measured by sensor of geophysical field, depending on the field and methods of its measurement can be represented as realization of time function, for example, with measuring the geomagnetic field or relief fields. The field is measured at point of space at time moment, and a set of discrete samples creates the time function. With observation by a frame the field value at each time moment is measured at certain surface area (surface field), such as optical field gives two-dimensional set of samples of contrast of the ground surface in the frame, described by the spatial signal. The set of such frames over time will already by described by spatial-time signals.

General classification of fields is shown in Table 7.2.

Table 7.2.Classification geophysical fields

Physical origin	By spatial structure		According to the principle of formation		By presentation on a map				By type of measuring signal		
	Spatial	Surfae	artificial	Natural	Analytical model	Isolines	Regular grid	Irregular grid	time	Spatial	Spatial-time
Terrain relief		✓		✓		✓	✓	✓	✓	✓	✓
Optical		✓		✓			✓			✓	✓
Thermal		✓		✓		✓	✓		✓	✓	✓
Radar contrast		✓	✓				✓		✓	✓	✓
Magnetic	✓			✓	✓	✓			✓		
Gravitational	✓			✓	✓	✓			✓		

7.5.3. Variants of existing correlation-extreme navigation systems

TERCOM system is guidance system of cruise missiles by the terrain relief. This system (in some sources - *TAINS-TERCOM* or *Aided Inertial Navigation System*) was developed by McDonnell Douglas for cruise missiles ALCM and "Tomahawk". The main components of the system are radio altimeter AN/APN-194, INS LN-35 and autopilot.

During the flight TERCOM system measures the vertical profile of terrain along the actual flight trajectory using radio altimeter (to measure geometric height) and pressure altimeter (for reference of the profile). Subtracting the current height measured by radio altimeter from pressure level, the system determines the profile of the relief along the flight trajectory and begins searching the similar one in computer memory that contains profile with known coordinates (Fig. 7.16).

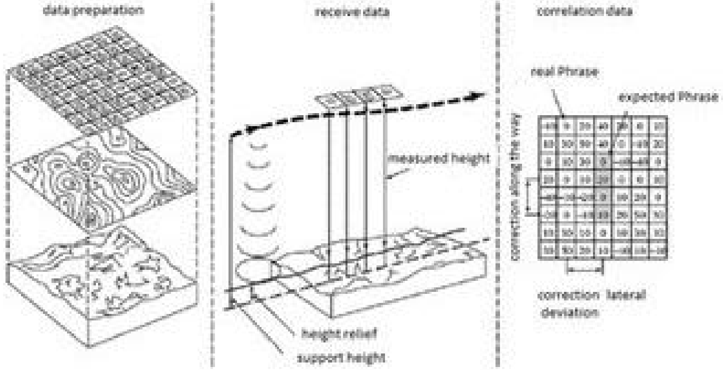


Fig. 7.16

Search is done by sequential comparison of field sensor signal with the digital map, first by viewing back and forth from the predicted location, and then across the trajectory within the matrix of 64×64 elements (at element size ζ from 30×30 m to 240 m $\times$ 240 m).

The procedure of comparison is the minimization of the absolute difference and can be represented approximately by the following expression:

$$\frac{1}{64}\sum_{i=1}^{64}\left|H_{true}(\zeta_i,\theta_i) - H_{\text{map}}(\zeta_i + j\zeta,\theta_i + v\zeta)\right| \to \min; \quad (1 \le j, v \le 64),$$

where H_{true} is elevation of relief along the actual flight path, H_{map} is relief elevation, selected from the airborne map; ζ_i, θ_i are object size.

Data fusion of INS with TERCOM system provides the accuracy of navigation solution rabout 50 meters.

TERCOM systems differ in the type of used algorithm of correlation search and filtering of navigation errors, namely such systems as *SITAN*, *VATAN*, *PTAN*. However, the use of these systems is possible only in areas with varying terrain relief and practically impossible above the water surface.

Systems of GGAINS type use abnormal gravitational and geomagnetic fields for correcting INS errors. Components of the gravitational field aboard the aircraft are measured using gravimeters, errors of which are within 0.5...0.6 MGal. The main sources of noise in the measurement of abnormal components of the Earth's gravitational field is vibration and overload during aircraft evolution which are eliminated by placing gravimeters on precision stabilized platforms and by further integration to INS structure. Components of Earth's magnetic field are determined by magnetometers with errors limited to 10...15 nT. The main types of magnetometers errors are instrumental errors associated with impact of artificial sources of magnetic field; errors of mapping of magnetic fields; random variation of the magnetic field; unbalance of magnetic fields of the vehicle, which generally limits the practical widespread use of magnetometers aboard.

The system GGAINS uses gravimeters and magnetometers as geophysical field sensors together with models of gravitational and geomagnetic fields to take into account regular components of fields and pre-recorded maps of gravitational and magnetic anomalies. Data fusion of CENS and INS is made by filter of adaptive estimation of errors (Kalman filter) for INS and by the method of weighted average for anomalies data.

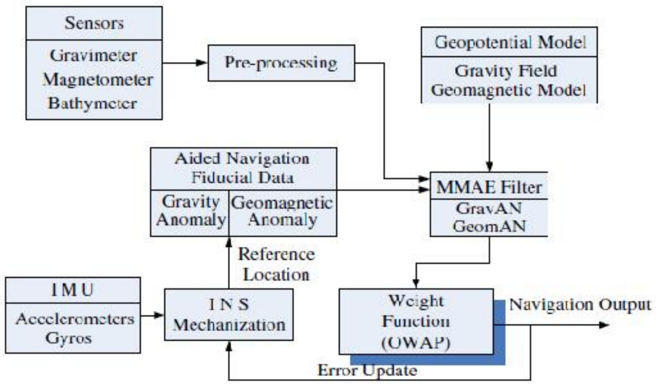


Fig. 7.17

Systems of VAINS type become of wide use because they can be related to the greatest number of systems, including systems that use radar contrast field Z RAC systems and MICRAD, systems with optical field of ground surface Z SMACI systems, OMFAC and others. The principal difference between these systems classes is in the use either optical information processing methods, providing higher speed, especially in real time modes, or digital methods. In particular, the system OMFAC is the optical correlator that uses coherent holographic methods of information processing.

The main advantages of the system OMFAC compared to other optical systems are the absense of moving parts in correlator and significantly larger memory volume. Increased memory allows the system to be used not only at the final, but also on the middle section of the flight trajectory, increasing also the possibility of maneuvering and approach to the target from different directions.

The principle of optical information processing is explained in Fig. 7.18, where *1* is plane of input signal, *2* is holographic lens matrix, *3* is memory device with matched filter, *4* is lens, *5* is photodetector plane.

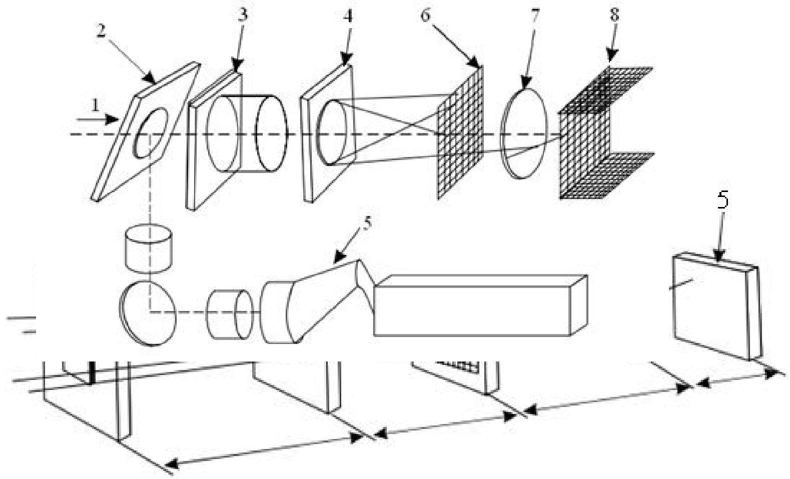


Fig. 7.18

At the plate a set of elementary Fourier transformers is coated. Every elementary hologram acts like a lens, deflecting the light beam passing through it in a certain point in the given focal plane. Memory device *3* is also photographic plate, on which Fourier holograms of template images are recorded. Through focusing lens *4* the resulting image which is the difference between the template and current images comes to photo detector *5* that detects the degree of deviation between them.

Functional diagram of OMFAC system is shown in Fig. 7.19.

Fig. 7.19

Here there are designations: current image *1*, semitransparent mirror *2*, converter *3*, holographic lens *4*, laser *5*, matched filter *6*, inverse Fourier transform lens *7*, photodetector *8*.

Since the current image of area *1* is obtained with incoherent sunlight, and the processing of information in the system uses properties of space-time coherence, then the converter *3* forms so-called equivalent image. Conversion of current image is performed in real time. Converter *3* is two-layer structure. It consists of thin layer of photoconductive material such as cadmium sulfide, and memory layer, such as, liquid crystal. With transparent conductive electrodes DC voltage 10...15 V is supplied to the converter.

With projecting the current image on the photoconductive material the resistance of photoconductor is reduced to seven orders. This electric potential is transmitted from the cadmium sulfide layer to the memory layer as a function of illumination of each point photoconductor. Then voltage from the converter is removed, and the coherent laser *5* radiation passes thought it. The laser beam passing through the converter is modulated by "image" amplitude, fixed in the liquid crystal, and enters the optical processor with matched holographic filters *6*.

To reproduce the resulting image the lens *7* of inverse Fourier transform is used, then the image is applied to photo detector *8* to get degree of mismatch between the template and current image.

The system can operate at incomplete input information, such as when clouds shade to 50% of the current image. With the angular displacement of template and current images of ± 3° and respectively with the scale ± 4° the level of output correlator signal is reduced to 3 dB.

7.6 Features of meteo radar station operation

Flight operation of systems. In the case of use with flight mission completeness the radar station is turned on before taxing to runway.

In flight, according to the instruction of the operation, RS is checked in the mode "GROUND". System serviceability is estimated by the appearance of radar images of the ground surface and by the ability to determine the range to the radar landmark and its heading angle.

In the mode “METEO” the size and degree of meteo formation risk are determined. With thunderstorm area detection the height of its formation relatively altitude of the aircraft is determined and the specific measures are taken to maneuver and avoid the storm front depending on its nature and development height.

It is forbidden to use radar stations as the basic tool of preventing the possibility of collision in the air or the danger proximity to ground surface and ground facilities.

While landing approach it is necessary to approach turn off the high-frequency antenna radiation, and with landing radar station must be completely turned off.

Technical exploitation of radar systems. Before turning on the system it is necessary to be sure that at distance of 100 m from the aircraft in sector 80⌠ there is no large objects with reflective surface and at distance of 30 m people are absent.

After turning on the system by pressing button «MAP/WX» the test starts (on the control display unit the word "TEST" appears and there is also the scale of test time). After test finishing (not more than 3 minutes) the word "TEST" changes to "RTW" (ready to work). There should be no reports of refusal on the airplane multifunctional display.

TEST QUESTIONS

1. What are the advantages and disadvantages inherent to comparative-overview navigation methods? What features can these systems be classified?

2. How is the problem of landmark recognition in manual overview-comparative navigation systems solved?

3. What purpose usually are overview-comparative point probing system usually used for?

4. What are two types of radar location?

5. Which factors does power of secondary radiation of radar field depend on?

6. What parameters of received radar waves is it possible to determine landmark location in space?

7. How is it possible to increase the resolution for radar of the ground surface surveillance?

8. How are the side-looking surveillance radars classified?
9. What are elements in structure of the simplest radar devices?
10. What is the main problem solved by radar in the analysis of the received signal?
11. What are the variants of surveillance at all? What surveillance methods are used in the two-dimensional radar?
12. What problems does "Buran A-140" solve?
13. What operation modes are implemented in "Buran A-140"?
14. What are the principles in the basis of CENS operation?
15. Give the classification features of the geophysical fields.
16. What field does CENS «TERCOM» use?
17. What methods of information processing does CENS «OM-FAC» use?