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PRECISION MECHANICAL INSTRUMENTS

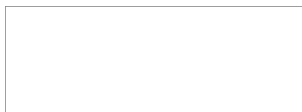
T.I. Vilgayeuskaya

AVIATION INSTRUMENTS AND AUTOMATIC PILOTS

(BRIEF COURSE)

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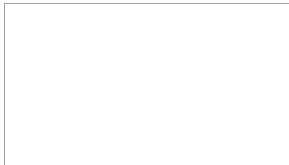


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This book is a textbook intended for students of the instrument-building technicians and has been compiled in accordance with the syllabus of the course in "Instruments of Precision Mechanics".

It sets forth the basic principles of operation of aviation instruments and autopilots, briefly describes the elements of design, and discusses the questions of the error of the instruments and the methods of eliminating such errors.

It gives an idea of the instruments that control the operation of the aircraft engine as well as that of piloting and navigation instruments and automatic equipment.



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PREFACE

Modern airplanes are equipped with instruments and automatic devices assuring continuous control of the regime of flight and solving the complex problems of automatic control and automatic orientation. The importance of instruments on aircraft is increasing every year. At the present time instruments have become one of the most important factors determining the general technical level of aviation. The development of aviation technology in turn has involved the improvement of the existing designs of aircraft instruments, the appearance of new designs, as well as the utilization of fundamentally new methods of measuring various quantities, that had not been previously in use. Aircraft equipment has undergone extensive quantitative and qualitative modifications. The number of items has increase, and entirely new forms of such equipment have appeared. Automatic devices for controlling flight and for operation of power plant, radio equipment, devices for piloting and landing aircraft under unfavorable meteorological conditions have been gaining ever wider use. In connection with the progress in the field of aircraft equipment the operational and tactical possibilities of utilizing aircraft have also expanded. The further development of aircraft equipment and, in particular, of instrumentation is proceeding along the lines of ever increasing automatization and increase in accuracy.

The present book has been written with respect to the syllabus of the course in "Instrument of Precision Mechanics" of instrument building technicums and contains materials on the section of the course entitled "Aviation Instruments and

Automatic Pilots".

The purpose of the book is to give an idea of the instruments controlling the operation of aircraft engines, the piloting-navigation instruments, and automatic devices. The book gives a description of the principle of operation of the individual instruments and, in the most general way, also presents the elements of design of instruments and gives a considerable amount of space to the errors of aircraft instruments. The question of errors is considered in greater detail, since the technical instrument builder must have a distinct idea of the causes for individual errors, of the methods for their total or partial elimination, and with respect to the possibilities of a given method of measurements, of the possible accuracy of operation of the instrument.

The book does not give a description of instruments that may be in question for aviation or of instruments actually in use but not typical for our modern USSR aircraft. Such instruments include: direct-current tachometers, tachometers with rectifiers, etc. The automation of the aircraft engine is not considered. The book does not include elements of calculation of instruments, question of installation, disassembly and operation of instruments. In cases where the reader requires more detailed study of some instrument (for example, in designing work) it will be necessary to consult the book by D.A.Braslavskiy, S.S.Logunov, and D.S.Pel'tor "The Calculation and Design of Aircraft Instruments" (Bibl.1) or one of the books given in the Bibliography at the end of this book.

I express my sincere gratitude to G.O.Fridlender for his valuable assistance with the book. I likewise express my appreciation to Ye.L.Veller, the editor of this book, for a number of valuable comments made in reading of the manuscript.

The Author

INTRODUCTION

The most important units of an aircraft are as follows:

1. The body, the main design part of the aircraft which houses the crew, power plant, equipment, and all units and devices forming a part of the design of the aircraft.

2. The power plant, including the engine with the systems of fuel feed, cooling, and lubrication, and also the propellers (for piston and turboprop engines).

3. The equipment, consisting of the instruments, mechanisms, units, and installations making it possible to control the aircraft to obtain optimum performance of its mission. The concept of aircraft equipment includes the technical means allowing:

- a. control of steering of the aircraft, flying speeds, and operation of the power plant, as well as mechanization and automation of these processes:

- b. assuring a more complete utilization of the flight-technical means of the aircraft in accomplishing its mission and increasing the safety of flight:

- c. providing the most pleasant working conditions for the crew and passengers.

The conditions of flight on modern aircraft, particularly on high-speed types would be so complex and would require such acute and sensitive sensory organs of the crew members, such an effort of memory and thought, such endurance, that in the best case it would lead to extraordinary fatigue, and in the worst case it would be entirely impossible to handle for the human organism, if technical equipment were not called into action.

In the complex environment of modern flight, even at zero ground visibility, the aircraft crew is rapidly able to solve, with the aid of instruments, the complex problems of position fixing, of orienting the aircraft with respect to the

ground, etc.

Aircraft equipment greatly simplifies the solution of many problems facing the aircraft crew. For example, the intercom system of aircraft and the means for outside radio communication allow a normal conversation both between the individual members and with crews of other aircraft or with the ground crew, despite the noise of the power plants.

The systems of control and measuring instruments, widely used on modern aircraft, and of transmission mechanisms allow the pilot to take the decisions required for a given condition of flight and to implement them without excessive efforts, by using various power transmission mechanisms and drives. In a number of cases, the latter task is considerably simplified by the use of appropriate automatic devices.

Thus, in spite of the progressive complexity of missions, the work of the crew is increasingly facilitated through further development and improvement of aircraft equipment.

The wide variety of modern aircraft instruments has been developed and improved over a period of many years, in the process of improvement of other aircraft equipment and in step with the continuous expansion and complication of the problems solved during flight. Some instruments were in existence considerably earlier than the first aircraft. For example, the magnetic compass and the methods of navigation by compass developed by the efforts of Russian scientists, such as Admiral I.F. Kruzenshtern (1770-1846), Lt. I.P. Belavents (1829-1878), Academician I.P. De-Kolong (1839-1902), and Academician A.N. Krylov (1863-1944), have found widespread application in aviation. Russian aircraft designers were the first in the world to apply the most advanced methods and instruments of the time with respect to navigation by compass as well as the methods and instruments of astronomic orientation of the aircraft. The Russian aircraft instrument building industry has always been, and still is, at a very high level.

The first navigation instruments were designed by Russian aeronauts as early as

1804, and the Academician Ya.D.Zakharov made a flight on a balloon equipped with a compass, an instrument for determining ascent and descent, and an optical telescope, i.e., instruments allowing the route of the balloon to be plotted on the map.

The famous Russian designer A.F.Mozhayskiy, in designing his airplane, thoroughly laid out not merely the design of aircraft and engine, but also took into account the purpose of the aircraft, i.e., he provided for equipment on board that was necessary for the completion of practical flights. Relying on the experience of navigation and ship building, he installed on his aircraft bank-and-turn indicators, altimeter, thermometers, a speedometer, and a compass. To him belongs priority in the design of the entire set of aircraft equipment.

In the 1890-s and at the beginning of the 20th century, Russian scientists designed a series of instruments for navigation and piloting. M.M.Pomortsev, in 1896, designed an instrument for determining the direction and angular velocity of clouds and, in 1897, an instrument to determine the velocity and direction of motion of a balloon, which was the prototype of modern sights that appeared considerably in other countries.

In 1898, the famous Russian scientist K.E.Tsiolkovskiy was the first in the world to propose the idea of an autopilot and to give its working diagram.

The founder of radio engineering, the inventor of the first radio transmitter and radio receiver, and the first person in the world to actually accomplish radio transmission and radio reception (1895) and to discover the principle of radio location, was the famous Russian scientist A.S. Popov.

The first heavier-than-air airplane in the world, the "Ruskiy vityaz'" constructed in 1913, was provided with a tachometer, clocks, altimeter, and compass.

The four-engine aircraft "Il'ya Muromets", constructed in 1914, was equipped with compasses, altimeters, speed indicators, clocks, and tachometers. On this aircraft the method of navigation by compass was successfully employed for the first time. At approximately the same time, A.N.Zhuravchenko designed an anemometer,

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V.A.Slesarev a speed indicator (which was a few years ahead of the American speed indicator "Pioneer"), V.P.Vetchinkin, an accelerometer, etc. In contrast to foreign designers, the designers of Russian aircraft always paid great attention to convenient working conditions for the crew and provided it with the equipment necessary for flights. Russian designers developed a number of very interesting instruments and devices, however, manufacture of these instruments in the necessary form and in adequate number proved impossible, since the development of aviation, and consequently also of the aircraft-instrument building industry, was not properly supported by the ruling class.

The Russian aviation industry and aircraft instrument building industry began their intensive development only after the October Revolution. In spite of the difficult conditions in the country at that time, the Soviet Government literally from the first days began to build an aviation industry and an aircraft-instrument building industry. Already in 1919 a Soviet plant "Aviapribor" turned out instruments for the air fleet, and in 1922 this plant changed to series production.

In 1923, this shop produced the following instruments: oil gauges, gasoline manometers, air thermometers, tachometers, altimeters, deflectometers, and speed indicators. The plant had many designers and research instrument builders whose names today are widely known (S.A.Kosdrovskiy, S.S.Tikhmenev, G.O.Fridlender, and others).

As a result of continuous close contact and work in collaboration with noted Soviet scientists, pilots, and designers, instruments and automatic devices were successfully used on aircraft and are still being created.

A Collective of Soviet instrument builders consisting of D.A.Braslavskiy, M.V.Kachkach'yan and M.G.El'kind has developed a number of instruments, including the first gyromagnetic compass in the world.

A.A.Andronov, B.V.Bulgakov, S.E.Khaykin, as well as the young Soviet scientist V.V.Solodovnikov, Ya.Z.Tsyppkin and others, have had exceptional success in the theory of automatic control and the theory of gyroscopic instruments. Utilizing the

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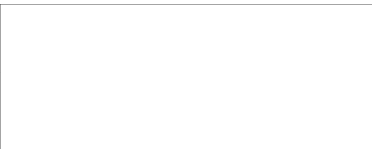
theoretical work of the scientists, the Soviet designers have constructed and are now constructing excellent models of instruments and automatic devices which are successfully used in aviation, and which lighten the work of the crew and assure safety of flight.

The use of automatic devices is particularly widespread for piloting aircraft (autopilots), automatic devices controlling aircraft power plants, and computers for automatic plotting of an aircraft course (automatic navigators).

Such intense development of the USSR aircraft-instrument building industry has been possible only on the basis of the success of Soviet scientist and designers in the theory of regulation, the theory and design of various types of automatic devices, electric measurement, gyroscopic and other instruments.

Such work in the field of metrology and in the study of the errors of measuring instruments have been of particular importance.

The further improvement in the technical level of our USSR aviation will confront Soviet instrument builders with a number of problems connected with the creation of new types of instruments and automatic devices and also with the introduction of the most recent achievements of science and industry and technology into production.



CHAPTER I

PURPOSE AND APPLICATION OF AIRCRAFT INSTRUMENTS

Depending on the purpose, the instrumental equipment of an aircraft is divided into the following groups:

1. Instruments controlling the operation of the aircraft engines;
2. Piloting and navigational instruments;
3. Automatic devices controlling the operation of the aircraft engines as well as automatic piloting devices (autopilots).

Section 1. THE POWER PLANTS of MODERN AIRCRAFT and the INSTRUMENTS

CONTROLLING THEIR OPERATION

Modern aircraft are equipped with piston engines (PE) turbojet engines (TJE), jet engines (JE), or turboprop engines (TPE). At the present time piston engines, air- or liquid-cooled, are still widely used. Jet engines are used on high-speed aircraft. Turboprop engines whose appearance has been relatively recent, are beginning to be used more and more. The necessity of using a propeller with such engines limits their application; such engines are unsuitable for high-speed aircraft.

The number and types of instruments controlling the power plant depends on the special features and type of the power plant. However, as shown below, their nomenclature varies only slightly.

Light-Fuel Piston Engines

The operation of such an engine is based on the conversion of the thermal energy of a combustible mixture burned in the engine cylinders into mechanical work, i.e., rotating the blades of a propeller, thus creating thrust.

For the exact setting and maintenance of the operating condition of an engine it is necessary to:

1. Know the fuel supply on the aircraft; its control is accomplished by the

fuel gage.

2. To supply air and fuel to the engine in a definite proportion and under a definite pressure. Indication is by means of the fuel gage and the vacuum gage (with forced air feed of the engine, i.e., with supercharging used for maintaining a definite ratio of oxygen to gasoline in the combustible mixture).

3. To assure the uninterrupted supply of lubricant to the friction parts of the engine. In this case it is necessary to control not only the pressure, under which the oil is supplied, but also its temperature. At a low temperature (below 10-15°C) the viscosity of the oil increases sharply, its rate of flow through the pipe lines is diminished, and its feed through channels of small cross section (for example, to the engine bearings) is impeded. At high temperatures, the viscosity of the oil decreases, it acquires fluidity, and adheres poorly to the clearance between the friction parts. At excessive temperatures, oil will burn, and the products of its combustion clog the wearing surfaces. Control of the oil system is accomplished by pressure gages and oil thermometers.

4. Maintain the temperature of the cylinders and pistons within the allowable limits.

During combustion of the fuel mixture the engine cylinders are strongly heated; to avoid overheating, cooling is used. Depending on the method of dissipating the heat, aircraft engines are conventionally divided into air-cooled and liquid-cooled engines. In air-cooled engines, the temperature is checked by means of a cylinder-head thermometer while in liquid-cooled engines, coolant thermometers are used.

Not only overheating but also overcooling of the cylinders is dangerous for engines, since in this case the rate of combustion of fuel-air mixture is reduced. An engine can lose its pickup, i.e., its ability to shift rapidly from one regime of operation to another. The loss of pickup is particularly dangerous in landing when, in some cases, the propeller rpm must be rapidly increased to prevent loss of air speed.

To maintain the necessary pickup, the gasoline must be evaporated at a sufficient rate in the carburetor. The rate of evaporation depends on the temperature of the carburetor, which is indicated by the carburetor thermometer.

5. Know the speed of the engine shaft. This value is measured by the tachometer.

The instrument indicating the composition of the fuel mixture is of great importance in the operation. However, attempts to design instruments of this type have not yet given the desired result.

The gas analyzers used for this purpose allow the composition of the fuel mixture to be determined from the composition of the exhaust gas. The considerable errors inherent in this instrument interfere with its widespread application.

Turbojet and Turboprop Engines

The operation of jet engines is based on the reactive action of the jet of gases formed by the combustion of fuel and expelled through a channel of small cross section having the form of a nozzle. For the combustion of fuel in jet engines atmospheric oxygen is used (air-jet engines) or special oxidizers (liquid-jet engine).

The performance of a jet engine is characterized by the rate of revolution of the turbine, the temperature of the gases in the jet nozzle, the temperature and pressure of the fuel and oil, the consumption of fuel and oxidizer, the temperature in front of the turbine, the Mach number at the entrance to the compressor, the static pressure, etc. These are the same parameters that are measured in the operation of piston engines, but the limits of measurements of many of these quantities are considerably wider for a jet engine. Since the instruments for controlling the operation of a jet engine are designed with allowance for the peculiar features of operation and for the range of operation, the possibility of using instruments based on completely different principles, not used in control instruments for piston engines, is not excluded.

The instruments used to control the operation of a turboprop engine (TPE) are

analogous to the instruments used for jet engines.

Instruments and Automatic Devices for the Power Plant

For reliable and economic operation of an aircraft power plant as well as for obtaining maximum thrust or power, it is necessary to provide under all conditions of flight, the most advantageous regime of operation of the power plant and to constantly check its operation by means of control instrument and automatic regulating processes.

To facilitate the work of the crew during flight every effort is made to use automatic equipment. The control instruments for operation of a power plant are subdivided into the following groups:

1. Instruments whose readings characterize the thermal regime and condition of the engine lubricant: the oil thermometer, coolant (or cylinder-head) thermometer, the working-gas thermometers (for jet engines) and the oil pressure gages.
2. Instruments indicating the power or thrust developed by an aircraft engine: vacuum gages, manometers, tachometers, thermometers, and gas analyzers.
3. Instruments indicating the fuel reserve and fuel consumption and the oil reserve: fuel gages, flowmeters, oil gages.

The automatic regulators of the regime of operation of aircraft engines include the following:

1. Automatic engine speed control.
2. Automatic coolant and oil-temperature controls and automatic cylinder-head temperature controls.
3. Automatic switches for supercharger speed.
4. Automatic boost pressure controls, etc.

Section 2. AIRCRAFT FLIGHT AND PILOTING INSTRUMENTS.

The Navigational Regime of Flight.

Every flight is connected with the fulfillment of an assignment defined by the

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navigational regime of flight, i.e. speed, altitude, and direction.

To maintain the regime of flight the pilot must continuously maintain the equilibrium of the aircraft in the air and check its position with respect to the ground.

These tasks are performed by the aid of instruments. The instruments controlling the operation of the power plant allow the necessary performance of the aircraft engine to be selected and maintained.

The piloting-navigation instruments make it possible to determine the position of the aircraft and its speed. According to the weather conditions, the pilot establishes and maintains the required navigational regime of flight by the aid of one or the other group of instruments.

Various causes, for example, gusts, variation in thrust of the propellers and other causes, may change the position of equilibrium or cause a deviation from the selected course. For this reason the aircraft crew must be continuously able to check the position of the aircraft in space and to restore it to the required attitude.

A deviation of the aircraft may occur with respect to the xx , yy , and zz axes (Fig.1a).

Two systems of coordinate axes are differentiated: The moveable system, invariably connected with the aircraft coordinate system $Oxyz$, with the initial coordinate in the center of gravity of the aircraft, is called the bound system of coordinates, while the xx , yy , and zz axes are called the principal axes of stability.

The equilibrium of the aircraft with respect to the xx axis is called transverse equilibrium and with respect to the zz axis, longitudinal equilibrium.

The longitudinal axis Ox , parallel to a wing chord, is directed forward and lies in the plane of symmetry of the aircraft.

The normal axis Oy perpendicular to the axis Ox , lies in the plane of symmetry of the aircraft. During horizontal flight of the aircraft, this axis is directed upward.

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The transverse axis Oz is perpendicular to the plane of symmetry of the aircraft (positive values are measured on the side of the right wing). The planes of the coordinates in the bound axes have the following designations:

- Oxy : the plane of symmetry of the aircraft;
- Oxz : the plane of the wings, or the principal plane;
- Oyz : the transverse plane.

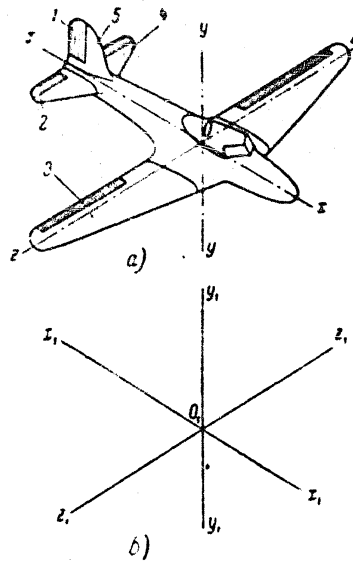


Fig.1. Coordinate Axes of Aircraft

- a- $Oxyz$ system fixed relative to the aircraft;
- b- $O_1x_1y_1z_1$ system fixed relative to the ground;
- 1- Rudder; 2- Elevator; 3- Ailerons
- 4- Stabilizer; 5- Fin.

The axes (Fig.1, b) fixed relative to the ground (the so-called ground axes) are selected in the following way: the axis O_1y_1 is directed vertically upward from below. The axes O_1x_1 and O_1z_1 are located arbitrarily in a horizontal plane and include an angle of 90° . The origin of coordinate is selected arbitrarily.

The position of the fixed axes with respect to the ground axes, and consequently also the position of the aircraft with respect to the ground, is determined by the angles θ , γ , and k (Fig.2). The angle θ between the xx axis and the plane of the horizon is called the angle of pitch. The angle γ of the rotation of the aircraft with respect to its longitudinal axis is called the angle of bank. The angle k between the plane of symmetry of the aircraft and the meridian (the line of intersection between the plane of the horizon and the plane of the geographical meridian) is called the true course of the aircraft. The angle k is measured from the air

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meridian.

If, as the initial direction, the magnetic rather than the geographic meridian is used, the course is known as the magnetic course instead of the true course.

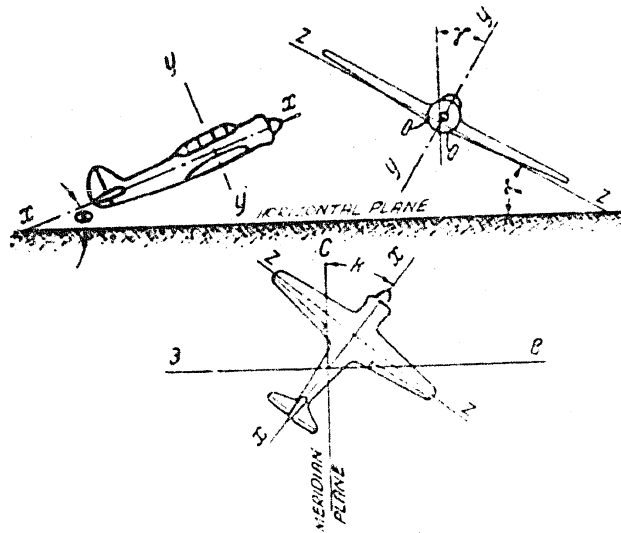


Fig. 2. Angles Characterizing the Position of the Aircraft with Respect to the Ground

θ , Angle of pitch; γ , Angle of bank; k , Aircraft course.

The longitudinal equilibrium of the aircraft depends on the angle of pitch, since this angle varies with any variation in the angle of attack of the aircraft (Fig.3) which, in turn, produces changes in the aerodynamic characteristics of the aircraft.

The angle between the direction of the projection of the velocity vector onto the plane of symmetry of the aircraft and the wing chord is called the angle of



attack. As the wing chord we take either a line tangent to the lower surface of the wing profile (tangent chord), or a line connecting the nose and tail of the profile (internal chord). The wing profiles and their chords that are most common at the present time are shown in Fig.4.

As already indicated, the angle of attack α affects the aerodynamic characteristics of the aircraft, i.e., the lift Y and the drag Q (Fig.5)

$$Y = \frac{c_y S_p v^2}{2}, \quad (1.1)$$

$$Q = \frac{c_x S_p v^2}{2}, \quad (1.2)$$



where S is the lifting surface of a wing in m^2 ;

ρ is the density of the air in $kg \text{ sec}^2/m^3$;

V the aircraft speed in m/sec ;

c_y the coefficient of lift;

c_x the coefficient of drag.

Fig. 3. Angle of Attack of Aircraft

α - angle of attack;

V - Velocity vector of the air

When the angle of attack increases, the coefficient of lift c_y increases and reaches its maximum value at a certain value of the angle of attack which is called the critical angle (Fig.6). Further increase of the angle of attack leads to a sharp decrease in c_y and the lifting force. One of the conditions of equilibrium of an aircraft in horizontal, rectilinear, and uniform flight is the equality of the weight of the aircraft G and its lifting force Y :

$$G = Y = \frac{c_y S_p v^2}{2},$$

whence

$$v = \sqrt{\frac{2G}{S_p c_y}} \quad (1.3)$$

Thus, the greater the value of c_y the smaller the speed necessary to maintain



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horizontal flight. Consequently, depending on the value of the speed V , an aircraft will perform a rectilinear horizontal flight at different angles of attack.

The angle of attack, to a considerable extent, determines the speed necessary

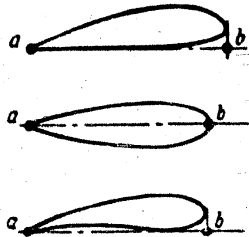


Fig. 4. Typical Wing Profiles.
ab- Wing chords

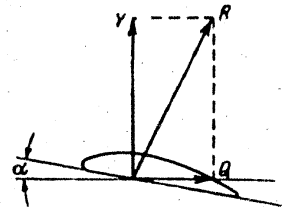


Fig. 5. Aerodynamic Forces Acting on the Wing

R- Total aerodynamic force; Y- Lift-
ing force; Q- Drag.

for maintenance of horizontal flight. Obviously, this is true also for any other regime of flight. To each regime of flight there corresponds a definite minimum

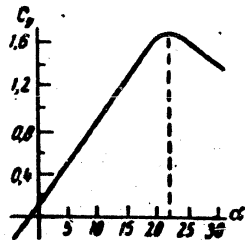


Fig. 6. Relation Between the Angle of Attack and the Coefficient of Lift c_y .

value of the flying speed V at which the aircraft is still able to maintain equilibrium and maintain the assigned regime.

Thus, it is necessary to know the aircraft speed V not only to calculate the time of flight, but also to maintain the longitudinal equilibrium of the aircraft, which depends to a considerable extent on the angle of attack.

Instruments Indicating the Regime of Flight

The aircraft speed is judged from the readings of the speed indicator. This instrument indicates the so-called air speed



i.e., the speed of the aircraft with respect to the air. As will be pointed out later, this same instrument indirectly makes it possible to judge the longitudinal equilibrium of the aircraft and, consequently, the value of the angle of attack. The flight altitude is indicated by the altimeter and the rate of ascent or descent by the climb indicator.

The course of the aircraft is controlled by means of compasses and the turn indicator.

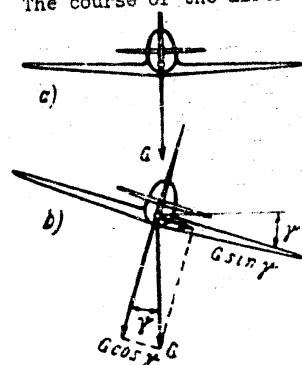


Fig.7 - Rectilinear Flight

- a - Without bank; b - With bank;
- c - Angle of absolute bank.

The latter indicates the presence of deviations of the aircraft from rectilinear flight, i.e., indicates changes in course. The lateral equilibrium of the aircraft depends on the angle of bank. In rectilinear flight, banking causes sideslipping (Fig.7).

In curvilinear flight (Fig.8), for example in turning, inertia forces* may cause wing-over and slipping. If when the aircraft executes a turn, the resultant of the gravity and centrifugal forces, directed

along the apparent vertical, coincides with a straight line perpendicular to the plane of the wings (cf. Fig.7 b), then a correct turn is being made. However, if the apparent vertical deviates from this straight line, there will be an outside or inside sideslip.

The lateral equilibrium of the airplane is checked by means of the bank indicator.

The direction of flight can be determined either from visible landmarks or by means of various compasses, or from airborne radio instruments. During the flight,

* In turning, the aircraft is subject to centripetal accelerations. In this case the centrifugal forces of inertia will act on the aircraft.

the crew of the aircraft is constantly using instruments and estimating the position of the aircraft from their readings.

The quieting aircraft instruments are particularly important in flight without visible external landmarks, when the human organism is subject to the action of inertia forces, and the pilot easily loses track of the actual position of the aircraft in space. In this case the imperfection of the human organs of equilibrium is revealed. If, for example, the aircraft is flying without visible external landmarks at a velocity V and makes an irregular left turn with a radius r and an angular velocity of turn ω , then the centrifugal force F that forces the pilot toward the right side of the ship is determined by the equation $F = mV^2/r = mV\omega$ where m is the mass of the pilot.

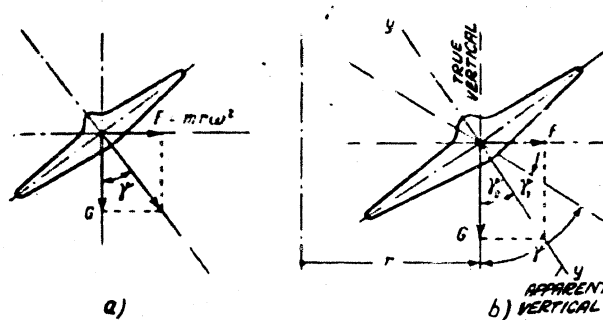


Fig.8 - Aircraft Turn

a - regular left turn; b - left turn with sideslip; r - radius of turn; ω - angular velocity of turn; γ - angle of absolute bank; γ_0 - angle between apparent and true vertical; $\gamma_1 = \gamma = \gamma_0$ - angle of relative bank.

Since the pilot does not feel the turn and believes that the aircraft is in rectilinear flight he reaches the conclusion that the aircraft is banking to the right, and that it is necessary to straighten it out; in this case he will straight-

en it out not with respect to the true vertical, but to the apparent vertical (Fig.8b). As a result the aircraft banks to the left and goes into a regular turn (Fig.9). The power necessary for the turn exceeds the power necessary for rectilinear flight, and therefore the aircraft begins to lose speed. If the aircraft has a turn-and-bank indicator on board, the pilot will immediately discover the change from rectilinear flight to a regular or irregular turn and will know how to restore the aircraft to the required attitude with respect to the ground.

The compass, altimeter, climb indicator, speed indicator, bank indicator, and turn indicator allow the pilot to judge by indirect methods the position of the aircraft with respect to the ground. The bank is determined from the bank-and-turn indicator while the pitch is determined from the speed indicator and the climb indicator.

If, at constant aircraft speed, the vertical speed changes, then the angle of attack and the angle of pitch of the airplane also change.

But the indirect method of determining the position of the aircraft considerably complicates piloting and places an excessive nervous strain on the pilot. In addition, the speed indicator and variometer as indicators of the longitudinal position of the aircraft give readings with considerable lag, while the readings of the magnetic compass are unstable under bumpy flight conditions. For this reason, instruments free from the above faults were designed which make it possible directly and accurately to determine the angles of longitudinal and lateral deviation and the course of the aircraft.

Such instruments are the gyroscopic instruments installed at the present time on all aircraft. They include the gyro-horizon, the directional gyro, the gyromagnetic compass, etc. The work of the pilot on an aircraft equipped with these instruments, in flight without visible landmarks is reduced to the observation of the instruments and control of the aircraft rudders. This is purely mechanical, monotonous, but very fatiguing work.

On an aircraft equipped with an automatic pilot, the pilot is considerably less burdened and his work becomes less fatiguing. But the simplest autopilot can be used only in rectilinear uniform horizontal flight; the more complex, so-called program

autopilots, which control take-off, landing, or maneuvers of the aircraft, are installed only on special aircraft.

The maneuver aircraft connected with a change of its speed in magnitude and direction is called evolution. They include: turning, diving, stunt flying, etc. In evolutions, the aircraft moves with accelerations varying in magnitude and direction, and consequently the forces of inertia act both on it and on the airborne instruments. The concept of overload is ordinarily used to characterize these forces.

Overload (indicated by the symbol n) is the ratio of the resultant acceleration a with which an aircraft is moving to the acceleration of gravity g .

$$n = \frac{a}{g} \quad (1.4)$$

High overloads can have a disastrous effect on the organism of the crew, on the aircraft, and on the instruments. The degree of overload is checked by the accelerometer.

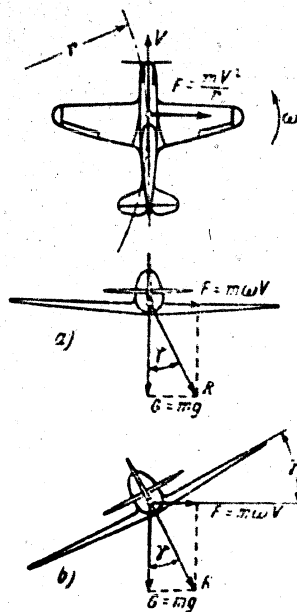
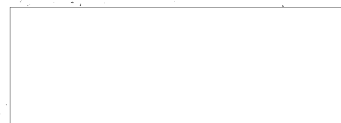


Fig.9 - Aircraft Turns

a - Without bank; b and c -
With bank by an angle



Section 3.

AIRCRAFT NAVIGATION

The work of the crew in flying the assigned route and reaching its objective is called aircraft navigation. On light aircraft designed for short trips this work is performed by the pilot. On medium and heavy aircraft which fly over considerable distances, a single person cannot do all the work; for this reason the pilot does the piloting of the aircraft while the navigator handles its orientation. In many cases the crew of an aircraft numbers 10 and more and includes several pilots, navigators, a board engineer, a radio operator, etc.

Methods of Orienting an Aircraft

For orienting an aircraft in the air the following methods are used:

a. The method of visual orientation, i.e., comparison of visible landmarks (railroads, bridges, etc.) with the map on board the aircraft. Such orientation is possible only when ground visibility is good.

b. Astronomical orientation reduced to the calculation of the position from angles measured between the directions to selected heavenly bodies and the plane of the horizon, allowing for the time of observation. Such orientation, which can be effected only if the visibility of the heavenly bodies is good, is performed by means of special optical instruments, for example, a sextant, optical sight, etc.

c. Radio orientation, reduced to the determination of the position of the aircraft from the directions to ground radio stations or the distances to them, as measured in flight. The method requires the existence of one or more ground radio stations whose position is known and is carried out with the aid of special radio instruments: radio compass, radio direction finder, etc.

Various methods of determining the position of the aircraft by means of radio methods are known. One of these will be discussed below.

By using the readings of the radio compass (or other radio instruments) it is

easy to find the angle α of the ground transmitter between the direction of flight to the radio station and the longitudinal axis of the aircraft. Knowing the course of the aircraft k (for example, from the readings of the magnetic compass), the true radio bearing of the aircraft $\varphi = k + \alpha$ can be found. By connecting all points on the earth's surface at which the true radio bearing has a certain value, we obtain a line of possible aircraft positions (position line), but still not its actual position.

By determining the value of the true radio bearing for two ground transmitters whose position is known, we obtain two position lines. The point of intersection of these position lines on the map will correspond to the position of the aircraft.

With such a method of orientation no radio transmitter with a directed emission is required.

Many methods of radio orientation require a directive transmission. The process of introducing radio engineering systems into the group of instrument equipment

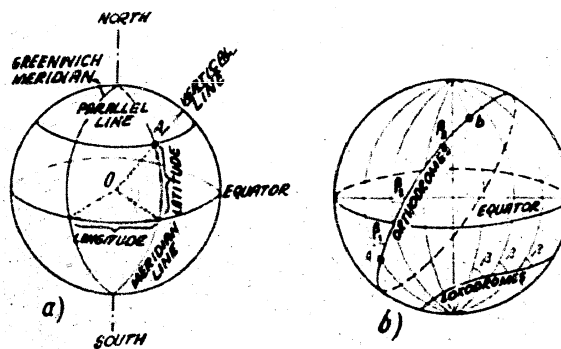
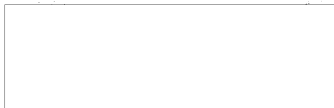


Fig.10 - Determination of the Position of an Aircraft

a - Geographical coordinates; φ , latitudes; λ , longitude; b, orthodromes and loxodromes

of aircraft has still not been completed. There is no doubt that in the future



these systems will acquire even greater importance. But it must be remembered that the introduction of radio navigation instruments does not exclude the use of instruments operating on other principles (mechanical altimeters, gyrocompasses, astronomical compasses, etc.)

This is explained in the first place by the danger of failure of the aircraft radio equipment due to interference by ground radio stations, and, in the second place, by the danger of being detected.

The operation of airborne radio equipment may reveal the aircraft and allow its approach to be detected sometimes at very great distances.

d. The method of calculating the course, in which the position of the aircraft is determined by calculating the value and direction of the segments of the course already traveled by the aircraft from the take-off point. Under the conditions of actual flight the aircraft flies along a certain curve connecting the initial and final points of the route, i.e., both the longitude and the latitude of the place vary during the time of flight (Fig.10a). In order to have the shortest possible route, the aircraft must fly along the arc of the great circle between the initial and final points of the flight course, called the "orthodrome", (Fig.10b). During such a flight, the course of the aircraft varies continuously, since the orthodrome intersects the meridians at different angles. This change of course is inconvenient in operation, but on long-distance flights, such maneuvers result in a marked shortening of the route and thus in a saving of fuel and time.

An orthodrome is divided into a series of segments within which the curvilinear segments intersecting the meridians at different angles $\beta_1, \beta_2, \beta_3$, etc. are replaced by segments each of which intersects the meridians it meets at one and the same angle β .

The curve so obtained is called a loxodrome. The flight route is plotted in advance; the proposed flight path is divided into separate sections in such a way that, within the limits of each section, the direction of flight remains the same.

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If, during the flight, the speed and time corresponding to each section are measured, then the true distances traveled by the aircraft can be found and these distances plotted on the corresponding scale on the map; in this way, the coordinates of the aircraft at a given moment can be determined. The method of route calculation is inconvenient in that it requires a very accurate determination of the course, speed, altitude, and time of flight.

Air-Navigation Control Instruments

It is commonly known that to measure the course of an aircraft it is necessary to know some fixed direction with respect to the ground, from which the calculation is made. This direction may be the geographic meridian, the magnetic meridian, etc. The course reckoned from the magnetic meridian is called the magnetic course; if reckoned from the geographic meridian, it is called the true course, while the angle between the geographic meridian and the direction of flight (course), measured clockwise from the northern direction of the meridian is called the true course angle (like the course, this angle may be either true or magnetic). Usually the course and the longitudinal axis of the aircraft do not coincide, since the speed of the aircraft with respect to the ground (the ground speed) is the geometric sum of two speeds, the speed of the aircraft with respect to the air (the airspeed) and the speed of the air with respect to the ground (the wind speed). The true course angle does not coincide with the true course of the aircraft and is equal to the sum of the true course of the aircraft and the angle of drift ω , i.e., the angle between the longitudinal axis of the aircraft and the course, the angle of drift due to the wind ψ_w and the aerodynamic drift ψ_a (which arises, for example, as a result of the unequal thrust of the propellers in multi-engine aircraft). These existing methods of determining the angle of drift give the total angle of drift $\psi = \psi_w + \psi_a$, and for this reason practical aerodynamic drift is not separately considered. The angle of drift is measured by the aid of special instruments, for example, navigation sights, etc. In determining it it is necessary to know

exactly the altitude and duration of flight. The angle of drift is considered positive if the aircraft deviates toward the right.

The course of the aircraft is determined by the aid of compasses.

It is more difficult to determine the speed of the aircraft than its course. The dead reckoning method requires the knowledge of the aircraft speed with respect

to the ground, i.e., the ground speed.

Until now, no instruments indicating the ground speed of the aircraft have been designed.

Manometric speed indicators, widely used in actual operation, indicate the airspeed of the aircraft or in the best case, the true air speed. A true air speed indicator allows for the so-called methodological error of the instrument, which is manifested as a result of the variation in density of the air with height, while the air speed indicator does not allow for this error.

The lack of an instrument indicating the ground speed makes it necessary to determine the value of this speed by navigation instruments.

The ground speed may be determined with a navigation sight, observing the rate of displacement of landmarks on the ground. In this case, the altitude must be known.

The magnitude and direction of the ground speed may also be found by constructing a navigational velocity triangle (Fig.11).

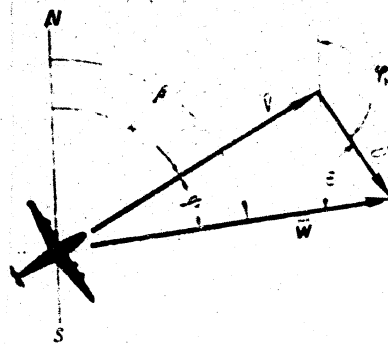


Fig.11 - The Navigational Velocity Triangle

V - Air speed; U - Wind speed; W - Ground speed; k - Course of aircraft; ψ - Direction of wind; β - Drift of aircraft; β - Course angle



The Navigational Velocity Triangle

One side of the navigational velocity triangle is the vector of air speed of the aircraft V , another side is the vector of wind speed U , while the third side representing the sum of these two vectors gives the vector of ground speed of the aircraft $W = V + U$. The magnitude of the air speed of the aircraft is taken from the readings of the true airspeed indicator or of the air speed indicator corrected with an air-navigation slide rule, making it possible to allow for the methodological error of the instrument. The wind-speed vector may be determined by the navigation sight using any one of several methods, for example, measurement of the wind from two angles of drift. It must be remembered that, in determining the angles of drift, the pilot is obliged to hold the assigned course and keep altitudes and speed constant.

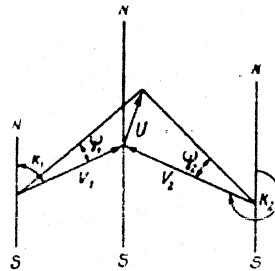


Fig.12 - Determination of the Wind

Vector U

k_1 and k_2 aircraft courses; ψ_1, ψ_2 angles of drift on courses k_1 and k_2 , respectively

The magnitude and direction of the wind are determined in the following way: Selecting some landmark on the ground and observing it for a definite time interval, the value of the angle of drift is determined, i.e., the direction of the course is found.

The same is done by changing the course of the aircraft by $40-50^\circ$, thus obtaining a new flight line of the aircraft. On plotting the flight line on the map the wind-speed vector is obtained (Fig.12).

In determining the wind vector, the navigator of the aircraft does not make the above-mentioned constructions on the map but uses a wind triangle instead, which allows him very rapidly and accurately to

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perform these constructions.

As already mentioned, the wind triangle was proposed by A.N. Zhuravchenko at the beginning of this Century and up to now has been widely used in aviation to obtain rapid and accurate solution of problems by the aid of the navigational triangle of speeds, thus making it possible to determine the speed and direction of the wind, the course speed, the angle of drift, the course angle, etc. i.e., to find the unknown elements of the velocity triangle.

When the ground is invisible, and when the use of optical sights is impossible, the course angle is determined by solving the navigational triangle. The course angle, the angle of drift, etc. are determined by calculation if the direction and speed of the wind are unknown (for example, from data obtained from the ground stations).

The Calculated Position of the Aircraft

The aircraft crew at every moment of time must know the position of the aircraft. This position is determined in terms of geographical longitude and latitude. The longitude is reckoned from the Greenwich meridian, to the west of it the western longitude λ_w , and to the east of it the eastern longitude λ_e . The latitude is reckoned from the equator (north and south latitude φ_n, φ_s). During flight along the meridian only the latitude changes by the value

$$\Delta \varphi = \Delta S_M \frac{360}{2\pi R}$$

where ΔS_M is the path traveled by the aircraft along the meridian; and

R is the radius of the earth.

During flight along a parallel line, only the longitude changes by the value

$$\Delta \lambda = \delta S_p \frac{360}{2\pi r} = \Delta S_p \frac{360}{2\pi R \cos \varphi}$$

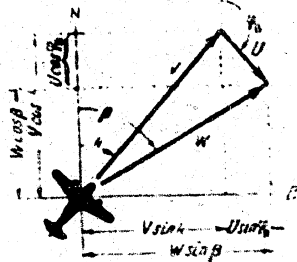
where ΔS_p is the path traveled by the aircraft along the parallel;

r is the radius of a circle of the given parallel $r = R \cos \varphi$

R is the radius of the earth;
 φ is the latitude of the given parallel.

In flight along an arbitrary trajectory the coordinates of the aircraft are continually changing.

Let us denote the latitude and longitude of the take-off point by φ_0 and λ_0 ;



current values of these coordinates by φ and λ ; the true course of the aircraft by k ; the direction of the wind by φ_w ; the wind speed by U ; the time of flight by t ; the component of the course velocity along the meridian by W_m ; and the component of the course speed along the parallel by W_p (Fig.13).

Then we may write

$$W_m = V \cos k + U \cos \varphi_w$$

$$W_p = V \sin k + U \sin \varphi_w$$

Fig.13 - Determination of the Calculated Position of the Aircraft

V - Airspeed; W - Ground speed;
 U - Wind speed; k - Course of aircraft; φ_w - direction of wind

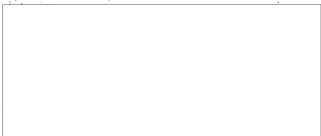
For an infinitesimal interval of time dt , the aircraft is displaced along the meridian or the parallel by the respective values dS_m and dS_p , which may be found from the equations

$$dS_m = W_m dt$$

$$dS_p = W_p dt$$

In this case the coordinates of the aircraft vary in the following manner:

$$d\varphi = \frac{360}{2\pi R} dS_m = \frac{360}{2\pi R} (V \cos k + U \cos \varphi_w) dt,$$



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$$d\lambda = \frac{360}{2\pi R \cos \varphi} dS_n = \frac{360}{2\pi R \cos \varphi} (V \sin k + U \sin \varphi_n) dt.$$

After integration of these equations with respect to time we find

$$\varphi = \frac{360}{2\pi R} \int_0^t (V \cos k + U \cos \varphi_n) dt. \quad (1.5)$$

$$\lambda = \frac{360}{2\pi R} \int_0^t \frac{V \sin k + U \sin \varphi_n}{\cos \varphi} dt. \quad (1.6)$$

During the time of flight, the navigator periodically determines the coordinates of the aircraft. This task is considerably simplified if there is a navigational coordinator on board the aircraft. This is an instrument which automatically solves the above equations and gives the values of λ and φ or an automatic navigator which solves the problem more fully and plots the entire course of the aircraft on the map.

Navigational and Piloting Instruments

For the purposes of piloting and aircraft running, the following instruments are used:

1. Speed indicators;
2. Compasses;
3. Altimeters;
4. Rate of climb indicators;
5. Gyro horizons;
6. Bank indicators;
7. Turn indicators;
8. Navigation coordinators;
9. Mach-number indicator;

10. Accelerometers.

In addition, in the majority of cases an automatic pilot is installed on the aircraft.

To assure reliability, many instruments are paired, for example, in addition to the magnetic compass a directional gyro, a remote reading compass or a remote indicating compass, a radio compass, etc, are installed on the aircraft simultaneously with the magnetic compass.

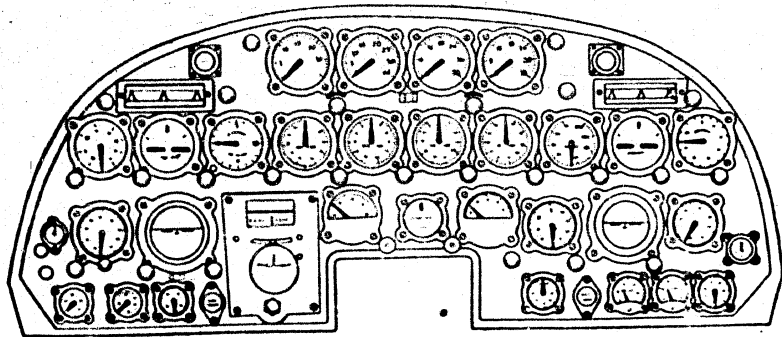


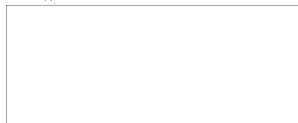
Fig.14 - Instrument Board of an Aircraft

The instrument board of an aircraft with instruments arranged so that the crew can conveniently observe their readings, is shown in Fig.14.

The central part of the instrument panel is occupied by the main piloting and navigation instruments: the speed indicator, turn indicator, and bank indicator, altimeter, rate of climb indicator, compasses, gyro horizon, and clock. Since the instrument boards are set up in the zone of direct observation of the pilot, any increase in their dimensions would unavoidably lead to impairing the view. To preserve free view, the instruments are compactly arranged, their outside dimensions are reduced, combination instruments are used, and finally the shape and design of the instrument boards is carefully selected.

Since the instrument panels are located at points of the aircraft where the effect of vibrations is very considerable, while the instruments on these panels are highly sensitive to vibrations, a shock absorption for instrument panels is provided.

In individual cases, in addition to the instrument panel, individual shock absorbing must be provided for some instruments, such as compasses.



CHAPTER II
BASIC REQUIREMENTS FOR AIRCRAFT EQUIPMENT

The development of modern aviation, the expansion and complication of the problems to be solved by aircraft crews in flight, require an increase in the number of aircraft equipment items installed on the aircraft. The number of objects in the aircraft cabin requiring the attention of the aircraft crew is in the tens or even in the hundreds. Some instruments require the performance of complex calculations, the use of special Tables, etc. (For example, radio equipment and means for astronomic aircraft navigation).

All aircraft equipment, including the instruments, is intended for the performance of various tasks under the specific conditions of flight, and must satisfy the so-called tactical-technical requirements resulting from these problems and conditions.

All tactical-technical requirements can be divided into functional, operational, physical-technical, assembly, etc.

Section 1.

TACTICAL-TECHNICAL REQUIREMENTS

Each instrument must satisfy its purpose, for example, measuring instruments must measure definite physical values with the necessary accuracy, drives must develop the necessary forces and moments, radio equipment must operate in a definite frequency range, etc.

Whether aircraft instruments answer their purpose is determined by the nominal values characteristic for their physical-technical quantities and indexes, i.e., its rated characteristics (nominal) and by the allowable deviations from these characteristics.

The rated characteristic of any technical installation is the quantity char-

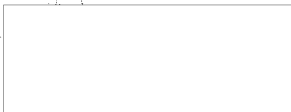
acterizing its basic technical parameters corresponding to its intended function. For example, the rated characteristic of a radio instrument includes the range of frequencies of the electromagnetic oscillations that can be used, the rated characteristic of electrical instruments includes the value of the voltage, current, etc. The selection of the rated characteristics must be done with great caution and care, and the rated characteristics must satisfy the actual conditions of operation. Otherwise the operation of a given instrument and of the aircraft units connected with it will give great difficulty, the quality of performance of the mission will be lowered, and there may even be danger to the aircraft and crew. At the same time, a selection of rated characteristics beyond the actual functional requirements may lead to complication of design, and to an increase in weight, dimensions, cost, etc.

The allowable deviations from the nominal are selected carefully, taking account of the service conditions. For example, it is often required of an instrument that it exhibit minimum deviation from the nominal over only a portion of the scale instead of over the entire scale. In this case, the remaining range of the scale may show somewhat higher reading errors.

Section 2.

PHYSICAL-TECHNICAL REQUIREMENTS

The physical-technical requirements to be met by aircraft instruments cover normal operation of the instruments under actual conditions of flight. In order to properly design and manufacture an instrument and to be able to compare various designs of instrument and various methods of measurements and, of course, to select the best ones, and finally to operate the instruments properly, it is necessary to study the conditions under which the instruments operate, and also the regularities to which these conditions are subject. It is above all necessary to familiarize oneself with the properties of the atmosphere.



The Atmosphere and its Properties

The atmosphere surrounding the earth is divided into three layers: the troposphere, stratosphere, and the zone of rarefied gases. Since at the present time flights are made mainly within the limits of the troposphere, it is particularly important to know its characteristics.

The lower layer of the atmosphere immediately adjacent to the earth's surface, is called the troposphere. The troposphere is characterized by extensive horizontal and vertical air currents. The parameters of the troposphere, i.e., the temperature, pressure, density, viscosity, etc. vary with height. The layer above which the temperature remains constant is assumed to be the upper boundary of the atmosphere. Since the atmospheric conditions vary substantially according to the season, weather, geographic location, etc., while the instruments and aircraft must function in different seasons and at various geographic points, it has been agreed that in calculations, designs, and tests of airplanes and their equipment, a certain fictitious and conventional atmosphere with certain definite parameters, will be used. This atmosphere is known as the standard atmosphere (SA). The standard atmosphere gives an arbitrary law of variation in pressure p , temperature T , density ρ , γ , etc. with the height, and also definite initial values of these quantities, p_0 , T_0 , ρ_0 , γ_0 , etc. corresponding to zero height.

The altitude with a corresponding initial temperature $T_0 = 15^\circ + 273^\circ = 288^\circ\text{C}$, initial pressure $P_0 = 760 \text{ mm Hg}$, mass density $\rho_0 = 0.125 \text{ kg-sec}^2/\text{m}^4$ and gravimetric density $\gamma_0 = 1.225 \text{ kg/m}^3$ is accepted as the level of the earth's surface, i.e., zero altitude H_0 . For the upper limit of the troposphere a height equal to 11,000 m has been assumed.

Within the boundaries of the troposphere, the temperature varies according to the law

$$T_H = T_0 - \eta H \quad (\text{II.1})$$

where T_H is the absolute temperature at the height H ;

τ is the temperature gradient, i.e., the change in temperature per unit height;

$\tau = 0.0065$ degrees/meter.

It is assumed in the calculations that the temperature may vary within the range from -60 to $+50^\circ\text{C}$.

Within the limits of the troposphere the pressure varies according to the law

$$P_H = P_0 \left[1 - \frac{\tau H}{T_0} \right]^{\frac{1}{R\tau}} \quad (\text{II.2})$$

where P_H is the pressure at the height H ;

R is the gas constant for air, equal to $29.27 \text{ m}^2/\text{C}$.

By solving equation (II.2) for H , we find

$$H = \frac{T_0}{\tau} \left[1 - \left(\frac{P_H}{P_0} \right)^{R\tau} \right] \quad (\text{II.3})$$

Equation (II.2) is termed the standard barometric formula, while eq.(II.3) represents the standard hypsometric formula.

In the theory of aircraft instruments the so-called Laplace formula is sometimes used. This has the form

$$H = 18,400 \left[1 + \frac{t_{\text{mean}}}{273} \right] \log \frac{P_0}{P_H} \quad (\text{II.4})$$

In contrast to the standard formulas, it is assumed in the derivation of the Laplace formula that the temperature does not vary with height and is equal to some mean value $t_{\text{mean}} = \frac{t_0 + t_H}{2}$ where t_0 and t_H are the temperatures at zero height and at height H , respectively.

The values for the density of the height H are calculated by the formulas



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$$\left. \begin{aligned} p_H &= p_o \left(1 - \frac{\tau_H}{T_o} \right)^{\frac{1}{R\tau} - 1} \\ \gamma_H &= \gamma_o \left(1 - \frac{\tau_H}{T_o} \right)^{\frac{1}{R\tau} - 1} \end{aligned} \right\} \quad (\text{II.5})$$

For heights exceeding 11,000 m, the temperature $T_H = 216.5^\circ\text{C}$, and the pressure varies by the law:

$$p_H = p_{11,000} e^{-\frac{H - 11,000}{RT_{11,000}}} \quad (\text{II.6})$$

(this formula has been obtained under the assumption of constant temperature in the stratosphere).

It is assumed of the SA that the air is dry, meaning that its humidity is not taken into account. In reality, the air contains a large quantity of moisture, which causes the corrosion of the instrument parts, leads to leakage of electric current in electrical instruments, creates water cushions in the bends of pipelines, considerably distorting the instrument readings, and, finally, at low temperatures may lead to icing, which is fatal not only for the instrument but for the entire aircraft.

The variation in T , p , γ , and other parameters of the atmosphere, the variation in humidity, etc., lead to errors in the instrument readings.

The function of the designers of an instrument includes the careful study of the probable variations in the parameters of the atmosphere under service conditions and prevention of possible errors in the instrument readings due to the influence of these varying conditions, or, at least, the minimizing of these errors.

Section 3.

OPERATIONAL REQUIREMENTS

The basic operational requirements to be met by aircraft equipment are simplic-



ity and convenient control and use during flight.

This is accomplished as follows:

1. By rational location of the instruments on the aircraft and on the instrument panel. The instruments are grouped according to their purpose and are placed in accordance with the requirements of a definite sequence of visual observations and manual operations. In placing the instruments, cross vision and, more important, reversed vision and motions must be avoided.

2. Protective blocking, signalization, and automation. Even the most expedient location of the equipment and control levers and the use of automatic blocking and signaling does not relieve the crew members of the need to follow the readings of certain instruments. A reduction in the strain on the crew is achieved by automation of the equipment.

3. Clearness and sharpness of the readings.

4. Convenience of approach, allowing periodic inspection, replacement of damaged instruments, etc.

In calculating and operating instruments, it is necessary to remember the influence of vibration, accelerations, and the like.

Vibrations and Overloads

Vibrations, inclinations, and overloads in the presence of unbalanced parts and units may lead to extensive errors in the instruments. For this reason, individual parts and units as well as the instrument as a whole are balanced. In addition, special care is taken to avoid the possibility of coincidence of the natural vibration frequency of an instrument with the frequencies of forced vibrations that arise on the aircraft.

All aircraft instruments are tested for resistance to vibration, i.e., for absence of distortions in the instrument readings under the influence of vibration, and for stability against vibration, by which we mean the trouble-free operation of the instruments during a certain definite period under an assigned vibration.

The approximate values of the vibrational overload for various units of an aircraft are given in Table 1. The instruments installed in the corresponding places of an aircraft are calculated on the basis of this Table.

Table 1

Maximum Vibrational Overloads of Various Aircraft Units

Name of Unit	Vibrational Overload
Aircraft engine	6-10
Engine frame	4-6
Fuselage	2.5-4
Non-shockproof instrument panel	1.5-2.5
Shockproof instrument panel	0.6-1.5

Dust

For protection against dust which is raised in large quantities during take-off or landing, careful hermetic sealing is provided for the instruments*.

In addition, wherever necessary, special filters are installed (for example, in the bearings of gyroscopic instruments).

Section 4.

ASSEMBLY REQUIREMENTS

The design of an aircraft and its equipment is intended to meet certain flight missions. These must include the realization of the closest possible design connection, which is characterized in particular by the assembly requirements, namely: the outside dimension and weight requirements, as well as matching the objects and systems of aircraft equipment with each other and with the design parts of the aircraft

* Hermetic closure has a considerably broader purpose than just protection from dust, for example, it isolates the internal cavity of an instrument from moisture, isolates the instruments from the action of the atmosphere, etc.

as a whole (combination instruments, connection of the autopilot and the gyroscopic sight, etc.).

The basic assembly requirement for aircraft instruments is the requirement for minimum dimensions (still preserving convenience of use and reliability of operation). The total weight of aircraft equipment does not exceed 5-8% of the gross weight. The established methods of calculations and design of aircraft instruments, the properties of the materials used in these instruments, the requirement that these instruments be given minimum dimensions, inevitably leads to a reduction in the instrument weight. Nevertheless, the designers and technologists, in designing and manufacturing aircraft instruments, must also strive toward maximum reduction in weight, while still preserving reliability and convenience of operation.

In addition to the above requirements, the individual items of aircraft equipment must also have the following features:

Remote reading, i.e., the instruments must include a device with which their readings may be transmitted to an indicating system located at a place convenient for observation or plotting. For example, the readings of all instruments controlling the operation of the aircraft engine must be transmitted to the instrument panel of the aircraft; many elements of the navigational regime of flight (the course of the airplane, the air speed, etc.) are of interest not only to the pilot but also to other members of the crew, and for this reason the values of these quantities must be simultaneously indicated by instruments installed on the instrument panels of individual crew members for example, pilot and navigator), etc.

The remote connection is designed to transmit the instrument readings. Since each value of the quantity to be measured must correspond to a definite instrument reading, the distance transmission must assure the unambiguous correspondence of the values at input and output.

In the aircraft instrument building industry various systems of transmission are used, operating on direct and alternating currents. The most widely used are

the potentiometric remote reading and the induction remote reading with all their modifications (selsyns, magnesyns, etc).

Safety from fire, which is ensured by so designing the instruments that they cannot cause a fire under any circumstance (this requirement is particularly important for instruments controlling the fuel-feed system of the engine).

Many of the above requirements are, in most cases, common to all instruments (for example, elimination of the effect of vibrations, resistance to humidity, etc.), while others apply only to some types of instruments (for example, instruments placed in the cabin do not have to be streamlined, etc.).

For this reason the requirements to be met by aircraft instruments are customarily divided into general and special requirements. All aircraft instruments, without exception, must meet the general requirements, while the special requirements take account of the specific nature of a given instrument.

Section 5.

GENERAL REQUIREMENTS FOR AIRCRAFT INSTRUMENTS

The general requirements are as follows:

1. Dependability and accuracy in operation are the basic requirements, on which proper utilization of the flight characteristics of the aircraft and the safety of flight depend to a considerable extent;
2. Convenience of operation, i.e., simplicity of observation, ease of handling, simplicity of repair, installation, etc;
3. Minimum outside dimensions, weight, and cost of the instruments, in accordance with the Standards;
4. Trouble-free operation of the instruments:
 - a) in the temperature range from - 60 to + 50°C;
 - b) under vibrations with a frequency ranging from 20 to 80 cps, and with an amplitude at which the overload attains 1.5 G (this requirement refers to the instrument pointers);

c) under a humidity ranging from 0 to 100%.

Section 6.

ERRORS AND CORRECTIONS OF INSTRUMENTS

The errors and corrections are equal in absolute value, but have different signs.

We will call the difference between the true value of the measured quantity and the reading of the instrument the correction. In order to determine the true value of the measured quantity from the reading of an instrument, the correction must be algebraically added to the instrument reading.

No instrument readings are free from errors. According to the causes responsible, errors are divided into systematic and instrumental.

Systematic errors are errors caused by the method of measurement selected. These errors appear as a result of using indirect methods of measurement in the aircraft instrument construction in the majority of cases, i.e., a method that does not measure the quantity of immediate interest for the aircraft crew, but some other quantity functionally related to it. This is explained by the fact that we still have not learned to measure directly the majority of values subject to measurement. For example, if we desire to measure the altitude, we use a barometric altimeter or a radio altimeter. In the former instrument, instead of the altitude itself, the pressure at this altitude is measured; in the second instrument, instead of the altitude, the time necessary for the passage of a signal sent from the aircraft to the ground and back to the aircraft, where this reflected signal is picked up by a receiving device, is measured. Both the pressure at the altitude in question and the time necessary for the passage of the signal transmitted by the radio altimeter depend not only on the altitude itself, but also on the state of the atmosphere, the season, etc.

It is impossible in the majority of cases to allow for all the factors that

distort the value of the measured quantity; for this reason, the instrument readings will necessarily include the error dependent on the method of measurement on which the operation of the instrument is based. In working out the design of aircraft instruments the selection of the method of measurement must be approached with extreme care and great caution. It is necessary to consider the type of problems, the given instrument, the function of the instrument, and its accuracy of operation under various operating conditions, etc. Having selected a method of measurement, it is then necessary to detect all causes that could be a source of systematic errors and as far as possible eliminate these errors in designing the instrument.

Since systematic errors are due to the method of measurement, it follows that these errors cannot be eliminated by careful preparation and calibration of the instrument.

This, however, does not exclude the possibility of introducing into the design of the instrument special compensating devices to eliminate or reduce these errors (for example, in the air speed indicator and the true air speed indicator). Errors that depend on the quality of manufacture of the instrument, the material used for the parts and units, etc., are called instrument errors.

These errors may be eliminated by improving the quality of manufacture and operation of the instrument.

Ensuring Reliability and Accuracy of the Readings

As already pointed out, the errors of aircraft instruments may be reduced if, in designing, manufacture, and installation, the peculiar features of operation of the instruments in question are taken into account.

Reliable operation of aircraft instruments under conditions of variable temperatures and pressures is ensured by the following:

- a) By placing the instruments away from the sources of heat on the aircraft preferably in heated cabins, in which as constant a temperature as possible is maintained;

- b) by using individual heating, heat dissipation, or cooling;
- c) by using materials of low sensitivity to temperature changes;
- d) by using special devices to compensate the temperature errors and the errors due to changes in pressure. Such devices are called compensators.

The moisture-proof and corrosion-proof operation of aircraft instrument is ensured by the following measures:

- a) By giving the instruments a form making it impossible for water to enter the body of the instrument;
- b) by using materials of the lowest possible hygroscopicity;
- c) by coating the surface with water-impermeable and anticorrosion lacquers and paints;
- d) by using special heaters and moisture collectors.

All parts of aircraft instruments and the instrument as a whole must have a mechanical strength sufficient to ensure reliable operation under flight conditions, where aerodynamic, vibrational shock, and other forces inevitably appear and lead to a vibration and displacement of the pointers of the aircraft instruments, and to the generation of resonant vibrations in the elastic elements, fastenings, etc., and to an accelerated wear of bearings and shafts, to interference with control, loss of hermeticity, etc.

Mechanical stability of aircraft instruments is attained by placing them at points of minimum vibration, by using shock absorbers, by preventing resonance, and by preventing self-loosening of attachments.

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CHAPTER III

INSTRUMENTS CONTROLLING THE OPERATION OF THE AIRCRAFT ENGINE

Section 1.

GENERAL DATA

Electrification of Instruments Controlling the Operation of the
Aircraft Engine

The gasoline piston engine is an extremely popular type of power plant for the modern aircraft. For this reason, the present chapter will consider mainly the instruments controlling its operation, i.e., manometers, thermometers, fuel gages, tachometers, etc. As already stated, the nomenclature of control instruments for other power plants used on aircraft, differs only slightly from that now under consideration.

In recent years, the control instruments for power-plant operation have been largely electrified. This is explained by the fact that electrical instruments give greater dependability of operation and are characterized by compactness, convenience, and simplicity of installation, and are easily converted to distant-reading instruments. This latter fact, in connection with the increasing size of many aircraft, is becoming more and more important.

These instruments are fed by the board electric system of the aircraft. The source of electric energy on the aircraft includes the aircraft generators driven by the aircraft engine, and storage batteries. The board electric system ordinarily has a voltage of 27 v.

To maintain the voltage at a constant value at varying load in the circuit and at varying engine speed, and to give a possibility of simultaneous operation of the generator and storage batteries as well as to prevent overloading, a junction box is used. Its basic parts are: 1) a voltage regulator; 2) a maximum relay to prevent overloads; 3) a minimum relay to ensure parallel operation of the generator and the

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storage battery.

For installation of the entire electric equipment on an aircraft, special electric wiring is laid the total length of which may reach several tens of thousands of meters.

In case of need, for example, to protect the radio installation from interference, a wiring is selected which, besides the rubber and cotton braiding, comprises a braiding of copper wire. In installing electric instruments it is necessary to provide for the protection of the wire from mechanical damage, the action of condensed moisture, heat, oil, etc, the absence of interference with radio reception, i.e., the shielding of the instrument itself and its wiring, the observation of fire-safety rules, low weight, small outside dimensions, and low cost.

Electric Methods of Measuring Nonelectric Quantities

Process in electrification of the instrument equipment was stepped-up considerably in connection with the development of automatic aircraft equipment. Almost all instruments installed on an aircraft for control of engine operation measure nonelectric quantities by electric methods of measurement. In this case, the measured nonelectric quantity is either converted into the corresponding values of electric parameters of circuits fed by an external voltage source, or is directly transformed into an electric quantity.

Examples of a conversion of the former type are instruments based on the changes in electric resistance (inductance, capacitance, etc.) of electric circuits, depending on the change of the parameter being measured. The resistance may vary with the temperature or as a result of the displacement of a movable contact under the influence of a change in pressure corresponding to the deformation of an elastic element, etc.

Examples of a conversion of the latter type are instruments based on thermoelectric, piezoelectric, and other effects.

The Basic Elements of Electric Measuring Instruments for Control of
Aircraft Engine Operation

In the general case, an electric measuring instrument measuring a nonelectrical quantity picks up the quantity to be measured and transforms it into another quantity which is convenient for remote transmission to the indicating part of the instrument. For this reason, such an instrument may be conceived as consisting of the following parts:

a) A sensitive element representing a part of the instrument that picks up the quantity to be measured and transforms it into a displacement. The design of this part of the instrument is to a considerable extent, determined by the character and limits of variation in the quantity being measured.

The transformation of one quantity into another may take place not only in the sensitive element, but also in special transducers the purpose of which consists in the transformation of the level of the quantity to be measured or the type of energy. For example, if the quantity being measured is the temperature, and its variation is measured by the aid of an ohmic resistance, then the transformer or transducer will transform the variation of thermal energy into a variation of electric energy. As a result of the variation in temperature, the value of the resistance of an electric circuit is also varied; consequently, with a constant source of emf, the value of the current flowing through this circuit will also vary. It is precisely this variation in current that is recorded by the indicating instrument, which is calibrated in units of temperature.

In the indirect methods of measurement, instead of the temperature, the measuring instrument measures some quantity functionally related to the temperature, such as the variation in current. In this case, systematic errors are unavoidable. Indeed, the current may vary not only under the influence of a variation in temperature, but under the effect of other causes, e.g., fluctuations in the feed voltage.

b) A remote transmission, which transmits an impulse, proportional to the quan-

tity being measured, to the indicating part of the instrument.

c) An indicator which reproduces the quantity being measured in the required units. Most often DC electric measuring instruments, magnetoelectric galvanometers, logometers, etc. are used as indicators.

d) A source of energy serving mainly to amplify the impulse received from the object being indicated.

In many instruments the individual elements are combined and are installed in a housing. In some cases certain parts may be omitted altogether. For example, in instruments where the parameter being measured is directly transformed into an electric quantity, there is no need for an outside source of electric energy. We assume that the electric instruments controlling the operation of an aircraft engine consist of the following elements: a pickup (receiver), a remote transmission, and an indicator.

In this case we must not forget that all the above functions of an electric measuring instrument are preserved and that the individual elements are present in a concealed form. For example, the function of a transducer may be performed either by the pickup or by the remote transmission.

The great variety of pickups designed to measure temperature, pressure, fuel supply, etc. is striking.

The pickups of various instruments, even those designed for measuring one and the same quantities, differ substantially and must meet entirely different requirements (for example, pickups of resistance thermometer and pickups of thermoelectric thermometers).

At the same time, the indicators of various instruments designed for the measuring of various physical quantities (temperature, pressure, fuel supply, etc.) are characterized by great uniformity. Electric measuring instruments of the type of galvanometers or ratiometers, are used as indicators for instruments controlling the operation of the aircraft engine, i.e., electric measuring instruments, which

are very widely used in measurement technology, are here employed.

Such measuring instruments also are used as indicators of many piloting-navigational instruments, such as the potentiometric remote-reading compass, the radio compass, etc.

Measuring instruments have very much in common with respect to their design, operating requirements to be met, installation, etc.; for this reason, the measuring instruments used on aircraft will be discussed here separately, so that no later reference to them is required.

Section 2.

ELECTRIC MEASURING INSTRUMENTS FOR CONTROL OF POWER-PLANT OPERATION

Magnetolectric Measuring Instruments

The operation of these instruments is based on the generation of a torque by the interaction of a permanent magnet and a current flowing through wires (the so-called frame or coil) located in the field of this magnet.

In these instruments, either the frames or the magnets may be moving. The use of instruments with moving miniature magnets has become possible only after the creation of high-coercivity alloys, which provide a high torque at relatively small dimensions of the moving magnet. A transitional stage is represented by instruments with a fixed intra-frame magnet. Since instruments with magnets inside the frame, movable or fixed, are considerably more compact than instruments with horseshoe magnets, they are becoming more popular than instruments with horseshoe magnets. This applies both to galvanometers and to ratiometers. If the torque is dependent on the current in a conductor, and the opposing moment is created by a special device (a coil, a permanent magnet, or the like) then the instrument is called a galvanometer. If the torque is dependent on the ratio of current in conductors, and the opposing moment is created by an electric coil whose function can be taken over by the frames, the instrument is called a ratiometer. A ratiometer can not have less

than two frames.

Magnetolectric Galvanometers

The design of a galvanometer with a horseshoe magnet (1) and the moving frame (2) is shown in Fig. 15. The inner core (3) provides a practically uniform radial magnetic field with the inductance B in the gap between the magnet and the core. The interaction of the conductor under the current (the frame) and the magnetic field leads to generation of the torque M .

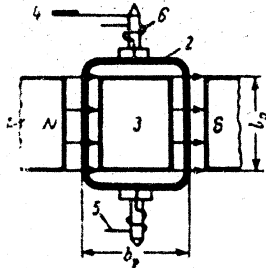


Fig.15 - The Moving System of a Galvanometer

1- Magnet; 2- Frame; 3- Core; 4- Balancing load; 5 and 6- Springs; l_p - Active length of frame; b_p : Width of frame.

The bearings represent the support on a pivot 0.3-2 mm in diameter; the pivot is made of special steel with jewel bearings in precision instruments and bronze bearings in less precise instruments (these are relatively seldom).

The form of the frame is determined by the purpose of the instrument. If the moment of inertia of the frame must be small, i.e., if the instrument is intended for recording rapidly varying phenomena, the frame is made in the form of narrow and long rectangles (in individual cases the frame may have the form of a single

The rotation of the frame is impeded by the springs (5) and (6) which provide a moment balancing the torque M , and serve at the same time as current supplies (so-called moving current taps). The balancing is effected by the counterpoise (4). To set the needle of the instrument to the zero division of the scale in the neutral position, a special corrector is used whose rotation changes the point of attachment of one of the hairs.

The bearings represent the support on a pivot 0.3-2 mm in diameter; the pivot is made of special steel with jewel bearings in precision instruments and bronze bearings in less precise instruments (these are relatively seldom).

conductor, in which case its natural frequency will be high and its moment of inertia low). The frame represents a coil, most often of copper (temperature coefficient $\alpha_t = 0.004$) or of aluminum ($\alpha_t = 0.00423$) wire, which is wound on the spider. The spider is made of metal in cases where high damping of the instrument is necessary, which is achieved by the action of eddy currents arising in the body of the spider. Sometimes the frame is made without a spider to lighten the moving system.

The magnets used are of the horseshoe type with special pole pieces, assuring the necessary law of distribution of magnetic induction in the gap and allowing the scale of the instrument to be expanded if necessary.

The magnets inside the frame of the instrument are given the form of cylinders (Fig.16). The outside dimensions of such a system may be as much as 15 - 30 mm. The value of the torque applied to the frame of the galvanometer may be found from

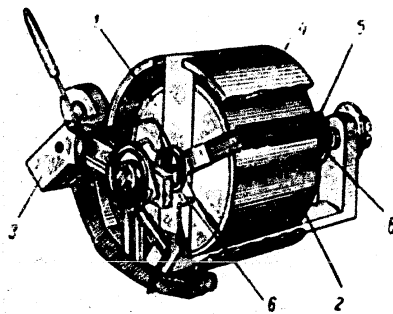


Fig.16 - Design of Galvanometer with Magnet Inside the Frame.

1.- Magnet; 2 - Iron ring; 3 - Armature;
4 - Outer magnetic circuit; 5- Movable frame; 6 - Current feed

frame. The same force acts on the other side on the frame, i.e., there exists a couple F_1 applied to the frame, the width of which is b_p .

The torque applied to the frame is equal to

the following considerations: The force of interaction F_1 between the current in the conductor and the magnetic field with inductance B_0 applied to one side of the frame, may be found from the equation

$$F_1 = 10^{-4} B_0 l_p i w, \quad (\text{III.1})$$

where l_p is the active length of the frame, i.e., the part of the frame covered by the radial magnetic field;

i is the current in the frame

w is the number of turns on the

$$M = 2F_1 \frac{b_p}{2} = F_1 b_p = 10^{-4} B_0 I_p b_p l w.$$

The quantity $S_p = l_p b_p$ is called the active area of the frame. By substituting S_p for $l_p b_p$ we obtain

$$M = 10^{-4} B_0 S_p I w. \quad (\text{III.2})$$

At an area Q_p of the cross section of the frame within the boundaries of this area the number of turns to be wound (w) depends on the size of these turns (q); the smaller the size, the greater the number of turns. Thus, to increase the torque M we must attempt to reduce q . The area of the cross section of the frame Q_p may be expressed by the equation

$$Q_p = \frac{w q}{k_3} \quad (\text{III.3})$$

where k_3 is the coefficient of filling of the frame by the conductor, allowing for the slack of the windings, the insulation, etc. and is equal to 0.7 - 0.8.

For known values of Q_p , k , and w , the value of q may be found from the expression

$$q = k_3 \frac{Q_p}{w}$$

and may be introduced in the expression for the resistance of a conductor

$$R = L \frac{\rho}{q}$$

where ρ is the specific resistance of the conductor material;

$L = pw$ is the length of the conductor;

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p is the perimeter of the frame.

Hence

$$R = \frac{p\omega^2}{k_s Q_p}; \quad \omega = \sqrt{\frac{k_s Q_p}{P}}$$

and

$$M = 10 \sqrt{B_o S_p} \sqrt{\frac{k_s Q_p}{P}} = 10 \sqrt{B_o S_p} \sqrt{P} \frac{k_s Q_p}{P} \quad (III.4)$$

where $\sqrt{P} = \sqrt{I^2 R}$ is the electric power supplied to the instrument.

If the power supplied to the instrument is small (a very frequent case), it is important to know the conditions of maximum sensitivity of the instrument, i.e., the conditions of obtaining the maximum value of the torque M for a given minimum power P_* ; in that case, it is precisely this value of the Power P_* that must be introduced in the expressions for ω , q , and M .

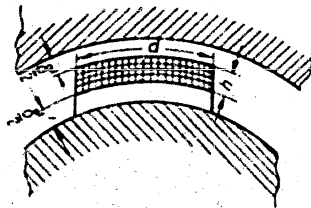


Fig. 17 - Galvanometer Frame in the Gap of the Magnetic System.

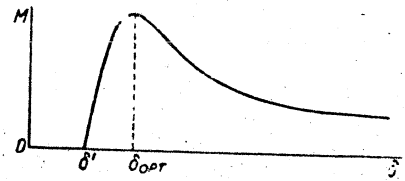


Fig. 18 - Relation of the Stationary Moment of a Galvanometer and the Value of the Air Gap.

$\delta/2$ - Gap between frame and magnet;
 δ - full air gap; M - Stationary moment.

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If, in spite of this, the torque is still too low, its further increase is achieved by selection of the optimum value for the gap in the magnetic system. If the thickness of the cross section of the frame is h and the length is d (Fig.17), then

$$Q_p = dh$$

We shall denote the gap between the frame and the iron by $\frac{\delta'}{2}$, so that the full air gap $\delta = \delta' + h$.

By replacing h by δ and δ' , we obtain

$$Q_p = d(\delta - \delta') \quad (\text{III.5})$$

The quantity δ' cannot be taken too small, since this may hinder the assembly of the instrument and lead to the frame being placed beyond the pole pieces.

The expression for the moment M may be rewritten in the form

$$M = 10^{-4} B_0 S_p \sqrt{P} \sqrt{\frac{k_z d}{p\rho}} \sqrt{\delta - \delta'}$$

With increasing value of δ the moment increases only to a certain limit after which it then begins to decrease.

The relation between the stationary moment and the value of the gap is given in Fig.18. At $\delta = \delta'$, the moment $M = 0$. At $\delta \gg \delta'$ the magnetic induction in the gap gradually decreases, which also leads to a gradual decrease of the stationary M .

The turning moment developed by the frame is opposed by the moment $M_1 = cd$ developed by the hairs, where α is the angle of rotation of the moving system and c is the coefficient of elasticity of the current conductor.

The position of the moving system is determined by the equation $M_2 = M - c\alpha$. At the instant of equilibrium, $M_2 = 0$, i.e.,

$$10^{-4} B_0 S_p \sqrt{P} \sqrt{\frac{k_z d}{p\rho}} = c\alpha \quad (\text{III.6})$$

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Galvanometer Errors and Preventive Methods

The galvanometer readings may be affected by the following:

a) Magnetic and electric fields created by other instruments; their action is neutralized by shielding.

b) Fluctuations in the feed voltage of the electric power system of the aircraft, of as much as $\pm 10\%$ which may lead to errors in the instrument readings of the same order. To eliminate these errors, the instrument is provided with a voltage stabilizer.

c) Friction in the bearings and unbalanced state of the individual unit and of instrument as a whole, which may lead to considerable errors.

d) Vibration. The influence of vibration on the position of the movable system is eliminated by the introduction of special dampers.

e) Fluctuations in the surrounding temperature leading to variation in the magnetic flux ϕ of the magnet, and consequently to variations in the value of the magnetic induction B_0 in the gap, to variation of the modulus of elasticity E , of the resistance of the frame R , etc. i.e., leading to instrumental temperature errors.

The variations of certain of the above quantities may be mutually compensated, entirely or in part (for example, variation of the magnetic inductance in the gap is partially compensated by variation of the modulus of elasticity); however, other variations remain uncompensated and require the introduction of special compensating devices.

The most widely used methods of temperature compensation for galvanometers are as follows:

1. The Method of additional resistance. The additional resistance R_a (Fig.19) is connected in series with the galvanometer; this resistance is made of wire with a very low temperature coefficient (nickel, Constantan, etc). Such a resistance may be considered temperature-independent. The current in the instrument, in this case, is equal to

$$I = \frac{U}{R_A + R} \quad (\text{III.7})$$

If $R_D \gg R$, then even considerable variations of R will not cause great temperature errors, but the sensitivity of the instrument is considerably reduced.

2. The Method of additional resistance with a negative temperature coefficient. The introduction of an in-series connected resistance with a negative temperature coefficient i.e., the introduction of a so-called neutralizer, is the most modern method of temperature compensation. A neutralizer produces almost no loss of sensitivity of the instrument, since the quantity R_D in this case may be considered insignificant. The compensation is effected by the selection of a law of variation of R_d in accordance with the variation of R . The loss of sensitivity will be smallest where the absolute value of the temperature coefficient β_t of the neutralizer exceeds the value of the temperature coefficient β_t of the material of which the re-

sistance of the galvanometer frame is made.

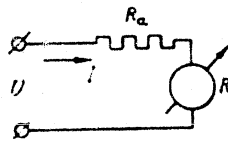
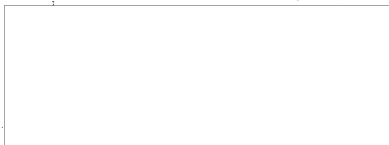


Fig.19 - Additional Resistance Method
 R_a - Additional resistance; R - Resistance of the instrument; I - Current flowing through the instrument; U - Feed voltage

Such resistances which are used more and more frequently, are made of celite ($\beta_t = 0.01$), tellurium with silver ($\beta_t = 0.012$), etc. Alloys exist in which the absolute value of the temperature coefficient of resistance is 10 to 12 times as great as the value of the temperature coefficient of the material of the frame, for example, silver sulfide. A study of this type of alloys was first made in the

USSR by Academician A.F.Ioffe.

3. Method of the thermosensitive shunt. Compensation of the error is effected by selecting the resistance r of the shunt (Fig.20), in such a way that on any vari-



ation in the temperature of the surrounding air, the current in the frame will remain constant.

The resistances R_1 and R_2 are made of Manganin or Constantan.

The resistance r is made of copper ($\alpha_t = 0.004$) or nickel ($\alpha_t = 0.0052$). With this method of compensation, the sensitivity of the instrument is likewise considerably decreased. The value of the current I flowing through the instrument, is determined by the relation

$$I = I_0 \frac{r}{R_2 + R_1 + r} \quad (III.8)$$

where I_0 is the current of the voltage source;

r is the shunt resistance;

R is the resistance of the instruments;

R_2 is the additional resistance.

By varying the resistances R_1 and R_2 , the instrument may be regulated within wide limits.

4. Method of the thermosensitive magnetic shunt. Figure 21 shows the thermomagnetic compensation of a galvanometer.

The magnetic flux of the magnet in this case consists of three components: the

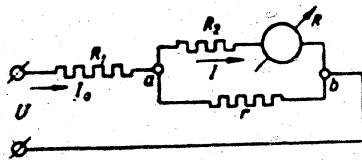


Fig. 20 - Scheme of Temperature Compensation by the Method of Thermosensitive Shunt

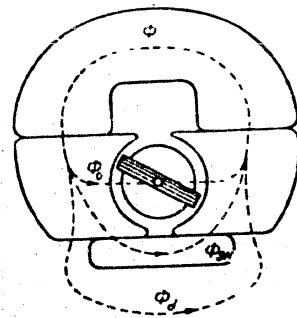


Fig. 21 - Thermomagnetic Compensation of Galvanometer



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working flux ϕ_0 , the flux in the shunt ϕ_{sh} and the flux of dispersion ϕ_d . The magnetic induction in the gap is B_0 . When the temperature is increased, the magnetic induction B_0 in the gap decreases, reducing the stationary moment cf. eq. (III.2) acting on the moving system.

A thermomagnetic shunt is made of material in which the resistance to the passage of magnetic lines of force is increased with increasing temperature (calmalloy, alloys of iron, nickel, chromium, etc). As the temperature increases, the magnetic flux from the shunt is forced into the working gaps of the magnetic system of the galvanometer, as a result of which temperature compensation is achieved.

The application of a thermomagnetic shunt involves a reduction in the magnetic flux in the gap, and consequently also a 20-30 % reduction in the sensitivity of the instrument.

Magnetolectric Ratiometers

Such ratiometers represent a permanent magnet in the field of which are placed two or more conductors (frames); a current is applied to the frames through special springs with very low elasticity. The value of the current applied to each of the frames is determined by the nature of the change in the measured parameter. The frames are connected in such a way that the direction of the current in them, and consequently the direction of the torques arising under the action of the magnetic field on each of the conductors, is different.

In ratiometers with two frames, the torques act in opposite directions to each other. There are no special devices in the instrument for creating a counter moment. In the galvanometer, the counter moment, the value of which depends on the angle of rotation of the moving system is created by spring-filled current feeders. In order to make the position of the moving system of the ratiometer stable, it is necessary that at least one of the torques acting on it be dependent not only on the quantity being measured but also on its position. Without satisfying this requirements, a stable position of the moving system of the instrument is impossible.

A stable position may be attained, for example by selection of a certain law of distribution of the magnetic inductance B_0 in the gap. In this case the frame becomes, as it were, an electrical spring.

The value of the moment M applied to the moving system of the instrument is determined by the ratio $M = M_1 - M_2$, where M_1 and M_2 are the moments acting on the first and second frames, respectively. These moments may be found from the equations

$$\left. \begin{aligned} M_1 &= 10^{-4} B_{01} S_{p1} w_1 I_1 = a_1 I_1 B_{01} \\ M_2 &= 10^{-4} B_{02} S_{p2} w_2 I_2 = a_2 I_2 B_{02} \end{aligned} \right\} \quad (\text{III.9})$$

where I_1 and I_2 are the currents in the first and second frames respectively;

B_{01} and B_{02} are the magnetic induction in the gap at the locus of the first and second frame, respectively; for symmetrical frames,

$$a_1 = a_2 = S_{p1} w_1 = S_{p2} w_2 = a$$

Consequently, in position of equilibrium, when $M_1 = M_2$, we have

$$I_1 B_{01} = I_2 B_{02}$$

i.e.,

$$\frac{I_1}{I_2} = \frac{B_{02}}{B_{01}} \quad (\text{III.10})$$

The values of the currents I_1 and I_2 vary in accordance with the variation of the quantity being measured. To assure stable equilibrium of the moving system, the induction in the gap when the system is displaced must vary by a law inversely proportional to the law of variation of the currents. Consequently, the frame with the highest current must rotate in such a way that the moment acting on it decreases with decreasing magnetic induction in the gap. In this case, the moment acting on the frame with the lower current must increase as a result of the increase in magnetic induction in the gap.

For horseshoe magnets, the required law of distribution of magnetic inductance in the gap may be obtained by experimental arrangement of the inner magnetic conductor (Fig.22) or by choice of the form of the pole pieces. In instruments with a

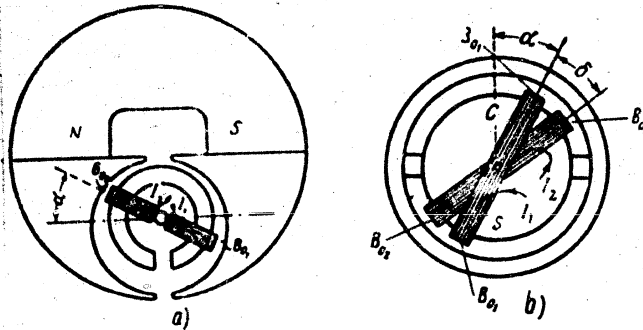


Fig.22 - Ratiometers with Two Frames

A- With a horseshoe magnet; b- With a magnet inside the frame; α - Angle of rotation of the moving system; δ - Angle between the frames; I_1 - Current flowing through the first frame; I_2 - Current flowing through the second frame

cylindrical magnet inside the frame the problem is simpler, since the induction in the gap of such a magnet is distributed by the cosine law

$$\left. \begin{aligned} B_{01} &= B_0 \cos \alpha, \\ B_{02} &= B_0 \cos(\alpha + \delta). \end{aligned} \right\} \quad (\text{III.11})$$

where δ is the angle between the frames;

α is the angle characterizing the mutual position of the moving system and the line of the magnet poles.

The equation of equilibrium $M_1 = M_2$ now takes the form

$$I_1 B_0 \cos \alpha - I_2 B_0 \cos(\alpha + \delta) = I_2 B_0 (\cos \alpha \cos \delta - \sin \alpha \sin \delta).$$



Whence

$$\cos \alpha (I_2 \cos \delta - I_1) = I_2 \sin \alpha \sin \delta,$$

$$\tan \alpha = \frac{I_2 \cos \delta - I_1}{I_2 \sin \delta} = \frac{\cos \delta - \frac{I_1}{I_2}}{\sin \delta}.$$

The angle of rotation of the moving system may be calculated by the aid of the expression

$$\alpha = \tan^{-1} \frac{\cos \delta - \frac{I_1}{I_2}}{\sin \delta} \quad (\text{III.12})$$

This equation shows that the value of the angle δ between the frames affects the value of the angle of rotation of the moving system, so that the angle δ must therefore be selected in such a way as to obtain the maximum value of the angle δ while still maintaining stability of the instrument readings. In aircraft instruments, this angle is most often taken between 8 and 30°.

Errors of Ratiometers and Methods of Eliminating Them

Fluctuations in Feed Voltage. In fixing the value of the angle δ , the position of the moving system is determined by the ratio of the currents I_1/I_2 in the frames which is practically independent of the fluctuations of the feed voltage.

Temperature Errors. As in the galvanometer, a change in the temperature of the surrounding air leads to temperature errors in ratiometers. To compensate instrument temperature errors, a neutralizer is sometimes connected in series with each of the frames; but more often an additional resistance is introduced into the general design of the instrument. This resistance is made of two materials, the resistance of one of them remains almost constant with varying temperature (Constantan), while the resistance of the other varies considerably (copper). This method of compensation is particularly widespread in bridge instruments.

A comparison of galvanometers and ratiometers permits the following conclusion.

Ratiometers have an advantage over galvanometers in that their readings are

almost independent of the fluctuations of the feed voltage, but the sensitivity of galvanometers is considerably higher.

Both in ratiometric (two-frame) and galvanometric instruments, the angle of rotation of the pointer does not exceed 90° .

To extend the scale of the instrument, specially shaped pole pieces are used, which vary the law of distribution of magnetic induction in the gap; in this case the angle of rotation of the pointer may go as high as 270° .

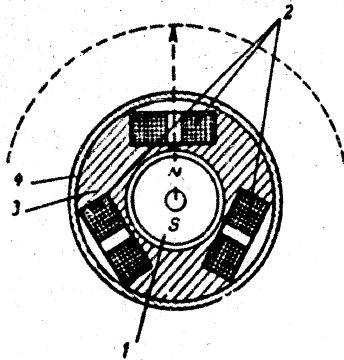


Fig. 23 - Three-Coil Ratiometer with Moving Magnet
1 - Magnet; 2 - Coil; 3 - Damper;
4 - Screen

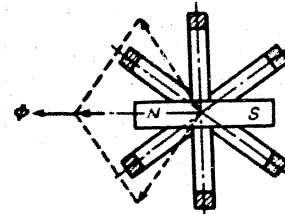


Fig. 24 - Scheme of Arrangement of Ratiometer Coil of Three-Frame Ratiometer Coils on the Rotation of the Moving System through 360°

To obtain a 360° scale, ratiometers with three frames are used. In some cases the frames are made movable, in other cases the magnets.

Figure 23 shows a three-coil ratiometer with a moving magnet. To reduce its weight, the miniature magnet is made from molded special magnetic powder. The reduction in the dimensions of the magnet inevitably leads to a reduction in the magnetic induction in the gap. This is compensated by a considerable increase in the outside dimensions and number of the windings of the frames which, being fixed, may be made fairly heavy and large. In its design, such an instrument is simpler



than an instrument with moving frames. Instead of the semi-axis of the moving frame mounted with high accuracy, the moving-magnet ratiometers use a continuous axis. The need for moving current feeds is thus eliminated, since the frames are fixed.

A drawback of this system is the poor damping owing to the absence of a moving metal skeleton of the frame. This may lead to oscillations of the pointers about the equilibrium position (at high sensitivity of the instrument and low steadying moment when the influence of the moment of inertia may become great).

To damp these oscillations, the moving magnet is placed in a copper or aluminum container in which eddy currents, damping the oscillations of the moving system, are induced. When the current is turned off, the instrument pointer is returned to the zero position by a special permanent magnet of very low power.

When the instrument is under current, the moving rotor of the magnet is set in the direction of the resultant vector of magnetic flux Φ (Fig. 24). If the frames are arranged at an angle of 120° , then the value of the magnetic induction for each of the frames may be determined from the formulas

$$\left. \begin{aligned} B_0 &= B_0 \sin \alpha; \\ B_0 &= B_0 \sin (120 + \alpha); \\ B_0 &= B_0 \sin (\alpha - 120); \end{aligned} \right\} \quad (\text{III.15})$$

where α is the angle of rotation of the instrument pointer.

With symmetrical frames, the magnitude of the moment acting on the moving system is equal to

$$M = a (I_1 \sin \alpha + I_2 \sin (120 + \alpha) + I_3 \sin (\alpha - 120))$$

where

$$a = 10^{-4} \frac{S W}{p}$$

In the equilibrium position of the moving system, $M = 0$.

Since

$$\sin (120 + \alpha) = \sin 120 \cos \alpha + \cos 120 \sin \alpha = \frac{\sqrt{3}}{2} \cos \alpha + \frac{1}{2} \sin \alpha$$

$$\sin (\alpha - 120) = \sin \alpha \cos 120 - \cos \alpha \sin 120 = -\frac{1}{2} \sin \alpha - \frac{\sqrt{3}}{2} \cos \alpha$$

we may write

$$I_1 \sin \alpha = - [I_2 \sin (120 + \alpha) + I_3 \sin (\alpha - 120)] = \frac{I_2 + I_3}{2} \sin \alpha + \frac{\sqrt{3}}{2} (I_3 - I_2) \cos \alpha$$

$$\tan \alpha = \frac{\sqrt{3} (I_3 - I_2)}{2I_1 - (I_2 + I_3)}$$

and

$$\alpha = \tan^{-1} \frac{\sqrt{3} (I_3 - I_2)}{2I_1 - (I_2 + I_3)} \quad (\text{III.14})$$

Equation (III.14) makes it possible to calculate the scale of the ratiometer.

The instrument readings, as in the case of the two-frame ratiometer, depend on the ratio of the currents in the frames.

The errors of three-frame ratiometers are similar in character and nature to the errors of two-frame instruments.

Section 3.

INSTRUMENTS FOR MEASURING PRESSURES AND TEMPERATURES

The Vacuum Manometer

The manometric instruments used a few years ago for checking the operation of a power plant have almost completely gone out of use, and for this reason these will be disregarded here.

The only manometric instrument still widely used at the present time is the vacuum gage, an instrument for measuring the pressure of the fuel mixture in the intake manifold of the engine. This instrument is used in engines with supercharging where, in order to maintain a definite composition of the fuel mixture,

regardless of the altitude, forced air supply (supercharging) is used.

The diagram of connection of the vacuum gage is given in Fig.25, and the kinematic diagram of the differential vacuum gage in Fig.26.

The sensitive element of the vacuum gage consist of two sylphons (2) and (3) which are rigidly attached to the hollow cylinder (1). The air is pumped out of the sylphon (2), while the sylphon (3) is connected to the intake manifold (2) of the engine.

The rigid center of the sylphon (3) is displaced as the pressure in it varies under the influence of the pressure difference $p - p_H$. The motion of the rigid center is transmitted over a transmitting mechanism to the instrument pointer (4).

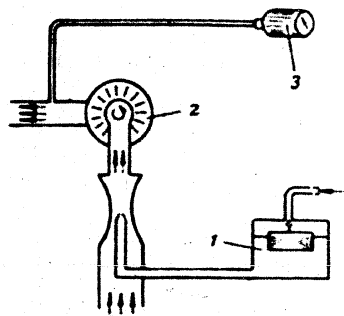


Fig.25 - Diagram of Connection of the Vacuum Manometer

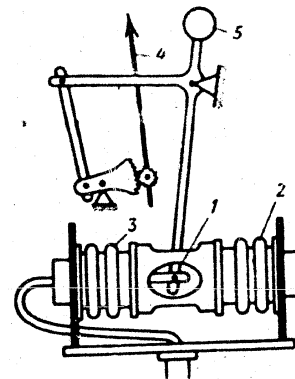


Fig.26 - Kinematic Diagram of Differential Vacuum Manometer

So long as the atmospheric pressure p_H remains unchanged, the sylphon (2) is not deformed, and only sylphon (3) operates. As the atmospheric pressure varies the value of the force acting on sylphon (3) from outside also varies and the sensitive element is deformed.

There are no errors in the instrument readings since, because of the variation of the atmospheric pressure and of the pressure in the body of the instrument, the

sylphon (2) is also deformed, thus compensating the deformation of the sylphon (3). The counterweight (5) is used to balance the instrument.

Errors of the Vacuum Manometer

As a result of the variation in the elasticity of the sensitive elements under the action of the surrounding temperature, so-called instrument temperature errors are created in the instrument. In order to eliminate these errors, a bimetal temperature compensation is provided, the design of which was described in the discussion on manometric navigational-piloting instruments (cf. Chapter IV, Section 1). Sometimes instruments with sensitive elements in the form of a corrugated box are used on aircraft. The design of such an instrument is similar to that of the altimeter, which is described in Section 1 of Chapter IV.

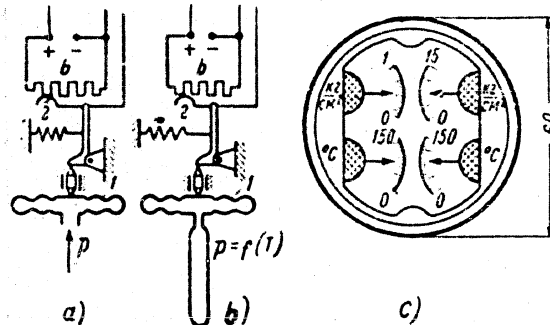


Fig.27 - Diagram of a Combination Instrument Operating on the Principle of Resistance Measurement

a - Pressure pickup; b - Temperature pickup; c - Indicator

Electromechanical combination instruments are widely used on aircraft to measure pressures and temperatures.

The principal advantages of such instruments are simple design, convenient installation, reliability, and compactness.

Compound Instruments for Measuring Pressures or Temperatures

Compound instruments whose indicator combines two to four instruments are widely used; these have separate pickups for each quantity to be measured. Such instruments lighten the work of the aircraft crew since they reduce the number of objects to be observed.

The operation of electromechanical instruments for measuring pressures and temperatures is based on the variation of resistances in relation to the values measured. A circuit diagram of one of such instruments and an over-all view of the indicator are given in Fig.27. The pressure p is supplied to the elastic element (1) having the form of a corrugated box or siphon.

If the pressure is being measured, then p represents the quantity measured (pressure of oil, gasoline, coolant, etc.). If the instrument is intended for measuring temperature, then the pressure p must be a function of the temperature being measured.

In this case the instrument must have a device for picking up the temperature and converting it into pressure. Such a device may be made, for example, in the form of a thermoscartridge filled with ligroin, the volume of which varies considerably with the temperature. The inner cavity of the thermoscartridge communicates with an elastic sensitive element having the form, for example, of a corrugated box.

As the temperature varies, the volume of the ligroin also varies causing a deformation (or flexing) of the elastic element.

The deformation of the elastic element (1) is transmitted over the transmitting mechanism to the slide block (2), wiping a potentiometer. Each position of the slide corresponds to a definite voltage applied to the indicator, designed as a galvanometer or ratiometer. Very frequently a combination ratiometer-galvanometer is used as indicator, in which case three or four indicators are combined in a single body (Fig.27c).

Compound instruments that measure only the pressure or only the temperature

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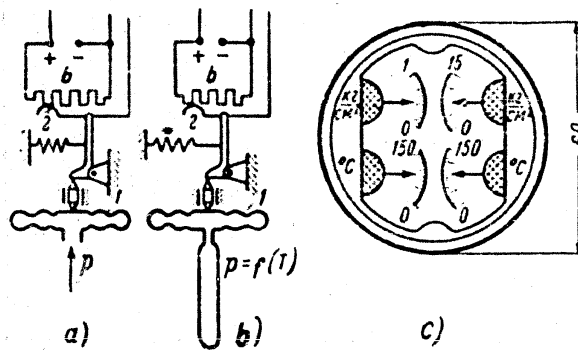


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Compound instruments that measure only the pressure or only the temperature

are widely used. An example of such an instrument is the combined differential manometer with double or even triple indicators (Fig.28).

In addition to these instruments, bimetal thermometers whose principal part is a bimetal coil are also used for measuring temperatures. The deformations of the coil corresponding to the variation in temperature permit estimating the temperature (such instruments are used for measuring the temperature of the outside air). Resistance thermometers and thermoelectric thermometers are also in use.

Resistance Thermometers

Resistance thermometers are used on aircraft for measuring the temperature of air, oil, coolant, carburetor mixture, etc.

The design of all above-listed instruments is the same and differs only within the limits of measurement and the range of the individual resistances. These instruments make use of the relation between the resistance of a conductor and the temperature, which obeys the law

$$R_T = R_0 (1 + \alpha_t \Delta T) \quad (\text{III.15})$$

where R_T and R_0 are the resistances at the temperatures T and T_0 , respectively, correlated by the expression

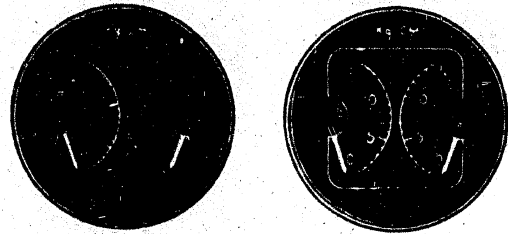
$$T = T - T_0;$$

α_t is the temperature coefficient of resistance; for copper and nickel,

which are used in these instruments, the coefficients are $\alpha_t = 0.004 - 0.0068$,

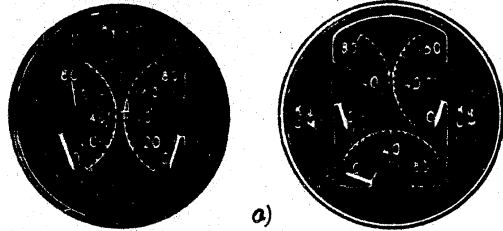
The instrument is arranged as a bridge circuit (Fig.29). One arm, representing the pickup, is placed at the point of measurement and is made of a thermosensitive material with a resistance R_T , whose value varies by about 0.4% for each Centigrade of temperature increase. Three other resistances are mounted in the indicator.

These are made of a material with a very low temperature coefficient (Constantan, Manganin). Usually, two of these resistances (R) are given equal value, while the



2 EDMU-2

2 EDMU-15



2 EDMU-80

3 EDMU-80

a)

Pickup

Indicator

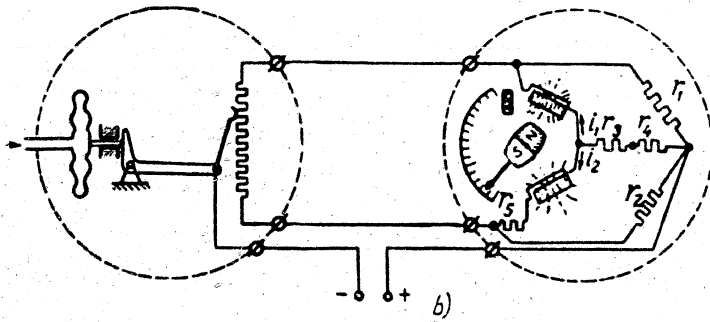


Fig.28 - Electric Remote-Reading Pressure Gage of EDMU Type

a- Outside view; b- Circuit diagram



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third (R_1) is so selected that, at the equilibrium state of the bridge, $R_{equil.} = R_1$. In one diagonal of the bridge, a galvanometer or ratiometer is connected (in this particular case a ratiometer), and in the other diagonal, the feed source. The resistance R_d steps down the voltage of the electric system of the aircraft to the required value. The resistance R_a is introduced to obtain currents of different value

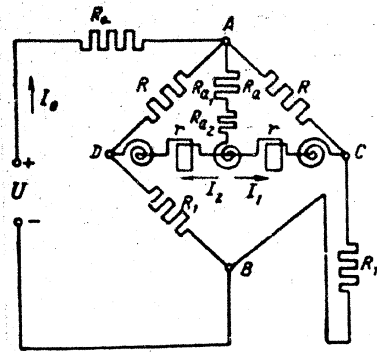


Fig.29 - Ratiometric Resistance

Thermometer

R_T - Resistance of the conductor; r - Resistance of the ratiometer frame; R and R_1 - Resistances of bridge arms; R_d - Additional resistances; $R_a = R_{a1} + R_{a2}$ - Compensating resistances; I_0 - Current used from voltage source; I_1, I_2 - Current in the ratiometer frame; U - Voltage, supplying the feed source.

and direction in the ratiometer frames.

This resistance is made of copper R_{a1} and Manganin R_{a2} , by which means temperature compensation is effected ($R_a = R_{a1} + R_{a2}$).

In the equilibrium state of the bridge, corresponding to a certain temperature $T_{equil.}$ the equation $RR_1 = RRT_{equil.}$ is valid; in this case, the current in the ratiometer frame is the same in value but differs in direction. Let us assume that the value of this current is I . For any other value of the temperature at the point of measurement, the current $I_1 = I + i$ flows to one frame of the ratiometer, while the current $I_2 = I - i$ flows through the other frame, where i is the current generated by the change in resistance R_T of the receiver with any change in the measured temperature. Depending on the ratio of the current I_1 to I_2 the pointer of the indicator assumes a certain position. The scales of such instruments never

er have divisions exceeding $160^\circ C$. This is explained by the fact that the copper

and nickel resistances used in the receivers cannot operate at high temperatures, because of a change in their characteristics as a result of oxidation. An external view of the indicator of the resistance thermometer is shown in Fig.30a, and the circuit diagram of the resistances on indicator panel.

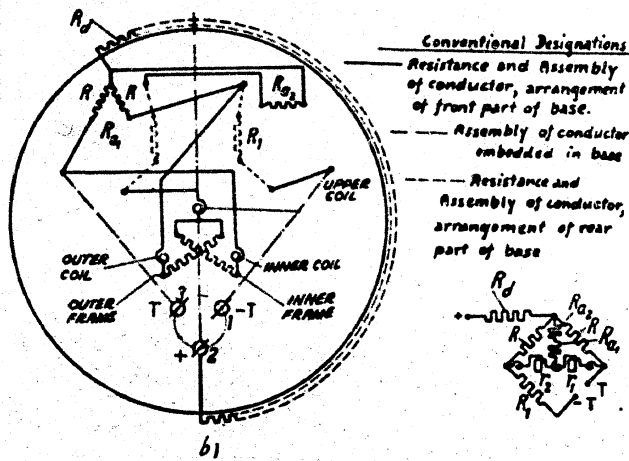
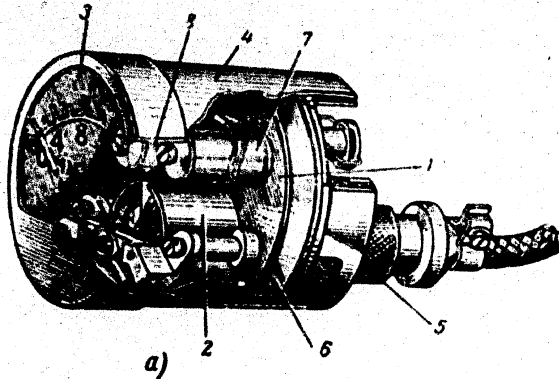


Fig.30 - Electrical Resistance Thermometer

a- Indicator

1- Panel; 2- Ratiometer; 3- Scale; 4- Body; 5- Plug contact;

6, 7, and 8- Resistances.

b- Circuit diagram of resistances on indicator panel.

circuit diagram of the resistances is given in Fig.30,b. Figure 31 gives the diagram of the resistance thermometer of the receiver. A copper wire, 0.05-0.07 mm in diameter, with enamel insulation, is wound on a paper-insulated hollow metal tube (2) which is

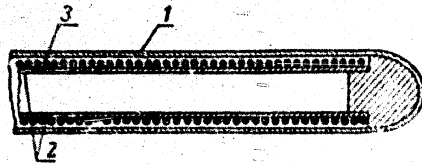


Fig.31 - Diagram of Resistance Thermometer Pickup.

1- Protective tube; 2- Inner tube;
3- Wire resistance

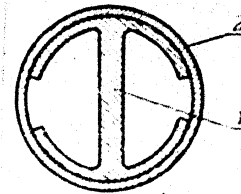


Fig.32 - Contact Plate of Resistance Thermometer

1- Plate; 2- Protective tube.

coated with lacquer and placed in the protective brass tube (1). During operation of the instrument, a good thermal contact must be provided between the thermosensi-

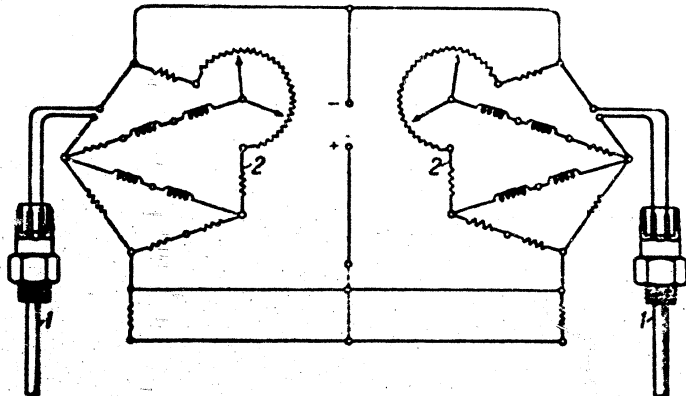


Fig.33 - Circuit Diagram of Universal Dual Electric Resistance Thermometer for Multi-Engine Aircraft.

...tive resistance (3) and the medium whose temperature is being measured. To improve

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0 this contact, a plate is sometimes used instead of the tube (2). This plate has the
1 form shown in Fig.32. The lugs of the plate (1) are pressed against the protective
2 tube (2). In designing the instruments the possibility of change in the resistance
3 R_T due to heating by the current itself, must be allowed for and eliminated.

4 To measure the temperature of water, oil, air, etc. on twin- and four-engine
5 aircraft, a universal dual resistance thermometer is used (Fig.33). The pickups (1)
6 of the thermometer do not differ from the pickups of ordinary resistance thermometers
7 and are installed at the point whose temperature must be controlled.

8 The indicator has two independent ratiometers (2) and two scales graduated from
9 - 50 to +150°C. The moving magnet is installed within two pairs of coils set at an
10 angle of 120°. The electromagnetic field of both pairs of coils acts on the magnet.
11 At low temperatures, the current is lower in the first pair of coils and higher in
12 the second pair. In this case, the action of the second pair predominates and de-
13 flects the magnet with the pointer downward. When the temperature increases, the
14 current increases in the first pair and decreases in the second pair, deflecting the
15 needle upwards.

16 The variable resistance is intended for regulating the sensitivity and for set-
17 ting the instrument to zero.

18 The indicator and pickup are linked to the conductor by means of plug connec-
19 tors.

20 Errors of the Resistance Thermometer

21 The cause of the systematic temperature error is a heating of the resistance of
22 the thermosensitive elements by the current passing through it.

23 Instrument errors are caused by the following:

- 24 a) Influence of the temperature of the surrounding medium on the characteris-
25 tics of the elastic elements, frame resistances, etc.
26 b) Wear, irregularity of the gaps, scale errors, and errors due to unbalanced
27 parts.

The error from wear appears to be the result of the relatively high friction in the axle journals, the hinge joints, etc. The error from the gaps is attributed to the weakness of the hair selecting the gaps in the transmitting mechanism of the instrument.

The scale errors are the mechanism control errors and scale markings, i.e., disparity between the angle of rotation of the pointer with the scale divisions.

The unbalanced state of the parts of the transmitting mechanism may cause a change of the readings when the instrument is inclined and may be due to fluctuations of the pointer under vibration.

c) Oscillations of the feed voltage.

According to the specifications for the instrument, the error in their reading under constant vibration and normal temperature must not exceed $\pm 2.5\%$ of the nominal scale value; at $T = + 50^{\circ}\text{C}$, this error may go as high as $\pm 5\%$ and reach $\pm 6\%$ of the nominal value at $T = - 60^{\circ}\text{C}$.

The error of a dual resistance thermometer with ratiometric indicators does not exceed 6°C over the entire scale.

Thermoelectric Thermometers

Thermoelectric thermometers are used for measuring high temperatures, for example the temperature at the cylinder heads and in air-cooled internal combustion engines, the temperature of the exhaust gases in jet engines, etc.

The operation of thermoelectric thermometers is based on the utilization of the thermal emf generated, in two cold junctions of dissimilar conductors when a temperature difference ΔT (Fig.34) arises at the soldering point. This phenomenon was first discovered in 1756 by the Russian Academician F.U.Epinus and later, in the middle of the 19th Century, studied in detail by M.P.Avenarius. In the USSR, P.I. Bakhmet'yev, I.I.Borgman, and others have done and are still doing large-scale work in this field.

If the temperature at one end of the junction is constant, then the value of

the thermoelectromotive force, for a given material of the thermocouple, depends only on the temperature of the other end.

In aircraft instruments, the temperature in the pilot cabin where the indicator is installed and the so-called "cold" junction of the thermocouple is located, is

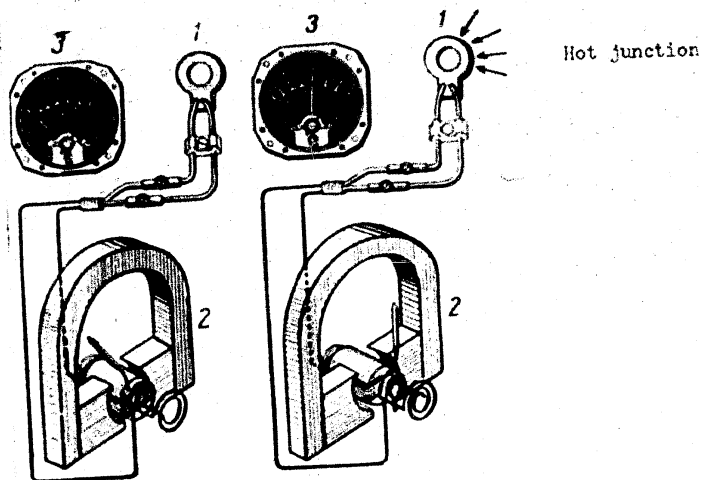


Fig.34 - Diagram of the Thermoelectric Thermometer

1- Pickup disk; 2- Galvanometer; 3- Indicator

considered constant. Consequently, the instrument readings depend on the tempera-

Material of Thermocouple	ΔT in °C	Thermoelectromotive force in mv
Nickel-nichrome	1000	42
Chromel alumel	1000	40
Iron-constantan	500	27
Chromel-copel	500	49

ture of the "hot" junction which is connected with the pickup of the instrument. The



most widely used materials for thermocouples are given in Table 2.

At $\Delta T = 300^{\circ}\text{C}$, the thermoelectromotive force of a chromel-copel thermocouple is equal to 20 mv, and the thermoelectric current is directed from the positive (chromel) electrode to the negative one (copel). Since the thermoelectric currents

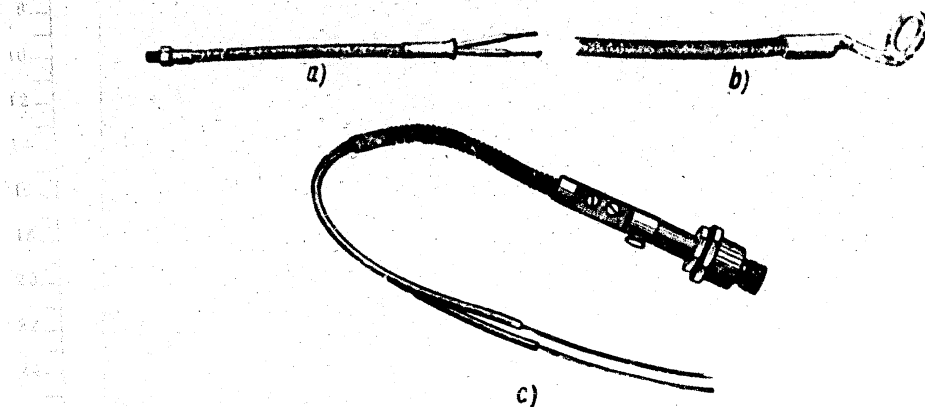


Fig.35 - Forms of Thermoelectrodes

- a- Thermocouple with junction to be screwed into a metal wall;
- b- Thermocouple in form of a disk; c- Sparkplug with thermocouple.

generated during the operation of the instrument are small, the measuring instrument must possess a high sensitivity, and for this reason a galvanometer with an intra-frame magnet is used as the measuring device.

The form of the thermoelectrodes may vary widely; according to the point of installation it may consist of a disk with two thermoelectrodes, two ends screwed into a metal wall, etc. (Fig.35).

Errors of Thermoelectric Thermometers

Systematic error arises as a result of the inconstancy of the temperature of the "cold" junction and the resultant variation in thermoelectromotive force.

To eliminate this error, either additional thermocouples creating a counter emf

can be introduced, or the design of the indicator can be supplemented by a device to correct its readings according to the value of the surrounding temperature. Such a device might be a bimetal spiral (1) which, as the temperature varies, changes the position of the end of the hair (2); see Fig.26. The moment of the hair is

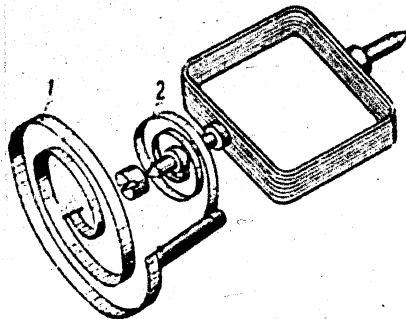


Fig.36 - Bimetal Compensation of Thermoelectric Thermometer
1- bimetal coil; 2 - Hair

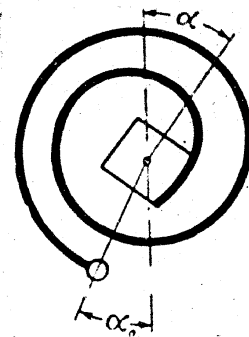


Fig.37.- Hair with Variable Point of Attachment

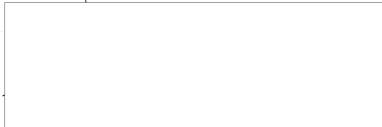
α - angle of twist of the hair spring due to change in value measured;
 α_0 - angle of deflection of hair spring due to deformation of the bimetal springs.

$M_1 = c (\alpha - \alpha_0)$; see Fig.37.

The instrument errors are caused by the variation in resistance of the galvanometer, conductors, etc. with any variation in temperature of the medium surrounding the indicator. Compensation of the temperature error is effected by means of a neutralizer.

In addition, instrument errors also arise as a result of wear, of incorrect gaps, etc.

The total error of the instruments (TTsT - 9) under constant vibration must not



exceed:

$\pm 8^{\circ}\text{C}$ at $t = 20^{\circ}\text{C}$

$\pm 15^{\circ}\text{C}$ at $t = 50^{\circ}\text{C}$

$\pm 24^{\circ}\text{C}$ at $t = - 60^{\circ}\text{C}$.

The Thermoelectric Thermometer for Measuring the Temperature of
Exhaust Gases of a Jet Engine.

A peculiarity of measuring the working gases in turbojet engines consist in the fact that the range of measurement is great and the temperature at various points of

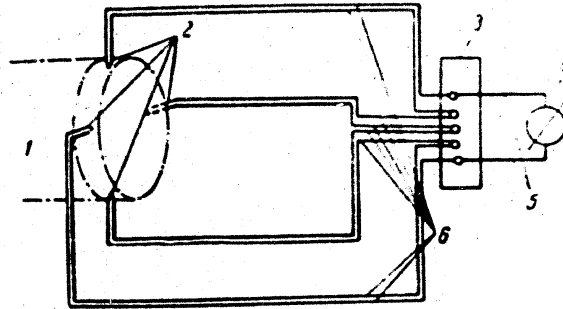


Fig.38. - Circuit Diagram of Thermoelectric Thermometer for Turbojet Engines
1- Jet nozzle; 2- Thermoelectric pickups; 3- Junction box; 4- Indicator;
5- Copper leads; 6- Thermoelectrode leads.

it may differ. In order to form a correct idea of the temperature, it is measured at several points and the mean value of these measurements is then determined.

A circuit diagram of the thermoelectric thermometer is given in Fig.38.

At four points of the reaction nozzle, thermoelectric pickups (2) (hot junction) are installed; the cold junctions of the thermocouples are placed in the junction box (3), where they are connected in series with each other. The two free ends are connected to the galvanometric indicator (4) by the copper conductors (5).

A thermoelectromotive force proportional to the temperature of the point being

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measured is generated in each thermocouple. At the ends of the leads (5) connected to the indicator the total thermoelectromotive force of the four thermocouples is active. The indicator is calibrated in values of one fourth of the total temperature of a single thermocouple and its readings correspond to the mean temperature of the four points of measurements.

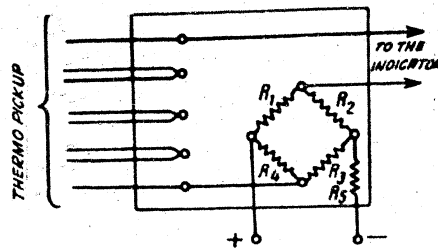


Fig.39 - Bridge Compensation of the Variations in Temperature of the Cold

Ends of the Junctions

The resistances R_1 and R_3 do not depend on the temperature;

The resistances R_2 and R_4 vary with the temperature.

The variation in temperature of the cold end of the junction leads to a systematic temperature error which is compensated in one way or another. In some cases, a bridge temperature compensation, whose diagram is given in Fig.39, is used.

Two arms of the bridge R_1 and R_3 are made of Manganin whose resistance is practically independent of the temperature, while the other two arms R_2 and R_4 are made of copper, i.e., of a material whose resistance varies with the temperature. At zero temperature, all the resistances are equal, and the bridge is balanced. A variation in temperature disturbs the balance of the bridge and produces an additional (compensating) voltage in its diagonals, which is algebraically added to the thermoelectromotive force of the thermocouple. The range of temperature measurements is from 300 to 900°C. In designing such instruments, particular attention is given to the corrosion resistance and electric strength of the pickup elements which operate

at high temperatures; it is likewise very important to ensure accuracy of measurement of high temperatures. This is accomplished by selection of thermocouples of low sensitivity to low temperatures (below 300°C).

Section 4.

INSTRUMENTS FOR MEASURING THE QUANTITY AND CONSUMPTION OF FUEL

Float Fuel Gages

Instruments designed for measuring the quantity of fuel in the aircraft tanks are called fuel gages. The design of oil gages indicating the oil supply on aircraft is similar to that of fuel gages. The same arrangement may also be used to measure the supply of oxidizer in liquid-jet engines, etc. Instruments based on measuring the volume of the fuel by means of a float are widely used.

Since the fuel tanks are located at a considerable distance from the instrument panel, the instrument must be of the remote-reading type. At the present time not only fuel gages measuring the fuel supply in a single tank are used, but also instruments indicating the supply of fuel in all the gasoline tanks, the so-called "summing" fuel gages.

The operation of remote-reading fuel gages is based on the transformation of a nonelectric quantity, namely the height of the fuel level, into an electric quantity acting on a galvanometric or ratiometric indicator.

A float fuel gage is designed on a bridge arrangement and consists of a pickup and an indicator connected by a remote line (Fig.40).

The pickup (Fig.41,a) is installed in the fuel tank and consists of a float submerged in the tank, whose lever (1) is displaced as the fuel is consumed. The displacements of the lever are transmitted to the slide (2), wiping the potentiometer (3), thus varying the ratio of the resistances R_1 and R_2 of the potentiometer arms. The corrugated box (4), through which passes the rocker lever (5), prevents penetration of fuel vapor into the potentiometer from the fuel tank. Thanks to the

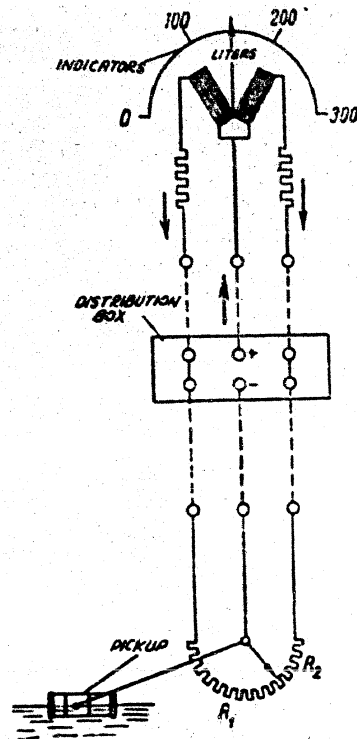


Fig. 40 - Scheme of Float Fuel Gage
 R_1 and R_2 - Resistances of potentiometer arms.

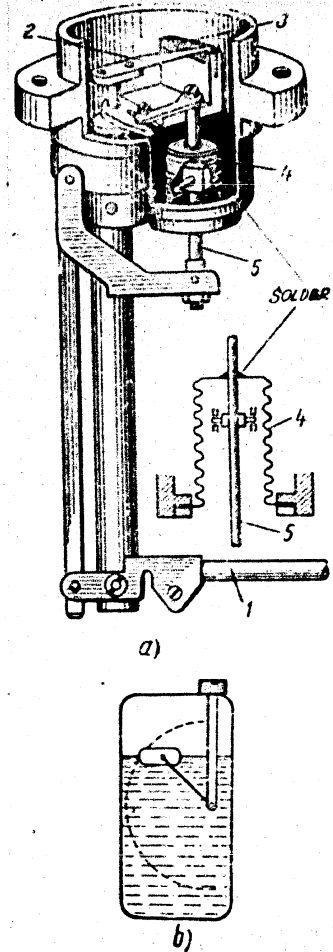


Fig. 41 - Pickup of Float Fuel Gage
 a- Operating mechanism of pickup;
 1- Float lever; 2- Potentiometer slide wire; 3- Potentiometer; 4- Corrugated wire; 5- Rocker arm
 b- Pickup of float fuel gage in long narrow tank

elasticity of the box (4), the lever (5) is displaced when the float lever (1) shifts, thus displacing, the sliding contact of the potentiometer.

The use of such a pickup is inconvenient in cases where the fuel tank has a long narrow shape (Fig.1,b).

The indicator of a float fuel gage is made in the form of a two-frame ratiometer (or of a galvanometer). The arrangement of the resistances on the indicator panel is shown in Fig.42.

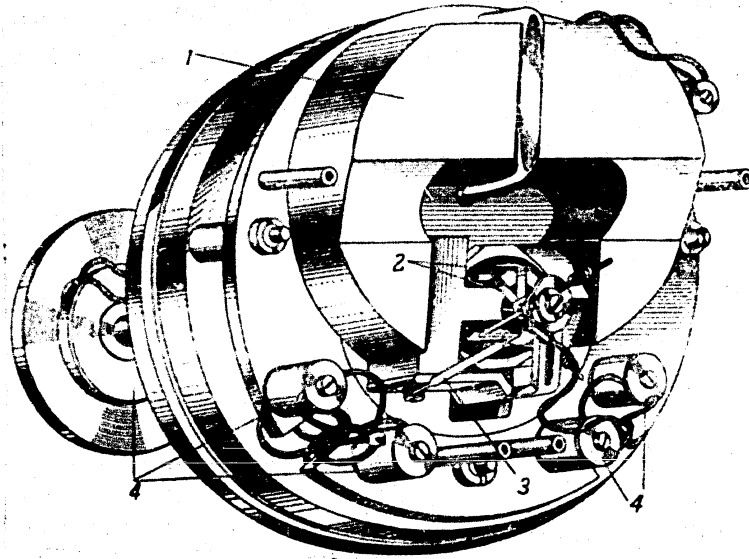


Fig.42 - Fuel Gage Indicator (scale removed).

1- Magnet; 2- Moving frames; 3- Instrument pointer; 4- Resistances

The fuel tanks of aircraft differ in shape, and for this reason the fuel gages serving an aircraft of a given type are assigned a definite mark; such a fuel gage is suitable only for that particular type of aircraft. Depending on the shape of

the tank, the shape of the pickup potentiometer also varies. To reduce the influence of the external magnetic field on the instruments and to eliminate the influence of this instrument on others, a magnetic shield is used in the form of an iron shell placed over the body of the fuel gage.

The diagram of the integrating fuel gage is given in Fig.43. The resistances R_{2I} , R_{2II} , R_{2III} are selected in such a way that the variation in their values with the motion of the slide corresponds to the fuel level in the tank. The instrument

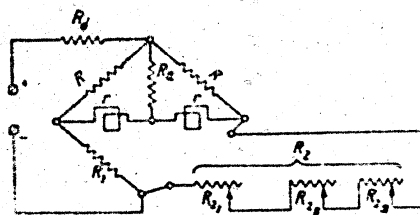


Fig.43 - Scheme of Integrating Fuel Gage

R_{2I} , R_{2II} , R_{2III} - Resistances of the pickups.

set includes a switch used for change-over of the fuel gage to the measuring of the quantity of fuel in all tanks. In this case, all the resistances R_{2I} , R_{2II} and R_{2III} are connected in series; when the quantity of fuel is measured in separate tanks, all pickups, except that installed in the particular tank in which the quantity of fuel is to be measured, are shorted.

Errors of Fuel Gages

a) Systematic errors are caused by the banking and pitching of the aircraft as

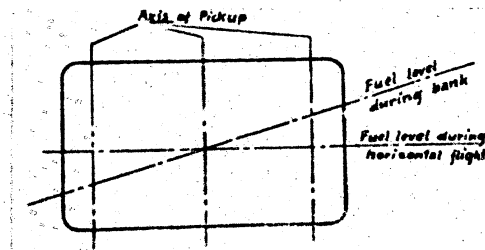


Fig.44 - Error due to Inclination of Tank

well as by the influence of accelerations of the aircraft, acting on the fuel in the tanks. In inclinations, when the tank is tilted together with the aircraft (Fig.44), the fuel gage reading for one and the same fuel supply will differ according to the position of the float. The smallest error corresponds to the case when the float is in the middle of the tank. A similar influence on the error of the instrument is due to accelerations which change the fuel level.

The cause of systematic errors may also be any deviation in shape, dimensions, installation of the tank from the design values.

b) Instrument errors of fuel gages may be divided into temperature errors, scale errors, errors due to wear, irregular gaps, fluctuations in the feed voltage, etc. These errors are compensated and eliminated by the same methods as those used in resistance thermometers. The total error over the entire scale of the instrument does not exceed 7% of the nominal value.

Electric Capacitance Fuel Gages

Capacitance fuel gages, based on the use of an alternating-current bridge (Fig.45) have recently come into wider use. The bridge is composed of the inductance (1) transformer winding), the fixed capacitance (2) and the variable capacitance of the pickup (3), which represents a capacitor. The galvanometer (4) is connected to a diagonal of the bridge across a rectifier.

The voltage in the bridge diagonal is amplified by an electronic amplifier. The bridge is fed with higher-frequency AC (400-1500 cycles) from a vacuum-tube oscillator which, in turn, is fed from the electric system of the aircraft through a vibrapack.

In the balanced state of the bridge, which corresponds to the instant where there is no fuel in the tanks (or when the tank is completely full of fuel), the bridge diagonal carries no current and the instrument pointer is at zero. Filling the tank with fuel (or in the latter case, consumption of fuel) leads to a change in the capacitance of the capacitor, since the dielectric constants of the liquid

well as by the influence of accelerations of the aircraft, acting on the fuel in the tanks. In inclinations, when the tank is tilted together with the aircraft (Fig.44), the fuel gage reading for one and the same fuel supply will differ according to the position of the float. The smallest error corresponds to the case when the float is in the middle of the tank. A similar influence on the error of the instrument is due to accelerations which change the fuel level.

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and air differ. The change in the capacitance of the pickup (which increases as the tank is filled with fuel) leads to an unbalance of the bridge. A current is generated in the diagonal, of a value proportional to the capacitance of the pickup and, consequently, to the fuel level in the tank. The maximum deflection of the pointer corresponds to a completely filled tank.

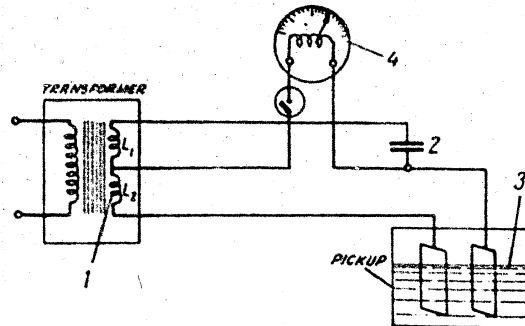


Fig.45 - Circuit Diagram of Capacitance Fuel Gage

1 - Inductance; 2 - Constant capacitance; 3 - Variable capacitance of pickup; 4 - Galvanometer

The pickup of the capacitance fuel gage shown in Fig.46, consists of a cylindrical capacitor with an inner electrode (1) and an outer electrode (2) (in high and narrow tanks, the wall of the tank itself may be used as the outer electrode) between which there is an insulating layer. Between the insulating layer and the outer electrode is the liquid whose level is being measured. An advantage of the capacitance pickup is its simple design and its small size, allowing a number of pickups to be placed in a single tank and permitting a mean readings. The readings of the capacitance fuel gage depend only slightly on the bank or on the vibration in the angle of pitch of the aircraft. In addition, the readings of such instruments are less affected by variations in the temperature of the surrounding air,

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2
3 since any change in temperature is accompanied by a simultaneous change in fuel
4 volume and in the dielectric constant.

5 A capacitance pickup can be successfully used in cases where the liquid is
6 electrically conductive or is chemically active (which is particularly important
7 for liquid-jet engines in which the supply of oxidant must be checked).

8 One disadvantage of capacitance fuel gages lies in the influence of the con-
9 necting leads on the accuracy of measurement of volume; as a result, special
10 shielded conductors or special connection diagrams must be used. Another disadvan-
11 tage is the relative complexity of the indicator.

12 The indicator of a capacitance fuel gage may be designed as a magnetoelectric
13 indicator connected across a rectifier, or as a directly connected ferrodynamic
14 indicator.

15 In recent times, electrocapitance fuel gages based on the utilization of
16 self-balancing AC bridges have come into use.

17 In such instruments, induction motors are used as indicators, operated by the
18 amplified voltage signals from the diagonals of the bridge.

19 Instruments Measuring Fuel Consumption

20 An indication of fuel consumption is necessary to evaluate the effectiveness
21 of the operation of an aircraft engine, as well as the consumption of oxidizer in
22 jet engines. There are flow meters that determine the total consumption of liquid
23 in kilogram or liters, the so-called integrating flow meters, and there are also
24 flow meters that determine the instantaneous hourly consumption of liquid in
25 liters/hour.

26 The flow meters used at present are based mainly on one of the following
27 three methods of measuring:

- 28 1. Measurement of the hydrodynamic pressure of the liquid (orifice meters).
- 29 2. Direct measurement of the rate of flow of liquid in a pipeline (volume

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0 meters).

3. Measurement of the units of equal volume of liquid fed in succession (rate of flow meters).

In the first method a throttling element, such as a Venturi tube, a diaphragm, or the like, is introduced in the stream

of liquid. The presence of this orifice leads to the formation of a pressure difference depending on the velocity of the stream. This permits calibration of the measuring instrument in units of rate of flow or of volume.

The manometer readings may be transmitted over considerable distances by means of a remote connection.

In rate-of-flow and volume flow meters a small vane is used as the sensitive element. The rate of rotation of the unloaded vane is proportional to the rate of flow of the liquid. The rate of rotation of the vane may be transmitted by a remote connection to an indicating instrument, calibrated in units of consumption.

If the rate of flow of the liquid is a constant quantity or is known in each case, a measured volume of fuel passes at each revolution of the vane.

The contact of an electrical chopper,

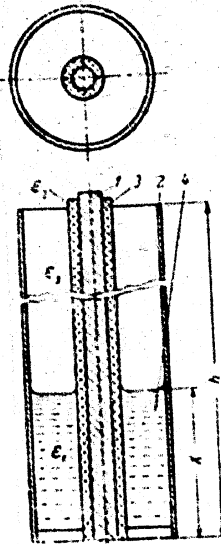


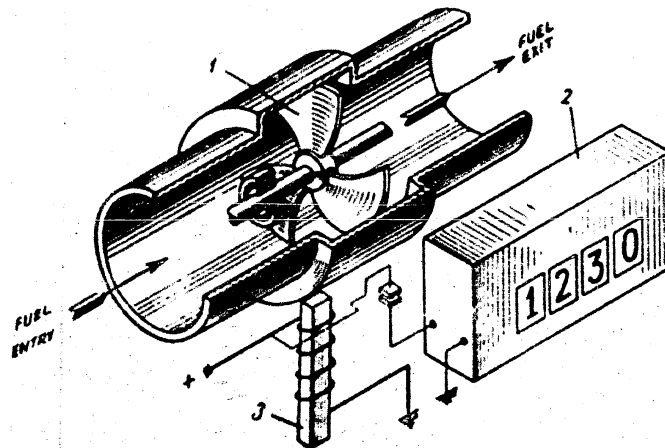
Fig.46 - Diagram of Pickup of Capacitance Fuel Gage

1 - Inner electrode; 2 - Outer electrode; 3 - Insulation; 4 - Fuel;

$\epsilon_1, \epsilon_2, \epsilon_3$ - Dielectric constants of the liquid, the insulating material, and the mixture of fuel vapor and air, respectively; h - Total height of the pickup cylinder; x - Height of the liquid level

0 sending pulses of electric current to an indicator, may be connected with the spin-
 2 dle of the vane by means of a crimped spring. The volume of fuel passed is propor-
 4 tional to the number of pulses. An instrument determining the quantity of fuel
 6 consumed, the so-called integrating flow meter, consists of a pulse emitter of mag-
 8 netic type and of an indicator-counter. Figure 47 gives the diagram of the pulse
 10 transmitter of a magnetic integrating flow meter. The magnetic transmitter consists
 12 of a vane enclosed in a tube.

14 The blades of the vane (1) are made of magnet steel and are magnetized. When
 16 the vane rotates, an alternating emf is induced in the coil of the pulse relay (3).
 18 The electric pulses received in the pulse relay are fed to a counter whose readings
 20 are proportional to the total number of pulses during the time of flight, i.e., to
 22 the volume of liquid that has passed through the measuring section. The pulse
 24 magnetic flow meter may also serve for the measurement of instantaneous consumption
 26



50 Fig.47 - Scheme of the Pulse Transmitter of the Integrating Magnetic Flow Meter

52 1 - Impeller; 2 - Indicator; 3 - Pulse relay

54 if a frequency meter determining the frequency of the pulses and calibrated in
 56

units of instantaneous consumption, is used as an indicator.

In addition to the above-described types of flow meter there are a large number of various types; compound flow meters intended for simultaneous measurement of the total and instantaneous consumption, are coming into wider use.

Section 5.

TACHOMETERS

Instruments designed to measure the rate of rotation of an engine, turbine, etc. are called tachometers.

The electromagnetic tachometer, which is widely used in aviation, consists of

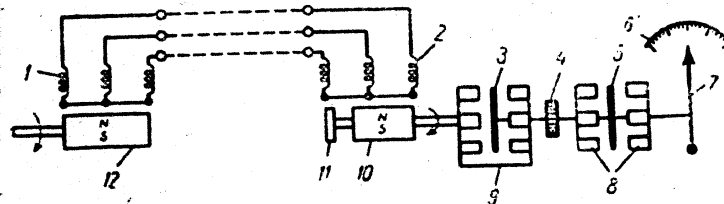


Fig.48 - Circuit Diagram of Electromagnetic Tachometer

1 - Stator winding of the pickup generator; 2 - Stator winding of the synchronous indicator motor; 3 - Sensitive elements; 4 - Spiral spring; 5 - Aluminum disc of magnetic damper; 6 - Scale; 7 - Pointer; 8 - Junction of magnetic damper; 9 - Magnetic junction of tachometer; 10 - Permanent magnet of rotor of synchronous indicator motor; 11 - Disk; 12 - Rotor of generator

a pickup installed in the immediate proximity of the aircraft engine (or turbine), of an indicator mechanically connected with it and installed on the instrument panels of the pilot and flight engineer, and of a system of conductors connecting the pickup with the indicator (Fig.48).

The pickup consists of a three-phase AC generator of the 4UG 1-48 type. The rotor (12) is a permanent magnet of a high-coercivity alloy "Alnico". The stator, built up of sheets of transformer steel 0.5 mm thick, carries a four pole three-phase winding (1) in star connection. A three-pronged plug-connector is used as lead-out for the stator winding. The three-phase current of the pickup feeds the synchronous electric motor of the indicator, whose rotational speed is proportional to the frequency of the feed current, and therefore, to the engine rpm.

The winding (2) of the stator of the electric indicator motor is analogous to the winding (1) of the generator. The rotor of the indicator motor is of the compound type, consisting of two cross-shaped permanent magnets (10) made of a copper-cobalt alloy with a high residual induction, and of a metal disk (11) made of non-magnetic metal. The rotor of the motor is shown in Fig.49.

When the magnetic field of the stator rotates rapidly, the magnet (2) is at first unable to follow it, because of inertia. However, the rotating field induces in the disk (1) currents that interact with the field, producing a mechanical moment directed toward the field of rotation. Under the action of this moment the rotor begins to rotate. When it approaches the synchronous speed, the magnet (2) will be

put into synchronous rotation. To facilitate the initial motion of the magnet at the low rates of rotation, it is installed on a bushing and is connected with the axle by the spring (3) which allows the magnet with the bushing to be rotated through 360° with respect to the rotor shaft. The purpose of the metal disk (1), rigidly connected with the rotor shaft, is to facilitate bringing the electric motor into syn-

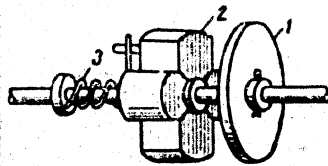


Fig.49 - Rotor of the Tachometer Indicator Motor

- 1 - Disk of nonmagnetic metal;
- 2 - Star magnet; 3 - Spring

chronism at high rotational speeds and to prevent the rotor from slipping when the

speed of the tachometer pickup changes suddenly.

The output end of the shaft of the electric motor carries, rigidly attached, the magnetic unit (9) of the tachometer (cf. Fig.48), consisting of two plates with molded-in cylindrical permanent magnets installed in such a way that the opposite poles of the magnets are opposite each other. The sensitive element is the aluminum disk (3) with a low temperature coefficient of resistance, which is installed in the air gap of the magnetic unit between the faces of the cylindrical magnets. The sensitive element is connected with the pointer (7), which is rigidly mounted to its spindle.

When the magnetic unit (9) of the tachometer rotates together with the rotor of the motor, eddy currents are produced in the sensitive element; their value is proportional to the rate of rotation of the rotor of the indicator motor and consequently also to the rate of rotation of the aircraft engine. The interaction of the eddy currents with the magnetic field of the magnets, pressed into the plate of the magnetic unit (9), produces a mechanical moment acting in direction of rotation of the magnetic unit (9) and proportional to its angular velocity. If the magnet rotates at a rate of n rpm, then the moment

$$M_d = a n$$

where a is the coefficient of proportionality, acts on the sensitive element (3).

This moment is resisted by moment M_n developed by the flat spiral spring (4), which is proportional to the angle of rotation of the moving system.

$$M_n = c \alpha$$

where α is the angle of rotation of the moving system;

c is the elasticity of the spring.

This means that, at a given rotational speed of the magnetic unit, the disk and the indicator pointer of the instrument will rotate through a definite angle

corresponding to the equality of the turning and opposing moments ($a_n = c\alpha$), i.e., the angle α of the indicator dial or the needle rotation is proportional to the number of rpm of the engine shaft:

$$\alpha = \frac{a}{c} n \quad (\text{III.16})$$

The tachometer is provided with a magnetic damper (8), which damps the oscillations of the moving system made analogous to the magnetic unit. The damping is affected by generation of eddy currents in the body of the aluminum disk (5). The damper considerably facilitates reading of the instrument.

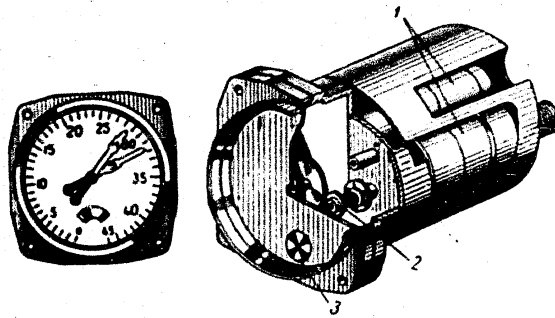


Fig.50 - Indicator of Dual Electric Tachometer

1 - Synchronous electric motor; 2 - Gear wheels; 3 - Synchronoscope

Multi-engine aircraft use dual indicators designed on the same principle as the above-described electromagnetic tachometer.

The indicator of the dual tachometer has two synchronous motors with a synchronous starter and two magnetic tachometers with pointers. The dials are rotated by means of geared couplings; one of the spindles is hollow (Fig.50).

The indicator is provided with a synchronoscope whose design is shown by Fig.51.

The principal part of the synchronoscope is an asynchronous electric motor (1)

fed by the two tachometer pickups. The rotor (2) of the motor has a three-phase winding and is fed from one of the pickups through contact rings and brushes. The winding (3) of the stator is also of the three-phase type. This stator is fed from the other pickup. The shaft of the rotor ends in the indicator disk (4).

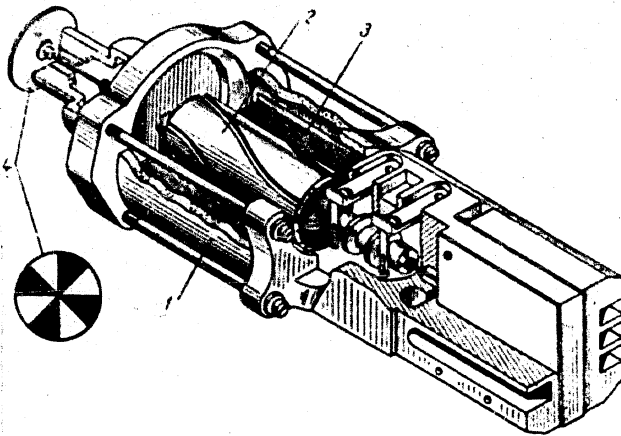


Fig.51 - Synchronoscope

1 - Induction motor; 2 - Rotor; 3 - Stator winding;

4 - Indicator disk

The phases of the rotor and stator of the synchronoscope are so connected that the directions of their rotating fields are different. If the frequency (speed) of both pickups is the same, then the magnetic fields of the rotor and stator of the synchronoscope mutually compensate each other in frequency, there are no rotating magnetic fields, and the rotor of the synchronoscope remains stationary. If the

speed of rotation (and frequency) of one of the pickups is higher than that of the other, then a resultant rotating field is formed with a velocity equal to the difference between the rates of rotation of the pickups. The rotation of the field takes place toward the pickup with the higher rotational speed.

The resultant field impels the rotar and thus indicator disk. The rotation of the disk may be observed through a slit on the indicator scale.

Errors of the Electromagnetic Tachometer

Instrumental Errors. The variation in the temperature of the ambient air causes a variation in the electric resistance of the sensitive element, in the magnetic induction in the gap of the magnetic unit, and also in the elastic properties of the hair. With increasing temperature, the induction in the gap decreases, leading to a decrease in the turning moment. This leads to the appearance of a temperature error compensated by a thermosensitive magnetic shunt whose resistance increases with the temperature; as a result, the magnetic flux passing through the shunt decreases, thus leading to an increase in the induction in the gap.

The instrument errors due to friction wear, elastic hysteresis, as well as to scale errors, etc. are the same as in other electric instruments.

The total error of the instrument at $t = 20^{\circ}\text{C}$ does not exceed 35 rpm.

Electric, ferrodynamic, and other tachometers are also used in aviation, but all of them are inferior in reliability of operation, convenience of use, cost, etc. to magnetic tachometer.

The DC electric tachometer is a system consisting of a DC generator (a collector generator with a permanent magnet in the stator) and an electrically connected indicator designed in the form of a magnetoelectric galvanometer.

These instruments have not found wide use because of the inadequate reliability of the tachometer pickups, whose transient electric resistance varies with the wear of the brushes and the fouling of the collector, leading to distortion of the readings.

0 The effort to increase the reliability of operation of the tachometer has led
2 to the design of electric tachometers with AC generators which can be made without
4 sliding contacts (for example, a synchronous generator with a rotor in the form of a
6 permanent magnet). Both the voltage and the frequency of such a generator are pro-
8 portional to the rate of rotation of the armature.

10 As an indicator for an AC tachometer, the following may be used: 1) a high-
12 sensitivity and precise magnetolectric galvanometer connected across a rectifier;
14 and 2) a ferrodynamic AC galvanometer in which the permanent magnet is replaced by an
16 electromagnet excited by a special winding through which there flows a current of
18 the same frequency as in the frame. The use of an induction galvanometer is also
20 possible.
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CHAPTER IV

PILOTING-NAVIGATIONAL INSTRUMENTS

SECTION 1.

BRIEF GENERAL INFORMATION ON MEMBRANE INSTRUMENTS

For the measurement of altitude, speed, and vertical speed of flight, membrane instruments are widely used on aircraft. Their operation is based on the measurements (by means of a corrugated box, manometric or aneroid) of a certain difference of pressure that is functionally connected with the quantity being measured.

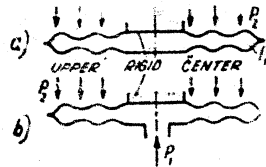


Fig.52 - Elastic Elements of Manometric Instruments

- a - Aneroid box; $p_1 = \text{const.}$;
- b - Manometric box $p_1 \neq \text{const.}$

A manometric box consists of corrugated membranes fused or welded at the ends (Fig.52). Usually a pin, connected with the region in which the pressure is being measured, is attached to the lower rigid (plane) center.

The upper rigid center is connected with a transmitting mechanism that moves a pointer. When there is a difference between the pressures inside and outside the box, both membranes tend to expand if the pressure inside is less than that outside. Since the lower rigid center is fixed, the displacement of the upper rigid center is equal to the sum of the bending of both membranes.

If the inner cavity of the membrane box is hermetically sealed, i.e., if there is no opening for the admission of air and all air has been exhausted, such a box is called an aneroid*.

* Sometimes gas-filled aneroid chambers are used.

The relation between the deformation of the chamber and the effective pressure difference is called the characteristic of the chamber.

The flexure of the membrane depends on the material, thickness, form, depth, and number of corrugations. The characteristics of the membrane may be rectilinear, decaying, or rising (Fig.53).

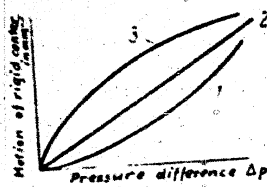


Fig.53 - Characteristic of Corrugated Membranes

- 1 - Rising; 2 - Rectilinear;
- 3 - Decaying

The deformation of the chamber is transmitted over a transmitting mechanism to the instrument pointer, and minor translational displacements (travel or stroke) of the elastic elements are transformed into measurable rotational motions of the indicating pointer. In some instruments the number of soldered membranes, i.e., the number of chambers, is increased

to increase the stroke. The transmitting mechanism is made in such a way as to allow regulation of the instruments by varying the length of the levers, the angles between them, etc. Owing to this feature the use of standard scales with divisions plotted in advance is possible. Most often reciprocating, jointed, horseshoe, and geared transmitting mechanisms are used in aircraft instruments. Whenever possible, a standardized mechanism is used, which simplifies manufacture, repair, and operation of the instruments.

The Barometric Altimeter

In flight, the aircraft crew must know:

1. The absolute elevation above sea level, corresponding to a pressure of 760 mm Hg and a temperature of 15°C.
2. The true altitude above the ground.
3. The relative height above a certain place selected in advance, for example, the take-off or landing place.

There are various methods and instruments for determining these heights. The absolute and relative altitudes are measured by a barometric altimeter. The true altitude is measured by a radio altimeter.

The barometric altimeter is based on the use of the relation between pressure and height, expressed by the standard barometric formula or the Laplace formula (cf. Chapter II).

$$H = \frac{T_0}{\gamma} \left[1 - \left(\frac{p_H}{p_0} \right)^{R/\gamma} \right]$$

$$H = 18400 \left[1 + \frac{1}{273} t_{mid} \cos \frac{p_0}{p_H} \right]$$

The barometric altimeter consists of a metallic barometer with an elastic element in the form of an aneroid chamber, in whose inner cavity the residual pressure amounts to 0.15 + 0.2 mm Hg while the pressure at the outer side is equal to atmosphere. The deformation of the chamber is greatest on the ground, where the atmospheric pressure is highest. On ascent to a certain height, the atmospheric pressure drops, the chamber is relieved of load and tends to be displaced upward. Figure 54 shows the kinematic diagram of the twin-pointer altimeters used at the present time.

To increase the accuracy of the readings, the sensitive element of the instrument is made in the form of two aneroid chambers (1), whose inner cavities are isolated from each other. With variation in altitude, the atmospheric pressure acting on the aneroid chamber from the outside also varies. The force of the atmospheric pressure is counteracted by the force of the elasticity of the chambers. To each atmospheric pressure there corresponds a definite value of deformation of the aneroid chambers. Any variation in atmospheric pressure causes a variation in the deformation of the chamber.

The deformation of the aneroid chambers is transmitted across a transmitting

mechanism to the instrument pointer. The transmitting mechanism consists of a combination of rocker and gear transmissions. The transmission of motion from the aneroid chambers (1) to the toothed sector (6) is effected by a reciprocating mechanism.

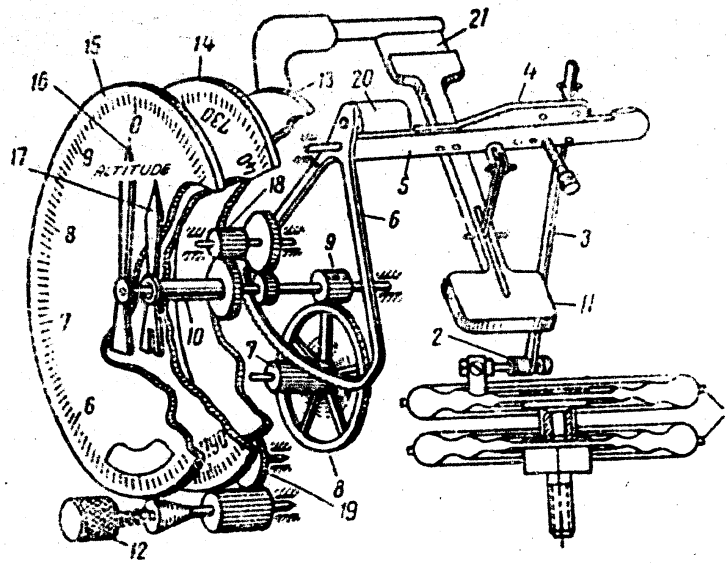


Fig. 54 - Kinematic Diagram of the Twin-Pointer Altimeter

- 1 - Aneroid chamber; 2 - Temperature compensation of the first kind;
- 3 - Tie rod; 4 - Bimetal plate (temperature compensation of the second kind); 5 - Shaft of a toothed sector; 6 - Sector; 7 - Gear; 8 - Large gear wheel; 9 - Gear; 10 - Shaft; 11 - Spring counterpoise; 12 - Rack;
- 13 - Base; 14 - Scale of barometric pressure; 15 - Instrument scale;
- 16, 17 - Pointers; 18 - Gear transmission; 19 - Gear wheel; 20 - Counterpoise; 21 - Spring

The toothed sector (6) engages the gear (7) to whose shaft the large gear wheel (8) is attached which in turn engages the gear (9). To the spindle of the

gear (9) the large pointer (16) is mounted.

Rotation of the gear (9) is transmitted at a gear ratio of 1:10, over the gear transmission (18) to the small pointer (17) which is mounted on a hollow shaft through which the shaft of the gear (9) passes. The large pointer indicates hundreds of meters and the small pointer thousands.

The aneroid chamber in the pickup mechanism is mounted on the rotating base (13). This base may be rotated by the aid of the rack (12), whose rotation is transmitted to the base (13) and the scale of the barometric pressure (14). The counterweight (20) is used to balance the drive mechanism.

In order to avoid influencing the readings of the instrument, the weight of the chambers themselves must be counterbalanced by the counterpoise (11) attached by the aid of the spring (21).

The spring of the counterweight is designed to hold the connections together. By rotating the shaft on which the spring is mounted, the initial position of the rigid center of the aneroid chambers may be displaced. This makes it possible to utilize the straightest part of the curve relating the stroke of the chambers to the altitude.

The instrument is placed in a standard hermetic body of 80 mm diameter. The body communicates with the atmosphere over the air-pressure intake. The scale of the instrument is graduated from 0 to 10,000 m.

Errors of the Barometric Altimeter

Systematic errors are caused by the use of an indirect method of measurements in the barometric altimeter, since this instrument does not measure the altitude itself but the pressure corresponding to a given altitude. The relation between the pressure and the altitude is determined by eqs.(II.3) and (II.4).

The altitude of flight H depends not only on the pressure corresponding to a given altitude p_H but also on the pressure p_0 on the ground and on the distribution

of temperature t_{mean} .

The deviations of these quantities from their calculated values, and the inaccuracy of the relations defining the value of H , P_H , T_H , t_{mean} , etc. lead to systematic errors.

The principal systematic errors of the altimeter are as follows:

1. Error caused by the variation in pressure on the airfield before take-off of the aircraft. At a variation in pressure on the airfield, the pointer of the instrument shifts from the neutral position and the instrument shows a certain altitude despite the fact that the aircraft has not taken off. This error is eliminated by rotating the pointer of the instrument by means of a rack and pinion, with respect to the fixed scale of the instrument, together with the entire mechanism, sensitive elements, and barometric scale.

It must be remembered that the use of a rack and pinion for changing the relative position of the pointer and scale and for setting the pointer to zero at various lengths of the chamber stroke (because of the various values of p_0), at zero altitude, requires the use of a scale that is uniform with respect to height.

With such a scale, at any altitude, one and the same angle of rotation $\Delta\alpha$ of the pointer corresponds to a definite increment of height ΔH ; despite the fact that the initial reading was displaced and the corrugated box shifted to a new position, this will not lead to an error.

When the pinion is rotated, the barometric scale and the pointers indicating the altitude are shifted toward each other, so that a loss of altitude will correspond to an increase in pressure.

After setting the pointers of the instrument to the zero division of the altitude scale, the true value of the pressure p_0 at the airfield is set on the pressure scale. This is accomplished by use of the gear wheel (19), assuring the required ratio between the rate of rotation of the base of the instrument and that of the pressure scale.

2. Error due to a variation in pressure at the airfield after take-off. If, at the moment of take-off the pressure at the airfield was P_0 , the pressure at an altitude H will be

$$P_H = P_0 \left[1 - \frac{gH}{r_0} \right]^{\frac{1}{\gamma}}$$

and the reading of the instrument will correspond to this pressure. Let us assume that, at the level of the airfield, the pressure has varied and has now become P_{01} ; this will lead to a variation in the value of the pressure P_H at all heights and thus to errors in the measurement of the altitude of flight H . This error is eliminated by rotating the barometric scale to pressure values, corresponding to the new value P_{01} . Together with the barometric scale, the entire mechanism and the pointer of the instrument are rotated and a correction is introduced into the altimeter reading. The value of this correction is calculated as follows:

If the calculated pressure on the ground is equal to P_0 , the actual value of the pressure on the ground is P_{01} , and the pressure at the altitude of flight is equal to P_H , then the altimeter will indicate the height in accordance with the equation $H = 18,400 \left(1 + \frac{t_{\text{mean}}}{273} \right) \log \frac{P_0}{P_H}$, while the actual value of the height

is $H_1 = 18,400 \left(1 + \frac{t_{\text{mean}}}{273} \right) \log \frac{P_{01}}{P_H}$. Since $\Delta H = H_1 - H$, it follows that

$$\frac{H_1}{H} = \frac{\log \frac{P_{01}}{P_H}}{\log \frac{P_0}{P_H}} \quad \text{and} \quad H_1 = H \frac{\log P_{01} - \log P_H}{\log P_0 - \log P_H}$$

or

$$\Delta H = H_1 - H = H \frac{\log P_{01} - \log P_H}{\log P_0 - \log P_H} \quad (\text{IV.1})$$

3. Error due to a change in the mean temperature of the column of air t_{mean} to the value t'_{mean} . The magnitude of the error may be found from the relations

$$\frac{H_1}{H} = \frac{273 + t_{\text{max}}}{273 + t_{\text{mean}}}; \quad H_1 = H \frac{273 + t_{\text{max}}}{273 + t_{\text{mean}}}$$

and

$$\Delta H = H_1 - H = H \frac{t_{\text{max}} - t_{\text{mean}}}{273 + t_{\text{mean}}} \quad (\text{IV.2})$$

The correction in the instrument readings can be made by means of the rack and pinion.

Instrument Errors. An altimeter, like any other membrane instrument, has the following characteristic instrument errors:

1. Scale errors due to imperfect adjustment of the mechanism and wrong scale setting, i.e., errors due to the noncorrespondence of the angle of rotation of the pointer with the divisions of the scale.
2. Errors due to friction, irregular gaps, unbalanced parts, nonherretic instrument body, elastic afterwork, and hysteresis.
3. Temperature errors due to change in the elasticity of the sensitive element with any variation in temperature. The variation in dimensions of the parts of the pickup mechanism may be disregarded, since such variations do not lead to substantial errors. The temperature errors may reach as much as 3% of the instrument readings. To eliminate these errors, a kinematic or a dynamic temperature compensation is used.

Instrument temperature errors are inherent to all instruments with elastic elements made of materials whose modulus of elasticity varies with the temperature of the surrounding air. For this reason the question of compensating these errors is of great interest. All data on the instrument temperature errors of altimeters and on the method of their compensation is applicable to all other instruments with elastic elements (speedometers, vacuum gages, etc.).

Kinematic Temperature Compensation. The main source of instrument temperature

errors is the change in the modulus of elasticity E of the sensitive element with variations in temperature. The law of variation in the modulus of elasticity may be considered to be approximately linear, i.e., it may be assumed that the bending of the sensitive element h varies according to the law

$$h = \frac{1}{E} f(p) = \frac{f(p)}{E_0 \cdot bT} \quad (IV.3)$$

where E is the modulus of elasticity;

E_0 is the value of the modulus of elasticity for $T = 0$;

b is the coefficient of proportionality;

T is the absolute temperature.

Let us assume that, at a given altitude, i.e., at a certain load on the sensitive element, the pointer is rotated through a certain angle. With increasing temperature the modulus of elasticity decreases, and, in spite of the fact that the load on the sensitive element remains unchanged, its course will vary so that the angle of rotation of the pointer will also vary. The higher the temperature, the greater will be the angle of rotation of the pointer at the same load. This variation in the angle of rotation of the pointer may be considered an increment in the transmission ratio of the instrument.

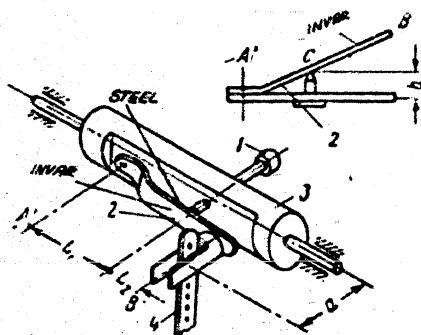


Fig.55 - Bimetal Temperature Compensation of the Second Kind

- 1 - Regulating screw; 2 - Bimetal plate; 3 - Shaft of toothed sector;
- 4 - Tie rod

In 1929, G.O.Fridlender proposed the correction of the readings of membrane instruments by means of a kinematic temperature compensation which would vary the transmission factor of the instrument by the

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necessary ratio. In a two-pointer altimeter this is accomplished by the bimetal plate (4), installed on the shaft (5) of the toothed sector (Cf. Fig.54).

Such a compensation is called a bimetal temperature compensation of the second kind. Its design is shown in Fig.55.

The bimetal plate (2) consists of two welded, soldered, or fused metals with different coefficients of temperature expansion.

A bimetal plate consisting of invar and steel is used in membrane aircraft instruments.

With increasing temperature, the bimetal plate bends in such a way that the metal with the higher temperature coefficient (steel) is located at the external (convex)side.

This property of a bimetal plate is utilized to change the transmission factor of the instrument. With increasing temperature, the rigidity of the chamber decreases, leading to an increase in the camber or curvature of the chamber under the same load, i.e., to a decrease in the altimeter reading.

To compensate the increase in camber, the transmission factor must be reduced. This may be accomplished by increasing the length of one of the driven arms or decreasing that of one of the driven arms of the transmitting mechanisms.

In an altimeter, a temperature compensation of the second kind is made on the driven arm. The increase in length of the driven arm, with increasing temperature, is affected under the action of the curvature of the plate (2). With decreasing temperature the curvature of the chamber and the length of the driven arm decreases.

To reduce the transmission factor of the instrument, the bimetal plate must be installed as shown in Fig.55, i.e., the steel part of the plate must be placed alongside the shaft (3) of the toothed sector. The bimetal compensation, based on the driven arm, will completely compensate the error at only two points of the scale; at all other points, the compensation is only partial.

In order to achieve complete temperature compensation over the entire scale,

the bimetal plate must be placed on the driving arm. However, for reasons of design this is not always possible. For example, in a barometric altimeter, a bimetal plate cannot be mounted on a driving member.

A bimetal temperature compensation of the second kind, reducing the temperature error over the entire scale of the instrument, may itself serve as a source of error. For example, with an unloaded chamber, a variation in temperature will not cause deformation of the sensitive element, while the arm of the toothed sector will change in length. This leads to a change in the transmission factor of the instrument mechanism and cause errors in its readings. It is obvious that this error will exist not only for an unloaded state of the chamber but over the entire range of operation of the instrument.

A working diagram of the temperature compensation of the second kind for various values of the angle ψ between crank and piston is shown in Fig.56.

Under the influence of a variation in temperature by ΔT , the driven arm receives an increment in length of Δa , and the shaft of the toothed sector rotates through an angle $\Delta \alpha$, whose value determines the error of the instrument reading. In this case the tie rod l is rotated by the angle $\Delta \beta$. The value of the angle $\Delta \alpha$ as indicated in Fig.56, may be determined from the relation

$$\Delta \alpha = \frac{l}{a} \Delta \beta \cos \psi \quad (\text{IV.4})$$

at $\psi = \frac{\pi}{2}$ the angle is $\Delta \alpha = 0$. For all values of ψ different from $\frac{\pi}{2}$, every change in temperature, even for the unloaded state of the chamber, will produce a displacement of the pointer.

It is impossible to satisfy the equation $\psi = \frac{\pi}{2}$ in the unloaded state of the chamber of a barometric altimeter. On the ground, at zero reading, of the instrument, the load on the chamber reaches a maximum, while its unloaded position corresponds to an altitude at which the pressure p_H is equal to the residual pressure inside the aneroid chamber. For this reason, we must assume that $\psi_0 \neq \frac{\pi}{2}$, which

leads to errors in the instrument reading due to the influence of the temperature compensation of the second kind. To eliminate this error a temperature compensation of the first kind is used. This type is installed on the upper rigid center of the corrugated membrane thus causing any variation in temperature to affect the origin of the reading.

In calculating the kinematics of the instrument, the operation of both compen-

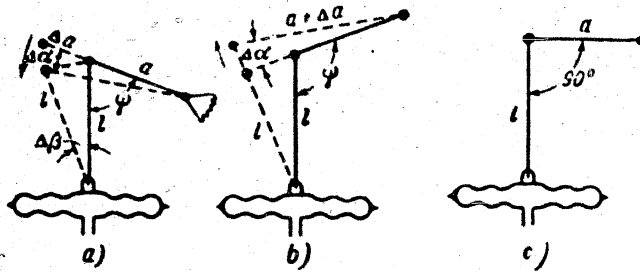


Fig.56 - Operation of a Bimetal Compensation of the Second Kind at Various Values of the Angle ψ

a - Angle $\psi < 90^\circ$; b - Angle $\psi > 90^\circ$; c - Angle $\psi = 90^\circ$

$\Delta\alpha$ - Angle of rotation of the shaft of the toothed sector under the influence of temperature change; Δa - Increase in length of crank; $\Delta\beta$ - Angle of rotation of tie-rod

sations is taken into account. Figure 58 gives the position of the crank drive of the altimeter corresponding to a certain value of the height H.

The camber h_1 of the compensation of the first kind is directed opposite to the stroke of the chamber.

The camber h_2 of the compensation of the second kind reduces (or increases) the length of the arm of the toothed sector. The angles between the separate members are taken in accordance with Fig.58, with the angle of rotation of the shaft

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of the toothed sector, determining the angle of rotation of the pointer, being de-

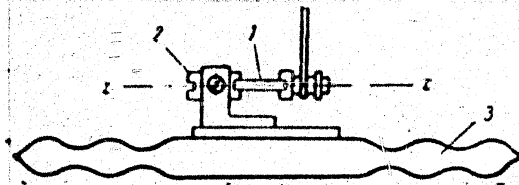


Fig. 57 - Bimetal Temperature Compensation of the First Kind

- 1 - Bimetal plate; 2 - Adjusting screw;
- 3 - Aneroid box

noted by α .

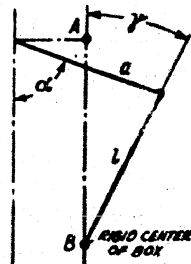


Fig. 58 - Reciprocating Drive of Altimeter

- α - Angle of rotation of the shaft of the toothed sector; l - Length of connecting rod (rocker); a - Arm of toothed sector (crankshaft)

The distance $AB = r$ is defined by the relation

$$r = a \cos \alpha + l \cos \gamma. \quad (IV.5)$$

With decreasing temperature, the elasticity of the box increases and its deformation diminishes by the value m_1 ; in this case the angles α and γ vary. In order to keep the instrument readings constant, it is necessary to hold the value of the angle α constant. In that case eq. (IV.5) takes the form

$$r - m + h_1 = (a - h_2) \cos \alpha + l \cos \gamma_1. \quad (IV.6)$$

Let us subtract eq. (IV.5) from eq. (IV.6).

Since the absolute value of the angle γ in the altimeter is close to zero, while the increments of this angle are small, it follows that the value of $l(\cos \gamma_1 - \cos \gamma)$

may be equated to zero. Then we have

$$h_2 \cos \alpha = m - h_1 \quad (\text{IV.7})$$

To find the value of the "stroke" of the compensation of the first and second kind (h_1, h_2), eq.(IV.7) is solved for a certain constant value of the temperature (when h_1 and h_2 are constant) and for two assigned arbitrary values of the pressure (usually for the pressure at ground level and the pressure at a certain altitude). If the temperature and pressure are known, it follows that the values

$$a_1, m_1 \text{ and } a_2, m_2$$

are also known.

After substitution in eq.(IV.7) we obtain

$$\left. \begin{aligned} h_2 \cos \alpha_1 &= m_1 - h_1 \\ h_2 \cos \alpha_2 &= m_2 - h_1 \end{aligned} \right\} \quad (\text{IV.8})$$

After finding the values of h_1 and h_2 from eq.(IV.8) we calculate the dimensions of the bimetal plates, starting from the condition of obtaining their required deformations. The dimensions of the plates are so selected that their stroke is somewhat longer than the calculated stroke, i.e., we take

$$h'_1 > h_1 \text{ AND } h'_2 > h_2.$$

The camber of the compensation of the first kind is determined by the equation

$$h'_1 = k_0 \Delta t \frac{L^2}{2} \quad (\text{IV.9})$$

where L is the length of the compensation of the first kind;

Δt is the increment of temperature;

$$k_0 = \frac{3}{2} \frac{a_1 - a_2}{a + b} \quad (\text{IV.10})$$

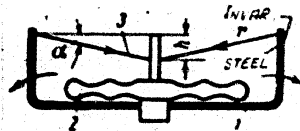
where a and b are the thickness of the layers of the bimetallic plate;
 a_1 and a_2 are the coefficients of linear temperature expansion.

The caniber of the compensation of the second kind is calculated by the equation

$$h_2 = h \left[1 + \frac{3}{2} \frac{L_2}{L_1} \right] + k_0 \Delta t \left[\frac{L_2}{2} + \frac{1}{1} L_1 L_2 \right]. \quad (\text{IV.11})$$

The regulation of the compensation of the first kind is accomplished by rotating it about the axis (Fig.57). To regulate the compensation of the second kind the

screw (1) is backed off or screwed in and, at the same time, displaced along the axis of the shaft (3); cf. Fig.55.



In the USSR the kinematic type of compensation is used considerably more often than the dynamic compensation. This is explained by the fact that the regulation of an instrument provided with such a compensation is much simpler.

Fig.59 - Power Temperature Compensation

- 1 - Bimetal braces; 2 - Membrane box;
- 3 - Spreader pins

Dynamic Temperature Compensation.

The operating principle of the dynamic temperature compensation is based on the force of action of the bimetal clamp (1) on the sensitive element of the instrument (2) over the spreader pins (3); (Fig.59).

The displacement h of the rigid center of the elastic elements, due to the deformation of the bimetal clamps is determined by the quantity $\frac{h}{r} = \sin \alpha$.

The compensation operates in the following way:

Let us assume that, at a certain altitude, the temperature of the surrounding

medium is increased while the atmospheric pressure remains constant. The box is more strongly compressed than before, and the instrument will give decreased readings. To eliminate this error, the bimetal clamps must be flexed, under the influence of the temperature change, in such a way that their force of compression against the rigid center of the aneroid box is reduced, i.e., that the displacement of the rigid center of the box at unchanged load also remains constant. The variation in the box at constant pressure, due to a change of its elastic properties with a change in the temperature of the surrounding air, must be compensated by a change in the concentrated load from the tie rods.

With increasing temperature, the aneroid box, at the same atmospheric pressure will be compressed more, while the compression of the spreader pins becomes less. At decreasing temperature, the picture reverses.

The direction of curvature of the bimetal braces, their shape and thickness, the position and number of the spreader pins may be varied according to the requirements of design.

By using a carefully designed temperature compensation and by replacing the single-pointer instrument by a twin-pointer type, the accuracy of the altimeter readings is considerably increased while the outside dimensions of the standard body, of 80 mm diameter, need not be changed. The allowable scale errors of modern altimeters at $t = 15^{\circ}\text{C}$ amount to 30 m at the beginning of the scale and 200 m at the end; at $t = + 50^{\circ}\text{C}$, the errors fluctuate within a range of 50 - 250 m and, at $t = - 60^{\circ}\text{C}$, amount to 80 - 300 m. However, such accuracy of readings is unsatisfactory for modern aviation. The problem of creating an exact and reliable altimeter continues to remain one of the most urgent problems of the aircraft-instrument building industry.

Airspeed Indicators

The presently existing instruments do not permit measuring the ground speed, i.e. the speed with respect to the ground allowing for the wind; instead, they indi-

cate the speed of the aircraft with respect to the air.

The airspeed indication (Fig.60) consists of a pickup (I) installed outside the aircraft (for example, under the wing), which picks up the static and dynamic pressure, and of an indicator II, installed in the cabin. The pickup consists of two

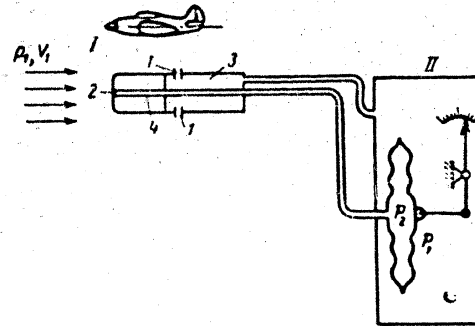


Fig.60 - Diagram of Aircraft Airspeed Indication

I - Air-pressure pickup II - Indicator

1 - Side vents; 2 - Front vents; 3 - Static pressure chamber; 4 - Full pressure chamber

chambers, for static and full pressure, respectively. In chamber (3) the pressure is equal to the atmospheric pressure. The air enters the static pressure chamber through the orifice (1) and enters the chamber (4) through the orifice (2) at the face of the pickup.

Let the pressure in the atmospheric layer at which the flight takes place be P_1 , the flying speed V , the density of air in the atmosphere γ_1 , the pressure in the total-pressure chamber P_2 , the air velocity in it V_2 , and the density of the air in

the total-pressure chamber γ_2 .

The principle of operation of the instrument is based on the Bernoulli equation, which has the form

$$\frac{p_1}{\gamma} + \frac{V_1^2}{2g} + z_1 = \frac{p_2}{\gamma} + \frac{V_2^2}{2g} + z_2 \quad (\text{IV.12})$$

The leveling heights $z_1 = z_2$ can be neglected. When contacting the face of the Pilot tube, the air stream is decelerated, its velocity becoming equal to zero, i.e., $V_2 = 0$, but as a compensation p_2 increases sharply. This pressure is called total pressure, while the pressure $p_1 = p_H$ is called static pressure. Then, eq.(IV.12) may be written in the form

$$\frac{p_1}{\gamma} + \frac{V^2}{2g} = p_2$$

At flying speeds up to 400 km/hour, the process of compression of the air in the pickup may be considered isochoric, and the specific gravity of the air to be constant, i.e.,

$$\gamma_1 = \gamma_2 = \gamma; \quad V_1 = V; \quad p_2 - p_1 = p_t - p_H = \Delta p;$$

In that case,

$$\Delta p = \frac{\gamma V^2}{2g}, \quad (\text{IV.13})$$

$$V = \sqrt{\frac{\Delta p 2g}{\gamma}}$$

At high speeds, the process of compression of the air is almost adiabatic, with the adiabatic index k ; in this case, $\gamma_1 \neq \gamma_2$, while

$$\frac{\gamma_1}{\gamma_2} = \left(\frac{p_1}{p_2} \right)^{\frac{1}{k}}$$

so that the Bernoulli equation now takes the form

$$\left[\frac{p_2}{p_1} - \frac{p_1}{p_1} \right]^{\frac{k}{k-1}} = \frac{V^2}{2g}$$

By solving this equation for V and replacing γ by $\frac{p}{RT}$, we find

$$V = \sqrt{\frac{2gRk}{k-1} \sqrt{T_H} \left[\frac{\Delta p}{p_H} + 1 \right]^{\frac{k-1}{k}} - 1} \quad (\text{IV.14})$$

where R is the universal gas constant equal to 29.27 for air;

g is the acceleration of gravity;

k is the adiabatic index;

T_H is the absolute temperature at the altitude H.

The total-pressure chamber is connected with the inner cavity of the sensitive element of the indicator, while the static-pressure chamber is connected with the body of the instrument.

In this way, the sensitive element of the indicator, consisting of a metallic manometer, is subjected to the pressure difference $\Delta p = p_2 - p_1$. Under the action of this pressure difference, the sensitive element, consisting of one or more corrugated boxes, is deformed. This deformation is transmitted over the transmitting mechanism to the instrument pointer.

Since the pressure difference Δp is correlated with the aircraft speed by eq.(IV.14), the scale of the indicator of the airspeed indicator may be calibrated in units of speed.

Speed Indicator Pickup. As noted above, the speed indicator pickup (Pitot tube) consists of the static-pressure chamber (5) and the total-pressure chamber (1); (Fig.61). The orifices (3) are placed at distances of four to five receiver diameters from the control point at a place where the air stream is not distorted. To

control icing, the pickup is provided with an electric heater. When the aircraft is on the ground, the pickup is covered with a special cover. The indicator of the instrument is manufactured in single-pointer and twin-pointer models.

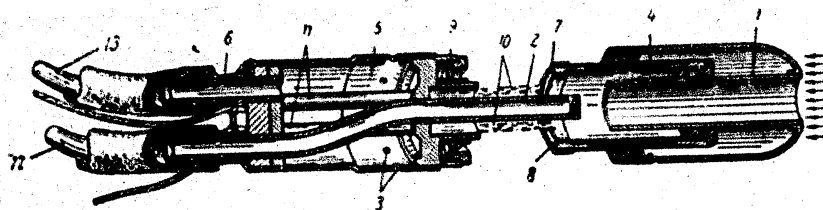


Fig.61 - Air Pressure Pickup

1 - Total-pressure chamber; 2 - Total-pressure tube; 3 - Open orifice of static chamber; 4 - Winding of electric heater; 5 - Static chamber; 6 - Static tube; 7 - Outer contact ring; 8 - Inner contact ring; 9 - Insulating bushing; 10 - Current conductor; 11 - Tube for electric conductor.

The diagram of the two-pointer speed indicator designed by the Soviet designer, M.G.El'kind, is given in Fig.62. The range of measurements is from 0 to 1000 km/hr. Since the instrument has two pointers, its scale must necessarily be uniform. To assure uniformity and increased sensitivity of the scale, the sensitive element is made of two manometric boxes (3) and the flat spring (8) with variable elasticity. Variable elasticity is obtained by means of screws which change the working length of a spring. With such a design of the sensitive element a characteristic, linear with respect to speed, may be obtained.

There is a transmission with a transmission ratio of 1:10 between the small and large pointer, so that the large pointer indicates tens and units of km/hr, while the small pointer indicates hundreds of km/hr.

To reduce friction and prolong the life, all spindles rest on agate bearings.

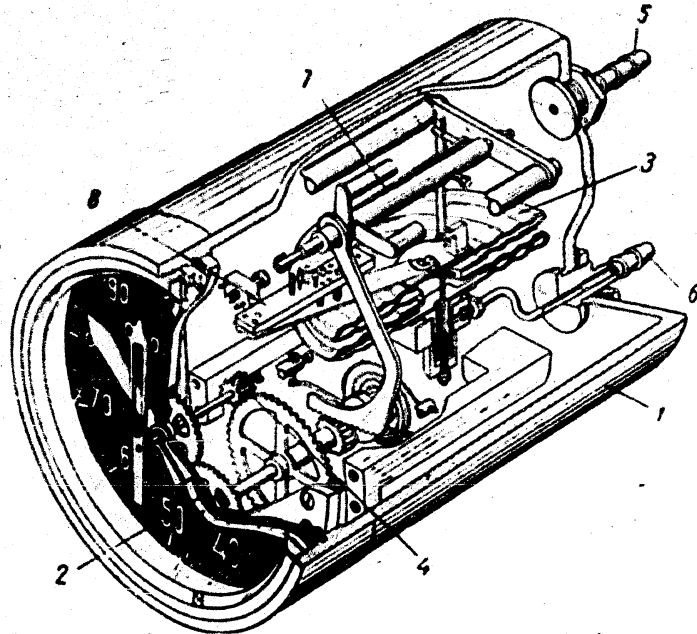


Fig.62 - Two-Pointer Speed Indicator

1 - Body of instrument; 2 - Scale; 3 - Manometric corrugated boxes; 4 - Gear transmission; 5 - Static pressure connecting branch; 6 - Total pressure connecting stud; 7 - Bimetal temperature compensation; 8 - Variable elasticity spring

Errors of the Speed Indicator

Sustematic Errors

1. Errors due to the Air Pressure Pickup. The form of the nozzle of the pickup, the form of the static orifices and their arrangement along the perimeter and along the generatrix of the pickup cylinder, the installation of the pickup on the aircraft, its arrangement with respect to the air currents encountered, all have a

considerable effect on the correctness of the total and static pressure intake. The errors due to inaccuracy of the pickup may go as high as 30 km/hr. In recent years work had been done, and is still going on, on designing the optimum type of air pressure pickup. Original and varied designs of a few pickups have been worked out including a few models of USSR pickups as well as those made by the firms Askania, Kollman, Pioneer, etc.

The best characteristics are exhibited by the Soviet pickups, in which the error due to changes in the angle of flow of the jets ranging from 0° to 25° , does not exceed 0.2%, while such error reaches 9% in the Kollman receiver and 11.5% in the Askania pickups.

Soviet designers have created pickups that operate well even at velocities close to sonic. The operation of an air pressure pickup is characterized by the aerodynamic coefficient ξ , which is equal to the ratio of the pressure drop sensed by the pickup to the calculated pressure drop, ranging from 0.96 to 1.2. This coefficient is introduced in the equation by which the flying speed V is determined;

$$V = \sqrt{\frac{2g\Delta p}{\xi}} \quad (\text{IV.13a})$$

$$V = \sqrt{\frac{2gRg}{(k-1)\xi} \sqrt{T_s}} \sqrt{\left(\frac{\Delta p}{P_s} + 1\right)^{\frac{k-1}{k}} - 1} \quad (\text{IV.14a})$$

The pickup is placed in such a position that its operation is unaffected by the propeller air stream and by the vortical motion of the air flowing around the aircraft.

2. Error due to the Influence of the Density of the Air. The operation of the

airspeed indicator is based on measuring the pressure difference $p = p_1 - p_2$ by a metal manometer.

The equation

$$V = \sqrt{\frac{2kRg}{(k-1)\rho} \sqrt{T_H} \sqrt{\left(\frac{\Delta p}{p_H} + 1\right)^{\frac{k-1}{k}} - 1}}$$

shows that the speed depends not only on p but also on p_H and T_H , i.e., on the density of the air. For this reason a systematic error appears in the instrument readings and is denoted as the error due to the influence of the density of the air.

As with the altimeter, this error is the result of the indirect method of measuring speed. In order to obtain the value of the airspeed of an aircraft without this error, the navigator of the aircraft, using an aerial navigation slide rule, applies a correction to the readings given by the airspeed indicator by the aid of the equation

$$V_{true} = V_{inst} \sqrt{\frac{p_0 T_H}{p_H T_0}} \quad (IV.15)$$

where V_{true} is the true airspeed;

V_{inst} is the speed shown by the speed indicator;

p_0, T_0 are the calibration values of pressure and temperature;

p_H and T_H are the pressure and temperature at the altitude of flight.

Instrument Errors of the Speed Indicator. Instrument errors of the speed indicator are analogous to the corresponding errors of the altimeter.

In the submerged condition of the box, the angle ψ (cf. Fig. 56) is taken as equal to $\frac{\pi}{2}$, and for this reason only a bimetal temperature compensation of the second kind is used in the instrument.

The allowable scale errors of the instrument at a temperature of $+15^\circ\text{C}$ do not

0 exceed ± 15 km/hr, while at temperatures $+ 50^{\circ}\text{C}$ and $- 60^{\circ}\text{C}$ these errors may reach
 2 ± 20 km/hr.

4 True Airspeed Indicator. At present, instruments in which the correction for
 6 the value of the density of the air in accordance with eq.(IV.14) is automatically
 8 applied by changing the gear ratio of the instrument, are widely used. Instruments
 10 showing the aircraft speed with respect to the air, corrected to the value of the
 12 density, are called true airspeed indicators, while the speed so shown is known as
 14 the true airspeed.

16 The correction for pressure is usually applied by the aid of an aneroid box,
 18 whose stroke depends on p_H ; the change in this stroke correspondingly changes the
 20 length of one arm of the drive. The correction for temperature may be applied by
 22 means of a thermometer, automatically or manually. The variation in the temperature
 24 of the surrounding medium deforms the sensitive element of a thermometer. This de-
 26 formation is converted into a change in the gear ratio of the instrument. Several
 28 different designs of such instruments exist. Figure 63 gives a diagram of a true
 30 airspeed indicator with a carrier drive mechanism. The deformation of the manomet-
 32 ric box (1) is transmitted over the shafts (2) and (3) to the toothed sector (8)
 34 and then to the pointer (4). The change in the pressure is compensated by the ane-
 36 roid box (5) which displaces the slide (9) along the carrier of the roller (2), in
 38 contact with it. Temperature compensation is achieved by the displacement of the
 40 slide (10) along the carriers connected with the rollers (2) and (3).

42 The thermometer (6) displacing the slide (10) must be of the remote-reading
 44 type and have a great steadying power. For this purpose a liquid thermometer should
 46 be used.

48 At present, the true airspeed indicators in use vary widely in design, and
 50 their readings are corrected for the value of the density of the air; they include
 52 anemometric instruments, wind vanes, and a whole series of manometric speed indica-
 54 tors. The use of these instruments considerably lightens the work of the navigator.
 56

The pilot does not need a true airspeed indicator; actually, cases in which he uses this instrument to judge the lift, it may even be harmful. We shall explain this by the following example: The difference Δp between the total and static pressure and the lift Y is proportional to the density of the air :

$$\Delta p = \frac{\gamma V^2}{2g} \quad (\text{IV.16})$$

$$Y = c_y S \frac{\gamma V^2}{2g} \quad (\text{IV.17})$$

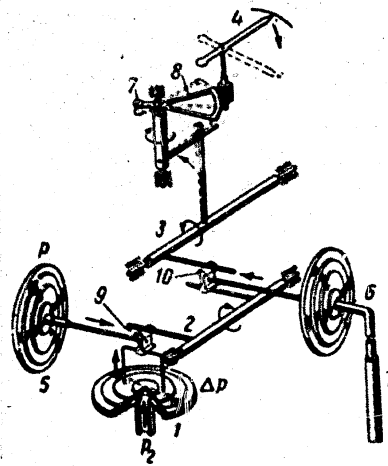


Fig.63 - Diagram of the Carrier of a True-Airspeed Indicator

- 1 - Manometric box; 2,3 - Shafts;
- 4 - Indicator; 5 - Aneroid box;
- 6 - Thermometer; 7 - Support;
- 8 - Toothed Sector; 9,10 - Slide blocks

is regulated by the toothed sector over the shaft (12), the holder (14), and the

For this reason, the change in the velocity V is a signal to the pilot of a change in the lift Y .

The true airspeed indicator should not react to variations in air density, while the value of the lift, which depends on it, does change. Thus the pilot cannot judge the variation in the lift from the readings of the true airspeed indicator. On modern high-speed and high-altitude aircraft, universal speed indicator of the CSI type are installed.

Compound Speed Indicator (KUS).

Figure 64 shows the layout of a compound speed indicator. The indicator hand (8)

shaft (15). The position of the box varies according to the density of the air* which corresponds to the given altitude H , causing an additional displacement of the spindle (15), and together with it, of the sector (16) and the indicator hand (18). This pointer indicates the airspeed corrected for the density of the air, i.e., the true airspeed. With diminishing density of the air, the aneroid boxes (20) expand,

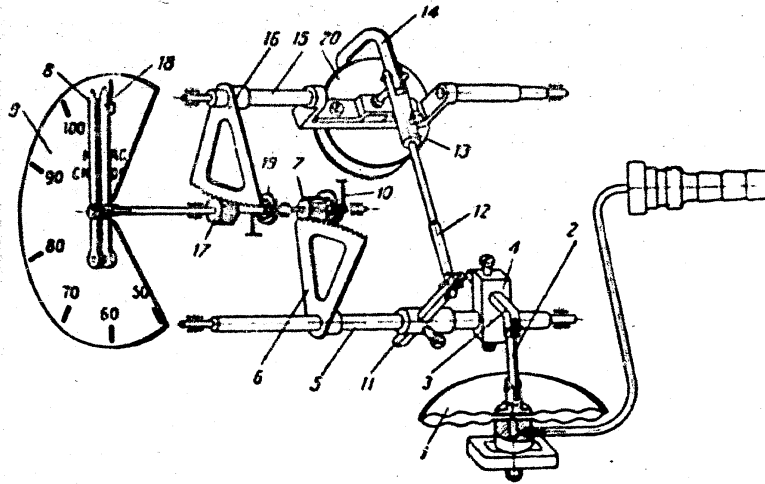


Fig. 64 - Kinematic Diagram of KUS-1200

- 1 - Manometric box; 2 - Tie rods; 3 - Carrier; 4 - Clamp; 5 - Spindles;
- 6 - Sector; 7 - Gear; 8 - Pointer; 9 - Scale; 10 - Hair; 11 - Carrier;
- 12 - Tie rods; 13 - Sleeve; 14 - U-holder; 15 - Compensated spindle;
- 16 - Sector; 17 - Gear wheel; 18 - Pointer; 19 - Air; 20 - Aneroid box

* The density of the air depends on pressure and temperature. The deformation of the box is a function only of the pressure. The instruments of the temperature may be taken into account if it is assumed that the temperature varies with height according to S_A . In that case, to each temperature will correspond its own pressure, and it may be considered that density is a function of pressure alone.

and the arm between the tie rod (12) and the spindle (15) is shortened. Because of this fact, at one and the same displacement of the movable rigid center of the manometric box (1), the sectors (16) and (18) rotate through a greater angle than the sector (6) and the pointer (8).

In this way, the pointer (18), indicating the true airspeed, will always lead the pointer (8) indicating the airspeed.

By means of the aneroid boxes (20), a correction for air density is applied, i.e., for the value of the pressure p_H and the temperature T_H .

In this way, the systematic errors in the KUS are taken into account by the aid of the aneroid boxes (20).

The instrument errors of the compound speed indicator are the same as in the airspeed indicator. The principal instrument error is the temperature error. To eliminate it, a temperature compensation of the second kind must be used in the design of the instrument. In the modern KUS, such a compensation is not used, since the deformations of the boxes (1) and (20), under the influence of the temperature compensate each other to a certain extent.

Mach-Number Indicators

In addition to the true airspeed indicators, instruments showing the ratio of the true airspeed to the speed of sound are also installed on aircraft. These instruments are known as Mach-number indicators:

$$M = \frac{V_{true}}{a} \quad (IV.18)$$

where a is the velocity of sound.

The velocity of sound does not vary with the pressure, but it does depend on the temperature

$$a = \sqrt{\frac{\gamma p}{\rho}} \quad (IV.19)$$

$$M = \frac{V_{true}}{a} = \frac{V_0 \sqrt{\frac{\rho_N}{\rho_0}}}{a_0 \sqrt{\frac{\gamma_N}{\gamma_0}}} = \frac{1}{a_0} \sqrt{\frac{2 \Delta p \gamma_0}{\rho_N}} \quad (IV.20)$$

If the value of the speed calculated by eq.(IV.14) is introduced into eq.(IV.20), then the equation for determining the Mach number takes the following form

$$M = \sqrt{\frac{2}{k-1} \left[\left(1 + \frac{\Delta p}{\rho_N} \right)^{\frac{k-1}{k}} - 1 \right]} \quad (IV.21)$$

The Mach number indicators are calibrated in Mach numbers from 0.3 to 0.95.

Figure 65 gives a simplified kinematic diagram of the instrument.

The deformations of the manometric box (4) and the angle of rotation α_1 of the spindle (1) are proportional to the quantity Δp

$$\alpha_1 = k_1 \Delta p$$

where k_1 is the coefficient of proportionality.

The angle of rotation α_2 of the spindle (2) depends not only on the quantity Δp but also on the deformation of the aneroid boxes (5). The angle α_2 is larger at a larger angle α_1 and a shorter arm h , whose length depends on the deformation of the aneroid box (5) under the action of the atmospheric pressure p_H at the given altitude

$$\alpha_2 = k_2 \frac{\alpha_1}{h}$$

Since $h = k_3 p_H$, it follows that

$$\alpha_2 = k_1 k_2 k_3 \frac{\Delta p}{p_H}$$

where k_2, k_3 are proportionality factors.

The angle α_3 of rotation of the pointer (3) is equal to

$$\alpha_3 = k \alpha_2 = k_1 k_2 k_3 k_4 \frac{\Delta p}{p_H} = k \frac{\Delta p}{p_H}$$

Thus the angle of rotation of the pointer is determined by the relation

$$\alpha_3 = k \frac{\Delta p}{p_H}$$

Equation (IV.21) may be represented in the form:

$$\frac{\Delta p}{p_H} = \left(1 + \frac{M^2}{5}\right)^{3.5} - 1$$

Consequently,

$$\alpha_3 = k \left[\left(1 + \frac{M^2}{5}\right)^{3.5} - 1 \right]. \quad (\text{IV.22})$$

However, this means that to each Mach number there corresponds a single definite angle of deflection of the instrument pointer, i.e., the scale of the instrument may be calibrated in equal numbers.

An exact determination of the speed of the aircraft, particularly the ground speed, is exceedingly important.

The lack of any ground speed indicator, and the errors inherent in the airspeed indicator considerably complicate the work of the crew. For this reason, designers and scientists are placing emphasis on the question of creating simple and reliable airspeed indicators and ground speed indicators.

Climb Indicators

The rate of change of altitude, i.e., the vertical speed of the aircraft V_y , is determined by an instrument known as a climb indicator.

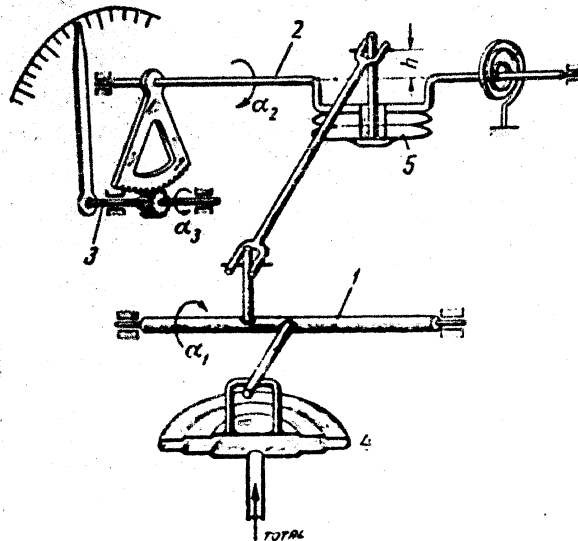


Fig.65 - Kinematic Diagram of Mach Number Indicator

- 1 - Axle whose angle of rotation depends on the flying speed;
- 2 - Axle whose angle of rotation is proportional to the flying speed and altitude;
- 3 - Spindle of pointer;
- 4 - Manometric box;
- 5 - Aneroid boxes

The operation of a climb indicator is based on the method of measuring, by means of a manometer, the pressure difference $\Delta p = p_c - p_H$, where p_c is the pressure inside a certain closed space, communicating with the atmosphere through a capillary (Fig.66), while p_H is the atmospheric pressure.

On the ground, and during prolonged horizontal flight, the pressure in the manometric box (2) and in the body (13) is the same, and the pointer (8) of the instru-

ment stands at zero. Every change in the vertical speed of flight means a change in altitude and, consequently, also in the pressure within the corrugated box. Instruments are in existence in which the body is connected with the atmosphere directly, and the corrugated box through a capillary. In this case an additional space

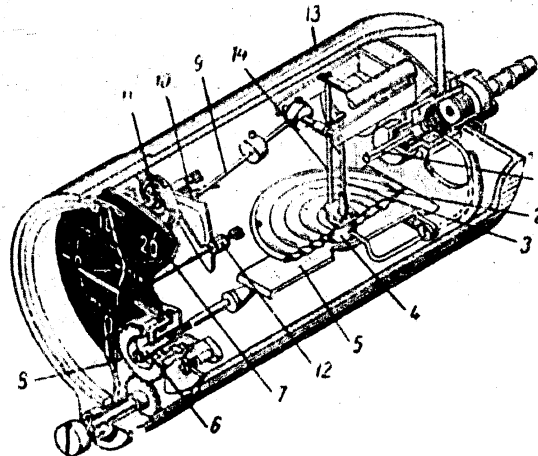


Fig. 66 - Diagram of the Climb Indicator

- 1 - Capillary; 2 - Manometric box; 3 - Tie rod; 4 - Base;
 5 - Spring plate; 6 - Rack and pinion; 7 - Scale; 8 - Glass;
 9 - Carrier; 10 - Sector; 11 - Hair; 12 - Gear; 13 - Casing;
 14 - Support

(a small tank), connected with the inner cavity of the corrugated box must be added to the design of the climb indicator. There is no basic difference in the operation of instruments of these two types.

When the aircraft climbs, the air emerges from the body of the instrument through the capillary (1); when the aircraft descends the air enters the body of the instrument through the same capillary. In both cases the pressure p_c of the instrument cannot become equal to the atmospheric pressure, and the pressure difference $\Delta p = p_c - p_H$ acts on the corrugated box (2). The deformation of the corru-

gated box is proportional to this pressure difference Δp . The rate of equalization of the pressure acting on the corrugated box depends on the capacity of the capillary device to transmit air.

At a steady vertical rate of climb or descent, when the altitude of the aircraft varies according to the law $H = V_y t$, where t is the time of climb or descent, the pressure difference Δp acting on the box is determined by the relation

$$\Delta p = \frac{V_y}{Rk} \quad (\text{IV.23})$$

$$k = \frac{\pi D^4}{128nlv}$$

where D is the diameter of the capillary in m;

l the length of the capillary in m;

v the volume of the instrument case in m^3 ;

V_y is the rate of climb or descent in m/sec;

$$n = \frac{\eta}{T}$$

η is the coefficient of viscosity of the air, kg-sec/ m^2 .

The viscosity of the air may, with a sufficient degree of accuracy, be considered a linear function of the ambient temperature, i.e., we may take

$$\eta = nT \quad (\text{IV.24})$$

where $n = 0.62152 \times 10^{-8}$ kg-sec/ m^2 deg.

The pressure drop acting on the sensitive element of the climb indicator is very small. At rates of climb from 30 to 75 m/sec, the pressure drop is equal to 10 - 75 mm of water column.

The indicator part of the instrument consists of a very sensitive manometer calibrated in units of vertical speed from 0 to ± 30 m/sec (or from 0 to ± 75 m/sec).

The special rack (6) is provided for setting the instrument at zero and for

changing the initial position of the sensitive element. In diving and steep climbing, the vertical speed considerably exceeds the calculated speed. In order to avoid irreversible deformation of the box, the supports (14), which limit the displacement of the box, are provided.

One of the most important parts of the instrument is the capillary device (1). In climb indicators designed to measure low velocities, this device is in the form of a glass tube 0.45 mm in diameter and 50 mm in length.

If the instrument is designed for high vertical speeds, such a capillary is unsuitable. In this case a battery of similar capillaries is used. In an instrument with a battery of capillaries, the climb indicator equation remains the same, but the so-called capillary characteristic k is equal to

$$k = \frac{m\pi D^4}{128n\pi v} \quad (\text{IV.25})$$

where m is the number of capillaries in the battery.

Errors of the Climb Indicator

Systematic Errors. The principal systematic errors of climb indicators are as follows:

1. Lag;
2. Temperature errors;
3. Error due to change in the volume of the manometric box.

Lag. During a change in the vertical speed of the aircraft, the instrument readings lag behind the variations of speed. The lag in the instrument readings remains marked until the values of the error become less than the threshold of sensitivity of the instrument.

The magnitude of the lag is determined by the time necessary for the reading of the instrument to become the true reading. In modern instruments, the time of lag is 4 - 6 sec. To reduce the time lag, the diameter of the capillary must be

made as large as possible. Increasing the diameter limits the sensitivity of the instrument, since a considerable increase of the diameter of the capillary requires the use of corrugated boxes of excessive sensitivity.

Temperature Errors. A variation in altitude is accompanied by a considerable variation in the temperature of the outside air T_H and in this case the temperature T_C in the instrument case also varies, but much more slowly.

As a result of this, the process of escape of the air through the capillary is nonisothermic and an error is created in the instrument. If the temperature at opposite ends of the capillary is T_C and T_H , respectively, then the equation of the climb indicator takes the form

$$\Delta p = \frac{V_y}{Rk} \frac{T_H}{T_C} \quad (\text{IV.26})$$

The error due to the nonisothermic nature of the process may be found by the equation

$$\Delta: = \frac{V_y}{Rk} \left[\frac{T_H}{T_C} - 1 \right] \quad (\text{IV.27})$$

The variation in the temperature of the air in the instrument case leads to an additional change of pressure there δp_c , that is, it introduces an error in the instrument reading.

The change of the air temperature in the casing and in the atmosphere is reflected in the temperature of the capillary, and consequently also in the viscosity of the air passing through the capillary, which varies according to the law

$$\eta = \alpha T_{\text{cap}}$$

where T_{cap} is the temperature of the air in the capillary.

These changes in temperature and pressure must be taken into account in the equations of the climb indicator.

If the temperature of the air at the ends of the capillary is denoted, in the

general case by T_1 and T_2 respectively, the temperature of the air at a height of H by T_H , and the temperature in the capillary by T_{cap} , then the equation of the climb indicator takes the form

$$\Delta p = \frac{V_y}{Rk} \frac{T_1}{T_2} \frac{T_{cap}}{T_H} \quad (IV.28)$$

If the capillary device is placed deep within the case of the instrument and a brass cylinder, in which the air is warmed by the temperature in the case of the instrument, is installed in its path, then the influence of the nonisothermic nature of the process may be eliminated even when the aircraft is descending and atmospheric air is rushing into the instrument; in this case T_1 , T_2 , T_{cap} all become equal to T_c , and the equation of the climb indicator can be written in the form

$$\Delta p = \frac{V_y}{Rk} \frac{T_c}{T_H} \quad (IV.29)$$

i.e., the error due to the influence of the ratio $\frac{T_c}{T_H}$ still remains in the readings of the instruments. This error can be eliminated by changing the transmission ratio of the instrument in accordance with the temperature ratio $\frac{T_c}{T_H}$. The temperature compensation may be made in the form of a bimetal plate, whose stroke varies with the temperature T_c in the instrument case, and which carries the plate of an aneroid box, whose stroke varies with the pressure T_c , this pressure in turn depending on the temperature T_H at the altitude of flight.

It must be remembered that the use of an aneroid box with a linear characteristic relative to altitude, as a device for measuring the atmospheric temperature is admissible only on condition of a standard distribution of temperatures by height. In this case it is assumed that the pressure in the instrument case is a unique function of the temperature of the outer air.

Error due to Change of Volume of the Manometric Box. When the vertical speed of flight varies, the stroke of the box also varies and thus, the volume of the instrument case. This leads to a change in the characteristic k of the capillary.

The error thus caused does not exceed 0.02% of the instrument readings.

The instrument errors of the climb indicator are the same as those of other manometric instruments.

The total error of the reading in a climb indicator calibrated to 30 m/sec, is about 1.5 m/sec.

Section 2.

AIRCRAFT COMPASSES

Instruments for determining the course of the aircraft are called compasses. These include compasses based on the utilization of various principles: magnetic, gyroscopic, induction, astronomic, radio compasses, etc.

The simplest and, until today the most widely used types, are the magnetic compasses.

Magnetic Compasses

The operation of the magnetic compass is based on the interaction of the earth's magnetic field with a magnet in the instrument. The magnetic pole of the earth is located near the geographic poles and is somewhat displaced with respect to them. It is considered by convention that the southern magnetism is concentrated in the northern hemisphere and the northern magnetism in the southern hemisphere. The magnetic needle comes to rest with its north end pointing north, i.e., its north magnetic pole which conventionally possesses southern magnetism, is considered to be located near the north geographic pole, and the south magnetic pole near the south geographic pole. The angle between the geographic and magnetic meridian is called the declination.

The declination varies with the geographic position, the season, etc. It is considered positive if the northern end of the needle is deflected to the east of the geographic meridian. The value of the declination is determined from special charts of magnetic declinations on which places with the same magnetic declination

are connected by lines called isogons; on these same charts, the magnetic anomalies are plotted. The magnetic field in almost all parts of the earth, except the magnetic equator, is inclined to the horizon, and for this reason the total force T of terrestrial magnetism, directed at an angle to the horizon, may be resolved into two components: a vertical component $Y = T \sin \theta$, and a horizontal component $H = T \cos \theta$ (Fig.67). The horizontal component H which establishes the magnetic needle in a direction north-south is the directing force of the magnetic

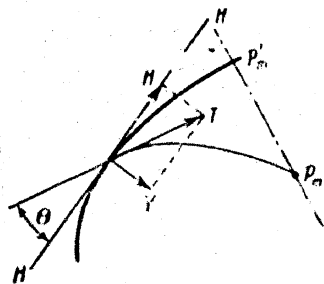


Fig.67 - Components of the Earth's Magnetic Field

HH - Plane of the horizon; P_m - Magnetic pole; P'_m - Projection of the magnetic pole onto the earth's magnetic surface; T - Total force of terrestrial magnetism; H - Horizontal component; Y - Vertical component; θ - Dip

compass. The angle θ between the total force of terrestrial magnetism and its horizontal component is called the inclination. A compass needle, freely rotating about a horizontal axis, takes a vertical position at the magnetic pole and a horizontal position at the equator. A magnetic needle set up from a point, when placed in the northern hemisphere, under the action of the force developed as a result of the interaction between the needle and the magnetic poles of the earth, tends to dip with its northern end downward. To eliminate this inclination, the southern end of the needle is made heavier. In the southern hemisphere, the northern end of the compass needle is made heavier.

The magnetic compass (Fig.68) makes it possible to determine the magnetic course of the aircraft. The principal part of the compass is the card with the magnet (1), resting on the steel point (3) on an agate sapphire bearing (4), attached to the column (5).

The presence of magnets compels the card to align itself along the magnetic lines of force of the earth, i.e., the card maintains an invariant direction with respect to the earth.

The card is installed in the bowl (6), filled with liquid, which reduces the weight of the card and helps to damp its oscillations.

The compass card consists of the limb (7) which is a thin brass or aluminum disk (in compasses with a horizontal scale) or a cone (in compasses with a vertical scale). In some compasses, divisions from 0 to 360° are marked on the disk. In this case, the instrument is read off by the aid of a course marker rigidly attached to the bowl (Fig.68a). The limb is connected with a hollow float (8), on which the magnet (1) and the damper (9), bearing numerals, are attached. In this case, the compass readings are read off from a scale rigidly attached to the bowl (Fig.68b). The liquid in the bowl must have the lowest possible viscosity to prevent excessive resistance to the motion of the card. The freezing point of the liquid must be below - 60°C, and the boiling point above + 50°C, while the density of the liquid must not vary substantially with the temperature. Liquids that completely satisfy these requirements have not yet been found; modern compasses use ligroin, whose density varies considerably with the ambient temperature; this leads to a change in the volume of the liquid and to the formation of air bubbles at low temperatures, or to an increase in pressure at high temperatures.

To eliminate these drawbacks, the instrument is provided with a compensating compression or overflow basin (10). At present, an overflow chamber consisting of an additional space is used into which air bubbles at lower temperatures and excess liquid at higher temperatures.

The compensating chamber somewhat increases the volume of the compass and thus ensures a wider range of compensation.

According to the purpose of the compass (for the navigator or for the pilot) the accuracy required for its readings differs. The navigator's compass, which is

also called the master compass, must be more accurate. The pilot's compass is called the course compass. In such compasses the readings are made by the aid of a course marker which is attached in a fixed position with respect to the compass bowl, and a movable card which takes the direction north-south (Fig.69).

There are many designs of magnetic compasses, but the layout is the same in all, with only the position and form of the scale (vertical and horizontal scales) or the system of reading differing.

The operation of all compasses follows the same general pattern: If the magnetic needle is removed from its equilibrium position and then released, it will return to its original position. The motion of the needle takes place under the influence of the following factors:

1. The turning moment M_{turn} developed by the magnetic card; the magnitude of this moment may be found from the equation

$$M_{\text{turn}} = MH \sin \varphi = 2ml H \sin \varphi$$

where M is the magnetic moment; $M = 2ml$;

m is the magnetic mass concentrated at one pole;

$2l$ is the length of the magnet;

φ is the angle between the direction of the magnetic lines of force and the axis of the magnet (a variable quantity);

H is the strength of the earth's magnetic field.

At small angles of declination of the needle, we may take $\sin \varphi = \varphi$; in that case,

$$M_{\text{turn}} = MH \varphi = 2ml H \varphi$$

2. The moment of resistance M_c of the liquid; at small angles φ , the magnitude of this moment may be found from the equation

$$M_c = K \varphi$$

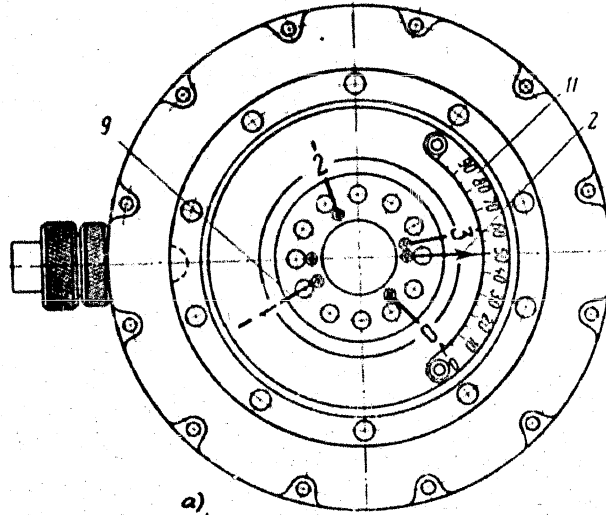
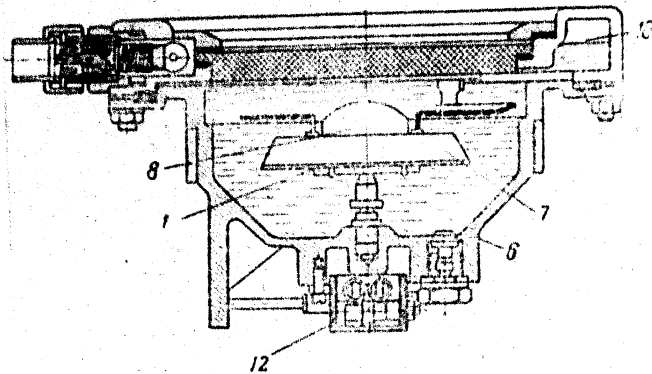


Fig.68 - Layout of the Parts of the Magnetic Compass

a - Diagram of compass; b - Compass card;

- 1 - Magnet; 2 - Pointer; 3 - Pivot; 4 - Bearing; 5 - Column; 6 - Bowl; 7 - Limb;
8 - Float; 9 - Damper; 10 - Overflow basin; 11 - Scale; 12 - Deviation of the parts

where K is the moment of resistance of the liquid per unit angular velocity of the card.

3. The moment of friction of the pivot on the bearing; this quantity is so insignificant that it is usually neglected.

4. The moment of the forces of inertia

$$M_I = J\ddot{\varphi}$$

where J is the moment of inertia of the card;

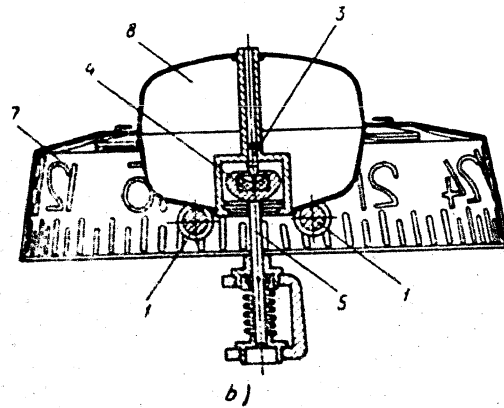
$\ddot{\varphi}$ is the acceleration of the card (a variable quantity).

The equation of the natural vibrations of the moving system of the compass has

the form

$$J\ddot{\varphi} + K\dot{\varphi} + M_I\varphi = 0 \quad (IV.30)$$

A solution of eq.(IV.30) shows that, according to the ratio of K to M_I , the



(Continuation of Fig.68)



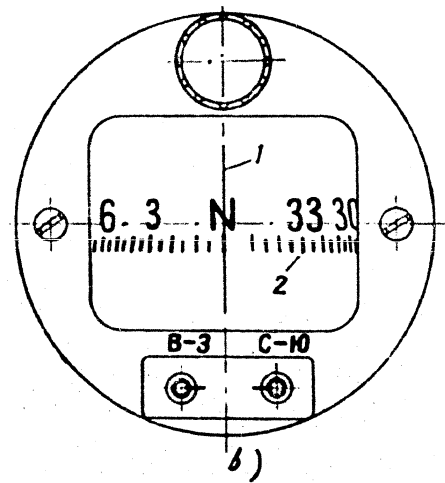
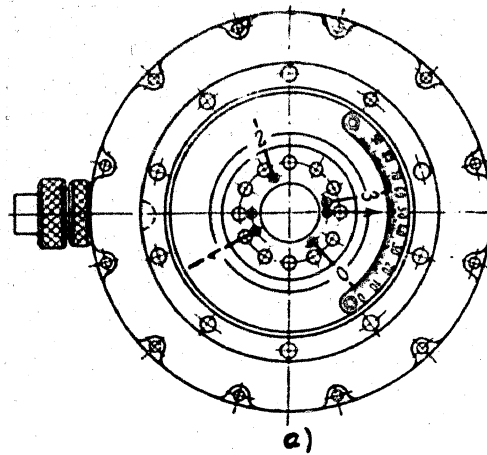


Fig.69 - External View of Navigator's Compass and
Pilot's Compass

a - Navigator's compass; b - Pilot's compass

1 - Course marker; 2 - Scale

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compass card, in returning to the position of equilibrium (toward the meridian), may execute oscillatory harmonic or aperiodic motions (Fig.70).

In modern aircraft compasses, a harmonic law of motion is assumed for the needle, and for this reason the compasses themselves are sometimes called periodic compasses.

Characteristics of Compasses. The operation of a periodic compass is characterized by a period of oscillation and by a damping decrement d ; the latter represents

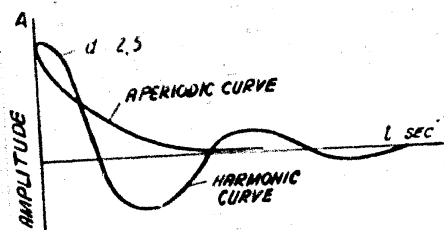


Fig.70 - Character of Motions of
Compass Card on Return to Equilibrium Position

the ratio of the amplitudes of two successive oscillations. The value $\ln d = D$ is called the logarithmic decrement of damping; the value of the damping decrement in modern compasses varies from 3 to 6.

In addition to these characteristics we must also know the following data:

- a. The damping time, i.e., the time necessary for the return of a card after deviation, to the meridian. In periodic compasses, the damping time varies from 15 to 30 sec.
- b. The entrainment, i.e., the angle through which the liquid entrains the card when the bowl is rotated together with the aircraft through 360° . The entrainment depends on the shape of the card, the viscosity of the liquid, the rate of rotation, etc. It is desirable to have the entrainment at a minimum value, since at high entrainment, even an minor change in the course leads to errors in the instrument readings. In modern compasses, for a 360° rotation at a rate of 0.1 revolution per second, the entrainment reaches $10-12^\circ$ (at normal temperature).
- c. The lag which depends on the friction in the bearing and is characterized by the angle at which a deviation of the card does not pass to the equilibrium positions. Under conditions of flight with vibration, the lag does not exceed 1° .

Errors of the Compass

The systematic errors of the compass comprise the deviation, as well as the banking and turning errors.

Deviation of the Compass. Besides the magnetic field of the earth, the magnetic field of the aircraft also acts on a compass installed in an aircraft. The earth's magnetic field produces the so-called directional force, i.e., the force holding the magnetic system to the direction of the magnetic meridian. The magnetic field of the aircraft, caused by various metallic aircraft parts, may be considered as consisting of two fields: a permanent field and a variable field.

The permanent magnetic field is produced by metals having a high coercive force, which are magnetically hard (these are conventionally called hard iron). The magnitude and direction of this field with respect to the aircraft does not vary when its course varies and, therefore, is called permanent.

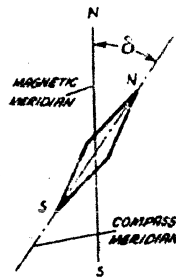


Fig.71 - Deviations of the Magnetic Needle

The variable magnetic field is produced by metals with a low coercive force which are rapidly magnetized and remagnetized; i.e., metals that are magnetically soft, and are conventionally called soft iron.

When the course of the aircraft changes, both the direction and strength of this field change with respect to the aircraft, for which reason it is called a variable field.

The aircraft magnetic field produces forces deflecting the magnetic system from the direction of the magnetic meridian. The angle δ between the direction of the magnetic meridian and the direction in which the magnetic system points under the influence of the permanent and variable magnetic fields of the aircraft is

called the deviation (Fig.71). The deviation is considered positive if the northern end of the magnet deviates toward the east.

The deviation of an aircraft is a variable quantity since the relative position of the restoring forces, i.e., the forces holding the needle in the position of the

magnetic meridian, and the forces causing the deviations, change with any change of the aircraft course.

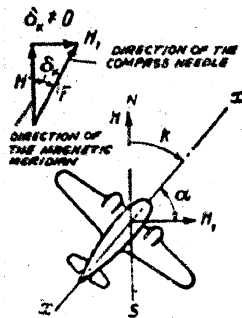


Fig.72 - Action of the Components of the Permanent Field of the Aircraft on the Magnetic Needle

H - Horizontal component of the earth's magnetic pole; H_1 - Horizontal component of the aircraft magnetic pole; F - Resultants H and H_1 ; δ_k - Deviation; k - Course; α - Angle between the vector H_1 and the axis of the airplane

If the horizontal component of the permanent magnetic field of the aircraft is H_1 , then the direction of the vector representing this force makes a certain invariable angle α with the axis of the aircraft, which angle is independent of the aircraft course (Fig.72). At the same time the direction of this vector with respect to the vector H, representing the horizontal component of the earth's magnetic field, does vary with the course of the aircraft and causes a variation of the magnitude and direction of the course, producing the deviation H_1^i (Fig.73).

If the aircraft is set on a course k on which the vectors H and H_1 coincide (Fig.73), then the deviation will be equal to zero, since this produces a deviation of the force $H_1^i = 0$. If the aircraft continues to turn clockwise, then the combined forces H and H_1 produce the resultant F (cf. Fig.72). The magnetic needle of the compass points in the direction of this resultant, i.e., it indicates the compass meridian. The angle between the compass meridian and the longitudinal axis of the aircraft is called the compass

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course, while the angle between the magnetic and compass meridians, i.e., the difference between the magnetic and compass courses, is called the deviation.

In this case the force H_1 , producing the deviation, varies in magnitude and direction with respect to the aircraft. On resolving the force H_1 in the direction of the aircraft axes xx and zz , we obtain the components H_{1x} and H_{1z} (Fig.74) whose directions are constant with respect to the aircraft, while their magnitude varies by the law

$$H_{1z} = H_1 \sin k$$

$$H_{1x} = H_1 \cos k$$

The magnitude of the deviation due to these forces is determined by the

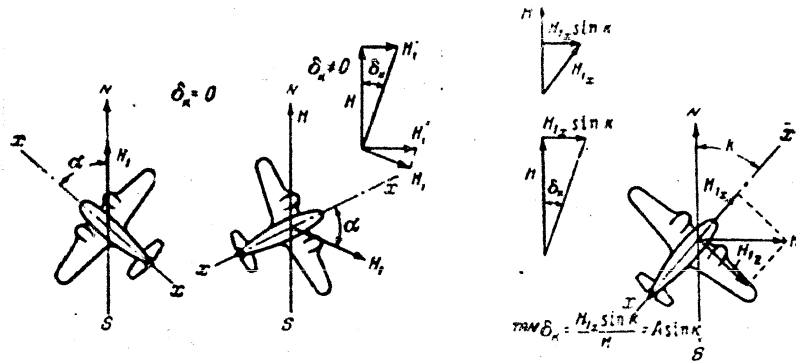


Fig.73 - Position of Vectors H and H_1 for Different Courses of the Aircraft

Fig.74 - Forces Producing the Semi-Circular Deviation $\bar{H}_1 = \bar{H}_{1x} + \bar{H}_{1z}$

equations

$$\Delta_1 = B \sin k$$

$$\Delta_2 = C \cos k$$

where k is the course of the aircraft;

B and C are the coefficients of deviation due to hard iron.

The forces H_{1X} and H_{1Z} and the deviation produced during two 360° rotations of the aircraft lead to zero and change their sign twice. For this reason, the deviations arising under the influence of the permanent magnetic field of the aircraft are called semicircular deviations (Fig.74).

The deviations of the aircraft compass due to the permanent hard iron may reach very large values (of the order of 10° and more).

In the design of all compasses, without exception, a special device, known as a deviation compensator, is incorporated, and fully or partially eliminates the deviation due to hard iron.

The variable magnetic field generated by the action of the terrestrial magnetism on soft iron (the instruments of hard iron and soft iron may be neglected since as a result of it, forces are produced which have the same direction as the forces from hard iron causing the semicircular deviation) gives the constant deviation A and the quadrant deviations D and E .

The quadrant deviations vary according to the law $D \sin 2k$ (Fig.75) and $E \cos 2k$ where k is the course.

Since there is little soft iron on an aircraft, the deviations due to it are usually small; therefore, no special devices to eliminate these deviations are incorporated in the compass. The deviation of a compass on an aircraft is determined by successively setting the aircraft to eight principal magnetic courses (0° , 45° , 90° , 135° , 180° , 225° , 270° , and 315°), simultaneously reading off the compass courses. The difference between the magnetic and compass course is the deviation for a given magnetic course. The curve of deviation is expressed mathematically by the formula

$$\Delta k = A + B \sin k + C \cos k + D \sin 2k + E \cos 2k \quad (IV.31)$$

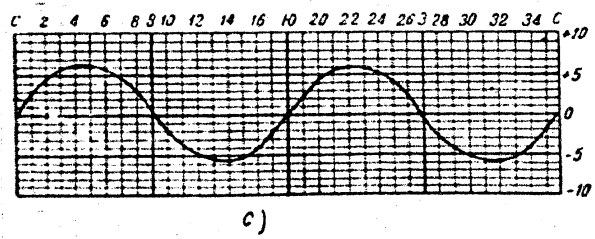
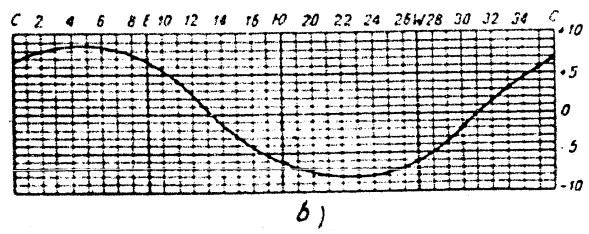
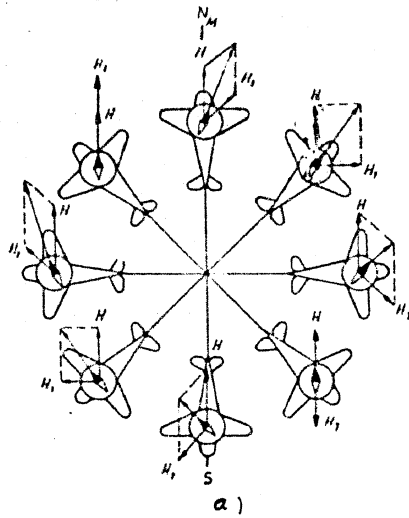


Fig.75 - Compass Deviations

a - Position of aircraft on various courses; b - Semi-circular deviations; c - Quadrant deviations ($D \sin 2k$)



where Δ_k is the deviation;

A is a coefficient characterizing the permanent deviation due to magnetism induced in soft iron, and the error of setting;

B,C are coefficients characterizing the semicircular deviation due to hard iron on the aircraft;

D,E are coefficients characterizing the quadrant deviations due to magnetism induced in soft iron.

In cases where the deviation of the compass exceeds 10° , it must be eliminated or reduced. The maximum value is reached by semicircular deviations on the aircraft, and for this reason special attention is usually focused on its elimination or reduction.

Elimination of Deviations. The permanent deviation, characterized by the

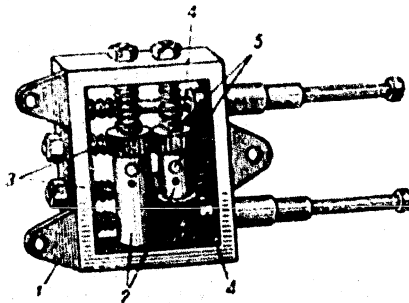


Fig. 76 - Deviation Compensator
 1 - Body; 2 - Transverse spindles;
 3 and 4 - Longitudinal spindles;
 5 - Compensating magnets

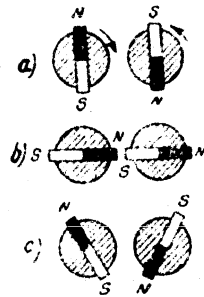


Fig. 77 - Principle of Operation of the Deviation Compensator
 a - Minimum action of the magnetic field; b - Maximum action of the magnetic field; c - Mean position

coefficient A, is taken into account by rotating the compass through the proper



angle with respect to the vertical axis.

The semicircular deviations, characterized by the expressions $B \sin k$ and $C \cos k$, are eliminated by a so-called deviation compensator, consisting of two pairs of permanent magnets (Fig.76) which produce moments equal in magnitude but opposite in direction to the moments produced by the semicircular deviations.

One pair of magnets (longitudinal magnets) is arranged parallel to the longitudinal axis of the aircraft. This pair of magnets is designed to compensate the force H_{Lx} . Another pair of magnets (transverse magnets) is arranged parallel to the yz axis of the aircraft. These magnets are designed to compensate the force H_{Lx} .

By rotating the rollers (2) and (2) the position of the compensating magnets can be changed, i.e., the action of their magnetic fields (Fig.77) can be increased or decreased, and positions of the magnets that keep the deviation to a minimum can be selected. Such a deviation compensator compensates only the semicircular deviation.

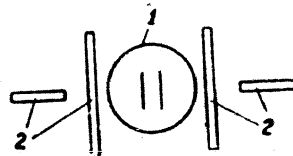


Fig.78 - Arrangement of Bars (2) of Soft Iron Around the Compass (1) to Eliminate the Quadrant Deviation

The quadrant deviation cannot be eliminated by permanent magnets. It may be compensated by means of strips of soft iron placed symmetrically at the sides, top, or bottom of the compass (Fig.78).

The quadrant deviation is usually not eliminated but allowance is made for it. If it is necessary to eliminate it, this can be done by means of soft iron bars.

After installing the compass on the aircraft, the semicircular deviation is

first removed after which the operation of the compass is periodically checked. The semicircular deviation is eliminated by the following procedure: The aircraft, while on the ground, is set to the course $k = 0$; on this course the deviation

is $\Delta_k = A + C + E$ so that the terms containing $\sin k$ and $\sin 2k$ vanish. The total force acts in the direction of the right wing of the aircraft. To eliminate the deviation along the course $k = 0$ by the permanent magnets of the deviation compensators, the force C_1' is produced, equal in magnitude to the force causing the deviation on this course but opposite in direction, i.e., in the direction of the left wing of the aircraft (Fig.79). In this case, not only the semicircular deviation:

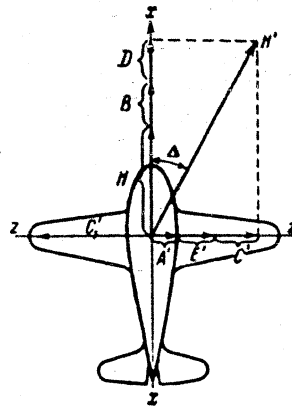


Fig.79 - Elimination of Deviation on a Northern Course

$$C_1' = A' + E' + C'$$

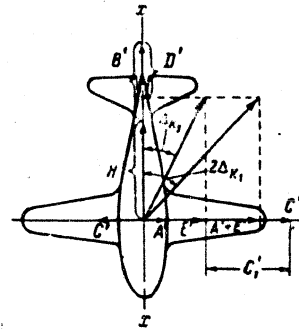


Fig.80 - Elimination of Deviation on a Southern Course

$$C_1' - C' = A' + E'$$

but also the constant deviation A and the quadrant deviation E on the course $k = 0$ is eliminated; but for that, the deviation is increased on the course $k = 180^\circ$.

When the aircraft is rotated through 180° , the deviation Δ_k , defined by the equation $\Delta_k = A - C + E$, is produced by the forces A' and E' acting in the direction of the left wing, and by the force C', acting in the direction of the right wing. The force $C_1' = A' + E' + C'$, produced by the deviation compensator, acts in the direction of the left wing (Fig.80). The forces C' and -C' mutually cancel each other; consequently, the force $2(A' + E')$ now acts toward the left and produces, the devi-

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ation $2\Delta_k = 2(A + E)$. The deviation compensator on the course $k = 180^\circ$ brings the deviation to the value $\Delta_k = A + E$; in this case, the deviation on the course 0 is likewise equal to $\Delta_k = A + E$.

If now the aircraft is again set to the course $k = 0$, the deviation will be equal to $\Delta_k = A + E$ and, in character, will be a quadrant and permanent deviation. With this order of work on the course $k = 0$, as well as on all other courses, only the quadrant constant deviation should remain.

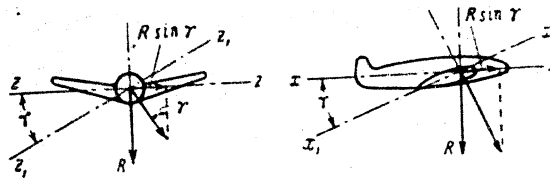


Fig.81 - Conditions Causing Banking Deviation.

To cancel the deviation $B \cos k$, the same operation is performed on the courses 90 and 270°. After the semicircular deviation has been canceled, the residual deviation is determined on eight principal courses 0°, 45°, 90°, 135°, 180°, 225°, 270°, and 315°, and eight equations of deviations $\Delta_0, \Delta_{45}, \Delta_{90}, \Delta_{135}, \Delta_{180}, \Delta_{225}, \Delta_{270}, \Delta_{315}$ are set up; in this equation, we find the value for which corrections have already been introduced into the instrument readings along any course.

During flight, the deviations may vary, for example, when the relative position of the equipment changes, when the bomb load is released, etc. This is one of the reasons why magnetic compasses are made remote-reading.

Banking Deviation. As already stated, the horizontal component H_1 of terrestrial magnetism gives a semicircular deviation, while the vertical coefficient Z in rectilinear horizontal flight does not cause a deviation, since it acts in a vertical plane and its projection on the plane of the coil is equal to zero. At

longitudinal and transverse banking, with the aircraft in rectilinear flight, the vertical component is inclined together with the aircraft, while the position of the compass card remains unchanged with respect to the plane of the horizon, i.e., the force R gives the projection R_1 onto the plane of the card. The value of this projection is determined by the equation

$$R_1 = R \sin \gamma$$

where γ is the angle of pitch or bank. The force R , during a longitudinal bank, is directed along the xx axis and, during a transverse bank, along the zz axis (Fig.81). From its character, the banking deviation will be semicircular just like

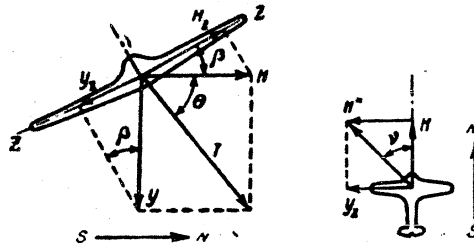


Fig.82 - Turning Error on the Course $k = 90^\circ$

β - Bank of compass card; θ - Inclination; T - Direction of magnetic pole; Y_2 - Projection of the vertical component to the direction of the axis zz ; H_2 - Projection of the horizontal component to the direction zz ; zz - Transverse axis of the aircraft; ψ - Turning error

that produced by the forces H_{1x} and H_{1z} . In most cases, the banking deviation is not canceled, despite the fact that it sometimes reaches considerable values. The banking deviation may be canceled by permanent magnets arranged in a vertical plane.



Turning Error. If, during revolutions of an aircraft, for example in the case of a turn, not only the aircraft but also the compass card is banked with respect to the plane of the horizon, then the vertical component of terrestrial magnetism will no longer be perpendicular to the plane of the card and will give instead the projection Y_z acting in the direction zz . This direction depends on the angle of bank β of the compass card* (Fig. 82). The compass card is then deflected from the direction of the magnetic meridian. The value of the error will depend on the course of the airplane. On the course 0° , the projection Y_z is directed along the transverse axis of the aircraft. The horizontal component of terrestrial magnetism is projected onto the direction of the xx axis. From the geometric composition of the forces Y_z and H , we find the resultant H' . The angle between this resultant and the magnetic meridian will give the value of the turning error ν_z

$$\operatorname{tg} \nu = \frac{Y_z}{H} = \frac{Y \sin \beta}{H} \quad (\text{IV.32})$$

On the course 180° , the picture is analogous. On the courses 90° and 270° , the projection of the horizontal component of terrestrial magnetism will act in the direction zz .

In this case,

$$H' = H \cos \beta$$

The force $Y_z = Y \sin \beta$ is likewise projected onto the axis zz . From the geometric composition of these two forces, we find the value of the directing force of the compass. On the course 90° it is equal to $H' = Y \sin \beta - H \cos \beta$ and is directed along the zz axis of the aircraft, coinciding with the magnetic meridian.

* The angle β of bank of the card does not always coincide with the angle γ of bank of the aircraft.

On replacing Z and H by T , i.e., by the total force of terrestrial magnetism, we may write

$$\begin{aligned} Z \sin \beta &= T \sin \theta \sin \beta; \\ H \cos \beta &= T \cos \theta \cos \beta; \\ H'' &= T[\sin \theta \sin \beta - \cos \theta \cos \beta]. \end{aligned}$$

For $\beta = 90^\circ - \theta$, the expression vanishes and $H'' = 0$, i.e., in this case there will be no directing force in the compass. If $Z \sin \beta > H \cos \beta$, then the compass needle will point south. The angle $\beta = 90^\circ - \theta$ is called the critical angle of bank.

At angles of bank less than the critical, on the courses 90 and 270° , the magnetic system points north and, although in this case the directing force H'' is decreased, there will be no error in the instrument readings.

At angles of bank above the critical value, the magnetic system tends to rotate through 180° ; in this case there will be an error in the instrument readings. The angles between the magnetic and compass courses is called the turning error.

The error bears this designation because it appears during turns of the aircraft.

The analytic expression of the turning error has the form

$$\nu = k_m - k_k$$

If, in this expression, we replace k_k by known values of the total force of terrestrial magnetism T , the inclination θ , the bank β and the magnetic course k_m , we obtain the expression

$$\nu = k_m - k_k = k_m - \arctg \left[\frac{\sin \theta \cos \beta}{\cos k_m} \right]. \quad (\text{IV.33})$$

The turning error on an arbitrary course of the aircraft is shown in Fig. 63.

The instrument errors of the compass are caused by lag, entrainment, inaccurate manufacture, and unbalance of the compass card, consisting of a pendulum. Under the

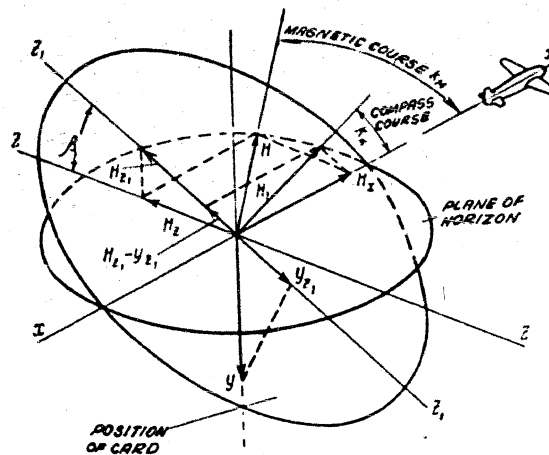


Fig.83 - Turning Error on an Arbitrary Course

action of periodic disturbances (for example in bumpy flight), the magnetic system will oscillate about the equilibrium position.

Remote-Reading Compasses

The presence of large ferromagnetic masses on airplanes, and the inconstancy of the magnitude and direction of the forces forming these masses (as a result of the change in the position of the control levers, turrets, etc.) leads not only to a variation in the deviation, but also to a dependence of the deviation on factors not amenable to estimation. The impossibility of eliminating these factors has led to the development of remote-reading compasses in which the pickup, ordinarily consisting of a magnetic compass, is installed at the point on the aircraft where the

influence of the ferromagnetic masses is smallest (for example, in the tail of the fuselage or in the wings of the aircraft). The compass readings through a remote transmission are communicated to an indicator placed on the instrument board.

Among the remote-reading instruments, the most widely used are the following: remote-reading gyromagnetic compass (RGMC), remote-reading potentiometric compass (RPC), remote-reading induction compass (RIC), etc. (cf. infra).

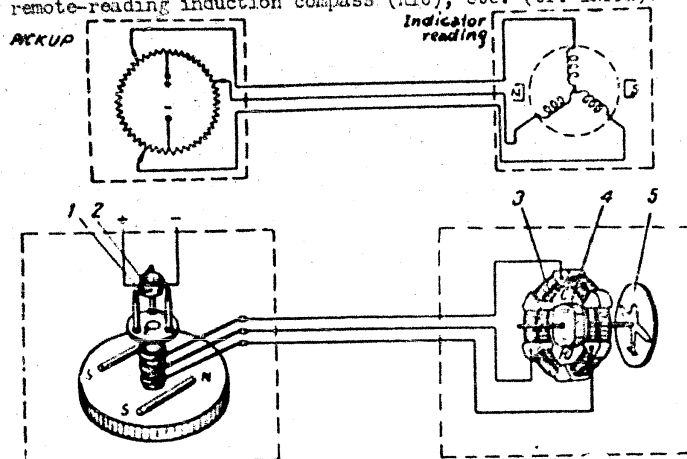


Fig.84 - Principal Diagram of Remote-Reading Potentiometric

Compass

- 1 - Brushes; 2 - Ring potentiometer; 3 - Stator; 4 - Permanent magnet-rotor; 5 - Pointer

Remote-Reading Potentiometric Compass. Figure 84 gives a diagram of the remote-reading potentiometric compass.

The pickup consists of a magnetic compass with a magnetic system which is considerably more powerful than that of the ordinary magnetic compass and consists of four magnets of a length of $l = 110$ mm and a diameter of $d = 10$ mm, with a magnetic moment of 120,000 units.

At a deviation of 90° , such a system develops a moment of 2 G/cm while the magnetic system of an ordinary compass develops only 0.03 G/cm. Such a great

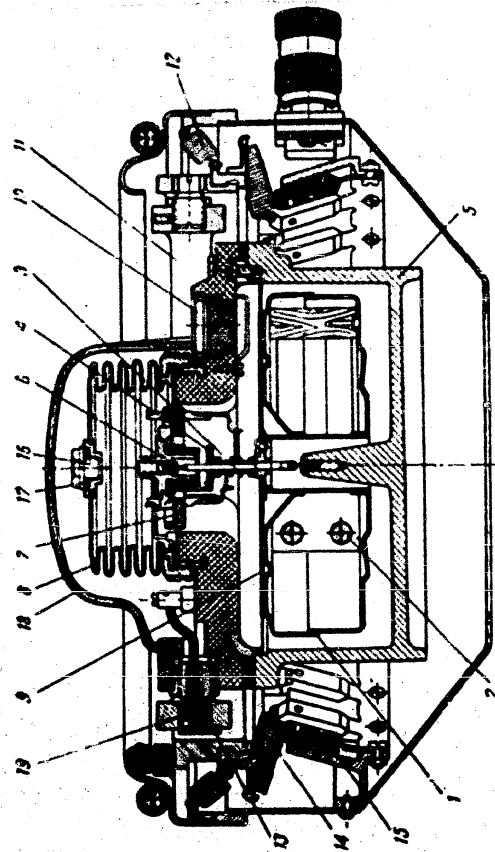


Fig.85 - Pickup of the Remote-Reading Potentiometric Compass

- 1 - Compass card; 2 - Permanent magnet; 3 - Collectors; 4 - Stage with brushes; 5 - Body;
- 6 - Ring potentiometer; 7 - Bridge; 8 - Sylphon; 9 - Scale; 10 - Inspection hole; 11 - Cardanic suspension; 12 - Damping; spring; 13 - Journal; 14 - Spring; 15 - Ring; 16 - Screw; 17 - Packing;
- 18 - Casing; 19 - Electric wiring

moment is necessary to overcome the friction of the brushes on the potentiometer. The three brushes (1), which are spaced at an angle of 120° are rigidly connected with the magnetic system of the compass. They consist of three sliding

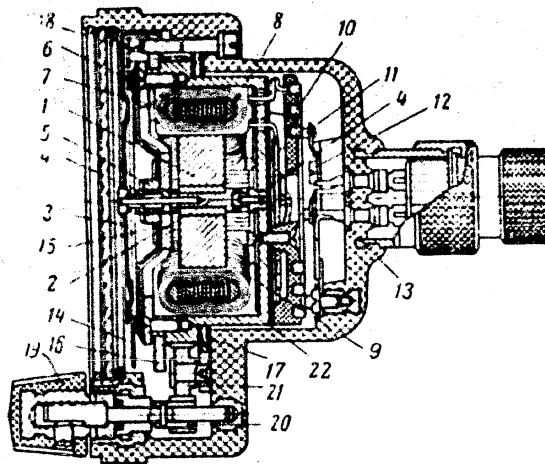


Fig.86 - Design of the Remote-Reading Potentiometric Compass Indicator

- 1 - Moving magnet; 2 - Spindle; 3 - Pointer; 4 - Bearing of spindle;
- 5 - Locknut for regulating longitudinal play of spindle; 6 - Permalloy core of stator; 7 and 8 - Brushes; 9 - Stator winding; 10 - Three contact rings connected with the stator coils; 11 - Brushes; 12 - Screws for attaching brushes; 13 - Plug contact, 14 - Scale; 15 - Plate; 16 - Ring in which the inner case rotates; 17 - Flat annular spring; 18 - Fixed sight; 19 - Rack; 20 and 21 - Gear drive; 22 - Inner case

current-collector contacts and slide along the potentiometer (2), attached to the compass case. The potentiometer is fed by the electric system of the aircraft. The magnetic system, together with the brushes, maintains a constant position in space. The potentiometer varies its position together with the aircraft during its evolu-

tion.

When the aircraft turns, the position of the potentiometer with respect to the magnetic meridian changes, and consequently it also changes with respect to the brushes. In this way, depending on the course of the aircraft, the potential difference between the brushes of the pickup is changed.

The voltage taken by the brushes from the potentiometer is supplied to the three frames (the windings of the stator 3) of an indicator consisting of a ratiometer with a moving magnet. A ratiometer with moving frames may also be used. The position of the moving system of an indicator depends only on the distribution of currents in the winding, i.e., on the position of the brushes on the potentiometer, with only one definite position of the moving system of the indicator corresponding to each position of the brushes of the pickup. From the scales of the pickup and indicator, graduated from 0 to 360° , the course of the aircraft with respect to the magnetic meridian is read off. The design of the magnetic pickup is given in Fig.85 and that of the indicator in Fig.86. To reduce the vibration, the instrument is carefully shock-absorbed. The weight of the pickup is 5 kg and that of the indicator about 1 kg.

The error of the instrument is of the same order as that of an ordinary magnetic compass; the lag is $\pm 1^{\circ}$, the entrainment at an angular velocity of 1 rpm does not exceed 12° ; the damping time for temperatures from $+ 50$ to $- 60^{\circ}\text{C}$ does not exceed 20 sec.

The error of the remote transmission of the compass runs up to $\pm 2^{\circ}$.

Section 3.

NAVIGATIONAL COORDINATORS AND AUTOMATIC NAVIGATORS

The automatic determination of the geographic coordinates of a moving object is possible in principle by utilizing the property of a gyroscope of keeping the position of its axis in space constant, regardless of the displacements of the

moving objects and the earth's rotation, and also by calculating the path from the variation in speed and course of the aircraft.

Gyro devices require exceptionally high accuracy, which is technically still difficult to achieve. This forces us for the time being to give preference to instruments based on the method of path calculation. In this case, the following equations are continuously and automatically solved:

$$\varphi = \frac{360}{2\pi R} \int_0^t (V \cos k + U \cos \varphi_B) dt;$$

$$\lambda = \frac{360}{2\pi R} \int_0^t \frac{V \sin k + U \sin \varphi_B}{\cos \varphi} dt.$$

where U is the windspeed;

φ_B is the wind angle, i.e., the angle between the meridian and the wind vector.

The principal parts of the automatic navigator (Fig.87) are as follows:

A - The ground speed pickup. Since such an instrument does not exist at the present time, the indicator part of a true-airspeed indicator is used instead. Some instruments have devices that apply corrections for the wind speed. When the pickup of the true wind speed is used, the readings of the automatic navigator may show an error whose value is smaller the higher the flying speed.

B - Compasses.

C - Coordinator.

The airplane aircraft course or, more accurately, the values of $\sin k$ and $\cos k$ are picked up by the compass. (this is most often a magnetic compass) together with the coordinators.

D - Multipliers, which multiply the values of the airspeed by $\sin k$, $\cos k$, $\frac{1}{\cos \varphi}$, and likewise the wind speed U by $\cos \varphi_B$ and $\sin \varphi_B$ (φ_B being the wind angle).



E - A summing device giving the sums:

$$V \sin k + U \sin \varphi_B$$

$$V \cos k + U \cos \varphi_B$$

F - Integrators making it possible to obtain the values of latitude and longitude.

G - Latitude and longitude indicators.

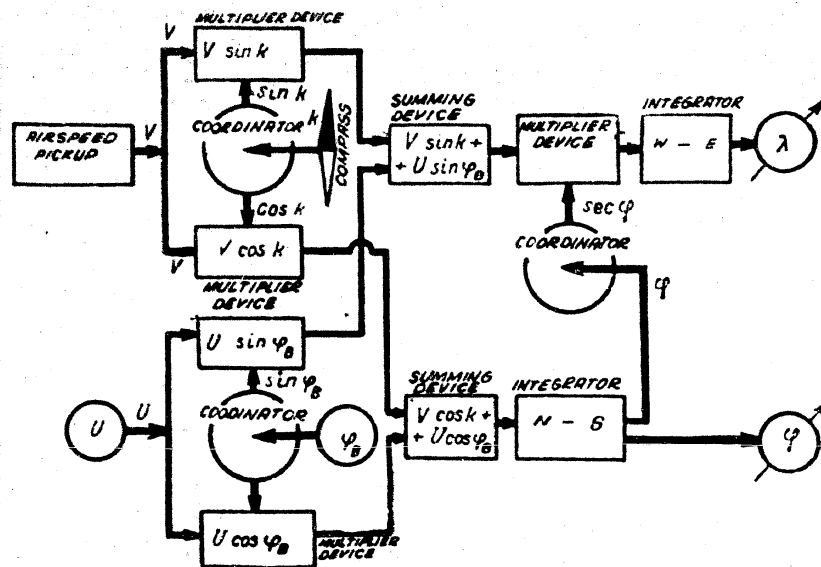


Fig.87 - Block Diagram of Automatic Navigator

Since, in practice, the aircraft must often fly in a direction laid out in advance, the position of the aircraft can be determined in an arbitrary system of coordinates (Fig.88).

Knowing the values of L and D and the points of take-off, the value of the geographic coordinates of the aircraft can be found.



The idea of the automatic navigator is best realized in an instrument that indicates the position of the aircraft directly on the map. Such automatic navigators are particularly convenient for short flights.

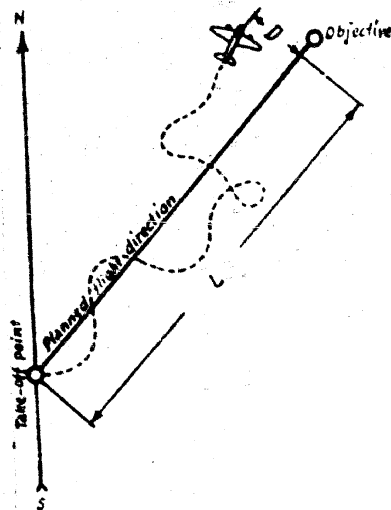


Fig.88 - Determination of the Position of the Aircraft in Arbitrary Coordinates

L - Distance between initial and final points of flight; D - Deviation from assigned route

As far back as 1932, the Soviet engineers D.A.Rozhanskiy and Yu.B.Kobzarev developed and worked out the principles on which the operation of modern radio location instruments are based.

The principles of the determination of the direction based on radio methods, were worked out by M.V.Shuleykin, A.N.Shchukin, M.Ye.Starik, and others.

Radio sondes of the P.A.Molchanov system are used for meteorological explora-

For long flights such an instrument requires the use of special maps on which, within the limits of each sheet, the scale is kept constant while the lines of the various courses (loxodromes) are rectangular.

Section 4.

RADIO INSTRUMENTS

Modern radio instruments for determining the altitude, the distance traveled and the location of the aircraft, as well as radar devices for determining the ground speed, angle of drift, and wind speed, are all based on the principle of radio telemetry, developed by the great Soviet scientists A.I.Mandel'shtam and N.D.Papaleksi, and by many Soviet engin-

tion of the air.

Radio instruments are being more and more widely used. But their introduction does not exclude the use of mechanical and electromechanical instruments, although the latter are often considerably less accurate. This is explained by the danger of interference in modern radio instruments, as well as by the impossibility of using the operating principles of radio instruments in the design of certain instruments (for example, of absolute or relative altimeters). The fact that the operation of aircraft radio instruments depends on the operation of a radio transmitter on the ground is a major disadvantage of such instruments.

Among radio navigation equipment, the following are most widely used: radio compasses, radio semi-compasses, receivers of radio beacons, radio altimeters, course, glide, and marker radio receivers, radio receivers of the hyperbolic and circular systems of radio navigation, radio telemeters, radio automatic navigators, etc.

Radio altimeters and radio compasses are very widely used in aviation.

Radio Altimeters

Aircraft radio altimeters solve one of the most complex problems of piloting and navigation, the problem of determining the true altitude of flight. Knowledge of the true altitude is necessary for proper landing, for flying over mountains, or for determining the height over a given objective. The knowledge of the true altitude is particularly important in blind flying (in fog, in clouds, at night, etc.).

According to the range of altitudes measured, radio altimeters are subdivided into the following classes:

- 1 - Low-altitude altimeters, used for determining altitudes to 1500 m.
- 2 - High-altitude altimeters, for altitudes above 1500 m.

Low-Altitude Radio Altimeters. Figure 89 shows the operating principle of a low-altitude altimeter. The transmitter installed on the aircraft generates electro-

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magnetic oscillations whose mean frequency (of the order of 440 megacycles) can be regulated by a modulator. The modulation frequency is of the order of 120 cycles. The antenna of the transmitter continuously emits electromagnetic waves toward the earth, and these waves, after being reflected from the ground, are received by the antenna of the receiver.

The frequency of the emitted signal differs from the frequency of the reflected signal by a quantity equal to the change in frequency of the transmitter during the time of passage of the signal to the ground and back. By virtue of this fact, a beat is produced in the detector of the receiver, in which the direct and reflected signal are combined. As a result, a low-frequency voltage is tapped from the detec-

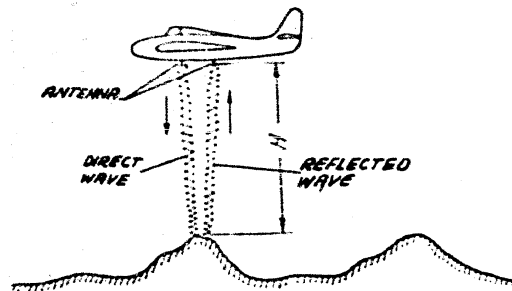


Fig.89 - Operating Principle of a Radio Altimeter

tor equal to the difference between the frequencies of the emitted and reflected signals.

This voltage is amplified by a low-frequency amplifier and is then fed across an amplitude limiter to a frequency meter, where it is converted into direct current, whose magnitude is directly proportional to the beat frequency.

An altitude indicator with a scale calibrated in meters is connected across the output of the frequency meter. The direct-current voltage received in the frequency meter does not depend with complete linearity on the altitude. There are methods of transforming this actual relation into a law of direct proportionality. The

voltage received is fed to an electric differentiator whose output voltage is proportional to the derivative of the altitude. Consequently, a voltmeter measuring the voltage may be calibrated in values of vertical velocity, i.e., a climb indicator may be designed on this principle.

Figure 90 shows the processes taking place in the altimeter.

The solid line shows the variation in the signal frequency entering the receiver directly from the transmitter; the broken line shows the variation in the frequency reflected from the ground. At constant altitude, the frequency of the second signal varies by the same law as the frequency of the first signal, but with a lag by the time T . The higher the altitude of flight, the greater is the shift in time between these two curves and the greater the difference frequency between the direct and reflected signals, which characterizes the altitude of the aircraft. This frequency is called the beat frequency.

The beat frequency, that is the frequency of the low-frequency oscillation generated in the receiver is a constant quantity, except for short time intervals when it drops to zero. At the detector output, a voltage of the difference frequency

$$F_p = f_1 - f_2 = 4 \cdot 10^6 \Delta f \frac{H}{a} = kH,$$

is generated, where F_p is the beat frequency in cycles;

Δf is the difference between the maximum and minimum frequency of the transmitter;

Δf is the modulation frequency in cycles;

H is the flight altitude in meters;

a is the velocity of propagation of the radio wave;

k is the proportionality factor.

As a rule, the low-altitude radio altimeter has two measuring ranges, and a switch is provided for change-over.

The first range from 0 to 150 m is used in take-off and landing; the second

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range, to altitudes of 1500 m.

The accuracy of such an instrument is $5\% \pm 2$ m.

The use of a frequency-modulated altimeter for measuring high altitudes is impossible since in this case we would require a very high-power transmitter, which would be capable of absorbing all the energy of the electric system of the aircraft.

High-Altitude Radio Altimeter. For measuring altitudes of flight above 1500 m,

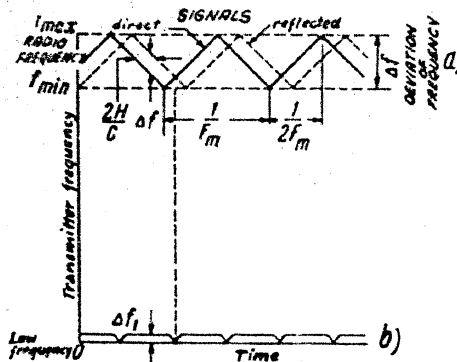


Fig.90 - Principle of Operation of Radio Altimeter

a - Variation in frequency of signals; b - Beat frequency

pulse radio altimeters are installed on aircraft. Such a radio altimeter consists of a radio pulse transmitter and a radio receiver both in a single unit, of a transmitting and receiving antenna (of symmetrical vibrators) and of an indicator unit.

Figure 91 is a block diagram of a high-altitude radio altimeter. The powerful radio transmitter with a narrow directional radiation emits short pulses spaced at uniform time intervals τ with a strictly constant high-frequency repetition (the repeat frequency is 100 or 1000 cycles, and the pulse period is 0.5 or 1 microsec). This regularity of generation is assured by using a special instrument called a chronizator.

The pulses so generated are fed into the modulator of a high-frequency vacuum-tube oscillator. The latter generates high-frequency pulses in the decimeter wave

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band and radiates them into the lower hemisphere to the antenna.

The time intervals between the pulses are many times the duration of the pulses themselves. The pulses of the radiation are propagated with the speed of light, c .

In the intervals between the pulses, the radio receiver operates and picks up the reflection of the signal from the ground, with a lag of time T equal to the time required for the pulse to reach the object and return to the receiver after reflection. The time T depends on the flight altitude H :

$$T = \frac{2H}{c}; \quad H = \frac{cT}{2}. \quad (\text{IV.35})$$

The direct proportionality between the time T and the altitude of flight allows

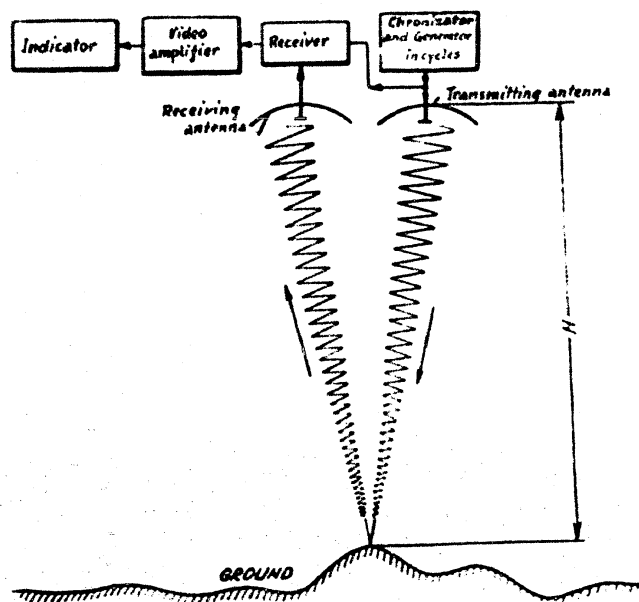


Fig.91 - Diagram of High-Altitude Radio Altimeter

the distance H to be determined by measuring the time T between reception of the pulses and reception of the reflected signals.

Since the velocity of propagation of the pulses is very high (300,000 km/sec), the time T so obtained is very small (1 microsecond corresponds to 150 m). At the same time, the accuracy of measurement of these short time intervals must be very high.

The receiver of a radio altimeter is constructed on the conventional superheterodyne principle. The signals, after detection, are amplified in an amplifier and are then fed to the indicator.

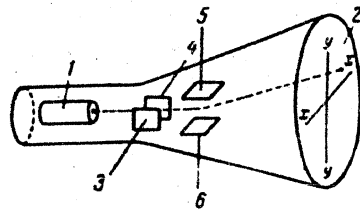


Fig.92 - Diagram of Cathode-Ray Tube

- 1 - Electron gun; 2 - Screen;
3,4,5,6 - Metal plates

The simplest and most perfected instrument for measuring time is the cathode-ray tube, which is similar to the tubes used in oscillographs and television sets. The diagram of such a tube is shown in Fig.92. The electron gun (1) throws a narrow beam of electrons on the screen, which is covered by a substance that fluoresces when electrons impinge on it. At the point of impact of the electron beam, a bright luminous spot is formed.

If the direction of the electron beam is rapidly and periodically varied, the luminous spot will be rapidly displaced on the screen, forming a luminous line on it. At a large number of such displacements per second, taking place in one and the same order, the eye perceives the trace of the electrons on the screen as a stable image, similar to the flashing of images, invisible to the eye, on the cinema screen. The electron ray passes between two pairs of metal plates, 3,4,5, and 6 which are arranged in mutually perpendicular planes. If a positive voltage is impressed on one of the plates and a negative voltage on the opposite plate, the electrons are attracted by the positive plate and repulsed by the negative plate. As a result, the electron beam and, consequently, the lumi-

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nous spot will be displaced in the corresponding direction.

In this way, by the aid of plates (3) and (4) the luminous spot can be shifted on the screen along the xx axis, and, by means of plates (5) and (6), along the yy axis.

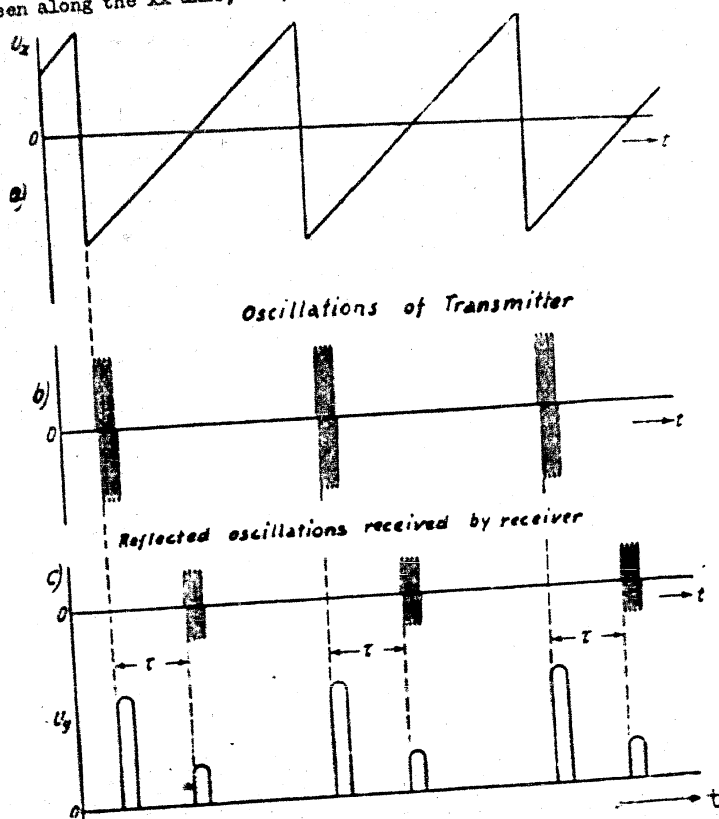


Fig.93 - Variation of Voltages on the Deflecting Plates of a Cathode-Ray Tube in the Pulse Radio Altimeter

The alternating voltage U_x , varying by a sawtooth law, as shown in Fig.93a is supplied to the plates (3) and (4). After each abrupt change in voltage, the luminous spot will be in the extreme left position. Then, as the voltage U_x varies smoothly, the spot is displaced to the right. When the voltage U_x becomes equal to



zero, the luminous spot is at the center of the scale. On further variation of the voltage, an ever increasing positive potential is created at the plate (4) and the electron beam is displaced to the extreme right-hand position. In this position, the voltage again abruptly changes its polarity to the opposite value, and the ray, being then attracted to the plate (3), passes to the extreme left position, after which it

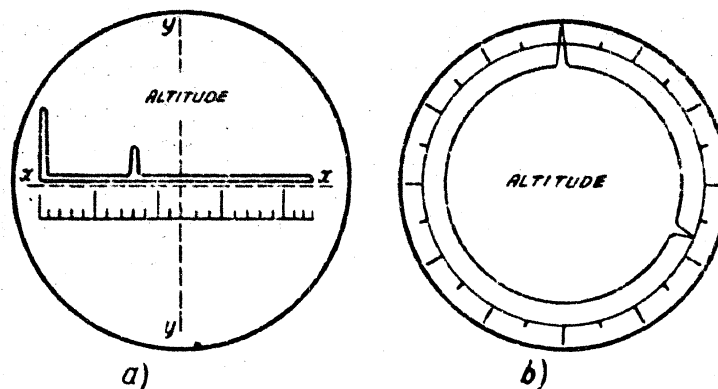


Fig.94 - Indicators of High-Altitude Radio Altimeter

a - Cathode-ray meter; b - Circular-motion electron meter

again begins smoothly to move to the right, and so on.

The pulse signals of the radio transmitter are emitted each time the voltage U_x undergoes an abrupt change. In the instrument, the signals of the transmitter and receiver are rectified, resulting in two voltage pulses as shown in Fig.93b, for each cycle of variation in the voltage U_x . It is these pulses of U_y that are fed to the plates (5) and (6) of the cathode-ray tube. In this way, during the time of its motion along the xx axis, the electron ray is twice thrown into the direction yy , as shown in Fig.94a. The distance between the obtained image and the straight line is proportional to the time T between the pulses, and to the altitude H .

By using another method of controlling the electron ray it is possible, instead

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of a rectilinear displacement of the light spot along the horizontal axis, to obtain a circular motion. In this case the initial mark and the mark corresponding to the measured altitude take the form of peaks or pips on a circle (Fig.94b). The existing models of high-altitude radio altimeters have an accuracy of measurement of the order of $0.25\% \pm 20$ m.

Radio Semi-Compasses and Radio Compasses

The methods of determining the position of an aircraft based on the use of radio semi-compasses and radio compasses are very useful to aviation in many cases, particularly in high latitudes, where the use of magnetic compasses is difficult or impossible.

The radio semi-compass and the radio compass allow the direction of the axis of the aircraft to be determined with respect to a ground radio station. The radio compass differs from the radio semi-compass in that the latter automatically indicates the direction of the ground radio station, while the former requires the intervention of the navigator.

By using a radio semi-compass the angle Π between the direction of flight toward a given radio station and the longitudinal axis of the aircraft can be determined.

By means of the radio semi-compass and a magnetic compass, we may determine the angle of true radio bearing $\Pi + \Pi$, i.e., the angle between the meridian at the position of the aircraft and the direction to the radio station (Fig.95). By connecting all points of the earth's surface at which the angle of true radio bearing has one and the same value, we obtain the position line, i.e., the line of possible positions of the aircraft. The radio semi-compass does not indicate in exactly what position on this line the aircraft is located. In order to determine the aircraft position auxiliary methods must be used; for example, the true radio bearing of two ground transmitters may be found and in this way two position lines may be obtained,

whose point of intersection determines the position of the aircraft. The diagram of the radio semi-compass is given in Fig.96. The instrument has two antennas, one loop and one open vertical antenna. The relation between the strength of reception

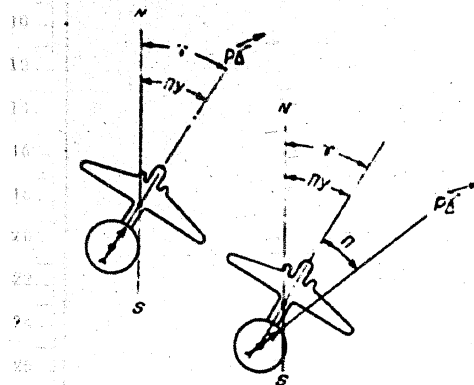


Fig.95 - Reading of the Radio Compass
P - Ground radio station; γ - Course angle; β - Bearing

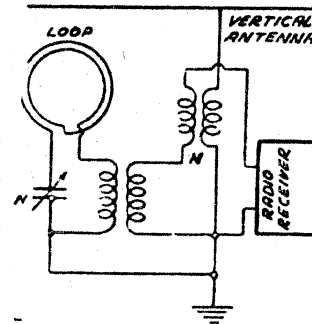


Fig.96 - Summation of Signals from Loop and Vertical Antennas

by the loop and the angle γ between the plane of the loop and the direction to the radio station is given in Fig.97. The signal becomes loudest when the plane of the loop is directed toward the transmitter, and is equal to zero at two diametrically opposite positions of the loop, 90° and 270° , when the plane of the loop is perpendicular to the direction of the radio station. In order to determine which of the positions of the loop is the correct one, the open antenna is used.

The high-frequency signal voltages from both antennas are combined in the common circuit (Fig.96). By choosing the mutual inductance M , the voltage of the open antenna (a constant quantity) is made equal to the maximum intensity of the voltage from the loop (on the coincidence of its loop with the direction to the radio trans-



mitter). By the aid of capacitor C, the phases of the voltage from the vertical antenna and the voltage from the loop are made to coincide when its plane coincides

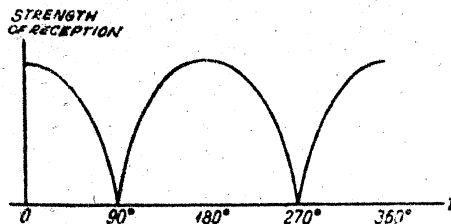


Fig.97 - Diagram of the Directivity of the Loop

with the direction to the transmitter. This coincidence is possible only in one of the two positions of the loop at which audibility is zero, since in the diametrical-

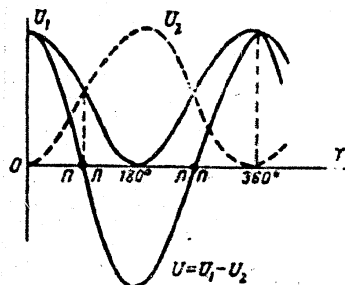


Fig.98 - Obtaining the Voltage Difference $U_1 - U_2$ for the Indicator of the Radio Semi-Compass

ly opposite positions the sign (phase) of the voltage changes (Fig.98).

By adding the voltages of both antennas we obtain a total characteristic from which it will be seen that at $\gamma = 180^\circ$, the total voltage is zero.

Only in this position of the loop will the signals from the transmitter on the ground fail to be heard in the ear-phones of the receiver. At $\gamma = 360^\circ$, the total voltage reaches its maximum.

This method of determining the two directions is not the only possible one. Very often the following procedure is used instead: If the positions of the ends of the loop are interchanged, then the sign of the voltage coming from the loop antenna is reversed; in this case, the total voltage is equal to 0 at an angle of $\gamma = 0$, instead of at $\gamma = 180^\circ$.

In the radio semi-compass the ends of the loop are periodically switched over so that the voltage U impressed on the indicator consists of the difference between the two voltages U_1 and U_2 , depends on the angle γ of rotation of the loop (Cf. Fig. 98), and has a positive value at angles from 0 to 90° and at angles from 270 to 360° , and a negative value at angles from 90 to 270° .

At a voltage of $U = 0$, the pointer of the instrument is at the midpoint of the scale. When U is positive, the pointer deviates to the right, and at negative value of U , to the left; in the former position of the loop, when $\gamma = 90^\circ$, the reduction in the angle, i.e., the rotation of the leftward rotation of the loop toward the zero position, leads to a rightward deflection of the instrument pointer. In the second position, when γ is 270° such a rotation leads instead to a leftward deflection of the pointer. Consequently, the first position will be the correct one, and the radio station is actually located to the right of the perpendicular to the plane of the loop. In order to determine the line of possible positions of the aircraft, the loop antenna is used, since in this case the minimum of reception will be sharper. To find the correct position, both antennas are used.

By the aid of the radio compass the same problems are solved as by means of the radio semi-compass, except that in that case the correct position of the loop of the radio compass is automatically found by means of the following system, which sets the loop in a certain way with respect to the radio station.

The automatic radio compass consists of a radio receiver which automatically locates the loop antenna, an automatically rotating loop antenna, an open nondirectional antenna, a bearing indicator, a control board, and a relay box. In tuning the receiver to the wave of the radio station, the output signal from the radio receiver is supplied to the mechanism that rotates the loop, forcing it to turn this loop until it coincides with the direction toward the radio station. After setting the loop in the direction of the radio station, signals in the receiver disappear and the rotations of the loop stops. The loop is connected with a synchronous elec-

tric transmission from the indicators of the radio compass, whose pointers change their position when the loop is rotated and stop together with it, showing the bearing of the radio station.

If the aircraft continues changing its course until the pointer of the indicator is at zero, and then keeps the pointers of the indicator at zero, its flight in the direction of the radio station will be insured.

Figure 99 shows the block diagram of the radio compass.

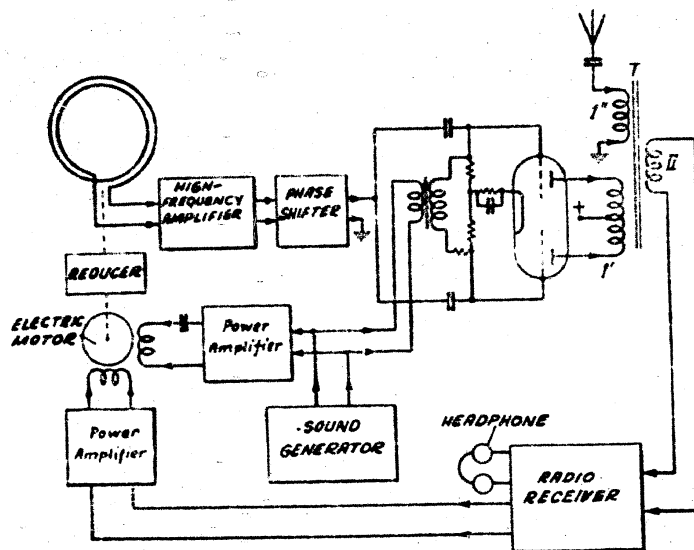


Fig.99 - Electric Block Diagram of the Radio Compass

In the loop antenna, the incoming electromagnetic oscillations set up an emf whose value is smaller the closer the plane of the loop is to a position normal to the direction of the incoming waves. At the moment when the plane of the loop is perpendicular to the direction of the wave, the emf of the loop is equal to zero. On passage through the zero position, the emf of the loop changes its phase by 180° .

with respect to the emf of the nondirectional antenna. In addition, the emf of the loop has a 90° lag relative to the emf of the open antenna.

The emf of the loop, after amplification in the high-frequency amplifier stage,

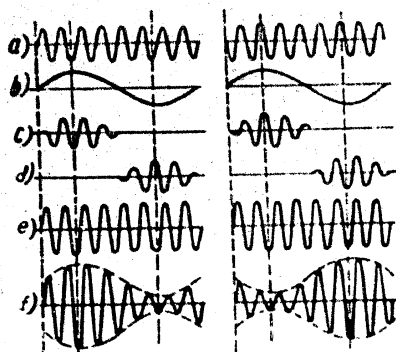


Fig.100 - Diagram of Voltage in the

Channel of the Balancing Modulator

a - Signal of loop after a 90° phase shift; b - Modulator to low-frequency voltage in balancing modulator; c - Output of first half of signal from balancing modulator; d - Output of second half of signal from balancing modulator; e - Signal from nondirectional antenna; f - Resultant signal after combination of modulated signal and signal from nondirectional antenna

This direction is fed to the grid of the modulator tube (a double triode).

Figure 100b shows the variation of the voltage from the low-frequency oscillator

is fed to the phase shifter where it is given an additional phase shift by 90° . Now the emf of the loop may either coincide with the emf of the nondirectional antenna or (when this loop passes to the zero position) it will have a phase differing by 180° from the emf of the antenna. The balance modulator on which the emf of the frame is impressed, is designed so as to indicate its phase and, consequently, also the direction of any deviation from zero position.

The low-frequency (60-100 cycles) oscillator feeds two generated voltages with opposite phases to the grid of the modulator tube, a balanced modulator, to which the emf of the loop is also fed.

Figure 100 is a diagram of the voltage in the channel of the balanced modulator.

The graph (Fig.100a) shows on the left and right the variation of voltage in the loop when it is deflected to the left or right of the neutral position.

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on one of the grids of the tube. When this voltage is positive, a high-frequency current corresponding to the voltage on the grid, passes through the plate circuit of this section of the tube, as shown in Fig.100c.

At this time, the voltage on the second grid of the modulator tube is negative, and there is no current flow in the plate circuit of the second section of the tube. When the low-frequency voltage on the first grid becomes negative, the current in

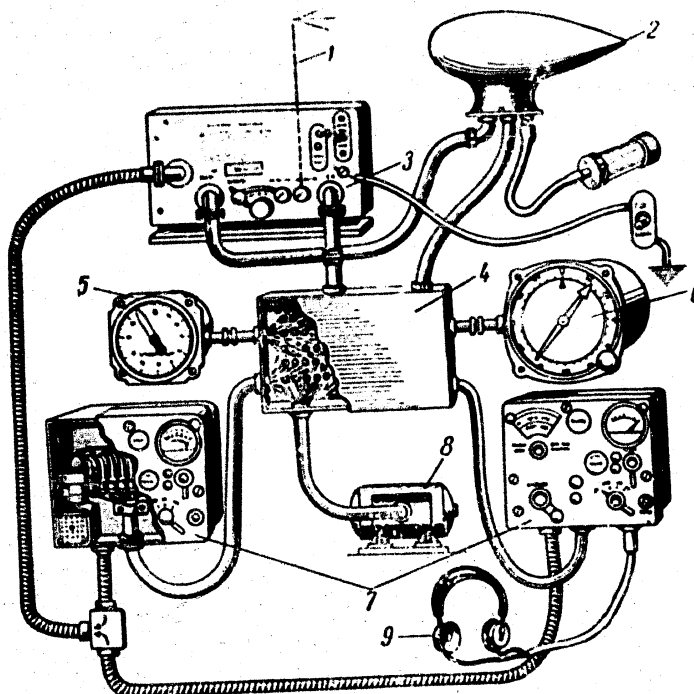


Fig.101 - The Radio Compass Set

- 1 - Directional antenna; 2 - Nondirectional antenna; 3 - Radio receiver; 4 - Relay box; 5 - Bearing indicator; 6 - Bearing indicator, pilot; 7 - Control panels;
- 8 - Inverter; 9 - Headphone

the plate circuit of this section is interrupted. The voltage from the second

grid then becomes positive, and a high frequency current flows in the plate circuit of the second section, as shown by Fig.100d. The current of this section of the tube, in the winding 1' (cf. Fig.99) of the transformer T, has a direction opposite to the plate current of the first section of the tube. For this reason, the high-frequency current in Fig.100c is shown as opposite in phase to the high-frequency voltage coming from the loop antenna (cf. Fig.100a).

In the secondary winding of the transformer T, the unmodulated oscillations of the same frequency coming from the nondirectional antenna across the primary winding 1" are added to the modulated high-frequency oscillations from the loop. The current in this winding is shown in Fig.100e.

The result of the combination of the oscillation shown in Fig.100a,c,d, and e is shown in Fig.100f. These are modulated oscillations with the modulation phase of their amplitude being determined by the direction of the deviation of the loop from the zero position.

The total high-frequency voltage is amplified in the receiver, after which it is detected and is again amplified at low frequency. At the output of the receiver a low-frequency voltage, corresponding to the law of modulation of the input signal, is obtained. The phase of this voltage likewise depends on the direction of the deflection of the loop.

At zero position of the loop, there is no high-frequency voltage. In this case, there is also no low-frequency voltage at the output of the receiver.

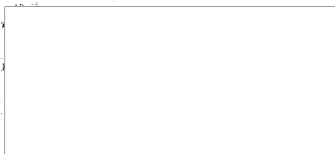
The voltage taken from the output of the receiver is amplified in a magnetic power amplifier and is fed to one of the phase windings of a two-phase asynchronous motor. The second phase winding is fed by alternating current from an inverter. The motor rotates the loop until it is set in zero position, after which the low-frequency voltage at the receiver output disappears and the motion stops. When the loop is deflected toward the other side, the low-frequency voltage changes its phase by 180° , as a result of which the direction of rotation of the motor is

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reversed, and the motor again returns the loop to the zero position.

The radio compass is fed with direct current (tube filaments, control signal, etc.) from the aircraft 27-volt system, while the alternating current feed (supplying the circuit of the radio receiver, motor, etc.) is supplied by a 115-volt 400-cycle inverter.

There is a control panel for the remote control of the compass. Figure 101 shows the complete set of the radio compass.



CHAPTER V
GYROSCOPIC INSTRUMENTS

Section 1

ELEMENTARY THEORY OF THE GYROSCOPE

The Concept of the Gyroscope

In technology, a flywheel (rotor) 1 (Fig.102) rapidly rotating about its axis of symmetry, held by one or two likewise movable rings (frames) 2, called gimbals, is known as a gyroscope.

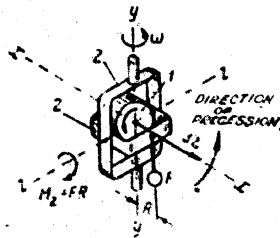


Fig.102 - Gyroscope with Three Degrees of Freedom
1 - Rotor; 2 - Frame

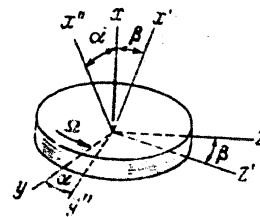
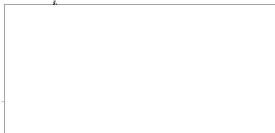


Fig.103 - Angles Determining the Direction of the Rotor Axis in Space
 α - Azimuth; β - Altitude; Ω - Velocity of the natural rotation of the gyroscope

Depending on the structure of the suspension, the number of degrees of freedom of the gyroscope varies. With two movable frames the gyroscope has three degrees of freedom, since the rotor may rotate about three mutually perpendicular axes xx , yy , zz . If one frame is made immobile, the gyroscope will have two degrees of freedom.

The axis xx of rotation of the rotor is called the principal axis, and the rotation of the rotor about this axis is known as the natural rotation of the gyro-



scope. The gimbals allow the rotor axis to occupy any position in space determined by the azimuth β and the elevation β (Fig.103). If the moments of the forces of gravity with respect to all three gyroscope axes are equal to zero, the gyroscope is called astatic. This is possible in two cases:

1. When all three axes of the gyroscope xx , yy , and zz intersect in a single point which remains fixed when the gyroscope moves and coincides with the center of gravity of the gyroscope;

2. When the center of gravity of the rotor lies on the xx axis of natural rotation; the center of gravity of the system rotor - inner ring is located on the zz axis of this ring; and the center of gravity of the system rotor - inner ring - outer ring is located on the yy axis of the outer ring.

If the moment of the forces of gravity with respect to the gyroscope axis is not equal to zero, then such a gyroscope is called a gyro pendulum.

A gyroscope is known as a free gyroscope if no external forces or moments act on it.

In considering gyroscopic phenomena we are dealing with the rotary motions of a body which are characterized by the direction of the axis of rotation, the sense of the rotation, and the angular velocity of rotation Ω , i.e., they may be characterized by the vector Ω of this velocity. We will conventionally consider the vector Ω to be placed in such a way that, when looking from the point of the vector, the motion of the rotor is counterclockwise. In this case it is a dextrarotary system. The length of the vector Ω is laid off on a definite scale and corresponds to the magnitude of this velocity.

The value of the angular velocity is measured in 1/sec. If the number of revolutions per minute n_m or per second n_s is known, then the quantity Ω may be found from the equation

$$\Omega = \frac{2\pi n}{60}$$

(V.1)

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For approximate calculation, the equation

(V.2)

is used.

The rotary motion is due to a moment of force or a couple.

The moment of the force is calculated by the formula

$$M = FR \sin \alpha$$

where F is the force;

R is the arm of the force;

α is the angle between the direction of the force and the axis of rotation.

If the force acts at an angle of 90° to the axis, then

$$M = FR$$

The moment of force is represented likewise in the form of a vector placed along the axis of rotation and directed in such a way that, when looking from its tip, the force applied to the body tends to turn the body counterclockwise. The kinetic energy (vis viva) of a rotating solid body is expressed by the formula

$$T = \frac{1}{2} \sum mV^2 = \frac{\Omega^2}{2} \sum mr^2$$

but since $\sum mr^2 = \int r^2 dm = J_x$, it follows that

$$T = \frac{J_x \Omega^2}{2}$$

where m is the mass of an elementary particle of the given body;

V is the velocity in m/sec;

Ω is the angular velocity of natural rotation;

r is the distance of the given particle from the axis of rotation;

J_x is the moment of inertia of the body with respect to its axis of rotation,

in gm-cm-sec².

Knowing the mass of the body and its form we may determine the moment of inertia of the body with respect to any axis. If the body is of complex form, it is divided into simpler bodies, and their moments of inertia with respect to one and the same axis are found. Then, by summation, the moment of inertia of the whole body is found with respect to the same axis. In cases where the form of the body is so complex that the moments of inertia cannot be calculated by this method, the moment of inertia of the body is found experimentally.

The moment of inertia of a rapidly rotating symmetrical gyroscope with respect to the proper axis will be denoted in what follows by the letter C, and with respect to the other two axes by A.

If a rotating body is exposed to the effects of several forces, the sum of the moments of which, with respect to the axis of rotation, is equal to zero, then the body will move without angular acceleration.

If the sum of the moments applied to the body is not equal to zero, then the body will move with an angular velocity which is measured by the increment of angular velocity of rotation in unit time.

The displacement, velocity, and acceleration of a body, relative to a fixed system of coordinates are called absolute, and those relative to a moving system of coordinates are called relative. The motion of a moving system of coordinates with respect to a fixed system is called transport motion and is characterized by the transport velocity V_t and by the acceleration a_t .

The absolute velocity V of a certain point A which is in complex motion is equal to the geometric sum of the relative velocity V_o and to the transport velocity V_t .

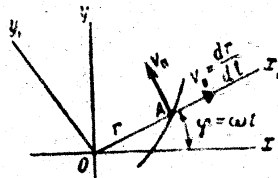
$$\bar{V} = \bar{V}_o + \bar{V}_t$$

By analogy, the absolute acceleration $\bar{a} = \bar{a}_o + \bar{a}_t$.

If a transport motion is rotary, then the rotary acceleration \bar{a}_c , which is also termed the Coriolis acceleration, must be added to this acceleration:

$$\bar{a} = \bar{a}_o + \bar{a}_t + \bar{a}_c$$

Let us assume that Oxy is a certain fixed system of coordinates (Fig.104) and



Ox_1y_1 is a moving system of coordinates, the system of coordinates Ox_1y_1 being in uniform rotation at the angular velocity ω about the point O . Then the angular displacement φ of this system in the time T will be

$$\varphi = \omega t$$

Fig.104 - Determination of Absolute Acceleration at Rotary Transport Motion

Motion

- Oxy - Fixed system of coordinates;
- Ox_1y_1 - Moving system of coordinates;
- ω - Angular velocity of rotation of moving system of coordinates; r - Distance of point A from origin of coordinates; V - Absolute velocity of point A ; V_r - Relative velocity of point A ; V_t - Transport velocity of point A

Let us assume that, in turn, the point A on the axis Ox_1 is displaced with respect to this axis by the velocity $V_o = \frac{dr}{dt}$, where r is the distance between point A and the origin of coordinates. The trajectory of the point A , performing a complex motion, will be a spiral. The absolute velocity V of the motion of the point A represents the geometric sum of the relative velocity V_o and the transport velocity $V_t = \omega r$ of the velocities of this point

$$\bar{V} = \bar{V}_o + \bar{V}_t = \bar{V}_o + \bar{\omega}r$$

The projection of the absolute velocity onto the fixed system of coordinates may be found from the equations

$V_x = V_0 \cos \varphi$
 $V_y = V_0 \sin \varphi$

In order to find the projection of the absolute acceleration onto the axes x and y, we use the derivatives of these equations with respect to time

$$a_x = \frac{dV_x}{dt} = \frac{d(V_0 \cos \varphi)}{dt} = -V_0 \sin \varphi \frac{d\varphi}{dt} = -V_0 \omega \sin \varphi$$

$$a_y = \frac{dV_y}{dt} = \frac{d(V_0 \sin \varphi)}{dt} = V_0 \cos \varphi \frac{d\varphi}{dt} = V_0 \omega \cos \varphi$$

Since $\omega^2 r = a_t$ (transport acceleration) we find the equation for the instant of reading t, when the axes Ox_1 and Ox coincide: $\varphi = 0$; $a_x = a_0 - a_t$; $a_y = 2 V_0 \omega$.

We obtain the absolute acceleration a as the geometric sum of the three accelerations

$$a = a_x + a_y = a_0 + a_n + 2V_0\omega. \tag{V.3}$$

The acceleration $2 V_0 \omega$ is called the rotary or Coriolis acceleration a_c . The direction of this acceleration is obtained by rotating the vector of relative velocity V_0 (in our case it is directed along the axis Ox) through 90° in the direction of the transport rotation ω . In the example we are discussing the rotary acceleration is directed along the axis Oy .

If the vector of relative velocity V_0 is not perpendicular to the vector of the transport angular acceleration ω and forms the angle α with it (Fig.105) then, by resolving this velocity into a parallel and a perpendicular component with respect to the axis of transport rotation, we find that the rotary acceleration is due only to the perpendicular component of this velocity $V_0 \sin \alpha$. The value of



the rotary acceleration is determined by the expression

$$a_c = 2 V_0 \sin \alpha$$

The direction of the acceleration a_c is determined by turning the component $V_0 \sin \alpha$ through 90° in the direction of the transport rotation.

Thus, on considering the complex motion of a body with a transport rotary motion, it is necessary to take account of all three accelerations a_0 , a_t , a_c .

Principal Properties of a Rapidly Rotating Gyroscope with Three Degrees of Freedom

Since the solution of the problem of the motion of the gyroscope with respect to a fixed point under the action of assigned external forces is very complex, let

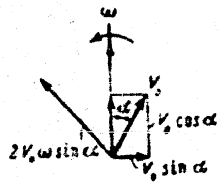


Fig.105 - Rotary Acceleration of the Body in the General Case of Motion
 α - Angle between the direction of relative velocity and the vector of transport angular velocity

us consider instead the inverse problem: Let us assign the motion and find what forces or moments cause that motion, and whether this is the result of the action of external forces or moments or takes place as a result of inertia.

Let the rotor of a gyroscope with three degrees of freedom rotate about its natural axis xx at an angular velocity Ω (cf. Fig.102).

At the same time the axis Ox and, with it, the whole system of coordinates Oxz , connected with the gyroscope, rotates with the angular velocity ω about the axis Oy (Fig.106a).

The relative motion of the gyroscope is characterized by the velocity Ω , and the transport motion by the velocity ω , if $\omega \ll \Omega$.

The relative linear velocity of any point B of the rotor (Fig.106b) is equal to $V_0 = \Omega r$, where r is the distance of this point from the axis of rotation.

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For points lying on the periphery of the rotor, the velocity V_o is determined by the expression

$$V_o = \Omega R$$

where R is the radius of the rotor.

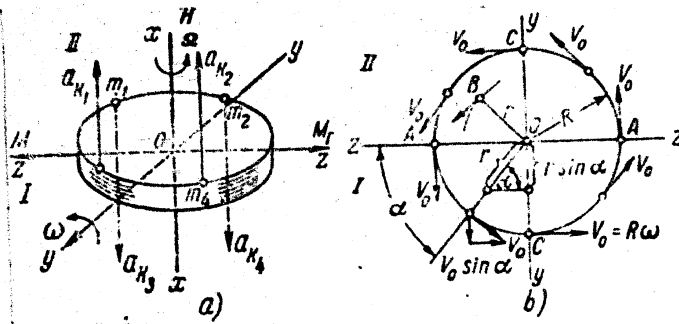


Fig.106 - Precessional Motion of the Gyroscope

Ω - Velocity of relative motions of the gyroscope; ω - Transport velocity of the gyroscope; R - Radius of rotor; V_o - Relative linear velocity of any point of the rotor; m_1, m_2, m_3, m_4 - Masses of points lying on the rotor rim; B - Arbitrary point on the rotor body

The direction of the velocity V_o is shown in Fig.106b.

The angular acceleration generated by the motion of the gyroscope under consideration is caused by the component of this velocity along the axis Oz , i.e., $V_{Oz} = V_o \sin \alpha = \Omega r \sin \alpha$.

The value of the angular acceleration is determined by the relation

$$a_t = 2 V_o \omega \sin \alpha$$

For all points of a body located in the region I, the acceleration is directed upward while for points located in region II, it is directed downward (with the

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selected directions of Ω and ω).

At the points AA, these velocities are zero [$V_{Oz} = 0$], while at the points CC the accelerations reach their maximum value, since $V_{Oz} = V_o = QR$.

It is clear from this that the forces producing the rotational acceleration and causing the motion of the rotor about the axis Oy with the angular velocity ω , form a couple acting with respect to the axis Oz, about which, by hypothesis, there is no motion.

The vector of the moment M of this couple is directed from right to left. Thus, at a transport motion of the rapidly rotating gyroscope about the axis Oy, with an angular velocity ω , due to the couple acting about the axis Oz, the points m of the rotor will have an acceleration of the direction shown in Fig.106a. Since these accelerations are not the result of the action of internal forces (the reactions of internal connections) but appear as a result of the interaction of the components of the relative velocity V_o of the gyroscope, directed along the ZZ axis (the velocity V_{Oz}), and of the transport velocity ω , it follows that the gyroscope motion under consideration, about the axis Oy, is the result of the action of external forces or moments. This motion is termed precessional, and its velocity is called velocity of precession. The precessional motion of a gyroscope, at first glance, seems to be paradoxical, since the external moment M acts about one axis of the gyroscope, while this motion takes place about another axis of the gyroscope. It will be easy to convince oneself from the following example that this contradiction is an apparent one.

A body being displaced in a horizontal plane in a vacuum at the initial instant of time with the velocity V , under the action of the force of gravity begins to vary its trajectory of motion (which will curve downward along a parabol). In this case, the direction of the velocity vector of the motion of the body will likewise vary, and this variation will be slower, the higher the initial velocity of the body.

Let us return to the gyroscope rotor. Assume that a gyroscope (Fig.107) with a fixed point O corresponding to the center of gravity, is subjected to the moment of the couple M . It may be assumed that forces directed upward act on the points of

the rotor in the region I, and forces directed downward on those in the region II.

The point A of the rotor is displaced with the velocity V_{Oz} (the linear relative velocity of this point $V_0 = \Omega R$). Under the action of the downward forces, the trajectory of the point A must likewise curve downward, and the projection of its velocity vector V_{Oz} onto the plane xOz likewise occupies the position V_{Oz1} inclined to the plane of the horizon. Consequently, the point of the periphery A rotates and moves into the position D .

This means that the whole rotor, together with it, has rotated about the yy axis in such a way that its axis now occupies the position z_1z_1 (Fig.107b).

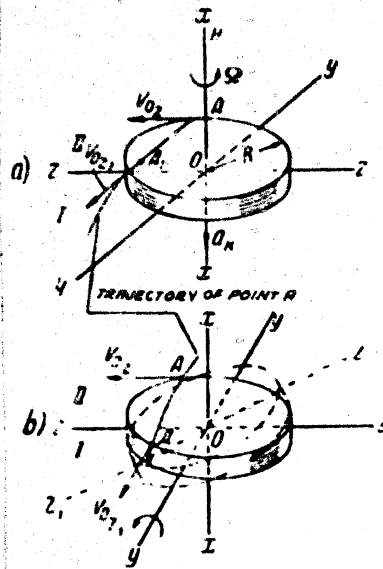


Fig.107 - Trajectory of an Arbitrary Point of a Gyroscope Rotor in Precessional Motion.

Thus, under the action of the moment of the external forces M , acting about the zz axis of the gyroscope rotating at high relative velocity $V_0 = \Omega R$ about the xx axis, the gyroscope will begin to precess about the yy axis (at an angular velocity ω).

Let us find the relation between the moment of inertia of the rotor C , the angular velocity of its proper rotation Ω , the angular velocity of precession ω , and the external moment M . The direction of the rotational acceleration is shown in Fig.106 and the value of the force applied to the point D of the gyroscope which

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causes this acceleration, may be found by the formula

$$P = a_c m = 2 V_0 m \omega \sin \alpha = 2 \Omega \omega r m \sin \alpha.$$

The moment M_d , produced by the force P is found from the equation $M_d = Pl$, where l is the arm of the moment equal to the distance of the point D from the zz axis, and consequently $l = r \sin \alpha$.

Then

$$M_d = 2 \Omega \omega r^2 m \sin^2 \alpha = 2 \Omega \omega m y^2$$

since $r \sin \alpha = y$.

All the particles of the gyroscope produce a total moment equal to

$$M = 2 \Omega \omega [m_1 y_1^2 + m_2 y_2^2 + \dots + m_n y_n^2] = 2 \Omega \omega \sum m y^2,$$

where $m_1, m_2, m_3, \dots, m_n$ are the masses of the particles of the gyroscope;

$y_1, y_2, y_3, \dots, y_n$ are corresponding coordinates of these points;

$\sum m y^2$ is the moment of inertia of the gyroscope with respect to the plane Oxz .

Since the gyroscope rotor usually takes the form of a body of revolution (roughly in the form of a cylinder), its moments of inertia with respect to the planes Oxz and Oxy are equal, i.e., $J_{xz} = J_{xy} = A$, while the moment of inertia C with respect to its axis of rotation is $C = 2A = 2 \sum m y^2$. In this case, the external moment is

$$M = C \Omega \omega = H \omega \tag{V.5}$$

since $C \Omega = H$ (kinetic moment of the gyroscope). The expression (V.5) is called the equation of precession.

On rapid rotation of the rotor, the vector of the kinetic moment of the gyroscope, H , practically coincides with the vector of the angular velocity Ω , and at

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the same time with the proper axis of the gyroscope.

The direction of the precession of the gyroscope under the action of the external moment may be found by the following method: Together the precessional motion of the gyroscope, its vector of kinetic motion rotates, tending by the shortest path to coincide with the vector of the external moment M (cf. Fig. 106).

Let us consider the case where the angle between the vectors Ω and ω , does not equal $\frac{\pi}{2}$, i.e., when, for example, the axis of transport rotation ω makes the angle α with the axis Ox of natural rotation of the gyroscope (Fig. 108).

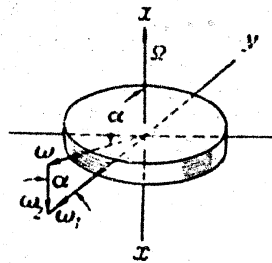


Fig. 108 - Motion of the Gyroscope under the Influence of the Moment of External Forces

be written in the form

$$M = H \omega \sin(\omega \hat{\Omega}) \quad (V.6)$$

It follows from eq. (V.6) that:

1) The precession caused by the action of the external moment M takes place at constant velocity ω , and at the instant of cessation of the action of the external moment, the precession of the gyroscope likewise ceases.

2) If the kinetic moment H of the rotor is large, while the moment of external forces M is small, then the velocity of precession $\omega = \frac{M}{H}$ will be very small; it

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may, therefore, be considered that the gyroscope keeps the direction of its proper axis in space practically constant. The higher the value of $H = C \Omega$, the more accurately will the constant direction of the axis of the gyroscope rotor be maintained. For this reason, an effort is made to increase this moment as much as possible in gyroscopic instruments, thus increasing the inertia of the gyroscope.

3) Under the action of the external moment M , the gyroscope precesses in the direction perpendicular to the plane of the couple. Consequently, the external moment is balanced by some moment, which is obviously the moment of the inertia forces. Indeed, the moment that impedes the rotation of the gyroscope in the direction of the action of external forces is basically the moment of the inertia forces due to the rotational accelerations a_c . This moment is equal to the external moment, but with reversed sign. It is termed the gyroscopic moment*.

$$M_g = -M = H\omega \sin(\Omega\omega) \quad (V.7)$$

The direction of the vector of the gyroscopic moment M_g may be found by rotating that component of the vector ω which is perpendicular to the rotor axis about this axis in the direction of its rotation through 90° . To determine the direction of the gyroscopic moment, we ordinarily use the following rule: The gyroscopic moment M_g created when the gyroscope is rotated at the instantaneous angular velocity ω is equal to $M_g = -H\omega \sin(H\omega)$. This moment is perpendicular to the plane in which the vectors H and ω are located, and so directed as to tend to bring the vec-

* In this case, we neglect the centripetal accelerations due to rotation at the angular velocity ω . The forces of inertia of this acceleration form a couple which acts as though it tended to reduce the angle $\Omega\omega$. The external moment producing the centripetal accelerations must be algebraically added to the moment producing the rotational accelerations.

tor ω (Fig.106a). If the friction in the bearings of the frames in the astatic free gyroscope were assumed equal to zero, then such a gyroscope would maintain its position at rest, and in that case the direction of its axis would remain fixed in space, regardless of the rotation of the rotor.

In reality, such an ideal suspension without friction is not feasible, and the character of the motion of a gyroscope with a fixed rotating rotor would be different. It is demonstrated in mechanics that even if no external moments at all act on the gyroscope, the rotor axis will still remain fixed in space or will describe a circular or elliptical cone, according to the initial conditions, and that the period of that cone will be

$$T = \frac{2\pi}{H} \sqrt{J_x J_y} \quad (V.8)$$

where H is the kinetic moment of the gyroscope;

J_x, J_y are the moments of inertia with respect to the Ox and Oz axes of the system of gimbals together with the rotor.

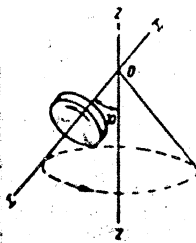


Fig.109 - Regular Precession of the Gyroscope

The motion of the axis of the rotor takes place without the action of an external moment, i.e., by virtue of inertia. In theoretical mechanics this is known as regular precession, and in technology as nutation. This motion is executed at high frequency, corresponding roughly to the velocity Ω of the natural rotation of the rotor and is manifested in the form of motion of the axis of symmetry of the rotor along a cone about the vector of kinetic moment (Fig.109). As a result of friction in the bearings of the frames, the nutational oscillations are rapidly damped while the precession remains.

Returning now to the question of the inertialess nature of the precessional motion of the gyroscope, the following must be remarked:

As already stated, the precessional motion of the gyroscope is immediately interrupted at the instant at which the causative moment ceases to act.

The motion due to inertia, however, takes place in the form of nutational oscillations, which are so small in the rapidly rotating gyroscope that they cannot be detected by the naked eye. The above-enumerated properties of the astatic gyroscope (with three degrees of freedom) and particularly its peculiar stability in the selected direction, i.e., the property of slowly varying its initial position under the action of external moments, which is inherent to the gyroscope to a greater extent than to any other body, opened excellent prospects for the use of the gyroscope in technology.

At the present time, gyroscopes are being used as systems for maintaining a fixed direction with respect to the vertical, the meridian, the fixed stars, etc. which allow us to determine the angles of inclination with respect to the plane of the horizon, the course of an aircraft or a ship, etc. and also as systems for stabilizing a given object, such as an aircraft, in a definite position (automatic pilot).

Figure 110 shows the utilization of the free gyroscope for observing the rotation of the earth.

If no external forces or moments act on the axis of a free gyroscope, it will maintain a constant position in space. Owing to the diurnal rotation of the earth, the position of this axis with respect to the meridian will vary. To an observer on the earth, moving together with it, it will seem that the axis of the gyroscope also rotates and describes, during one rotation of the earth, a complete cone whose axis is parallel to the axis of rotation of the earth. The observer sees the apparent motion of the gyroscope.

Thus, even if it were possible to make an ideal free gyroscope, it could not

be used for determining a direction that was constant with respect to the earth, since its axis maintains its direction constant with respect to space. For example, if the axis of such a free gyroscope at any point of the earth were directed along the vertical, then at all other points of the earth it would indicate the former direction in space, i.e., a direction different from that of the local vertical.

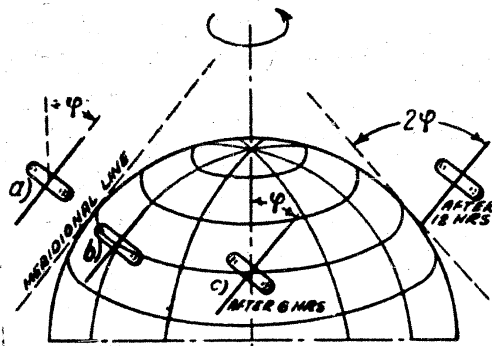


Fig.110 - Apparent Motion of the Free Gyroscope

In addition, if such a gyroscope, under the influence of any forces or moments, were to deviate from the initial position, then it would no longer return to it but would instead tend to maintain its new position. Such a gyroscope does not possess selectivity in the choice of its direction with respect to the selected direction and is "indifferent" in the same way to any position with respect to the earth.

The deviation of the gyroscope from the initial position may take place under the influence of the moments of inertia forces which arise in the presence of all types of acceleration, on displacements of the object, friction, residual unbalance, etc. The rate of this deviation (precession), if the kinetic moment of the gyroscope $H = C\Omega$ is properly selected, is small. For this reason, in cases when a definite direction must be maintained for a limited and short period of time (2 - 10 min), it is entirely possible to use an ordinary astatic gyroscope, provided it is properly set at the initial moment.

In cases where the gyroscope is being operated for a long period, or when it is impossible to guarantee the accuracy of its setting at the initial moment, it is necessary, even during short-period operation, to give the gyroscope axis a selectivity with respect to the earth, i.e., the power of assuming only a single selected direction.

Two methods are used for giving selectivity to gyroscopes with three degrees of freedom:

The first method is based on the displacement of the center of gravity of the system away from the fixed point. For example, in order to give selectivity with respect to the vertical, the center of gravity in the system is shifted along the rotor axis (Fig.111), i.e., a gyroscopic pendulum is used.

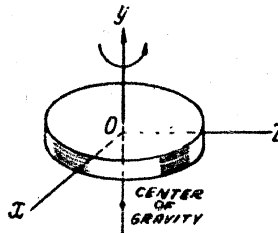


Fig.111 - Giving a Gyroscope, with Three Degrees of Freedom, Selectivity of Displacement of its Center of Gravity.

The restoring moments, i.e., the moments restoring the axis of the rotor to the selected direction, are obtained by a very simple method; however, in doing this, the dimensions of the instruments are increased and in addition, the oscillatory motions of the proper axis of the gyroscope, generated in such instruments, require the use of damping devices which considerably complicate the design.

For this reason, a second method is used in aviation, based on the utilization of special mechanisms (correcting devices) which return the proper axis of the gyro-

scope to the selected direction, and retain it at this direction.

The operation of the correction device is based on the following two principles:

1. During the deviation of the gyroscope, the correcting moment, i.e., the moment produced by the correcting device, is applied to the gyroscope in such a way that it causes the rotor axis to move in the same plane in which the deflection took place, but in the opposite direction.

Consequently, the moment that eliminates the deviation of the rotor axis must act in the plane perpendicular to this deviation.

2. The fact of the deviation of the gyroscope from the selected direction is determined by comparing the position of the gyroscope with the position of any other device possessing selectivity with respect to the direction selected.

This device is sometimes termed a sensitive correcting element. The selection of a sensitive correcting element depends on the fixed direction the gyroscope must maintain. For example, if the gyroscope must be set along the vertical direction (gyro vertical), then the sensitive element is made in the form of a pendulum.

The action of the correcting device is dependent on the comparison of the position of the gyroscope and the sensitive element.

As a result, a system is obtained that combines selectivity (the sensitive correcting element) and high inertia (a gyroscope with a very low rate of precession).

Owing to this, brief and random deviations of the sensitive element of correction do not noticeably affect the position of the axis of a gyroscope with a low rate of precession. At the same time, the precession of the gyroscope that is due to moments of friction, unbalances of the gyroscope, etc. is eliminated in time by means of the correcting device.

The diagrams of gyroscopic instruments with various correcting devices as well as the characteristics of such corrective devices, are described in Sections 3 and 4

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The diagrams of gyroscopic instruments with various correcting devices as well as the characteristics of such corrective devices, are described in Sections 3 and 4

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of Chapter V.

The Gyroscope with Two Degrees of Freedom

By attaching an outer frame (Fig.102), to a gyroscope with three degrees of freedom, i.e., by depriving it of one degree of freedom (displacement about the yy axis; here $\beta = 0, \dot{\beta} = 0, \ddot{\beta} = 0$) we get a gyroscope with two degrees of freedom (Fig.112a).

Let us apply to the gyroscope the moment M_z acting about the zz axis.

If the gyroscope had three degrees of freedom, then the external moment would be balanced by the gyroscopic moment, and a precessional moment would appear about the axis yy, in accordance with the equation

$$M_{zG} = -H \dot{\beta}$$

Since in our case $\dot{\beta} = 0$, there is no gyroscopic moment with respect to the

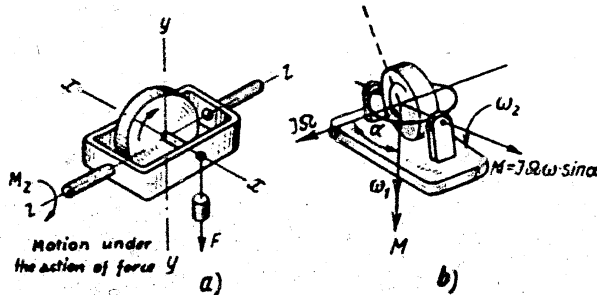


Fig.112 --Gyroscope with Two Degrees of Freedom

zz axis. Consequently, under the influence of the external moment M_z , a gyroscope with two degrees of freedom, in spite of the existence of a natural rotation, will behave like any other body, i.e., it will rotate with acceleration about the zz axis.

Let us now suppose that no external moments about the zz axis act on a gyro-

scope with two degrees of freedom, i.e., that $M_x = 0$, and that the whole instrument is installed on a moving base, which rotates about the yy axis at the velocity ω . In this case the gyroscope will, as it were, recover its lost degree of freedom.

In this case, the external moment M_y , coinciding in direction with the angular velocity of rotation of the base, is applied to the gyroscope.

A precessional moment about the zz axis now arises and tends to bring the vectors H and M_y into coincidence by the shortest route.

The zz axis of rotation of the rotor tends to become parallel to the axis of rotation of the whole system so that the direction of rotation of the rotor Ω will coincide with the direction of ω .

Such a displacement of the gyroscope leads to a variation in the mutual position of the axis of rotation of the moving base and the proper axis of the rotor (Fig. 112b); the angle α between these axes varies with the rotation of the gyroscope in the range from $\frac{\pi}{2}$ to 0.

If we prevent the rotation of the gyroscope frame, the gyroscopic moment $M_{zg} = -H \omega \sin(90 - \alpha) = -H \omega \cos \alpha$ will act on the bearings, preventing the frame of the gyroscope from rotating. At a constant kinetic moment of the gyroscope, this moment is proportional to the angular velocity of rotation of the moving base and to the sine of the angle between the axis of the rotor and the axis of rotation. This peculiarity of a gyroscope with two degrees of freedom is utilized in instruments measuring the angular velocity of rotation.

Section 2.

MAIN PARTS AND ELEMENTS OF THE GYROSCOPE

In their structure, gyroscopic instruments are subdivided into gyroscopes with three degrees of freedom, and high-speed gyroscopes with two degrees of freedom.

Most gyroscopes consist of the following main units:

- 1) base or body of the instrument;

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- 2) rotor;
- 3) bearings;
- 4) gyroscope gimbals;
- 5) rotor motor, if the instrument is electric;
- 6) correcting devices;
- 7) arresting devices;
- 8) power source.

Let us consider in more detail the rotors and bearings, which are necessary elements of any gyroscopic instrument.

The rate of precession due to the action of the moments of external forces (friction in the bearings, in the contacts of the servosystem, etc.) depends on the magnitude of the kinetic moment H .

To reduce the rate of precession, the moment of inertia must be increased by increasing the weight of the rotor, and the distribution of the principal mass must be as far as possible from the axis of rotation, or the rotational speed of the rotor must be increased.

The possibility of increasing the rotational speed of the rotor is limited by the wear on the bearings, which at high rotor speeds (10,000 - 25,000 rpm) operate under severe conditions. For gyroscopic instruments, special types of ball bearings are used, which absorb axial and radial loads with equal facility, and have minimum clearances to avoid impairment of balance. The rotor bearings, which are usually called main bearings, operate under more severe conditions than the bearings of the gimbals, since the spindle resting on them rotates at a velocity many times that of the outer and inner frames.

For this reason, the rotor bearings have a shorter service life than the frame bearings.

The moments of friction generated at the main bearings and frame bearings have different effects on the gyroscope. The moments of friction in the main bearings

0 affect only the power expended to rotate the rotor, while the friction in the bear-
 2 ings of the suspension leads to the appearance of a precessional moment of the gyro-
 4 scope and reduces the accuracy of the instrument.

6 The displacement of the center of gravity of the gyroscope with respect to the
 8 various axes likewise has a different effect on the behavior of the gyroscope. The
 10 displacement of the center of gravity along the yy axis (cf. Fig. 102) does not pro-
 12 duce a moment. Its displacement along the zz axis by the quantity i causes the
 14 appearance of a moment $i Pa$ acting with respect to the xx axis and absorbed by the
 16 bearings of the outer gyroscope frame.

18 On displacement of the center of gravity of the gyroscope along the xx axis by
 20 the amount of the clearance $i c$, a moment $i Pc$ is produced relative to the zz axis,
 22 causing a precessional moment with respect to the yy axis at the angular velocity

$$\omega = \pm \frac{Pc}{H} \quad (V.9)$$

24
 26
 28
 30 Consequently, a clearance in the bearings of the gimbals is allowable, but in
 32 the main bearings it must be reduced to the lowest possible value.

34 All the gyroscope bearings must be precisely manufactured and must have a long
 36 service life. In addition the friction in the suspension bearings must be as small
 38 as possible, and there must be no axial clearances in the main bearings.

40 The material of which the bearing is made must be very hard and resistant to
 42 wear. In this respect, structural pearlite steels with 1 - 2% Cr, 1 - 1.7% C and
 44 not more than 0.06% P and S, have proved satisfactory.

46 ShKh-15 steel is used for the rigs.

48 According to the design of the inner frame, rotors are subdivided into open
 50 and closed types.

52 In closed-type rotors, the inner frame consists of a shell within which the
 54 rotor rotates. To maintain the rotor in continuous rotation at a constant angular
 56

velocity Ω , either a pneumatic or electric gyromotor is used.

The pneumatic gyromotor (Fig.113) consists of the rotor (2) with a rim on which a number of grooves are milled. Blasts of air from the two nozzles (1) attached to the shell, of which the inner frame consists, strike these grooves. The air brings the rotor into rotation. Such gyromotors were very widely used a few years ago and are still met in pneumatic gyro instruments. At the present time, however, electric gyromotors operating on direct or alternating current are widely used. In both cases, they are inverted type electric motors, that is, the stator is located inside the rotor; this gives an increased moment of inertia and consequently also an increased kinetic moment.

Direct-current gyromotors are convenient in that they can be directly connected to the aircraft DC system and have only two supply leads. A shortcoming of DC gyromotors is the rapid wear of the friction parts (collector and brushes), sparking, etc.

Alternating-current gyromotors (Fig.114) are more often used; they require no lead to the rotor, having a short-circuited squirrel-cage type winding and no collector.

Alternating-current gyromotors require a special three-phase generator.

Pendulum, induction, electromagnetic, or other correctors are used to stabilize the rotor axis with respect to the earth. The feed is through the bearings to the stator winding. The current-input devices must not produce an appreciable additional moment causing precession of the gyroscope, and must not limit the displacement of the gyroscope with respect to the axes of suspension.

When the range of displacements of the gyroscope is limited, so-called momentless filaments, i.e., spiral springs with a very low moment, may be used.

Supply leads are often designed according to the diagram shown in Fig.115, b. Such leads do not restrict rotation, but they can be used only with a two-wire feed system in the form of axial contacts at both ends of the rotor shaft. When

displacement is limited, several axial contacts may be used.

So-called comb contacts are often used in three-phase motors (Fig.115 b).

The pneumatic instruments receive their feed from vacuum or pressure pumps,

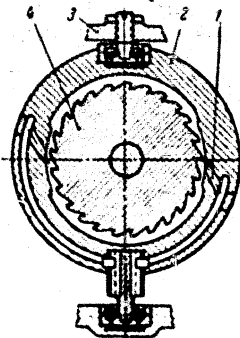


Fig.113 - Pneumatic Gyromotor

1 - Nozzle; 2 and 3 - Gyroscope frames; 4 - Rotor

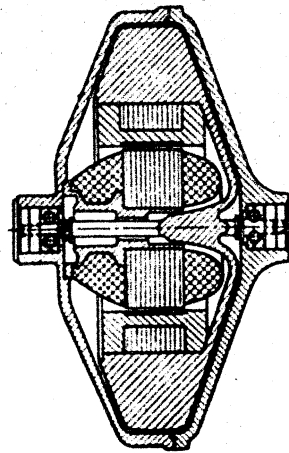


Fig.114 - Alternating Current Gyromotor

35 Venturi tubes, etc.

40 Electric gyroscopes are fed either directly from the aircraft DC system, or
41 across a converter yielding higher frequency three-phase current (at about 500
42 cycles).

46 The accuracy of the readings of gyro instruments depends to a considerable ex-
48 tent on the relation between the moments of friction in the bearings of the frames
50 and the moments of the correcting devices. The moment of the correction devices
49 must be 3 to 5 times as great as the sum of the moments of friction. To reduce the
54 friction in the bearings, vibrating, rocking, or rotating bearings, as well as
56

elastic bearings, are used.

Since the rotor of a gyromotor rotates at very high speed, great care must be used in its manufacture.

The rotor, and thus the weight of the gyro unit, is exactly balanced, since any unbalance leads to an additional load on the bearing and causes an error in the

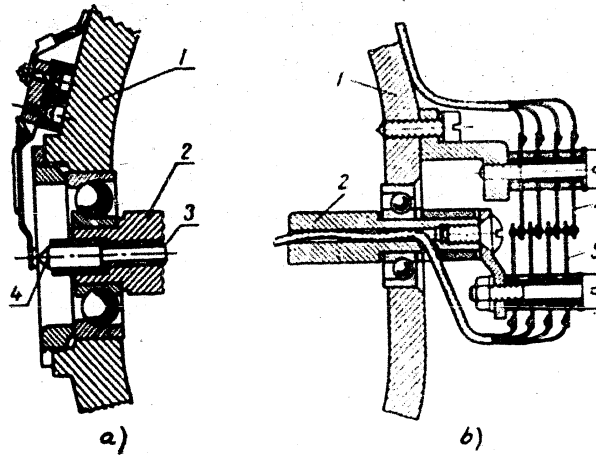


Fig.115 - Current Leads to a Gyromotor through the Bearings

a - Axial lead; b - Comb lead

1 - Frame of gyroscope suspension; 2 - Shaft; 3 - Insulation; 4 - Contact;
5 - Fixed contact plates

readings of the instrument. The rotor must be made of a material of high specific gravity which is uniform and strong, since considerable stresses arise in the body of the rotor when it rotates at high speed. Brass, steel, and bronze are usually employed.

If the circular velocity V of the rotor, the radius R and the density of the material ρ are known, then the stress σ in the circular rim of the rotor is found

from the conditions of equilibrium between the centrifugal force $F = \frac{mv^2}{R}$ and the elastic tensile forces $P = \sigma S$. An element of the rim with an infinitesimal central angle $d\alpha$ is separated. The mass of this element is $m = \rho SR d\alpha$ (Fig. 116).

On projecting all forces in the direction of the force, F , we obtain the equation of equilibrium in the form

$$F = 2P \cos\left(90 - \frac{d\alpha}{2}\right) = 2P \sin \frac{d\alpha}{2} \approx P d\alpha. \quad (V.10)$$

In view of the smallness of the angle $d\alpha$, we may take the quantity $\frac{d\alpha}{2}$ instead of $\sin \frac{d\alpha}{2}$. By substituting the values of P and F in eq. (V.10) we get

$$F = \frac{mV^2}{R} = \rho S V^2 d\alpha = \sigma S d\alpha;$$

then

$$\rho S V^2 d\alpha = \sigma S d\alpha.$$

whence

$$\sigma_1 = \rho V^2.$$

Since the material of the rotor, and consequently the assigned allowable stress σ_b is known, it follows that the maximum allowable circular velocity is

$$V_{max} = \sqrt{\frac{\sigma_b}{\rho}}. \quad (V.12)$$

In a rotor, having the form of a solid disk of a radius R and a constant thickness, rotating at a velocity V , the maximum tensile stresses will be concentrated at the center of the disk.

The stresses σ_2 at the outer periphery are equal to $\sigma_2 = \frac{1-\mu}{4} \rho V^2$, where

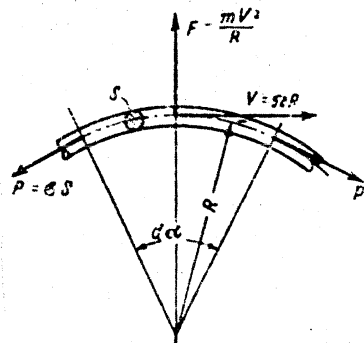


Fig. 116 - Tension Stresses in the Rotor Body

S - Area of rim cross section; P - Tensile force; F - Centrifugal force; V - Linear velocity; σ - Stress on rotor rim.



$\mu = 0.25 - 0.3$ is the Poisson coefficient; the stress at the center of the disk is

$$\sigma_1 = \frac{3+\mu}{8} \rho V^2. \quad (V.13)$$

If the rotor has a round opening of a radius r at the center, then the maximum tensile stresses σ_3 at the inner periphery are equal to

$$\sigma_3 = \frac{3+\mu}{8} \rho V^2 \left(1 + \frac{1-\mu}{3+\mu} \alpha^2 \right), \quad (V.14)$$

where $\alpha = \frac{r}{R}$.

In all cases considered, the value of the stress will be less than $\sigma = \rho V^2$ (since the coefficient for ρV^2 is less than 1), and only for a thin ring, when $R \approx r$, will the stress be $\sigma_1 \approx \sigma_3 = \rho V^2$.

This calculation does not allow for the strain in the body of the rotor created during stamping of the shaft nor for the varying thickness of the rotor.

In connection with the non-uniformity of the material and the inaccuracy of manufacture, the rotor may prove to be unbalanced both statistically and dynamically. Static unbalance is expressed in the failure of the center of gravity to coincide with the axis of rotation of the rotor, and to its being displaced from it by a certain distance ϵ (Fig.117).

On rotation of a statically unbalanced rotor, the centrifugal force $F = m\Omega^2\epsilon$ is generated, creating a pulsating load.

To eliminate this shortcoming, static balancing of rotors is used, detecting the unbalance on a special installation, and then drilling holes in the body of the rotor on the heavier side.

Dynamic unbalance is expressed in the production of a moment (couple of forces) by the centrifugal forces that appear on rapid rotation, even though the center of gravity of the body lies on the axis of rotation. Such unbalance may be obtained,

for example, as a result of a skewing of the rotor shaft during the stamping, as a result of nonuniform material of the rotor, etc. Rotors are first subjected to static balancing, followed by dynamic balancing.

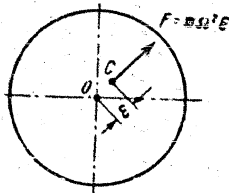


Fig.117 - Static Unbalance of Rotor

After static balancing, the rotor is fixed in the frame and set in rapid rotation. If vibration occurs, this indicates dynamic unbalance.

Plastilline or wax are used for balancing the rotor (Fig.118), after which holes are drilled at places opposite the point of application of the wax; the wax

is then removed. For balancing it is necessary that the weight Q of the metal removed by drilling satisfies the equation $Qr_1 = Pr_2$, where P is the weight of the plastilline and r, l, R, L are the dimensions shown in Fig.118. The static and dynamic unbalance are tested on special instruments.

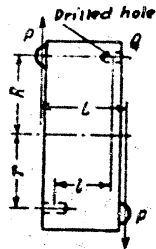


Fig.118 - Dynamic Balancing of Rotor

As already stated, the increase of the kinetic moment of the rotor is obtained by increasing the moment of inertia and the rate of rotation.

The increase in the rate of rotation is limited mainly by the quality of the ball bearings.

It must be remarked that, in a pneumatic instrument operating in a vacuum, the variation of the parameters (p, T , etc.) of the atmosphere with height have a great influence on the operation.

If the instrument operates on compressed air, the form of the blades of the miniature turbine and the guide nozzles (the body of the rotor may be considered a

miniature turbine) exert a great influence on the rate of rotation of the rotor. In the electric feed not only the power supplied is extremely important, but also the method of manufacturing the electric motor that turns the rotor, which affects the rotor efficiency. In designing a new gyroscopic instrument, the designer in most cases does not calculate the gyromotor, but is guided by the standards in force at the plant.

In designing gyro instruments it is necessary to allow for the ventilation losses, i.e., losses due to friction between the rotor and the air, on which as much as 80-95% of all the energy required by the gyroscope may be expended.

To reduce these losses, the rotor is given a lenticular streamlined shape, polished, and placed in a special chamber whose inner surface is likewise polished.

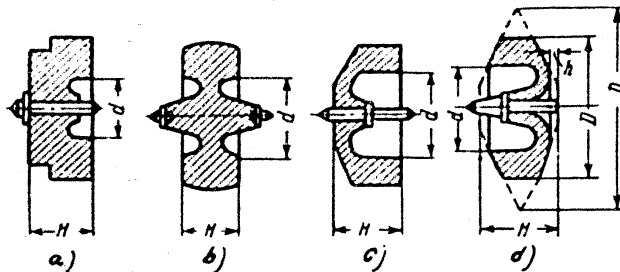


Fig.119 - Rotors of Gyroscopic Instruments

Figure 119 shows the most common type of rotors. The rotors shown in Fig.119 Fig.119 a, b are characteristic for pneumatic low-power gyroscopes. The rotors in Fig.119 b and c are sometimes used in low-power electric instruments. The rotor in Fig.119 d is characteristic of electric gyroscopes rotating at high speed.

The dimensions and characteristics of rotors of a few gyro instruments are given in Tables 3 and 4.

The rotors of pneumatic gyroscopes are made of aluminum-nickel bronze with a specific gravity of 7.7.

The rotors of electric gyroscopes are made of high-grade steel with a specific gravity of 7.8 - 8.2.

In designing a gyroscopic instrument, the specifications for it and the necessary accuracy of operation of the instrument, together with its dimensions, are prescribed.

Table 3

Characteristics of Rotors of Gyro Instruments

Model of Rotor	Weight in grams	Moment of Inertia in gm-cm-sec ²	Angular Velocity rpm	Kinetic Moment 1/sec	Instrument Type
1	700	3.5	20 000	2094	7 330 Anschuetz V-02-50
2	2 330	56	20 000	2094	117 000 Anschuetz three-rotor gyro
3	450	3	10 000	1050	3 150 Sperry aircraft horizon
4	400	2	13 500	1410	2 800 Electric turn-and-bank indicator

Table 4

Dimensions of Rotors

Rotor Model	D_n	H	$\frac{D_n}{H}$	D	$\frac{D_n}{D}$	h	$\frac{H}{h}$	$\frac{D_n}{h}$	$\frac{D}{h}$	d	$\frac{D}{d}$
1	84	81	1,038	60	1,40	11	7,36	7,64	5,45	35	1,77
2	146	62	2,355	130	1,122	3	20,65	48,6	43,35	51,5	2,525
3	-	26	-	55	-	-	-	-	-	15	3,67
4	-	35	-	55	-	-	-	-	-	40	1,37

Section 3

THE TURN-AND-BANK INDICATOR

When an aircraft makes a turn, the magnetic compass is deflected under the influence of the vertical component of the magnetic field of the earth to such an extent that, in some cases, the instrument completely fails to show the turn or even shows a turn in the opposite direction (turning error). For this reason, the magnetic compass is not used in aircraft turns and is replaced by a turn indicator, which is a compound instrument consisting of a gyroscope with two degrees of freedom and a pendulum bank indicator. The turn-and-bank indicator shows the rotation of the aircraft about the vertical axis as well as its sideslip.

The turn indicator (Fig.120) is a glass tube curved along a circle with a radius

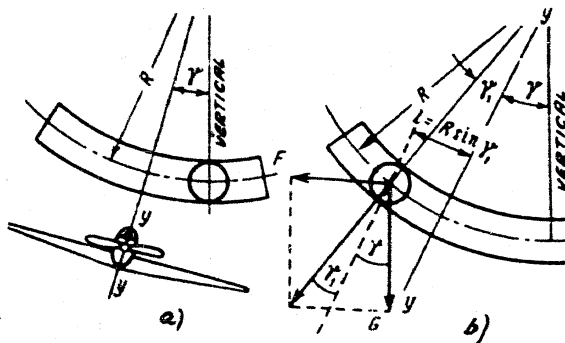


Fig.120 - The Bank of an Aircraft

- a - Operation of the turn-and-bank indicator in a regular turn;
 - b - Operation of the turn-and-bank indicator in a turn and bank.
- γ - Actual (absolute) bank to the right; γ_1 - Apparent bank to the left.

of R and filled with liquid; the tube contains a heavy ball moving freely inside it. The behavior of the ball is analogous to the behavior of a pendulum of the length R and mass $m = \frac{G}{g}$, where G is the weight of the ball and g the acceleration of gravity.

When the tube is inclined, the ball slides downward, and the line connecting the center of gravity of the ball with the center of gravity of the circumference of the tube coincides with the direction of the vertical. In a regular turn this line coincides with the direction of the apparent vertical, i.e., in a regular turn the ball will occupy the central position in the tube (Fig.120 a).

During a turn, the forces of gravity $G = mg$ and the centrifugal force $F = mV^2$ are applied to the ball (Fig.120 b). The displacement of the ball from the middle of the tube at an angle of sideslip γ_1 , is equal to $S = \frac{2\pi R}{360} \gamma_1$ i.e., the value of

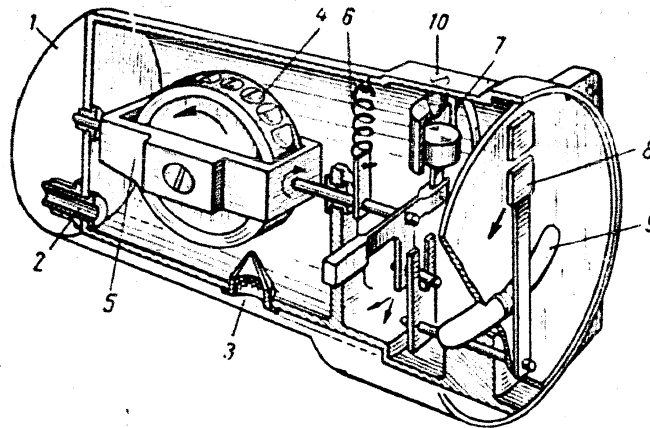


Fig.121 - Diagram of Turn Indicator

- 1 - Body; 2 - Pin; 3 - Port for admission of air; 4 - Rotor;
 5 - Frame; 6 - Spring; 7 - Damper; 8 - Pointer; 9 - Bank indicator;
 10 - Adjusting screw.

the displacement is proportional to the angle of slip and to the radius of the tube. The tube is filled with liquid to damp the oscillations of the ball that take place under the effect of accelerations. The damping of the ball is stronger the smaller the clearance between it and the inner wall of the tube (which is taken as about 0.5 mm), and the higher the viscosity of the liquid. Toluene is often used as a

damping liquid.

Figure 121 gives a schematic diagram of the turn indicator in combination with the bank indicator. The sensitive element of the instrument is a gyroscope with two degrees of freedom (rotation about the xx and zz axes shown in Fig. 112 a).

When the gyroscope is inclined about the longitudinal and lateral axes of the aircraft, no precessional motions occur. The rotation of the aircraft with respect to the yy axis causes a gyroscopic moment to appear and the frame of the gyroscope to rotate about the xx axis in accordance with the equation $M_y = -H\omega \cos \alpha$. The moment M_y is balanced by the moment developed by the steel spring (6) connecting the gyroscope frame with the body of the instrument. As a result of the action of these two moments, the instrument pointer (8) is deflected.

The moment developed by the spring is equal to $M_{sp} \approx c\delta$, where c is the moment developed by the spring on a 1° rotation of the frame, i.e., the coefficient of rigidity of the spring; and δ is the angle of rotation of the gyroscope frame about the xx axis. The gyroscopic moment is

$$M_x = -J\Omega \omega \sin \alpha = -H\omega \sin \alpha,$$

where J is the principal moment of inertia of the gyroscope;

Ω is the angular velocity of rotation of the rotor;

ω is the angular velocity of rotation of the aircraft;

α is the angle between the axis of rotation of the aircraft and the rotor axis.

In the equilibrium position,

$$J\Omega \omega \sin \alpha = H\omega \sin \alpha = c\delta.$$

or

$$c\delta = J\Omega \omega \cos(\delta - \gamma),$$

but

$$\cos(\delta - \gamma) = \cos \delta \cos \gamma + \sin \delta \sin \gamma.$$

In practice the angle δ of inclination of the frame does not exceed 15° ; we may therefore write

$$\cos(\delta - \gamma) \approx \cos \gamma + \delta \sin \gamma.$$

Since, in this case,

$$\cos \delta \approx 1, \quad \sin \delta \approx \delta,$$

we may write

$$\delta \dot{\phi} = J \omega \Omega (\cos \gamma + \delta \sin \gamma)$$

and

$$\delta = \frac{J \omega \Omega \cos \gamma}{c - J \omega \Omega \sin \gamma};$$

(V.15)

The angle δ of rotation of the gyroscope frame about the xx axis depends not only on the angular velocity ω of rotation of the aircraft, but also on the kinetic moment H of the gyroscope, the angle of bank γ and the elastic properties of the spring c . Since some of these values, for example the angle γ , may vary during flight and these variations are not taken into account by the instrument, the turn indicator shows only the direction of the turn but not the angular velocity of the aircraft.

In order to make the motion of the pointer smooth without sharp fluctuations, the frame of the gyroscope (5) is connected by a special tie rod with the damper (7), consisting of a cylinder rigidly connected to the body of the instrument, and a plunger connected with the gyroscope frame (5).

The wall of the cylinder has a capillary opening connecting the inner cavity of the cylinder with the atmosphere and covered by the adjusting screw (10). On sharp rotations of the frame, i.e., on rapid displacement of the plunger of the damper, the air cannot pass through the capillary, and a deceleration force, counteracting the oscillations of the gyroscope frame and, consequently, also those of the pointer

of the instrument, is produced. The degree of damping is regulated by varying the area of the cross section of the capillary opening by means of the adjusting screw (10), whose position may be changed by turning its head, which is located on the outside of the instrument body. This method is not the only way of producing a damping force.

The gyroscope rotor may be actuated both by a pneumatic or an electric motor. Recently, instruments with an electric gyromotor have come into wide use, and compound electric instruments are being used more and more, which, in a single case, combine a turn indicator, a gyro horizon, and a bank indicator.

Figure 121 shows the diagram of the pneumatic turn indicator. The rotor rotates in two radial ball bearings which are lubricated through a special opening in the instrument body. The distance between the centers in the ball bearings is regulated by means of special screws placed on the instrument body. The frame, together with the rotor, is carefully balanced.

The pointer (8) is attached to one axis with a yoke, provided with a counterweight for balancing the entire unit.

Since the instrument does not measure the rate of turn cf. [eq.(V.15)], its scale has only three divisions: the center one corresponding to a zero rate of turn, and two extreme divisions (without figures).

On the face side of the instrument the bank indicator, consisting of a pendulum, is mounted.

The position of the pointer of the turn indicator and the ball of the bank indicator, during various evolutions of the aircraft, are given in Fig.122.

In rectilinear flight and during a regular turn, the ball of the bank indicator occupies the central position. The pilot decides from the position of the bank-indicator ball whether the aircraft is sideslipping in flight. From the position of the pointer of the turn indicator, he judges the turns. The pointer occupies the central position in rectilinear flight, is deflected to the right on a right

turn and to the left on a left turn.

Principal Characteristics of the Pneumatic Turn Indicator

Diameter of case	80 mm
Length of case	115 mm
Moment of inertia of rotor	0.5 gm-cm-sec ²
Rigidity of spring	160 gm-cm/rad = 2.8 gm-cm/deg
Rate of rotation of rotor (at normal vacuum equal to 50 mm Hg)	7000 - 8000 rpm
Angle of stagnation	± 1°
Deflection of pointer at turning rate of 6° per sec (360° per min)	26 - 30°
Radius of curvature of bank indicator tube	140 mm
Diameter of ball	11 mm
Weight of instrument	650 gm

Since the instrument has only one numbered division (at zero), the error of the instrument, and in particular, the scale error, is tested at the zero division.

The scale error must not exceed ± 1°.

The pointer oscillations at vibrations of the instrument at a frequency of 40 cycles and an amplitude of 0.15 mm must not exceed ± 1°.

Figure 123 gives the diagram of the electric turn indicator. The rotor of the gyroscope is rotated by means of a DC electric motor with parallel excitation. The rotor speed is 8000 - 10,000 rpm. The gyroscopic moment is balanced by a spiral spring, and oscillation of the pointer is prevented by an air damper of the plunger type (as in the pneumatic turn indicator).

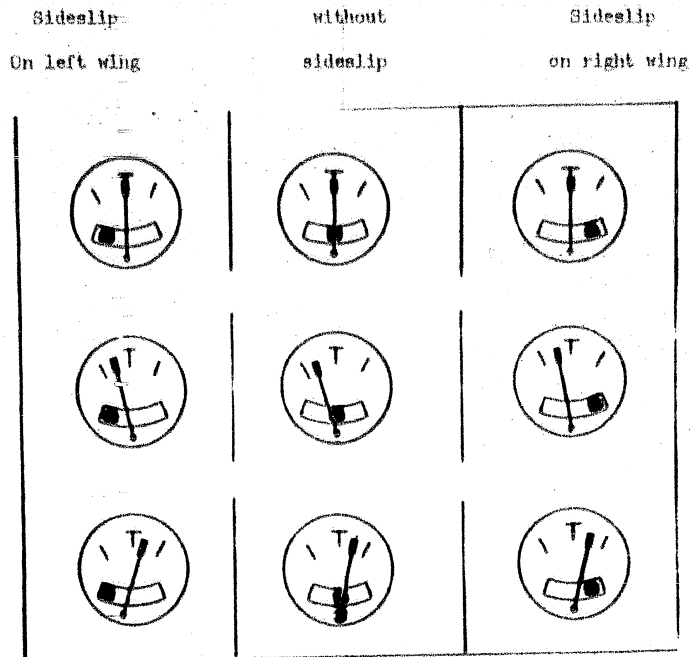
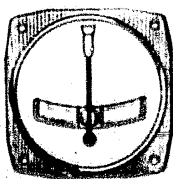


Fig.122 - Pointer Readings of Turn Indicator at Various Positions of Aircraft



- A - Rectilinear flight
- B - Left turn
- C - Right turn

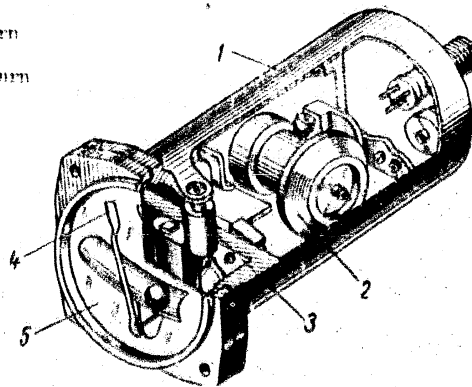


Fig.123 - Diagram of Electric Turn Indicator

- 1 - Body; 2 - Gyroscope unit; 3 - Damper; 4 - Pointer; 5 - Scale

Section 4

GYRO-RECORDERS

Gyroscopes with two degrees of freedom are used not only in turn indicators but also in gyro-recorders, i.e., in instruments for recording angular velocities.

Figure 124 gives a schematic diagram of the gyroscopic part of the instrument. The instrument records the value and direction of the angular velocity of the aircraft with respect to its three axes.

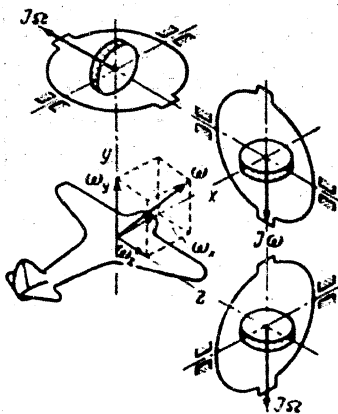


Fig.124 - Diagram of Gyro-Recorder
of Angular Velocities

Each of the gyroscopes registers the value and direction of one of the components of this velocity. The gyroscopes are therefore arranged, respectively, along the three aircraft axes.

As a result of the action of an external moment, proportional to one of the components of the angular velocity of the aircraft, the frame of the corresponding gyroscope is rotated (just as in the turn indicator).

For recording various angular velocities, the set of the gyro-recorder includes interchangeable frames, springs, and dampers.

The gyroscope rotors consist of DC motors (AC gyromotors may also be used). To ensure constancy of the angular velocity of the flywheel rotor, a centrifugal governor is used. The record is made in most cases on a paper chart by perforations or with special inks. The paper is moved by an electric motor. The time marker is most often provided in the form of an electromagnet which receives an impulse from an electric clock.

Section 5

GYRO HORIZONS

Principle of Operation of the Gyro Horizons

The gyro horizon is designed for determining the position of the aircraft with respect to the plane of the horizon. The instrument permits holding the aircraft in horizontal flight even when the natural horizon is invisible, or indicates the degree of bank and pitch on turns and changes of altitude.

The artificial horizon is a gyroscopic instrument with a pneumatic or electric drive for the rotor. Depending on the drive used in the rotor, a gyro horizon is called pneumatic or electric. The principal part of the instrument is a gyroscope with three degrees of freedom, which, as already stated, is indifferent to the position in which it is installed and may be used for determining an assigned direction. As is commonly known, a prerequisite for using an astatic gyroscope as a device for maintaining a given position constant, is to give it selectivity.

One of the most widely used methods of giving a gyroscopic instrument the necessary selectivity is the introduction of a radial correction*.

The action of the radial corrector is based on the following principle:

1. When the axis of a gyroscope deviates from the desired direction, a correction moment is applied to the gyroscope in such a way as to cause it to move in the same plane as the deviation in question, but in the opposite direction. In other words, the deviation of the gyroscopic unit is eliminated, and the gyroscope is returned to its initial position along the shortest radial direction. This is accomplished by rotating the plane of action by the correcting moment through 90° with respect to the plane in which the deviation of the gyroscope occurs.

* The term "radial correction" was introduced by B.V. Bulgakov, who performed the fundamental investigations on gyroscopic devices equipped with such correction devices.

2. The fact of the deviation and its value is established by comparison with a certain element possessing selectivity with respect to the position selected. This element is termed the sensitive element of the correction system. For example, when selectivity is imparted in the vertical direction, an ordinary pendulum is used; to give selectivity in a horizontal plane, magnets are often employed, etc.

In addition to the radial corrector, other systems of correction are also used in which the return of the gyroscope to the selected position may also take place over a distance that is not necessarily the shortest one.

The question is very often asked as to how the necessity of using a gyroscope is to be explained. Would it not be better to use only a sensitive element?

The point is that a gyroscope with a radial or other form of correction is a system that possesses not only selectivity but also stability in the selected direction (the precessional motion of the gyroscope takes place at a very low speed, while a sensitive element, for example, a pendulum, reacts very rapidly to the action of disturbing forces).

The gyroscope used in the gyro horizon is given selectivity with respect to the vertical by use of some system of correction, making this gyroscope a gyro vertical.

Figure 125 gives a schematic diagram of the pneumatic gyro horizon. In any positions of the aircraft, the principal axis of the instrument $I - I$ (the axis of natural rotation) maintains its vertical direction, so that the angle between the plane of rotation of the rotor and the longitudinal axis of the aircraft, xx , is equal to the absolute pitch θ , while the angle between this plane and the lateral axis of the aircraft, zz , is equal to the absolute bank γ (Fig.126).

An air correction with pendulum slide valves is used in the pneumatic gyro horizon. The instrument can operate either on vacuum or on pressure.

The air driving the gyroscope rotor passes through the bearings and channels in the outer frame (8) (cf.Fig.125), through the bearings and ducts in the inner frame (2), which consists of a closed shell, and in two jets, issuing from the

nozzles, mounted to the shell, strikes the grooves of the rotor (1), setting it in rotation.

The inner frame (2) in the lower part ends in the hollow cylinder with four windows (openings) (5) for the discharge of air, which are covered by the pendulum shutters (4). When the shell is in the vertical position, corresponding to the vertical position of the rotor axis, each shutter covers half of the opening of the

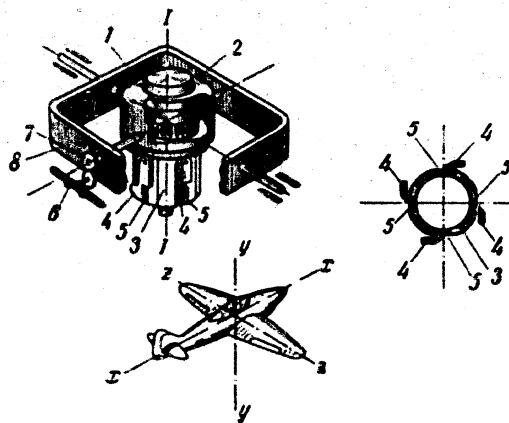


Fig.125 - Diagram of Pneumatic Gyro Horizon

1 - Rotor; 2 - Inner frame (shell); 3 - Correction chamber; 4 - Pendulum slide gate; 5 - Opening of correction chamber; 6 - Silhouette of aircraft; 7 - Gear transmission; 8 - Outer frame

window, and the air is ejected in four equal jets through these openings. The reactive pressures of the jets, in this case, are mutually balanced, no external moments are applied to the gyroscope, and its axis maintains its initial position (Fig.127,a).

The slide valves are arranged in pairs on the spindles, one of which is parallel to the xx axis, the other to the zz axis.

If, under the influence of any external moments, the gyroscope begins to precess and deviate from the vertical in any direction, for example, with respect to the z axis, while the slide valves remain in the vertical position, then the pair of openings of the correction at chamber (3) will now no longer be closed to the same

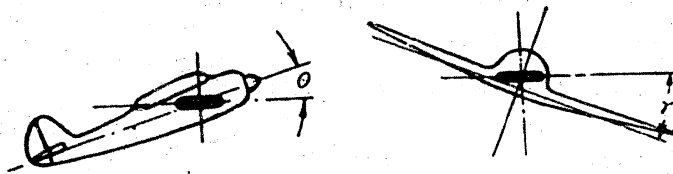


Fig.126 - Measurement of Angle of Bank and Angle of Pitch by Means of the Gyro Horizon

extent, i.e., one window will be open wider, while the opposite window will be closed more. The equilibrium of the reactive forces perpendicular to the rotor axis will now be disturbed.

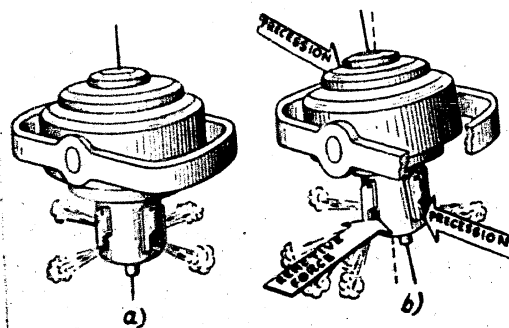


Fig.127 - Pendulum Pneumatic Correction for Gyro Horizon

a - Axis of rotor is vertical; b - axis of rotor deviates from the vertical

The resultant reactive force will provide a correction moment which will act

perpendicular to the direction of the gyroscope deviation.

Precession will now take place and return the axis of the gyroscope to the vertical position (Fig.127 b).

The sensitive element of the correction is a pendulum installed on the aircraft, which, under the action of accelerations, produces oscillations, causing the precession of the gyroscope. However, since the rate of precession is very small, the axis of the gyroscope will hardly deviate from the vertical. All gyro verticals operate on this principle, including gyro horizons. The most widely used designs for gyro horizons are the diving gyro horizon AGP and the electric gyro horizon of the AGK-47B type.

The AGP-2 Diving Gyro Horizon

The sensitive element of a gyro vertical is an astatic gyroscope whose rotor is actuated by their jets (cf. Fig.125). The axis of rotation of the rotor, which is made integral with the rotor, rests on two ball bearings installed in the shell (2) of the rotor. The shell, together with the rotor, is able to rotate with respect to the outer frame (8) which, together with the shell, is able to rotate freely in the body of the instrument at any angle. The shaft of the rotor is vertical. The axes of the outer and inner gimbals lie in the horizontal plane. In the AGP, the axis of rotation of the outer frame coincides with the lateral axis of the aircraft (in some designs of the gyro vertical, the axis of the inner frame coincides instead with the longitudinal axis of the aircraft).

The diving gyro horizon can operate on vacuum or on pressure.

From the upper part of the shell, the air enters the correction chamber through special orifices and from there enters the body of the instrument through the orifices (5), which are covered by pendulum shutters.

The body and frames are cast of silumin.

The rotor is a solid brass ring with a moment of inertia of the order of about

1.3 gm-cm-sec², and the rotor speed is 14,000 - 15,000 rpm.

The rotor rests on radial ball bearings which are lubricated by special inserts saturated with oil. The same bearings are installed in the frames. The clearance in the rotor spindle is selected by means of calibrated inserts, placed under the upper spring bearing.

A spring bearing is used to avoid possible compression of the aluminum frame by the steel axis of the rotor when the outside temperature rises, since the coefficient of temperature expansion of the frame is considerably greater than that of the rotor spindle. The lower bearings forms a tight fit.

The clearance in the suspension bearings are maintained by tightening the centers. To reduce the escape of air through the bearings, special air gaskets are installed in the form of condensing rings of German silver. The front face of the in-

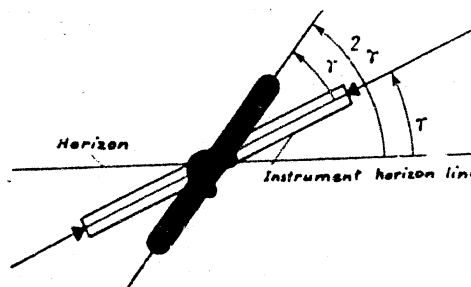


Fig.128 - Reading the Angle of Bank from the Position of the Aircraft Silhouette

Left: Horizon; Right: Instrumental horizontal line

strument case carries a screen with a luminous line representing the line of the horizon. Through a vertical slit in the screen passes the spindle of a movable index (the aircraft silhouette) placed between the screen and the glass of the instrument, and serving as indicator of the position of the horizon.

When the aircraft banks at the angle γ , the index pin of the horizon line like-

tilts at the angle γ . If the aircraft silhouette is attached directly to the spindle of the gyroscope unit, then, when the aircraft banks at the angle γ , the aircraft silhouette, together with the gyro unit, will remain fixed in space and indicate the bank at an angle γ on the opposite side with respect to the horizon.

To ensure correct reading of the bank of the aircraft, the silhouette of the aircraft is connected with the inner frame over a pair of gear wheels with a transmission ratio of 1:1 (in some instruments, a different design of transmission mechanism is used). When the aircraft banks at an angle γ , the aircraft silhouette on the scale of the gyro horizon rotates through an angle γ in the same direction (Fig. 128), while its position with respect to the index of the horizon line will correspond to the true angle of bank of the aircraft.

The right side of the instrument carries a scale attached to the outer frame, which is in the form of a ring with graduations within the limits of $\pm 90^\circ$; on this scale the angle of pitch is read off. Since, at high angles of pitch, the aircraft silhouette goes outside the face of the instrument, the scale is painted in two colors: blue for climb and brown for dive.

The diving gyro horizon allows the execution of aerobatics. To prevent the spindles of the frame and rotor from coinciding at banks over $75 - 80^\circ$, stops are placed above and below on the rotor shell.

The Errors of the Gyro Horizon

1) Errors due to Aircraft Accelerations. The accelerations produced in rectilinear flight, as well as those during turns, cause a deviation of the shutter slides of the correction and a precession of the gyroscope axis, which deflects the rotor axis from the vertical. Because of the low rate of precession, equal on the average to $6^\circ/\text{min}$, the gyroscope is unable to deviate significantly from the vertical if the action of the force of inertia deflecting the shutter is brief. In cases where the action of the inertia force is prolonged, the deflections of the gyroscope axis may reach considerable values.

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The magnitude of the deviation of the gyro rotor axis depends not only on the time of flight under acceleration but also on the nature of the acceleration. For example, at constant longitudinal acceleration, when one of the windows of the correction chamber is completely open, and when the increment of the flying speed is $\Delta V = 360 \text{ km/hr}$, the error in the position of the rotor axis may reach about 8° . If, however, the value of the acceleration at which the shutters are completely opened is variable, then this error may be greater than 10° .

If the flying speed is increasing, the direction of the acceleration will coincide with the direction of the velocity; in this case, the shutters will deviate in the opposite direction (backward). The rotor will deviate forward, away from the pilot, in such a way that the instrument indicates a climb of the aircraft although it is actually flying horizontally. On reduction of the flying speed the gyro horizon will show a descent.

Considerable errors in the reading of the instruments also arise in the case of a turn. After prolonged horizontal acceleration during a turn, the lateral pair of pendulum slides assume a position on the apparent vertical. As a result, the precession of the gyroscope takes place in the direction of the apparent vertical.

Since the lateral plane of the aircraft is rotated about the vertical during a turn, at a velocity equal to the angular velocity of the turn, the direction of precession of the gyroscope in space will also vary continuously, causing the position of the gyroscope, with respect to the vertical, to vary also. As an ultimate result, the axis of the rotor will deviate from the vertical forward in flight, and inside of the bank. The angle of deflection of the rotor axis from the vertical during a turn may reach $4 - 5^\circ$ or more. To reduce this error, the gyro unit of the gyro horizon is arranged in such a way that, at a vertical position of the slides and zero position of the pointer, the upper end of the rotor axis will be deflected from the vertical forward in the direction of flight by $2 - 3^\circ$.

During a turn, this longitudinal inclination of the rotor axis changes to a lat-

0 eral inclination, and at some definite value of the angular velocity of turning, it
 2 will completely compensate the precession of the gyroscope, while for all other val-
 4 ues of the angular velocity, the compensation will be only partial. The angular ve-
 6 locity of turning ω_c of the aircraft, at which the turning error is completely com-
 8 pensated, is determined by the relation

$$9 \quad \omega_c = \frac{\omega_0}{\beta_0} \quad (V.16)$$

11 where ω_c is the angular velocity of turning;

13 ω_0 the angular velocity of precession;

15 β_0 the angle of inclination of the rotor axis.

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 19 2) The errors caused by the flying speed of the aircraft, which appear during
 21 rectilinear flight as a result of the curvature of the earth's surface (while the
 23 aircraft rotates about its lateral axis zz), as well as the errors resulting from
 25 the diurnal rotation of the earth, have a negligible value, not exceeding 2 - 4",
 27 and are usually disregarded.

29 3) The instrument errors of the gyro horizon due to friction in the suspension
 31 bearings, friction in the shutter pins, vibrations, and unbalance, are reduced to a
 33 minimum by selecting a suitable design and by efficient work in the manufacture and
 35 assembly of the instruments.

37 For example, to reduce the moments of friction in the bearings of the frames
 39 acting in a direction opposite to that of the relative rotation, bearings of the
 41 vibrating, rocking, rotating, or elastic type may be used.

43 To reduce the influence of vibration on the gyroscope it is necessary that for
 45 each frame, the shaft of one bearing and the bearing of another act as supports;
 47 this causes the moments of friction in the bearings under vibration to have oppo-
 49 site directions so that not the sum but the difference of the moments of friction
 51 in the bearings will act on the gyroscope.
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The AGK-47B Electric Combination Gyro Horizon

The altitude-dependence of the rotor speed is one shortcoming of pneumatic gyro horizons, that is, the dependence of rotor speed on the variation in atmospheric pressure. Electric instruments fed from the aircraft electrical system do not have this drawback.

The AGK-47B gyro horizon consists of a combination of three instruments mounted in a single case:

- 1) a gyro horizon in the form of a gyroscope with three degrees of freedom, indicating the position of the aircraft with respect to the plane of the horizon;
- 2) a turn indicator in the form of a gyroscope with two degrees of freedom, indicating the rotation of the aircraft about its vertical axis;
- 3) a bank indicator, consisting of a pendulum indicating the sideslip of the aircraft in rectilinear flight and during turns.

Figure 129 shows the readings of the gyro horizon during various evolutions of the aircraft.

Figure 130 gives the kinematic diagram of the gyro horizon.

The gyro unit consists of a rotor installed in gimbals.

The axis of rotation of the rotor is inclined forward in the direction of flight by an angle of 2° . This reduces the errors of the gyro vertical during a turn.

The axis of rotation of the outer frame of the gimbals is parallel to the lateral axis of the aircraft, while the axis of the inner frame is parallel to the longitudinal axis of the aircraft.

Pitch and bank scales are attached to the face side of the outer frame; the aircraft silhouette attached to the gimbals serves as the horizon indicator; the index of the horizon line is attached to the instrument case.

Figure 131 shows the design of the AGK-47B gyro horizon. The unit (1), and the body of the gyromotor (2) form the gimbals. The gyromotor is a three-phase electric motor with a speed of 20,000 rpm. The stator is attached to the top of the instru-

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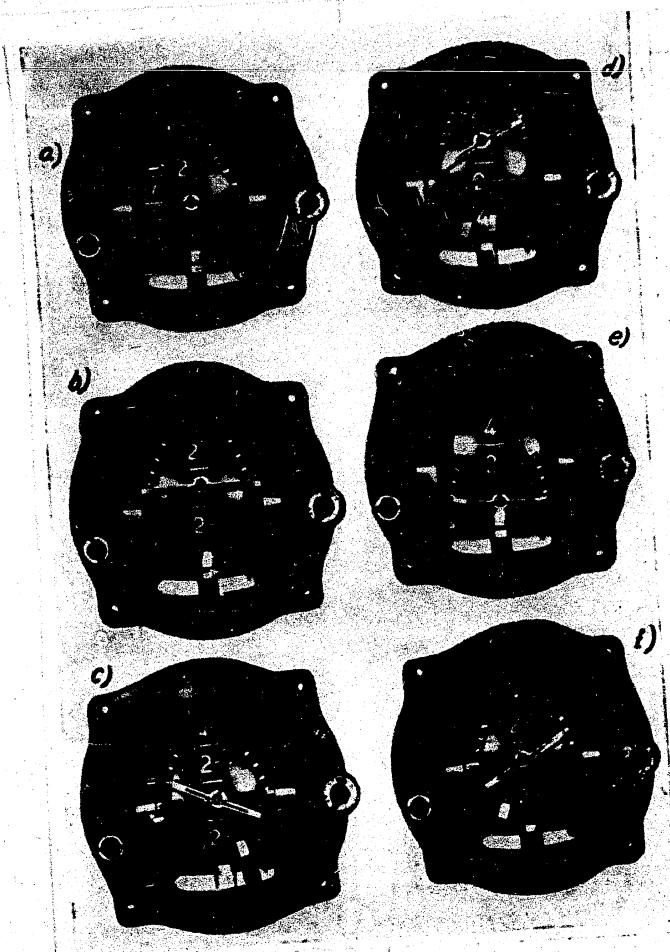


Fig.129 - Gyro Horizon Readings during Aircraft Evolutions
a - Horizontal flight; b - Climb; c - Right bank with sideslip; d - Left bank with climb; e - Glide; f - Left turn

ment. The stator block is made of dynamo iron, carrying, in twelve slots, the three-phase two-pole winding in star connection. The rotor block, which is likewise made of dynamo iron, has a short-circuit squirrel-cage winding, cast of aluminum alloy. The rotor turns in radial bearings with brass separators. The bearings are lubricated by a felt wick soaked with MVP oil.

Radial bearings are also used in the gimbals. The gyromotor and correction device are supplied with current across the gimbals, using contact rings and brushes. The block of the contact rings consists of three polished silver rings with insulated bushings, used instead of a wrist-pin bearing. The contact brushes consist of

flat rings, to the ends of which platinum-iridium alloy wires are soldered, which make contact with the silver contact rings. A brush pressure of 1 - 1.5 gm ensures reliable contact.

The position of the rotor axis of the gyro horizon is determined and maintained by the aid of a correcting device consisting of a liquid switch and two solenoids placed in the gyromotor body (Fig.132).

Each solenoid has two windings located to the right and left of its geometric center. An armature (core) is placed inside the winding and can be displaced along the axis of the solenoid. The correcting moment is produced by the weight of the armature, when it is displaced with respect to the center of the solenoid.

The magnitude of the moment depends on the magnitude of the displacement of the armature.

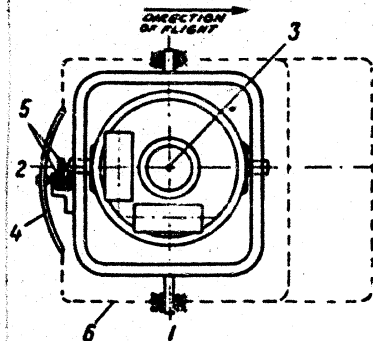


Fig.130 - Kinematic Diagram
of the Gyro Horizon

- 1 - Axis of outer frame of gimbals;
- 2 - Axis of inner frame of gimbals;
- 3 - Axis of rotor; 4 - Pointer of gyro horizon; 5 - Gear transmission;
- 6 - Case

of the armature, when it is displaced with respect to the center of the solenoid. The magnitude of the moment depends on the magnitude of the displacement of the armature.

0 The correction switch (Fig.133) consists of a copper vessel filled with an elec-
2 trically conducting liquid. Into the upper part of the vessel, which is made of an
4 insulating material, four red copper contacts are stamped. The assembled and ad-
6 justed switch is a self-contained unit which needs no further adjustment during oper-
8 ation or repair of the instrument. It is categorically forbidden to test the switch
10 with direct current, since this will put the switch out of order.

12 The static balancing of the gyro unit is performed by centering the cores in
14 the solenoids in such a manner that produces no moments with respect to the axes of
16 the gyroscope gimbals. Each of the solenoids has two windings. The windings of the
18 solenoids and the switch are connected with the switch of the correcting device ac-
20 cording to Fig.134.

22 The correcting device operates in the following way:

24 Any deviation of the rotor axis from its original vertical position, relative
26 to the lateral axis of the aircraft, results in an inclination of the switch with
28 respect to the horizontal plane, causing a change in the wetted area of the contact
30 surfaces along the longitudinal axis of the instrument. The forward contact (with
32 respect to direction of flight) is immersed deeper into the liquid than the rear
34 contact. These contacts are connected in the circuit of the windings of the sole-
36 noid, placed parallel to the lateral axis of the instrument. When the equality of
38 the resistances of the circuits of the solenoid windings is disturbed, a redistri-
40 bution of the currents flowing in the solenoid windings occurs, and the armature is
42 displaced toward the winding through which the greater current flows. In its new
44 position, the solenoid armature produces a moment acting on the gyroscope and caus-
46 ing its rotor axis to return to its original position.

48 A rotation of the rotor axis about the longitudinal axis of the instrument
50 causes a change in the liquid coverage of the conductors of the switch contacts a-
52 long the lateral axis.

54 In the original calibration of the gyro horizon at the factory, the caging mech-
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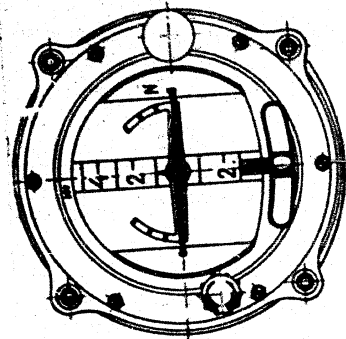
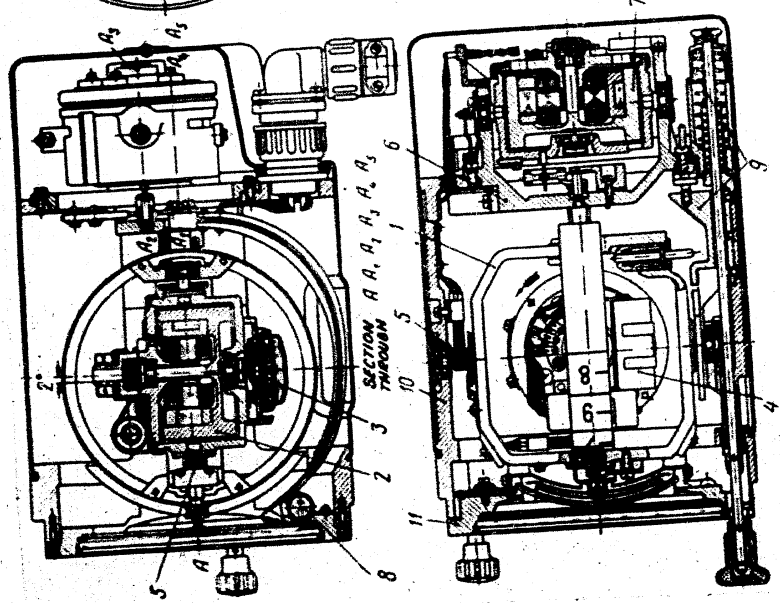


Fig.131 - Design of the ACK-47B.
1 - Gimbal unit of gyro horizon; 2 - Gyromotor of gyro horizon; 3 - Switch of correction device; 4 - Solenoids; 5 - Leads for gyromotor; 6 - Turn indicator; 7 - Gyromotor of turn indicator; 8 - Bank indicator; 9 - Caging mechanism; 10 - Case; 11 - Rear cover.



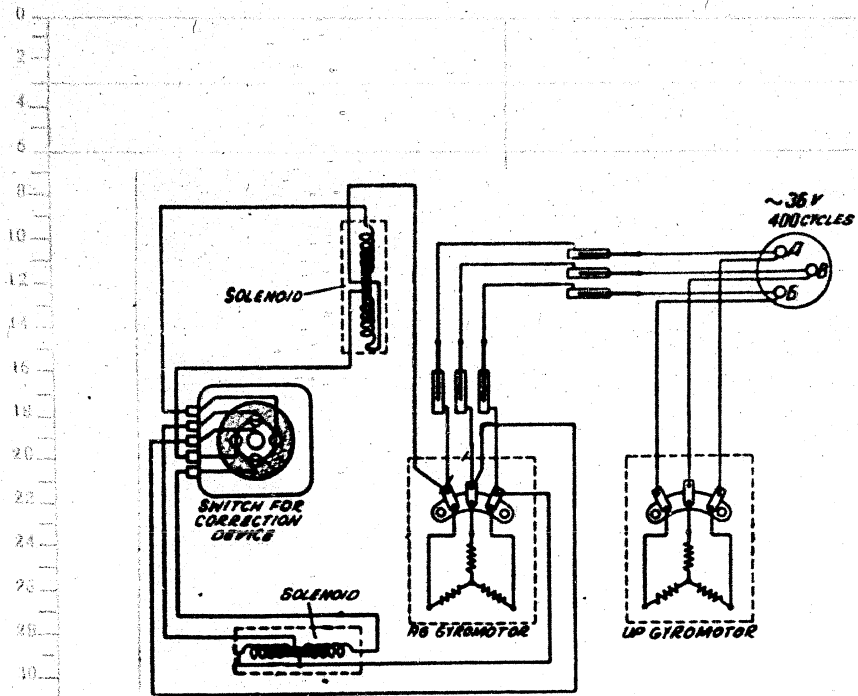


Fig.132 - Circuit Diagram of AGK-47B

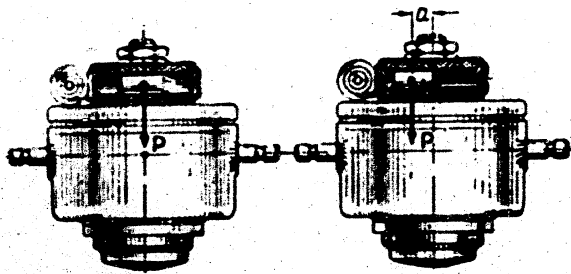
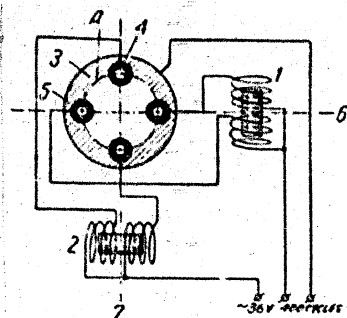


Fig.133 - Schematic Diagram for Generation of Correcting Moment

anism (9) is used for rapid elimination of errors in the readings after deviation from the vertical, and also for preventing damage in shipment (cf. Fig.131). This



device consists of three cams, push rods, pawls, a stop, working and return springs, and a signal blinker.

When the caging knob is moved, the force developed by the working spring, is transmitted through the push rod and cam

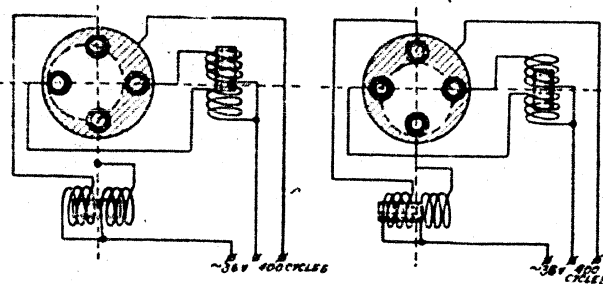


Fig.134 - Wiring Diagram of the Correcting Device

a - Central position; b - Rotation of the gyro-horizon axis about the lateral axis of the aircraft; c - Rotation of the gyro-horizon axis about the vertical axis of the aircraft.

A - Boundary of the liquid meniscus

1 - Solenoid of longitudinal correction; 2 - Solenoid of lateral correction; 3 - Switch; 4 - Switch contact; 5 - Insulating bushing; 6 - Longitudinal axis of instrument; 7 - Lateral axis of instrument

seated on the spring axle of the gimbals, to the gyroscope unit.

The profile of the cam is cut in a logarithmic spiral, thus causing the force

acting on the cam to create a moment about the outer axis of the gimbals. Under the action of this moment, the gyro unit begins to precess until the axis of rotation of the gyroscope coincides with the outer axis of the gimbals. In this position, the gyroscope loses one degree of freedom, its gyroscopic moment disappears, and the frame, under the pressure of the plunger, rotates freely about the outer axis of the gimbals until the plunger engages the cam slot. As soon as the plunger enters the

cam slot and fixes the gyro unit with respect to the outer axis of the instrument, the arrest of the gyro unit relative to the inner axis of the gimbals begins.

The turn indicator, which is designed as a separate and complete unit, is attached to the rear flange of the instrument case. The axis of rotation of the rotor is parallel to the longitudinal axis of the aircraft (Fig.135).

The gyroscopic moment of the gyroscope, which arises when the aircraft turns, is balanced by the spring (2). To extinguish the oscillations of the system the air damper (3) is installed. The degree of damping is regulated by a screw which covers the orifice by which the inner cavity of the damper communicates with

the atmosphere (as in the ordinary turn indicator).

The instrument is fed by a PAG-IF type converter. One converter can feed two instruments (for instance, two gyro horizons or one gyro horizon and a DGMK-2 remote-reading compass).

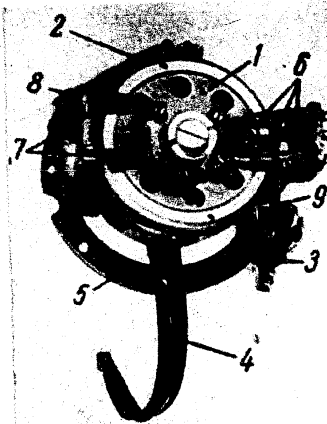


Fig.135 - The AGK-47B Turn Indicator

- 1 - Gyromotor; 2 - Spring balancing the gyroscopic moment; 3 - Damper;
- 4 - Pointer; 5 - Frame; 6 - Leads;
- 7 - Bearings; 8 - Adjusting lever;
- 9 - Damper lever

The gyro horizon causes considerable distortion in the character of the magnetic field in the space surrounding it; therefore magnetic compasses must not be installed closer than 200 mm to the AGK-47B gyro horizon.

Total reading error of the AGK-47B gyro horizon:

in rectilinear flight	not more than 1°
after coming out of a turn with a 20° bank at a flying speed of 400 km/hr	not more than 2°

The principal characteristics do not depend on the flying height.

Electric gyro horizons have come into wider use. There is a great variety in the designs of these instruments.

In principle their operation does not differ from that of the AGK-47B. Only the design of the gyro unit and the method of correction differ.

The primary advantage of the air radial correction is its simplicity. This explains the attempts to introduce it in electric gyro instruments.

A substantial shortcoming of the air radial correction is the fact that its efficiency depends on the flying height. When the rotor is fed by air this drawback is not too noticeable, since the rotor speed is reduced simultaneously with a reduction in effectiveness of the correction. When the instrument is electrically powered the speed of the rotor does not depend on the flying height and the lowering of the effectiveness of the correction makes it too sluggish at high elevations.

Consequently, the transition to electric rotors whose speed can be considerably higher than that of pneumatic rotors and is practically independent of the flying height, requires the development of electrical methods of correction.

Induction correction may serve as an example of electric correction.

Figure 136 a gives the induction diagram of the electric correction of one of the frames of a gyro vertical. The signal pickup consists of the contact roller (1), connected with the frame being corrected (2) and the contact plate (3), connected with the sensitive element of the correction and divided into two parts by an insu-

lating gap. When the pickup is in the coinciding position, the contact roller is in the center of the insulating gap. When it is in disagreement, the roller is displaced from the center.

A system of two induction coils (4) and (5) is used as pickup for the correction moment. Slots in these coils engage the conductor disk (6), connected with the

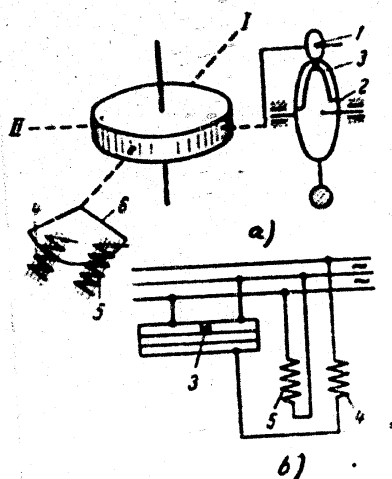


Fig. 136 - Diagram of Electric Inductive Correction

- a - Schematic diagram; b - Circuit diagram; I, II - Axes of gimbals;
- 1 - Contact roller; 2 - Frame to be corrected; 3 - Contact plate; 4, 5 - Induction coils; 6 - Conductor disk

axis of rotation of the corresponding frame. One of the coils constantly carries the voltage of one of the phases of a three-phase alternating-current line

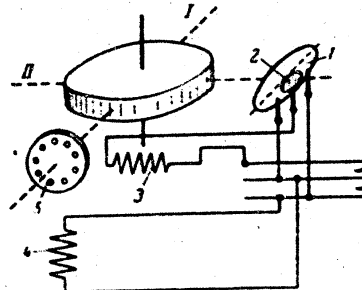


Fig. 137 - Diagram of Electric Correction with Mercury Switches

- I and II - Axes of gimbals;
- 1 - Ampoule; 2 - Mercury; 3 and 4 - Winding of induction motor; 5 - Short-circuited rotor of induction motor.

(cf. Fig. 136 b). The second coil is connected across the signal pickup to one of the other two phases of the line, depending on which half of the contact plate makes contact with the contact roller. In this way, as soon as the contact roller touches the insulating gap, i.e., when the signal pickup is in the coinciding state, current

will flow through only one of the coils, and there will be no moment on the conductor disk. When the roller passes to one side of the plate, current will flow through both of the coils. There is a 120° phase shift between the alternating magnetic

fluxes produced by the coils. On the conductor disk, as a result of the interaction of the eddy currents and the magnetic fluxes, which are phase-shifted with respect to each other, a moment is produced, constant in magnitude and having a sign depending on the side of the plate on which the roller is located. The same type of correction can be obtained by mercury switches (Fig.137). The signal pickup, in this case, is an ampoule with a droplet of mercury having three contacts and so located that the plane of its rocking coincides with the plane of rocking of the frame to be corrected. When the position of the gyroscope axis and the pendulum coincide, the mercury droplet is in the center, while when they are in disagreement it is in one of the extreme positions. In the latter case, the mercury droplet connects two contacts, thus actuating one of the extreme contacts. The moment pickup is an induction motor with a short-circuited rotor mounted to the axis of rotation of the corresponding

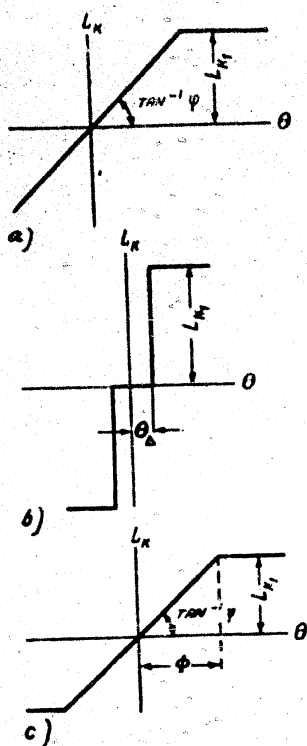


Fig.138 - Characteristics of Correction Device

L_K - Correcting moment; θ - Angle of discrepancy in signal pickup; ϕ - Sector of proportional part; L_{K1} - Moment of correction of constant part of characteristic; φ - Steepness of characteristic; θ_{Δ} - Zone of insensitivity

(i.e., diagonally opposite) frame to be corrected. One of the phases of the motor is always supplied by voltage from one of the phases of the three-phase feed line.

When two contacts are closed by the mercury, the second phase of the motor is turned on, causing a rotary field there. The direction of this field depends on which of the extreme contacts is closed by the mercury switch. In this way, a moment of either sign and of constant magnitude, is developed on the rotor.

Gyroverticals include widely varying devices for performing the correction; these may be divided into three groups, according to their characteristics:

1. Proportional correction (Fig.138 a).
2. Constant correction (Fig.138 b).
3. Mixed correction (Fig.138 c).

Section 6

COURSE GYROSCOPES

The disadvantages of magnetic compasses and the impossibility of installing gyroscopic compasses on an aircraft, owing to their large size, has led to the creation of so-called directional gyro turn indicators (DGT), gyromagnetic compasses (GMK) and, finally, to the creation of remote-reading gyromagnetic and induction compasses.

THE DIRECTIONAL GYRO TURN INDICATOR

As with gyro horizons, there are both pneumatic and electric directional gyro turn indicators in existence. The pneumatic type consists of an astatic gyroscope with three degrees of freedom, mounted in gimbals. To maintain its perpendicular position, the special flanges (2) are provided between the principal axis of the gyroscope and the axis of the outer frame of the rotor (1) of the instrument. The air jet from the rotor, striking against these flanges, produces a correcting moment (Fig.139).

Since the astatic free gyroscope has no directional force, the axis of the gyro-

scope, in flight, begins to deviate more and more from the initial direction, mainly under the influence of friction in the suspension bearings. The instrument readings will be correct for 10-15 minutes. A correction of the instrument is then necessary,

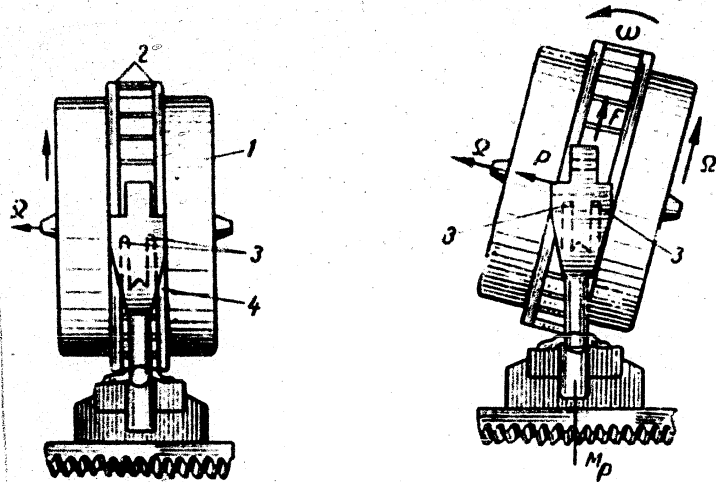


Fig.139 - Rotor of a Pneumatic Directional Gyro Turn Indicator

1 - Rotor; 2 - Flanges; 3 - Nozzles; 4 - Tubes conducting air to the nozzles 3; Ω - Angular velocity of rotor; F - Air jet spinning the rotor; p - Pressure of jet on flange caused by the moment M_p and the precession

and is made manually by the pilot.

The term gyro semicompass is also used for such an instrument and is explained by the fact that this instrument cannot completely replace the compass; the gyro semicompass provides only a possibility of holding the course with an accuracy within $2 - 3^\circ$ for a limited time interval (10 - 15 min).

Figure 140 gives the diagram of a pneumatic gyro semicompass. The rotor is driven by an air jet issuing from two nozzles, one on each side of the rotor. The moment of inertia of the rotor is about 0.8 gm-cm-sec^2 , and its speed is 10,000 to

12,000 rpm. The rotor rests on ball bearings mounted in the inner frame. The outer and inner frames likewise rest on ball bearings. The scale of the instrument (2) is attached to the outer frame (3) of the suspension. The course line from which the readings are taken is placed on the glass mounted in the instrument case.

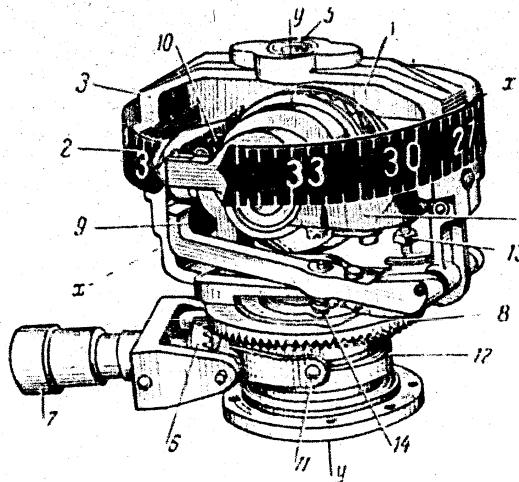


Fig.140 - The Gyro Unit of the Directional Gyro Turn Indicator

1 - Rotor; 2 - Card; 3 - Outer frame; 4 - Inner frame; 5 - Bearing; 6 - Small cog wheel; 7 - Caging knob; 8 - Frame cog wheel; 9 - Caging lever; 10 - Nozzle; 11 - Caging fork; 12 - Caging bushing; 13 - Balancing stud and nut; 14 - Spring plunger

The stop lever (9) is attached to the outer frame and is actuated by the caging knob (7). When this knob is pushed "in", the yoke (11) engages the conical cut-out in the face of the cog wheel (8); in this case the yoke is set in the horizontal direction and lifts the bushing (12), turning the lever (9) upward. The lever (9) comes to rest on the inner frame and does not allow it to be displaced. At the same time the cog wheel (6) engages the cog wheel (8), so that turning the knob (7) will

rotate the entire gyroscopic unit is rotated by the required angle. No precessional motion occurs when this is done, since the inner frame is fixed by the stop lever, that is, the gyroscope is deprived of one degree of freedom. When the knob is pulled "out", the gyro unit is released and the instrument begins to operate.

In most cases, after setting the aircraft on the required course, the pilot sets the gyro card, mounted to the inner frame, to zero division, and then only observes the deviations. This relieves him of the necessity for remembering the flight-path angle.

Errors of Directional Gyro Turn Indicators

Error due to Diurnal Rotation of the Earth. The principal axis of the free gyroscope maintains a constant direction in space; in this case, owing to the rotation of the earth, an apparent motion of the gyroscope about the xx and zz axes occurs (Fig. 141). The angular velocity of the apparent motion depends on the geographic latitude and longitude of the place and on the angle of inclination of the principal axis to the horizon; the

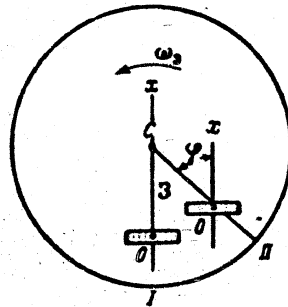


Fig. 141 - Apparent Motion of the Rotor of the Semicompass with Respect to the Earth

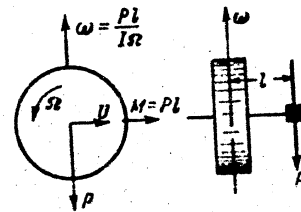


Fig. 142 - Compensation of the Earth's Rotation by Displacement of the Center of Gravity of the Rotor

angular velocity is equal in value to the vertical component of the velocity of rotation of the earth. For example, for Leningrad, the velocity of the apparent precession of the gyroscope due to the earth's rotation is equal to 13°/hour.

The gyro unit of the semicompass is balanced so as to eliminate, as far as possible, the apparent motion of the free gyroscope about its axis. This is accomplished by displacing the center of gravity of the rotor by a certain value l with respect to the center of the rotor (Fig. 142) under the action of the weight P . If the magnitude of the moment $M = Pl$ produced by this weight is so selected that the precession due to this moment is equal in magnitude but opposite in direction to the apparent motion, there will be no errors in the instrument readings. This motion, however, can be completely compensated only for the given latitude and at a constant rotor speed, since the rate of precession, due to the fact that $M = Pl$, is determined by the relation

$$\omega = \frac{Pl}{J\Omega} = \omega_e \sin \varphi. \quad (V.17)$$

where φ is the latitude;

ω is the rate of precession;

ω_e the velocity of rotation of the earth.

2. Error due to Variation in Latitude. This error appears in flights at latitudes different from that for which the error due to the earth's rotation had been eliminated by balancing. For example, a semicompass balanced for Leningrad will deviate by 2.4° /hour in flights to the Crimea.

3. Errors due to Variation in Longitude and Latitude (without allowing for the rotation of the earth). In flights along a parallel, the position of the gyro axis in space will vary by the value

$$\Delta\psi = \frac{S}{R} \operatorname{tg} \varphi. \quad (V.18)$$

if the influence of the earth's rotation is neglected (Fig. 143). For example, in a flight along the 60° parallel of latitude, the readings of the semi compass will be $1^\circ 56'$ too low for a distance of 100 km eastward.

A variation in latitude causes a variation in the vertical component of the

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angular velocity of the earth's rotation. For this reason, if a semicompass is balanced for a certain latitude φ , there will be an error in its readings due to unbalance at all other latitudes. This error is relatively small (of the order of about $2.4^\circ/\text{hour}$).

4. Error due to Friction in the Bearings. This error may reach $3 - 5^\circ$ in 15 min, which necessitates frequent corrections of the semicompass readings with another instrument.

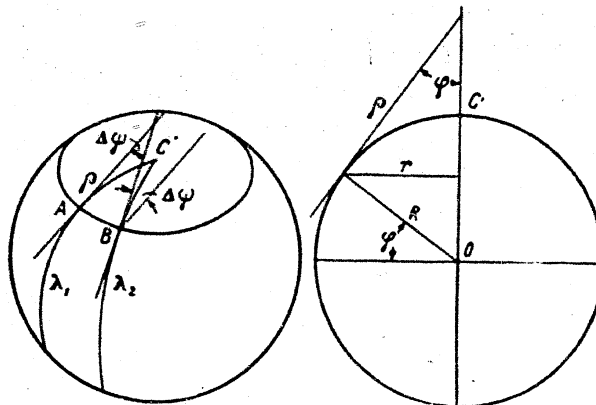


Fig.143 - Error Due to Variation in Longitude

R - Radius of earth; r - Radius of parallel; φ - Latitude of the place;
 C - North Pole; A, B - Two points at the same latitude; λ_1, λ_2 , Longitudes
of these points; $\Delta\psi$ - Angle between the meridians of the points A and B ;

$$AB = r (\lambda_1 - \lambda_2).$$

5. Errors due to Unbalance of the Suspension.

6. Cardanic Error. The axis of the outer frame to which the card is mounted is attached to the instrument case, parallel to the vertical axis of the aircraft. When the aircraft is inclined, this axis and the gyro card are inclined together

with the instrument, while the position of the rotor axis remains constant. This leads to a geometric error in course reading, called the Cardanic error.

Principal Characteristics of a Directional Gyro Turn Indicator

Moment of inertia of rotor	0.8 gm-cm-sec ²
Moment of friction in bearings	0.3 gm-cm;
Speed of rotor	12,000 rpm
Maximum deviation in 15 min	3 - 5°
Weight	1.5 kg
Allowable pitch and bank	not over 45°

Like the gyro horizon, a gyro semicompass can operate either on pneumatic or electric power.

Recently electric gyro semicompasses have come into wider use.

The gyro semicompass is used as the sensitive element of an automatic course indicator, which is one of the principal units of an automatic pilot.

Gyromagnetic Compasses

The readings of the gyro semicompass must be periodically corrected by the readings of a magnetic compass, selecting for this a time when the magnetic compass has the smallest errors. In the gyromagnetic compass this correction is performed automatically and continuously and converts the instrument from a semicompass into a compass.

Two types of correction by a magnetic compass are used.

1. Remote control, when a remote magnetic compass is used for the correction.
2. Direct control, when a magnetic system located on the gyroscope is used.

The former variant of the correction gives readings of higher accuracy, but the correction device, in this case, is bulkier. The latter variant gives the most compact size from the design point of view. The most successful design of a gyromagnetic compass, constructed according to the latter method, is the GM-2 gyromagnetic

compass proposed in 1932 by the Soviet designers D.A. Braslavskiy, M.G. El'kind, and M.M. Kachkach'yan.

The GK-2 Gyromagnetic Compass

Figure 144 gives a schematic diagram of the GK-2. The GK-2 gyromagnetic compass consists of a static gyroscope with three degrees of freedom, with its princi-

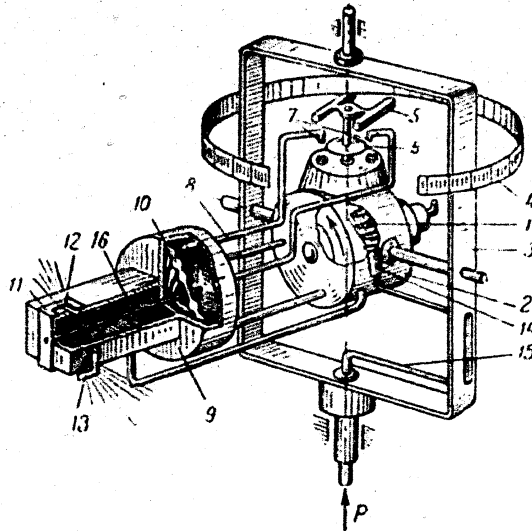


Fig. 144 - Schematic Diagram of the Gyromagnetic Compass

- 1 - Rotor; 2 - Rotor shell; 3 - Frame; 4 - Card; 5 - Magnet; 6 - Eccentric slide; 7 - Nozzle; 8 - Tubes; 9 - Pneumatic relay; 10 - Aneroid; 11 - Slide of magnetic correction; 12 - Orifice of correction chamber; 13 - Slide of pendulum correction; 14 - Rotor driven by nozzle; 15 - Air duct for rotor; 16 - Shaft

pal axis in the horizontal plane. The rotor (1) is enclosed in the shell (2), which is the inner ring of a Cardanic suspension. The shell is attached to the center

frame (3) whose axis of rotation coincides with the normal axis of the aircraft, yy. The outer frame carries a card graduated from 0 to 360°.

The shell (2) of the rotor carries the moving magnetic system (5) with the eccentric slide (6) attached to its axle, and covering the nozzle (7) of the magnetic corrector. The receiving nozzles are connected with the pneumatic relay (9) which is also installed on the rotor shell.

The pneumatic relay consists of an aneroid (10) whose inner part is connected with one of the nozzles; it transmits to its outer side the pressure from a second nozzle. The center of the box is connected over the shaft (16) with the slides (11), which cover the jet slide (12) and are located on the upper and lower parts of the chamber. In the neutral position of the aneroid, when the pressure difference is equal to zero, the slide covers the upper and lower orifice.

The two magnetic slides (13), which half-cover two side windows when the rotor shaft is horizontal, are also attached to the rotor shell.

The instrument is pneumatic, and its feed system is entirely analogous to the feed system of the gyro horizon.

After the rotor is started, a considerable part of the air enters directly into the correction chamber and only an insignificant part of it passes through the system of nozzles into the pneumatic relay. The magnetic correction operates in the following way:

If the axis of the rotor is parallel to the magnetic meridian, i.e., parallel to the axis of the magnetic system, the eccentric slide (6) covers the nozzles, the membrane occupies a neutral position, the actions of the air jets entering from the upper and lower orifices balance each other, and no new external moments are applied to the gyroscope.

If the rotor shaft deviates from the north-south direction, the nozzles of the pneumatic relay will be covered differently, the pressure difference on the aneroid will deform it and cause a displacement of the slide valve; in this case, one of the

orifices (12) of the nozzle will be covered more and the other less, the air jets will no longer balance each other and a moment will be applied to the gyroscope, causing its precession in a direction opposite to the original deviation of the rotor shaft. The magnetic correction, operating in the same way as in the gyro horizon, maintains the horizontal direction of the rotor axis.

In this instrument, a new version of the pneumatic radial correction device is used, in which the signal pickup is an air collector with two nozzles covered by eccentric slide valves, while the moment pickup is an air chamber with two pairs of windows, one of which provides a correction in the vertical plane (pendulum) the other a correction in azimuth (magnetic).

Errors of the GMK-2 Gyromagnetic Compass

1. Deviations created by the influence of the gyromagnetic force of the air-

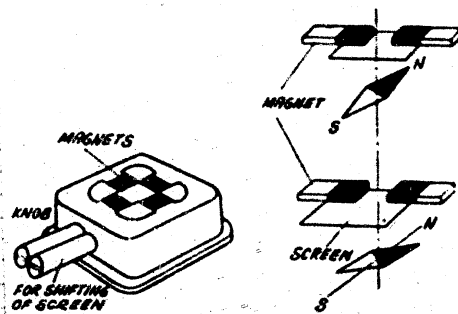


Fig.145 - Principle of Operation of the Deviation Instrument

craft on the magnetic part of the correction device; the deviation is reduced by the aid of a deviation instrument.

Figure 145 shows its principle of operation and its appearance.

The deviation instrument consists of four permanent magnets placed in the aluminum case of the instrument in the shape of a cross, with the south poles at the ends. One pair of magnets is parallel to the longitudinal axis of the aircraft, the other to the lateral axis. The magnetic

field of these magnets is closed by a screen of soft iron placed under the magnets.

The screen may be displaced in two mutually perpendicular directions. According to the location of the screen, the value and direction of the stray field will vary and will compensate the semicircular deviation of the instrument.

2. Stagnation in the magnetic and correction system; does not exceed 1° .
3. The banking deviation and rotational error are the same as in any magnetic instrument. The Cardanic error is the same as in the gyro semicompass.
4. Errors due to eddy currents; these arise in the rotor when it rotates in the magnetic field and interacts with the magnets, causing their deviation. To reduce the value of the currents, the rotor is milled perpendicular to its axis.

Principal Characteristics of the Gyromagnetic Compass

Moment of inertia of rotor	0.64 gm-cm-sec ²
Moment of friction in suspension	0.3 gm-cm
Stagnation	1°
Setting to meridian	accurate to 3°
Rotor speed	12,000 rpm
Rate of precession	10 - 15 rpm
Reactive moment of correction	2.5 gm-cm
Weight	about 2 kg
Period of rotation of rotor due to inertia	16 - 23 min
Deviation on inclination	not more than 5° in 2 min

Remote-Reading Gyromagnetic Compasses

Remote-reading gyromagnetic compasses are being more widely used in recent times (Bibl.12).

Figure 146 gives a schematic diagram of one of these remote-reading gyromagnetic compasses.

The DGMK remote-reading gyromagnetic compass is a gyro semicompass with readings corrected by a type PDK magnetic compass.

The full set of the instrument contains:

- a. a gyro semicompass;
- b. a type PDK magnetic transmitter;



- c. an amplifier;
- d. two indicators;
- e. a junction box;
- f. a knob for coincidence;
- g. a transformer.

On the axis of the outer frame of the free gyroscope (1) is attached the type PDK ring potentiometer (2) with two diametrically opposite points a direct-current voltage of 27 v is applied. Three brushes, 120° apart, slide on the potentiometer. The position of the brushes with respect to the conductors determine the values of the potential of each brush. Through the contact ring (9) and the current collecting springs the brushes are connected simultaneously with the potentiometer of the

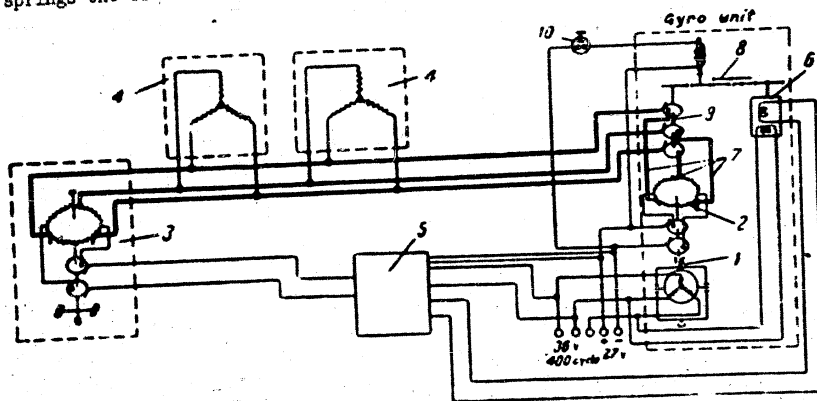


Fig. 146 - Schematic Diagram of the Remote-Reading Gyromagnetic Compass DGM-3
 1 - Outer frame of free gyroscope; 2 - Ring potentiometer; 3 - Magnetic pickup
 PDK-45; 4 - Indicators; 5 - Amplifier; 6 - Electric motor; 7 - Brushes; 8 - Re-
 ducer; 9 - Contact rings; 10 - Adjusting button.

magnetic pickup (3) and the two indicators (4).

If the angles between the brushes and the conductors of the gyro unit, and be-
 tween the brushes and conductors of the magnetic pickup, respectively, differ by 90°,

then the potential difference across the conductors of the transmitter is equal to zero. However, if the difference between these angles is not equal to 90° , a potential difference is produced across the conductors of the transmitter and is delivered to the input of the amplifier (5). The voltage tapped from the secondary winding of the output transformer of the amplifier is fed to the control winding of the electric motor (6) of the gyro unit, which rotates the potentiometer brushes of the gyro unit in the direction in which the angle of discrepancy will be reduced.

Thus, the readings of the indicator depend on the position of the brushes with respect to the potentiometer conductors of the magnetic pickup. In this case, the indicators show the compass course with an error of not over -2° .

A discrepancy between the position of the brushes of the potentiometer of the gyro unit and that of the magnet of the pickup may be due to either of two causes:

a. Precession of the gyroscope in azimuth.

Since the gyroscope is not corrected in azimuth it may precess under the action of the forces of friction and of a certain unbalance of the inner frame, which is always present. This precession of the gyroscope is compensated by the corresponding movement of the brushes of the potentiometer of the gyro unit. Since the rate of precession of the gyroscope, under no conditions, exceeds $1^\circ/\text{min}$, the precession of the gyroscope in azimuth cannot cause discrepancy in the system. In addition, this rate of deflection of the brushes also assures, regardless of the fluctuations of the magnetic system of the PDK pickup, that the indicators will show the mean compass course.

b. Shifting of the card of the magnetic pickup.

The dynamic errors of the magnetic system of the PDK, which arise under high load factors of the aircraft, cause a displacement of the card of the magnetic pickup, leading to discrepancies in the system. In this case the electric motor (6) of the gyro unit rotates the potentiometer brushes in a direction diminishing the discrepancy so that the readings of the indicator show an error. However, since the

high overloads, especially those created during turns, are usually of brief duration and since the gyroscope, during this time, maintains its position in space practically constant, while the rate of deflection of the brushes is small ($3^{\circ}/\text{min}$), the error for each minute of turn will not exceed 2.5° on the average.

The excessive errors inherent in the magnetic compass and the necessity for a precise determination of the aircraft course in solving a number of problems, force us to seek new solutions for determining the aircraft course. One such solution is the creation of instruments based on the direct measurement of the earth's magnetic field, i.e., the creation of induction compasses.

Remote-Reading Induction Compasses

The remote-reading induction compass consists of:

1. an induction pickup;
2. several of indicators (including one master indicator);
3. an amplifier;
4. an inverter;
5. a caging mechanism.

The induction pickup is a principal part of the instrument. It is placed in the aircraft wing or in the tail of the fuselage to isolate it from the interference of the ferromagnetic masses of the electric currents.

The sensitive element of the pickup consists of three pairs of rods (Fig. 147) made of a material with a high magnetic permeability (permalloy). Each pair of rods is provided with two windings, one of which serves for the magnetic excitation of the rods and the other for indicating the change in direction of the rod with respect of the earth's magnetic field.

The excitation windings are connected in series and are fed by 500-cycle alternating current.

The alternating magnetic flux produced by these currents in the rods of each pair have opposite directions. For this reason, they do not induce an emf in the

secondary windings but merely cause a change in the magnetic permeability of the rods.

Because of the periodic variation in the magnetic permeability, the horizontal component of the earth's magnetic field produces pulsating magnetic fluxes in the rods, leading to the appearance of an electromotive force in the secondary winding

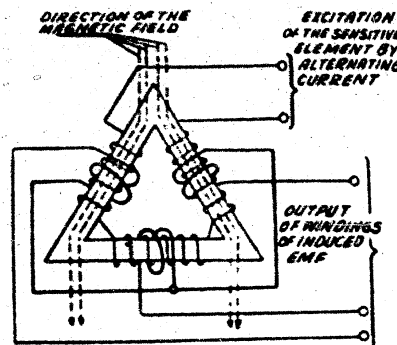


Fig.147 - Diagram of Sensitive Element of the Induction Pickup

of a magnitude which is a function of the arrangement of the rods with respect to the magnetic meridian.

The emf reaches its greatest value in the winding of the rod located parallel to the lines of the field (north-south). No emf appears in the windings perpendicular to the field (east-west) since their loops are not cut by the magnetic field.

To each course of the aircraft there corresponds a definite ratio of the emf of the three windings. This fact is utilized for measuring the aircraft course.

The sensitive element of the pickup is mounted on a gyroscope with three degrees of freedom. The body axis of the gyroscope is located in the vertical plane.

The gyroscope maintains the horizontal position of the sensitive element and, on any change in course, does not interfere with the rotation of the element together with the aircraft about the vertical axis.

Figure 148 gives a diagram for computing the course by such a compass.



The main indicator (4) contains a selsyn (2), which receives the emf signals from the sensitive element.

The secondary winding of the induction pickup feeds the stator winding of the selsyn (2). In the winding of the selsyn rotor an emf is thus induced by the resultant magnetic flux of all three stator windings. Consequently, the resultant emf of the rotor winding will have a different angular direction, depending on the ratio between the electromotive forces of the pickup windings (i.e., depending on the course). Since the emf of the selsyn rotor is small, it is amplified in the vacuum-

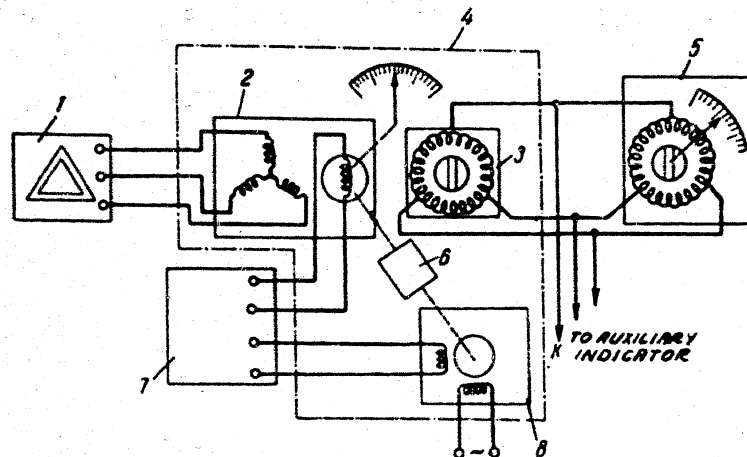


Fig.148 - Circuit Diagram of Remote-Reading Induction Compass

1 - Pickup; 2 - Selsyn; 3 - Magnesyn; 4 - Main indicator; 5 - Auxiliary indicator; 6 - Reducer; 7 - Amplifier; 8 - Asynchronous motor.

tube amplifier(7) from where the amplified signals are fed to one of the windings of the two-phase synchronous motor (8) of the mechanism of the course computer.

The secondary winding of the motor is fed by alternating current of the same frequency, obtained directly from the converter. The phase of the current supplied

to the secondary winding is constant.

The electric motor (8), over the reducer (6), rotates the pointer of the main indicator (4) and the rotor of the selsyn (2). When the rotor of the selsyn has rotated until its winding is perpendicular to the resultant magnetic flux of the stator, the emf in the selsyn disappears, the electric motor (8) stops, and the pointer of the instrument indicates the course of the aircraft.

Every change of course is accompanied by a change in direction of the magnetic flux; in this case the electric motor is again excited and shifts the rotor and the instrument pointer to a new position, corresponding to the new course.

The course computer mechanism, while spinning the rotor of the selsyn tube, also spins the rotor of the magnesyn (3) which transmits the readings to the auxiliary indicator (5).

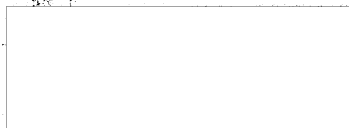
The auxiliary indicators have a receiving magnesyn with a pointer on the rotor.

Since the transmitter of the compass has no moving sensitive elements to cause inertia errors and is remote from the ferromagnetic masses producing deviations, it will give readings with a high degree of accuracy; this has led to its widespread use in medium and heavy aircraft.

The absence of a Cardanic error is a great advantage of the induction compass, with the sensitive element stabilized in the plane of the horizon.

In designing a remote-reading induction compass, it must be remembered that the very accurate gyro vertical used in it must be installed at the center of gravity of the aircraft (to ensure adequate accuracy of the readings) while the magnetic induction pickup must be placed in a location where the compass deviations are negligible. In practice, the magnetic deviations are not at a minimum at the center of gravity of the aircraft.

A drawback which limits the usefulness of the remote-reading induction compass is its complexity and bulkiness.



CHAPTER VI

AUTOMATIC PILOTS

Section 1

General Data

In flight, an aircraft may rotate simultaneously with respect to all three axes. Since the aircraft rotates with respect to its center of gravity, it is necessary for maintaining the longitudinal equilibrium that the sum of the moments acting on the aircraft with respect to the zz axis (cf. Fig.1), are equal to zero. The moments turning the aircraft with respect to the zz axis may be due to aerodynamic forces, to forces of gravity, etc.

If the longitudinal equilibrium of the aircraft is disturbed, the elevator must be deflected to restore it.

The equilibrium of the aircraft with respect to the yy axis is ensured if the sum of the moments acting in the plane of symmetry is equal to zero. This is called course equilibrium. The disturbance of this equilibrium may be due to a number of reasons: distortion of the planform of one half of the wing, deflection of the rudder, unequal thrust of the propellers in multi-engine aircraft, etc.

A disturbance of equilibrium results in turns of the aircraft to the right or left, with a simultaneous dip in the direction of the turn. This inclination is the result of the fact that the outside wing (with respect to the turn) traverses a longer path during the same period of time; consequently, its speed and lift will

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likewise be greater than of the inside wing.

Such turns of the aircraft are counteracted by the pilot by deflecting the rudder toward the side opposite the turn.

The equilibrium of the aircraft with respect to the xx axis is called lateral equilibrium and is maintained primarily by strict symmetry of the aircraft with respect to its plane of symmetry.

The reaction of the propeller-engine group, deflecting the aircraft toward the side opposite the sense of rotation of the propeller, is absorbed by aileron tabs (small movable control surfaces circumscribed in the dimensions of the aileron).

To restore the lateral equilibrium, both ailerons and tabs are used.

While banking, the aircraft tends to turn toward the side of the bank. For this reason, in banks and turns the pilot always moves rudder and ailerons simultaneously.

The ability of the aircraft to maintain a constant flight attitude without intervention by the pilot as well as its ability, during a short disturbance of equilibrium, to restore this attitude rapidly, is called the stability of the aircraft.

The stability of the aircraft with respect to the xx and yy axes is called lateral stability, while the stability with respect to the xx axis is known as transverse stability and the stability with respect to the yy axis as directional stability.

The additional concepts of dynamic stability and stability in roll are also often used.

An aircraft is called dynamically stable if, without interference by the pilot, it tends to adjust its plane of symmetry strictly in accordance with aerodynamic forces.

The tendency of an aircraft to come out of a roll is termed stability in roll; the more stable the aircraft, the harder it is for the pilot to change its attitude, and the poorer will be its controllability.

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To facilitate the work of the pilot in controlling the aircraft, automatic stabilizers known as autopilots are used.

Almost all present day automatic pilots have gyroscopic units.

Automatic pilots with gyroscopes, having three degrees of freedom, are very important for flying without landmarks or at zero visibility.

The automatic pilot is an automatic regulator designed to hold the aircraft at a predetermined attitude without intervention by the pilot. It is therefore necessary that the parameters characterizing the given attitude remain constant or, at least, that the deviations are as small as possible. For example, in rectilinear horizontal flight, the mean speed, altitude, and course must remain constant, and the amplitude of their variations must be small. In addition the oscillations of the aircraft about all three of its axes must also be eliminated.

Certain parameters characterizing the state of equilibrium of the aircraft relative to its axes may vary during flight (centering, weight, propeller thrust, etc); the automatic pilot must, without intervention by the pilot, hold the aircraft at the assigned attitude (or more exactly, at an attitude differing only slightly from the assigned one).

The automatic pilot must execute all principal maneuvers, i.e., right and left turns, in horizontal flight as well as in climbing and descending.

The automatic pilot must be reliable, independent in its characteristics of the surrounding medium, and simple in operation.

Since the simultaneous stabilization of the four principal parameters of flight: altitude, speed, course, bank, is possible only by the aid of four control elements, i.e., throttle control, elevator, rudder, and ailerons it follows that an automatic pilot which completely frees the pilot from controlling the aircraft must consist of four automatic devices, for the throttle control of the aircraft engine, the elevator, the rudder, and the ailerons. In most modern autopilots this problem has not been fully solved, and the only types of autopilots available at present are

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those manipulating the control surfaces of the aircraft.

Figure 149 shows schematically the main parts of the automatic portion of the automatic pilot.

The sensitive element, consisting of a gyroscope with three degrees of freedom, picks up any variation in the aircraft parameter regulated by it.

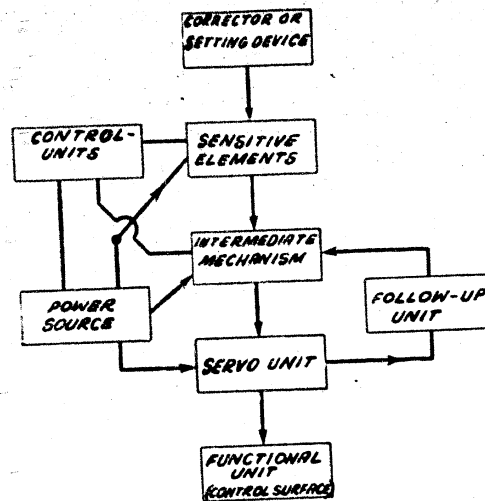


Fig.149 - Principal Parts of the Automatic Pilot

At present, the sensitive element for maintaining the selected direction in an automatic pilot, usually is a gyroscopic instrument in the form of a gyroscope with three degrees of freedom, which is less subject to the action of gusts than other instruments.

The correction in these instruments does not basically differ from the corrections used in gyro horizons, directional gyros, etc. Membrane instruments operated by air pickoffs are used for stabilizing the flying speed and altitude.

The intermediate mechanisms receive the impulses from the sensitive elements,

amplify and compound them where necessary (if one control surface is controlled by several sensitive elements), and control the operation of the servo unit which deflects the corresponding follow-up unit (the control surface).

Intermediate mechanisms consist of:

- a. a sensing element: the pickup;
- b. an amplifying element;
- c. a compounding element;
- d. a control element.

The power sources supply the necessary energy to all remaining elements. The most widely used are pneumatic, hydraulic, and electric energy. Electric energy is coming into increasingly wider use.

The follow-up unit coordinates the angle of control-surface deflection with the value of the deflection of the parameter being stabilized.

The indicator units allow the pilot to follow the operation of the automatic pilot and its components. The automatic pilot is turned on and off by control knobs.

The servo unit deflects the control surface of the aircraft by means of an external source of energy. According to the type of energy used, these are divided into pneumatic, hydraulic, electric, and mechanical. The pneumatic units, usually operating on a pressure of 1.5 - 5 kg/cm², have the smoothest action. Of all power-consuming elements of the automatic pilot, the servo units consume the most power.

The source of energy used generally is a pneumatic or hydraulic pump. According to their purpose, these pumps are divided into groups serving: a) the sensitive elements; b) the servo units; c) the intermediate mechanisms.

Section 2

PRINCIPLES OF CONTROL

Any automatic pilot operates on one of the following control principles:

1. Direct Control: a Direct-Acting Automatic Device. The sensitive elements of such automatic devices take the form of pendulums or free gyroscopes. Pendulum

automatic pilots have not come into wide use, since all of them operate at considerable lag and cause flutter of the aircraft (yawing).

Gyroscopic automatic pilots of this type also are not in wide use, because of the need for high-power gyroscopes.

Present-day automatic pilots do not use direct control principles.

2. Indirect Control without Follow-Up. This type of control leads to undamped oscillations of the control surfaces and the aircraft itself (auto-oscillations).

This method is not used on aircraft.

3. Indirect Control with Rigid Follow-Up (Fig.150). An automatic pilot with such a control consists of the free gyroscope (1), whose frame is connected over the tie rods (2) and (3) with the valve (4), controlling the operation of the piston servo unit (5); the rod (6) of the unit (5) is connected over a tie rod with the aircraft control surface (7). The lever (8) connects the valve (4) with the upper end of the piston rod of the servo unit

(follow-up). In this arrangement, the position of the valve depends on the position of the gyroscope and the position of the control surface.

On any deviation from the assigned flight attitude the Valve is displaced, for example, upward; in this case, the piston of the servo unit moves downward,

deflecting the control surface to the left. At the same time the piston rod of the

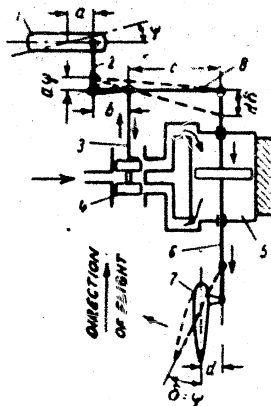


Fig.150 - Principle of Indirect Control with Follow-Up

- 1 - Gyroscope; 2 - Tie rod; 3 - Valve tie rod; 4 - Valve; 5 - Servo unit;
- 6 - Piston rod; 7 - Control surface;
- 8 - Follow-up clutch; a, b, c, and d - Arms of clutch; ψ - Angle of deviation of aircraft.

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servo unit moves the valve downward. The motion of the piston and the deflection of the control surface is interrupted, as soon as the valve returns to the neutral position.

Automatic pilots operating on this principle do not always completely restore the assigned state of operation. For example, in aircraft equipped with a longitudinal automatic device, on any change in centering due, for example, to the displacement of a passenger in the cabin, which will cause a nose-down (or nose-up) of the aircraft, the automatic pilot will deflect the elevator upward (or downward) and the equilibrium of the aircraft would be restored at the instant when the moment produced by the control surface balances the moment produced by the displacement of the center of gravity, i.e., at some new position of the aircraft. The angle at which the aircraft still deviates from its original position is called the residual error of control. For return to the original flight attitude, the pilot must apply corrections manually in accordance with his instrument readings.

4. Indirect Control with a Nonrigid Follow-Up, also Termed Isodromic.

In this case, the valve (4) is connected with the piston rod of the servo unit (5) over the damper (9), called a cataract (Fig.151) whose degree of damping is regulated by the cock (10). If the cock (10) is closed, the follow-up is converted into a rigid follow-up. The end of the lever (8) carries the restraining spring (11).

The action of the follow-up of the servo unit on the valve depends on the rate of relative displacement of the aircraft and the aircraft control surfaces. In the case of brief disturbances, this automatic pilot operates like an automatic device with a rigid follow-up.

On prolonged disturbances, such as due to a change in the thrust of the right propeller, the aircraft begins to rotate (in this case to the right) while the valve forces the rudder to deflect to the left. As in the preceding case, the aircraft assume a new position, different from its initial position, at which the moment of the deflected control surface compensates the moment of asymmetry of the thrust. We

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obtain a new position of equilibrium of the aircraft, but not of the automatic device, since the spring (11) in this case will be stretched and will displace the piston of the cataract (9) and the valve (4) upwards, for which reason the angle of deflection of the directional rudder to the left will increase still more. The equilibrium of the aircraft with respect to the yy axis will be disturbed, and the aircraft will begin to return to its course.

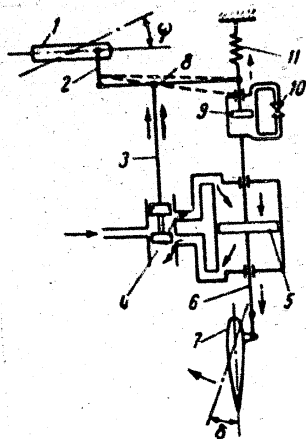


Fig.151 - Indirect Regulation with Nonrigid Follow-Up

1 - Gyroscope; 2 - Tie rod; 3 - Valve tie rod; 4 - Valve; 5 - Servo unit; 6 - Piston rod of unit; 7 - Control surface; 8 - Follow-up knob; 9 - Cataract; 10 - Cock; 11 - Spring; ψ - Angle of deflection of aircraft.

This process will continue until the spring and the valve are established in the neutral position corresponding to the assigned course. As a result, the aircraft will now be stabilized on this course. In this case, the rudder is deflected leftward by an angle such that the moment from the rudder will exactly compensate the moment due to the asymmetry of the thrust, and all elements of the automatic device, except the piston rod of the rudder machine will take neutral positions.

Section 3

STRUCTURE AND OPERATION OF AUTOMATIC PILOTS

The AP-45 Pneumatic Automatic Pilot

The operation of all three automatic devices of the AP-45 autopilot, longitudinal, lateral, and directional, is based on a single principle. The sensitive element which consists of an astatic gyroscope with three degrees of freedom, actuates a servo rudder over a valve, by means of a special pneumatic relay. This unit then

deflects the corresponding control surface.

The sensitive element of the course stabilizer represents a directional gyro, while the sensitive element of the lateral and longitudinal stabilizers is a gyro horizon. Both gyroscopic instruments are pneumatic and are analogous in design to the above-described directional gyro and diving gyro horizon, except for some additional parts.

Figure 152 gives a schematic diagram of the course stabilizer of the automatic

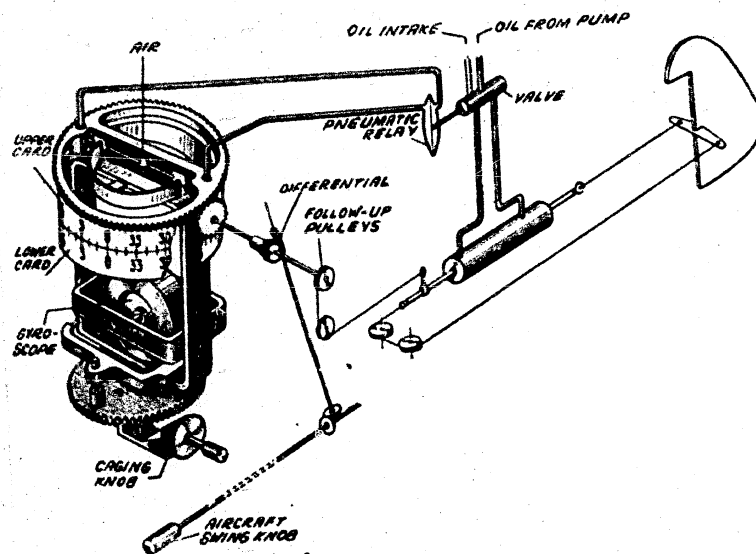


Fig.152 - Schematic Diagram of Directional Gyro Control of AP-45 Automatic Pilot

The directional gyro furnishes airplane headings while the aircraft is under manual or automatic control.

The caging mechanism operates as in an ordinary directional gyro.

The lower card is the same as in conventional directional gyros. The upper card, or follow-up card, is connected over cog wheels and a differential with the

aircraft swing knob and over follow-up cables to the aircraft rudder.

When the scale divisions of both cards coincide, the arc-shaped valve, connected with the gyroscope covers all nozzles of the air pick-off system, and the pneumatic relay occupies a neutral position. The air enters the pick-off from special pumps through a central orifice.

On any deviation from the course, the body of the automatic course device and the collector rigidly connected with it are deflected together with the aircraft, while the valve connected with the gyroscope, maintains a constant position in space. As a result, the nozzles are covered differently, causing the pneumatic relay to operate and the valves of the servo unit to displace the rudder in such a way that the aircraft begins to return to the initial position.

The follow-up system is in the form of cables connecting the piston rod of the servo unit with the air system, and displacing this pick-off system by a quantity proportional to the deflection of the rudder.

An automatic course stabilizer designed according to this layout is successfully used on aircraft, but has a number of drawbacks: 1) it does not ensure coordinated operation of the rudder when exposed to disturbing forces; 2) it has an inadequate service ceiling; 3) it is hard to regulate under various flight conditions (e.g., bumpy flight); 4) it requires various different types of power sources; 5) it has an inconvenient mechanical follow-up system, etc.

The sensitive elements of the longitudinal and lateral stabilizers of the aircraft are in the form of a gyro horizon with two pickups; one for longitudinal stabilization, the other for the lateral.

The deflection of the control surfaces is accomplished over a pneumatic relay and a servo unit.

The follow-up consists of a system of cables. The complete layout of the automatic pilot is given in Table 5.

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Table 5

Name of Part	Weight in kg	Name of Part	Weight in kg
Automatic course stabilizer	3.45	Drainage tank	0.34
Longitudinal-lateral automatic stabilizer	4.55	Pressure regulator	0.3
Mounting bracket	2.1	Electric remote control	1.7
Hydraulic unit	3.2	Oil pump	1.6
Servo unit	4.1	Weight of installation, in- cluding transmission and oil	≈ 30
Oil tank	1		
Two-pointer manometer	0.2		

The kinematic diagram of the automatic pilot is given in Fig.153, and an overall sketch in Fig.154.

A special pressure regulator is used to maintain constant pressure in the automatic system.

The oil for the hydraulic units is kept in a tank and is supplied through an oil pump over an oil filter to the oil valve and servo unit. The excess oil and the used oil are recycled to the oil tank through special manifolds.

Depending on the type of aircraft, the following variations in the layout of the automatic pilot are possible:

1. The sensitive element can be fed by vacuum.
2. Remote control units can be incorporated in the design of the instrument, with control knobs at the pilot and navigator seats.
3. Corrections from a magnetic compass and a radio compass can be provided.
4. Instead of compound servo units, separate units may be installed.

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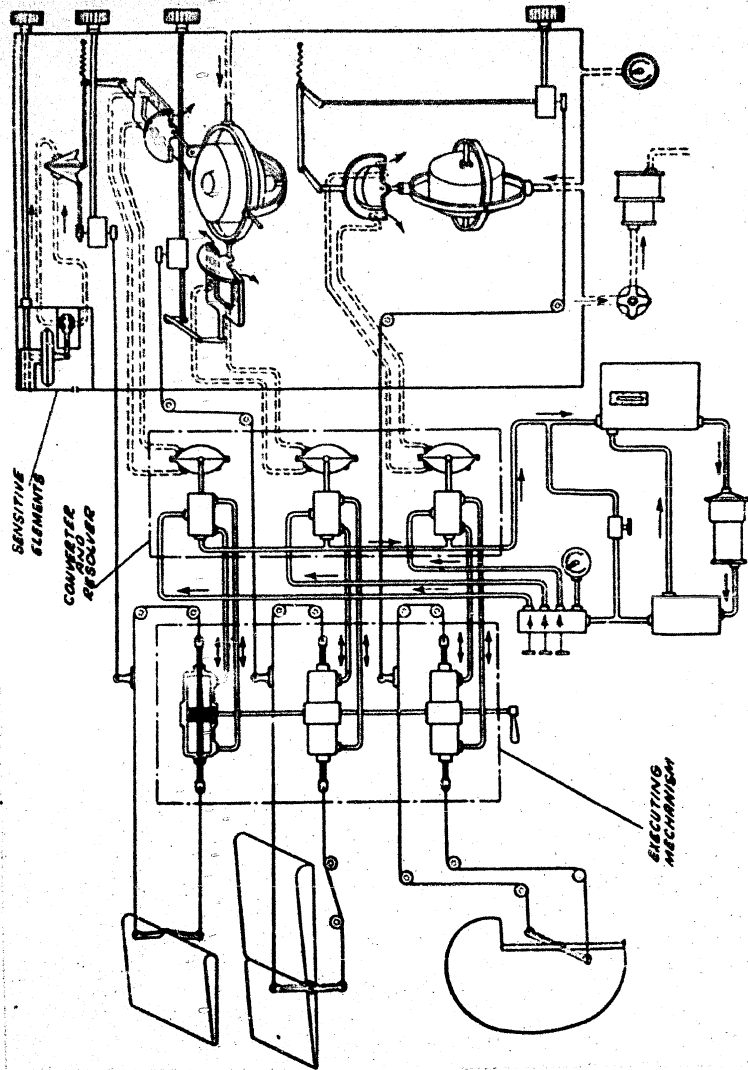


Fig.153 - Kinematic Diagram of AP-45 Automatic Pilot

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The automatic pilot is able to make a climb at an angle up to 35°, a dive at an angle up to 80°, and a turn with a bank up to 55°.

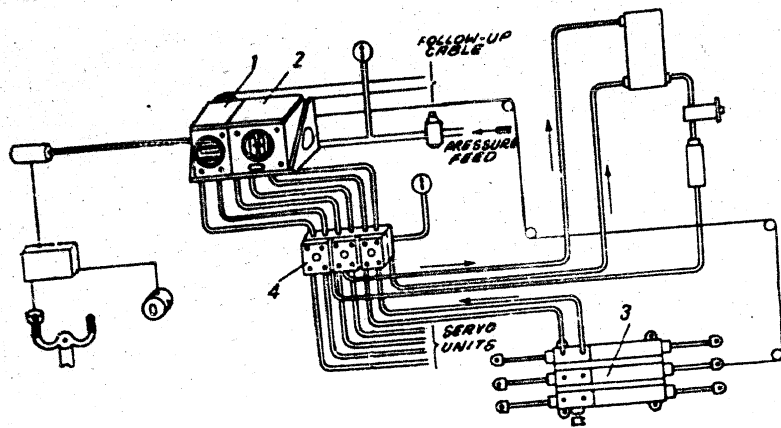


Fig.154 - General Sketch of the AP-45 Automatic Pilot

- 1 - Automatic course stabilizer; 2 - Automatic longitudinal and lateral stabilizer; 3 - Servo unit; 4 - Hydraulic unit

Characteristics of the AP-45

Service ceiling	9 - 10 km
Temperature range	from +50 to -35°C
Angular oscillations of aircraft	(0.5 - 1°)
Thrust of servo unit	85 kg
Power required	1.5 - 2 hp

The pressure regulator used when the AP-45 is fed from the supercharger of the aircraft engine maintains a pressure of 80 - 100 mm Hg.

The AP-45 pneumatic automatic pilot is used successfully in commercial aviation, and meets the following specifications:

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Accuracy of stabilization

about 1 - 2°

Temperature range

from -20 to +50°C

Service ceiling

about 7000 m

Besides this, reliability, low weight, low cost, and simplicity of operation must also be assured.

The Electric Automatic Pilot

All units of an electric automatic pilot are completely electrified, and their operation does not depend on the altitude of flight or the temperature. Electric heaters are installed on the units of the sensitive element and the servo units, and are turned on when the temperature drops below -20°C.

The automatic pilot controls the aircraft and stabilizes the sight in azimuth during bombing operations. The action of the control surfaces is coordinated, i.e., during a bank not only the ailerons but also the rudder and elevators are deflected. This assures high accuracy of stabilization.

The electric automatic pilot provides three aircraft stabilizations. The sensitive element of the directional stabilization consists of a free gyroscope and an oil damper by means of which the angular velocity of the aircraft with respect to the vertical axis is measured*. The sensitive element of the longitudinal-lateral stabilizers consists of a free gyroscope with pendulum corrections.

The complete set of the electric automatic pilot includes:

1. Course stabilizer unit;
2. Longitudinal and lateral stabilizer unit;
3. Servo units;
4. Control panel;

*Feeding a signal proportional to the angular velocity into the course channel improves the damping of the flutter.

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- 5. Amplifier;
- 6. Junction box;
- 7. Course indicator.

The absence of mechanical connections between the elements of the automatic pilot considerably facilitates its installation.

The slide (2) of the course control is connected to the block (4), bearing the wipers of the course and aileron potentiometers.

The displacement of the potentiometer slide wire is proportional to the angular deflection of the aircraft from the course and to the rate of such deflection.

Before the automatic pilot is turned on, the wiper of the course potentiometer must be in a neutral position

The pilot's indicator (7) consists of a galvanometric instrument operated on direct current and connected to the potentiometer. The galvanometer measures the voltage across the area from the midpoint of the potentiometer to the point of contact with the wiper. The value of this voltage is proportional to the change of course angle.

The load on the vertical axis of the gyroscope unit imposed by the forces of friction and inertia is so great that compensation is required.

For this, the design includes a correction device (Fig.156) in the form of the motor (1), which develops a moment with a sign opposite to that of the moment exerted on the gyroscope. The cog wheel (2), rigidly connected with the outer vertical frame of the gyroscope, is connected over intermediate cog wheels with the gears (3) and (4) sliding along its shaft; to these, the two friction disks (5) are rigidly connected. The two cog wheels (6), with the same number of teeth, are mounted coaxially with the disks.

The faces of these cog wheels, facing the friction disks, have a cork covering.

The Course Stabilizer

Figure 155 gives the kinematic diagram of the course stabilizer of the electric

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automatic-pilot.

The free gyroscope has its axis of natural rotation in the horizontal position; its rotor is fed by direct current and rotates at the rate of 7000 - 8000 rpm.

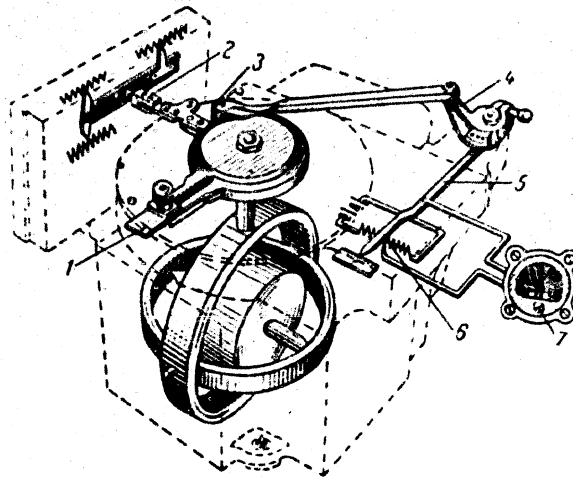


Fig.155 - Kinematic Diagram of the Course Stabilizer of an Electric Automatic Pilot

1 - Caging mechanism; 2 - Potentiometer wiper; 3 - Slide; 4 - Block;
5 - Tie rod; 6 - Pointer; 7 - Pilot's indicator

A friction clutch with a drum, above which a second drum is installed, connected over a clutch with the sight, is rigidly connected with the vertical axis of the outer frame. A ring with three levers is coupled to the friction drum. One lever is connected with the wipers (2) of the control potentiometer, the second with the caging mechanism (1), while the third connects the slide (3) with the indicator (7).

The left side of the case of the course stabilizer carries the box containing the control potentiometers, including the potentiometer of the course control and

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the double potentiometer for aileron control.

Both cog wheels (6) are actuated into continuous and mutually opposite rotation by the motor (1).

The brush (7), mounted on the gyroscope shell, slides along two pairs of thin plates (8) when the gyroscope rotates about the vertical axis.

Each inner plate is connected across a resistance with a winding of one of the electromagnets (9), which are connected in series and, when turned on, attract the friction disks with the cork surface, bearing the cog wheel (6).

The outer plates are connected directly with the magnets (9).

When an external moment acts about the vertical axis of the outer frame, the axis of the rotor precesses in a vertical plane.

When the rotor is inclined to a side opposite that indicated by the arrow A, the brush (7) closes the circuit of the right electromagnet, the right disk engages the gear wheel (6), and the frictional moment developed on the surfaces of this disk is superimposed on the moment directly opposite the load moment, whose direction is indicated by the arrow B.

If the frictional moment proves greater than the load moment, then the axis of the rotor will be displaced in a direction opposite to the original direction until the brush (7) rests on the insulator between the plates and the action of the frictional moment is interrupted. If the frictional moment proves to be insufficient, then the axis of the rotor will be inclined until the circuit of the right electromagnet is closed, this time over the upper plate (without the intermediate resistor), as a result of which the force of adhesion on the friction disks and, consequently, also the frictional moment, will increase. When the case of the course stabilizer unit is motionless, the cog wheel (2), the gears (3) and (4), and the friction disks (5) are all stationary.

The sensitive element of the longitudinal-lateral stabilizer is a gyro vertical (Fig.157) fed by 26-volt direct current. The speed of the gyro vertical rotor is

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7000 - 9000 rpm.

The gyro vertical has a friction-pendulum correction device.

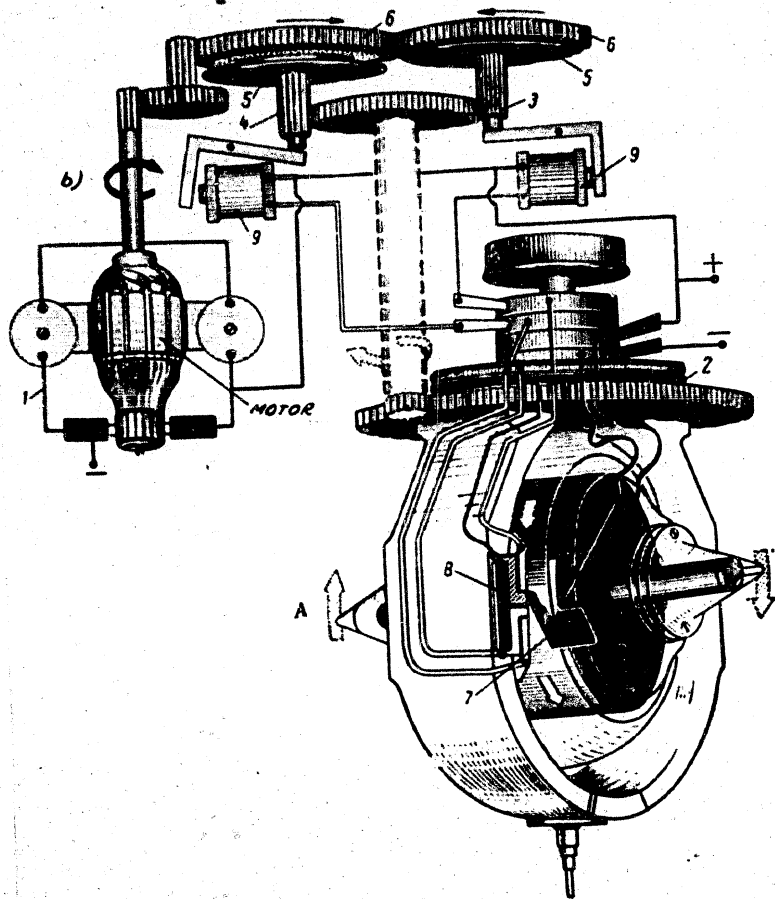


Fig.156 - Diagram of Correction of a Gyroscope for Continuous Rotation of the Correction Motor

- 1 - Motor; 2 - Cog wheels; 3 and 4 - Drive gears; 5 - Disks; 6 - Cog wheels;
- 7 - Brush; 8 - Plates; 9 - Electromagnets



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When the aircraft turns, the lateral correction is switched off, and the gyroscope will not undergo precession due to the action of inertia forces.

The brushes of the turn-and-bank potentiometers are attached to the gyroscope frames. These potentiometers are fed by alternating current from separate transformers.

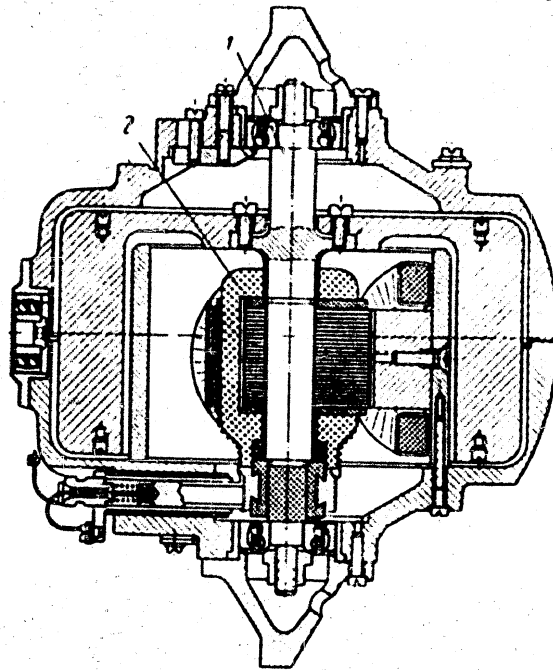


Fig.157 - Gyro Vertical

1 - Principal axis; 2 - Electric motor

The voltages from the brushes of all the potentiometers (of the course stabilizer and the gyro vertical) are fed to the control panel to which the voltages from the follow-up potentiometers on the servo units are also fed.

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There are six lamps on the control panel, two of which signal the operating of the automatic pilot, and a change-over switch, allowing the control of the aircraft to be turned over to any member of the crew. In addition, the control panel also has switches for individually disconnecting the servo units.

When the current is turned off, all the servo units are automatically disconnected, and the automatic pilot ceases to control the aircraft.

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