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TRANSLATION

MISSILE GUIDANCE

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CHAPTER I

GENERAL PRINCIPLES OF MISSILE GUIDANCE

Yu. Kh. Vermishev

1. General Information on the Principles of Automatic Control

Guided weapons differ in principle from other types of armament. The characteristic feature is the possibility of guiding a missile after it has been launched. Depending upon the designation of the missile, it can be controlled during the entire flight—up to the moment it strikes the target—or during the initial phase of the flight, which is necessary for putting the missile in a trajectory which guarantees striking the target. In most missile guidance systems, control commands are generated automatically when the missile deviates from a given trajectory.

The entire complicated set of means necessary for determining the coordinates of the target and missile, for generating commands, transmitting them to the guided missile and execution of these commands is called a missile guidance system, and, in essesnce, is an automatic control system.

Automatic control systems are characterized by the so-called

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closed circuit of action.

In a closed system of automatic control, the output parameter is continuously controlled and compared with the input signal. When the output parameter deviates from the input signal, an error signal is automatically generated and acts on the control organs, causing them to vary the output parameter until there is zero mismatch.

Automatic regulators which operate according to the closed-circuit principle are widely used in technology.

As an example, let us examine a device for controlling the angle of elevation of a gun barrel. The block diagram is shown in Fig. 1.

The device for controlling battery fire generates a control signal which accurately corresponds to the required angle of elevation of the gun barrel. This signal is supplied as a d-c voltage U_{in} to the sensitive element. On the gun fortification there is a potentiometer which picks up a d-c voltage, U_{out}, which is proportional to the true angle of elevation of the gun. This voltage is also fed to a sensitive element, in which the input and output signals are compared. As a result of the comparison of these signals, an error signal $\delta = U_{out} - U_{in}$ is generated, which is sent to the command-generation device.

Depending upon the sign and magnitude of the error signal, a command is formed which, after amplification, is sent to the drive motor, which sets the angle of elevation of the gun accordingly.

These devices are called servomechanisms.

In simple servomechanisms, there is usually no specially designed, separate element in which commands are generated; the command-generating elements are located in the amplifier.

These elements play an extremely important role: they ensure

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The more complicated the control system, the more functions the command-formation system must perform. As a rule, the following elements are found in automatic control systems:

1) an object of control, i.e., a device to be controlled;
2) a measuring device, required for determining the output parameter (a potentiometer in the case under examination);
3) a sensitive element for comparing the input and output signals and generating an error signal;
4) a control-command generator;
5) a control device, usually an amplifier and drive mechanism; in many cases energy converters can replace these.

In missile guidance systems, the object of control is the missile itself. The position of the longitudinal axis of the missile, its center of gravity relative to some given trajectory or any characteristic of its motion can be used as the output parameter.

The measuring devices for various systems are: gyroscopic systems, which fix the position of the axes of the missile; inertial devices, which measure the transverse acceleration of the center of gravity of the missile; ground radar systems, which determine the position of the missile in space; homing systems, etc.

Computers are usually used as the sensitive element, in which the error signal is formed. In these a control command can be formulated simultaneously.

The control elements are steering devices for the rudder drives and air and gas rudders.

The amplifier part of the control system in some cases is replaced by a command converter and a radio line for command transmission.

All missile guidance systems can be divided into three classes: internal control, remote control and homing.

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The Basic Characteristics of Guidance Systems

Properties of guidance systems. The requirements placed before guidance systems result from tactical missions assigned to one or another type of weapon which can be guided.

These requirements are highly diverse and encompass technical, tactical and exploitational characteristics of the individual elements of the system, as well as the system as a whole.

The group of requirements pertaining to continuous control processes includes requirements for stability and accuracy of operation.

The choice of missile-guidance parameters (the formulation of a system) comes down, chiefly, to determining the most rational characteristics of stabilization elements and devices for command generation, ensuring the maximum accuracy of operation of the system.

This task is extremely difficult and can be performed only within limits. One of the basic difficulties is the diversity and inconstancy of input disturbances acting on the guidance system. For example, in the case of linear target motion, certain parameters for the homing system must be chosen, and when the target is maneuvering, other parameters must be chosen.

In addition, the quality of target information affects the choice of parameters. When target information is received (for example, from a radar station), a spurious signal of fluctuating noise enters the computer simultaneously with the signals corresponding to the coordinates of the target. This noise is caused by reflection of radio signals from the target and the presence of inherent noise in the receiver. Fluctuating noise, which is a random process with variable sign (Fig. 3), causes error in determining the target coordinates—fluctuating error.

The parameters of the system must be chosen according to the spectral composition and the amplitude of the fluctuating noise.

Besides the effects due to the nature of input disturbances, when designing a guidance system it is necessary to take into consideration the variation of parameters of the system with respect to time. The automatic control systems widely used in technology usually have more or less stable parameters; missile guidance systems have parameters which sometimes vary within wide limits. The variation of parameters in this case is due to the fact that the characteristics of the missile (which is the object to be controlled) are functions of velocity, the density of the atmosphere, the degree of fuel burn-up and the change in the thrust of the engine.

These properties of missile guidance systems complicate considerably the calculation of the accuracy of its operation, but owing to the development of computers, these calculations have become fully realizable.

In addition, there exist simplified methods for calculating the basic parameters of the system; for example, calculation by using *frequency characteristics*.

The concept of the frequency characteristic. The frequency characteristics of an element or an entire system pertain to a set of two characteristics: the amplitude-frequency characteristic and the phase-frequency characteristic.

The frequency characteristics allow the degree of stability of a guidance system to be evaluated and the nature of the motion of a missile in the presence of various disturbances to be determined.

In order to determine the frequency characteristics experimentally, a sine-wave signal of determined frequency is fed into the device to be studied. For example, the signal can be a voltage U_{in}. The signal U_{out} at the output of the device is measured; it has the shape and frequency of the input signal, but a different amplitude and is shifted with respect to time (Fig. 4).

The ratio of output and input amplitudes $\frac{A_{out}}{A_{in}}$ gives the amplitude characteristic of the system for a given frequency.

Since in an element (system) to be studied there are inertial elements, the characteristic points of the sine-wave signal to it will be shifted with respect to time, and in angular measure it is called a phase shift $\Delta\psi$. Transition from time to angular measure is made by the formula

$$\Delta\phi = \frac{\Delta\tau \cdot 360}{T}.$$

where T is the period of one sine-wave oscillation; and $\Delta\tau$ the magnitude of the time shift.

The magnitude of the phase shift determines the phase characteristic of the device for a given frequency.

If the period of the sine-wave oscillation is changed (the frequency is the reciprocal of the period), we obtain different amplitude and phase ratios.

By arranging a sufficient number of these measurements for a series of frequencies, we can construct curves, one of which shows the dependence of the amplitude ratio upon frequency f, and the other shows the dependence of the phase shift upon frequency f. These are the amplitude-frequency and phase-frequency characteristics (Fig. 5).

In practice, we do not resort to such difficult experiments to find the frequency characteristics. The characteristics are usually calculated mathematically from a given system of equations.

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Note that not all systems have frequency characteristics. In some systems, an output signal of one or another shape will not correspond to an input sine-wave signal. In particular, in missile guidance systems, owing to the variability of parameters, frequency methods of study sometimes have limited application.

The concept of stability of a guidance system. By stability is meant the property of an automatic missile guidance system to return the missile to steady-state motion (for example, to a given trajectory) after the effect of disturbing forces has stopped.

The stability of a missile is characterized by oscillations of its center of gravity relative to a given trajectory (Fig. 6) or by oscillations of the axes of the missile relative to its center of gravity.

Besides the concept of the stability of a guidance system, there exists also the concept of stability margins. This characteristic gives quantitative representations of the possibilities of a guidance system and determines the allowable limits of deviation of the basic parameters of the system from normal. Calculation of stability margins is necessary, for example, when determining tolerances for adjusting on-board and ground equipment.

There are several methods for calculating the stability and stability margins of a guidance system.

One of the most widely used and most graphic methods is that of determining stability and the stability margin by the frequency characteristics. Let us examine the frequency characteristic of an automatic control system, i.e., a system with an open feedback loop (Fig. 7).

Since the system has inertial elements, the phase shift between the input and output signals increases with an increase in the frequency of the signal supplied. At some frequency, the phase shift can be equal

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to 180°. In this case the output signal has the opposite sign of the input signal. If in this case its amplitude is equal to or greater than the amplitude of the input signal, the system will be unstable with a closed feedback loop. If, with a phase shift of 180°, the output signal is lower in amplitude than the input signal, the system will be stable.

Thus there exist certain critical values for the phase-frequency and amplitude-frequency characteristics the exceeding of which destroys the stability of the system.

The phase stability margin is determined by the frequency f_1 at which the amplitude of the output signal is equal to the amplitude of the input signal, i.e., $\frac{A_{out}}{A_{in}} = 1$; numerically it is equal to the deficiency of the phase shift up to 180°. The amplitude stability margin is determined by the frequency f_2 at which the phase shift equals 180°; numerically the amplitude stability margin is expressed by the deficiency in the amplitude ratio up to 1. The greater their margins, the more stable the system.

The frequency characteristics shown in Fig. 7 are typical for a guidance system which is stable when closed.

Determining the reaction of a guidance system to various disturbances. In order to determine the dispersion characteristic of guided missiles, it is necessary to know how the guidance system will behave in the presence of various input disturbances, i.e., various fluctuating signals, target maneuvers, gusts of wind, etc.

In order to adjust and control on-board and ground equipment, it is necessary to know how one or another element of the system reacts to a standard input signal. These problems are usually solved by using the frequency characteristics.

Certain standard signals can be represented as the sum of sine-wave signals. They are expanded by the Fourier method. By knowing the frequency characteristic of the system, it is possible to find the reaction of the system to each sine-wave signal separately. Having added all signals received at the output, we find the resultant action of a standard input signal. Thus we can find, for example, the action of the system on a brief square pulse (Fig. 8).

By knowing the action of a guidance system upon this pulse signal, we can determine the reaction of the system upon a signal of any shape, including those which cannot be represented as the sum of sine-waves. For this, a pulse, rather than a sine-wave signal, is used as the elementary signal; the reaction on the pulse has already been determined. The input signal is not represented as a series of square pulses, as shown in Fig. 9. Having found the action on each individual pulse taking into account the moment of its transmission and having added all output elementary signals, we find the total action on the input signal.

From the frequency characteristics we can also determine the behavior of the system under the influence of random fluctuating signals. For this it is necessary to find beforehand the frequency composition of these signals, by expanding them into a Fourier series.

The inverse problem can also be solved: the frequency characteristic of the system can be found from known input and output signals.

3. Guidance System Errors

The effectiveness of guided missiles depends upon the degree of their dispersion from errors in the guidance system.

The correct choice of parameters and system elements leads to minimum error in guidance.

Three groups of errors are characteristic of automatic control systems:

1) instrument errors;
2) dynamic errors;
3) fluctuating errors.

Instrument errors are caused by inaccuracy in adjustment and operation of elements of a system, e.g., instability of d-c amplifiers, incorrect tuning of antenna systems, aerodynamic asymmetry of the missile, etc.

Instrument error is relatively constant in guidance of a single missile and can vary from launching to launching. Its magnitude can be characterized by the root-mean-square error σ, which is calculated as follows:

$$\sigma=\sqrt{\frac{\delta_1^2+\delta_2^2+\delta_3^2+\ldots+\delta_n^2}{n}},$$

where n is the number of experiments in which error was measured; δ_1, δ_2, ..., δ_n the errors in each concrete case.

Another characteristic of instrument error is its systematic value m:

$$m=\frac{\delta_1+\delta_2+\ldots+\delta_n}{n},$$

where δ_1, δ_2, ..., δ_n are errors, taking their signs into account.

If the systematic instrument error is not zero, then this means that there is a constant error in the equipment.

Dynamic errors result from the very operating principle of automatic control systems. In order that an actual system produce a control command, it is necessary that there be a certain mismatch between the required and actual positions of the object of control. (For example,

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in automatic missile guidance systems, the missile must be somewhat off the required trajectory in order for a command to be generated.) Therefore, even in systems without instrument errors, there will exist a certain error due to the dynamics of control. The more abruptly the input disturbances vary (e.g., target maneuvers, gusts of wind, etc.), the greater this error will be.

Fluctuating errors arise as a result of the action of fluctuating disturbances on the system (see Fig. 3).

Owing to the change in sign of this error, the system begins to cause the missile to oscillate, which leads to additional guidance errors (Fig. 10). A fluctuating error can also be characterized by its root-mean-square value σ_{rms}.

The fluctuating and dynamic errors can be calculated when the parameters of the system and the disturbances acting on it are known.

4. The Concept of Controlling Forces Acting on a Missile

Generating the lift force of the missile. In order to change the direction of a missile, a force perpendicular to the direction of motion must be imparted to it, i.e., perpendicular to the velocity vector. This is called the lift force.

The controlability of a missile is characterized by the lift force applied to its mass. According to Newton's second law, this force is equal to normal acceleration of the missile.

Acceleration, as is well-known, is an increment in velocity per unit time. Normal acceleration is acceleration perpendicular to the direction of motion of the missile; it does not change the velocity, but only changes its direction. Thus the greater the normal acceleration, the greater the possibility of changing the direction of missile

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flight, or, in other words, the greater its maneuverability.

The requirements for maneuverability of a missile can vary according to the type of missile.

For example, in horizontal flight relative to the earth's surface, the lift force must be equal to the weight of the missile and be directed vertically upward. When the missile is in a ballistic trajectory, lift force is completely absent.

The most complicated type of motion for guided missiles is motion in a trajectory of arbitrary curvature. In this regard, missile guidance to a moving target or from a moving object (e.g., air-to-air, air-to-ground, etc. type missiles) is characteristic. In this case the lift force must be in the direction in which the missile is to move. If the normal acceleration considerably exceeds the acceleration due to gravity, then the direction of the lift force will almost coincide with the plane of the maneuver.

Depending upon the conditions of use of a missile, either aerodynamic or gasdynamic means can be used for creating the lift force. When the flight of the missile is beyond the earth's atmosphere, it is obvious that only gasdynamic means can be used. When the flight is in the dense layers of the atmosphere, the lift force can be created by aerodynamic forces.

It should be noted that when the engine is operating, the total lift force can be made up of two forces: aerodynamic and gasdynamic. The share of the gasdynamic force increases with an increase in altitude or with a decrease in flight speed.

Let us refer to Fig. 11 in order to explain the mechanism of the formation of the lift force. In Fig. 11, the longitudinal axis X of the missile forms with the direction of velocity vector V a certain

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angle α, which is called the angle of attack. In this airframe diagram, the aerodynamic surfaces of the wing coincide with the axis of the missile, therefore, the angle of attack of the wing is the same as the angle of attack of the fuselage. Lift is created only when this angle is present.

The formation of aerodynamic lift is, in addition, connected with dynamic pressure q, the magnitude of which is directly proportional to the density of the atmosphere and the square of the velocity of the missile. If the angle of attack of the wing is zero, the plane of the wing is flown past by an air stream symmetrically and there is no lift force. If, as shown in Fig. 11b, there is an angle of attack, then the symmetry is destroyed and a lift force appears.

Gasdynamic lift is created when the direction of thrust of the engine does not coincide with the direction of velocity; in this case the lift force forms with the velocity vector a certain angle and gives a component which is perpendicular to the direction of the velocity. If the axis of the engine and the direction of the thrust coincide exactly with the axis of the missile, then this angle is equal to the angle of attack (Fig. 11a). If the angle of attack is not zero (Fig. 11b), then a thrust component appears which is perpendicular to the direction of velocity, i.e., a lift force due to thrust.

Aerodynamic diagrams of missiles. For motion in an arbitrary trajectory, a missile must be able to develop lift in any direction perpendicular to the direction of its flight.

This condition can be fulfilled in two basic schemes of positioning aerodynamic planes: by using a single wing or two mutually perpendicular aerodynamic surfaces (Fig. 12a and b).

The first aircraft-type plan is characteristic of aircraft missiles. For antiaircraft missiles of the Nike or Oerlikon type, or V-2 type ballistic missiles, the second plan is characteristics.

But such an arrangement is not obligatory. For example, the American Bomark antiaircraft missile uses the airplane aerodynamic plan.

The principal feature of the first plan is the fact that lift can be obtained only in a direction perpendicular to the plane of the wing. The sign of the force varies depending upon the sign of the angle of attack (Fig. 13).

The second plan allows lift to be obtained in any direction, depending upon the combination of angles of attack* in the control surfaces.

For example, if the angle of attack varies only in one plane, then the lift is formed in a direction perpendicular to it (Fig. 12a). If equal angles of attack are formed simultaneously in both planes, then by vector addition of the forces, the total lift will be directed at an angle of 45° to the planes (Fig. 14b). Correspondingly, it is possible to combine the angles of attack; in this case the direction and magnitude of the total lift force (force vector) will be changed. The possible positions of the ends of force vectors in a plane perpendicular to the velocity will be found not on a straight line, but in the region bounded by the square (Fig. 14c).

In guidance according to the first plan, for correct spatial

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* The angle in the vertical plane is called the angle of attack, in the horizontal plane it is called the slip angle. However, for symmetrical two-plane plans this is not always acceptable, since it is not at all necessary that one plane be vertical and the other horizontal. Therefore, we shall use the angle of attack for any case.

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orientation of the lift vector it is necessary to turn the missile about its longitudinal axis so that the plane of the wing is banked in the direction of the maneuver to be accomplished. This maneuver is accomplished simultaneously with the bank.

Thus a spatial maneuver is accomplished, as in ordinary airplanes, by the simultaneous formation of an angle of attack for creating the required lift force and bank, which ensures the required direction of this force.

Accordingly, computers, independent of their position, must generate two commands: one for banking the missile, and the other for forming an angle of attack.

Missile guidance in the case of a two-plane aerodynamic plan is accomplished in another way. Inasmuch as it is possible for the missile to create a lift force in any direction, it is not necessary to bank the missile beforehand. The direction and magnitude of the lift force are provided in this case by the appropriate combination of angles of attack in both control surfaces. In this case the computer of the system must generate two commands for the two symmetrical control surfaces.

It should be noted that the requirements for maneuverability in many respects determine the design of a missile.

Highly maneuverable missiles, which have a great margin of aerodynamic force, have relatively large wings.

But it should be mentioned that in general that the wings of supersonic missiles are considerably smaller than those of ordinary airplanes. This is due to dynamic pressure, which increases in proportion to the square of the velocity of the missile. Therefore, if the velocity of the missile is twice the velocity of an airplane, then its

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wing surface will be about four times smaller to obtain the same lift force.

In ballistic missiles, the trajectory of which is close to ballistic and, therefore, does not require high maneuverability, the wing surfaces are either very small or nonexistent.

In the latter case the lift force is created by the fuselage of the missile or by the thrust of the engine.

Controlling the lift force. The lift is determined by the angle of attack. Thus control of the lift force comes down to creating the necessary angle of attack.

Let us see how this control is accomplished. First of all, it is necessary to create moments which will take the missile out of a state of dynamic equilibrium when the angle of attack is zero.

Missiles are usually designed so that their flights are stable even without stabilizing apparatus, and any deviation of the angle of attack from the zero position generates a moment which stops this deviation. As a result, the missile flies stably, continuously orienting its longitudinal axis in the direction of the velocity vector. A necessary condition for such stable motion is placing the center of application of aerodynamic force of the missile behind (with respect to the direction of flight) the center of gravity. An ordinary feathered arrow which flies stably with its point forward answers this condition. If an arrow is made without a point or feathers, it will fly unstably when shot from a bow.

A mechanical analog of a stable missile can be represented as a rod supported at the center of gravity and held by springs from both sides (Fig. 15). If a force is applied to the end of the rod and we attempt to take the rod out of equilibrium, this will give rise to an

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opposite action from the spring. After the force on the rod ceases, it again takes the initial position.

Missile guidance by using an aerodynamic rudder is seen as follows. When the rudder turns, a small aerodynamic lift force is created on it perpendicular to the direction of motion. The lift force of the rudder is created in the same way as the lift force of the wing: when the plane of the rudder is parallel to the air stream, there is no lift; if an angle of attack is created, then there is an aerodynamic force perpendicular to the direction of motion of the missile (Fig. 16). Since rudders are placed on one end of a missile, i.e., far from the center of gravity, a large aerodynamic moment, equal to the product of the force and the arm, arises relative to this center of gravity. Under the influence of the aerodynamic force, the missile begins to turn about the center of gravity until there is no angle of attack, in which the moment of the elastic aerodynamic force of the wing compensates the moment from the rudder.

There are two basic plans for rudder positioning: the "canard" configuration and the normal tail plan. Thus a missile can be compared to a lever of either the first or second kind from the fulcrum to its center of gravity.

Mechanical analogs of both these plans are shown in Fig. 17a and b.

Since the force arm l_p is much greater than the arm of the aerodynamic lift force l_k, force from the side of the rudder Y_p must be as many times less than the lift force of the wing Y_k.

Characteristics of gasdynamic guidance. Gasdynamic guidance is the creation of the required lift force for controlling a missile by using a jet engine.

Gasdynamic control takes on principal importance at high altitudes

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and low speeds, when aerodynamic forces are either small or nonexistent.

The first result of the absence of aerodynamic forces is loss of the natural balance of the missile.

In principle, stabilization of a missile in space can be accomplished in two ways.

The first method consists of power gyrostabilization, the essence of which is the following. A heavy, high rpm gyroscope is installed in the missile. The axis of the flywheel is connected to the body of the missile and prevents its turning. This system is cumbersome and heavy, but it can be used.

The second method of stabilization is by using gas rudders. When the fuselage of the missile deviates from a given direction, sensitive elements record this deviation and an automatic control system using gas rudders turns the missile to the required position. The operating principle of gas rudders is analogous to that of aerodynamic rudders. The difference lies in the design. A gas rudder is made of heat resistant materials and is placed in a gas jet, which, flowing past the rudder, acts on it as would an air stream (Fig. 18). As a result, a gasdynamic lift force and a corresponding moment which turns the housing relative to the center of gravity are created on the rudder.

The lift force required to change the direction of motion when the axis of the engine is fixed relative to the fuselage of the missile is formed by turning of the entire body by gas rudders.

Inasmuch as it is assumed that the axis of the engine and, correspondingly, the direction of the thrust in this case pass through the center of gravity of the missile, then only a lift force will be created when the body turns, and there will be no moment relative to

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the center of gravity. Therefore, after the body of the missile turns by a definite angle, the gas rudders must return to a neutral position.

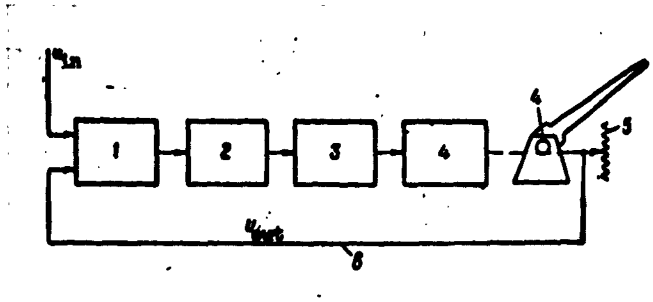
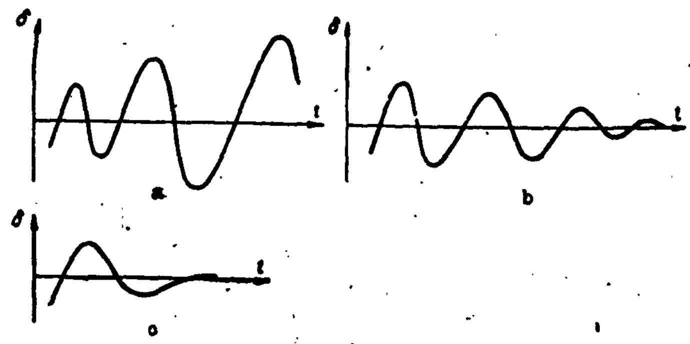


Fig. 1. Block diagram of automatic control system
1) sensitive element; 2) device for command generation; 3) amplifier; 4) barrel drive; 5) potentiometer for measuring the position angle of the barrel; 6) feedback loop.

Fig. 2. Processes in various automatic control systems:
a) unstable system; the amplitude of the mismatch signal has a tendency to grow; b) the system has low stability; c) the parameters of the command-formation system are correctly chosen; the process is quickly attenuated.



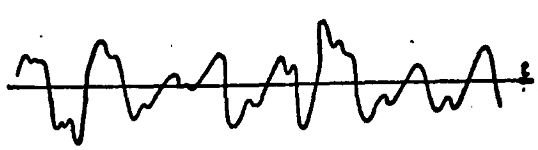
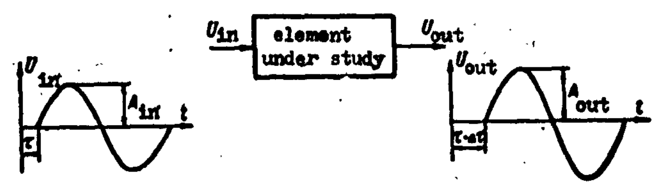


Fig. 3. Nature of change of fluctuating noise with respect to time.

Fig. 4. Input (U_{in}) and output (U_{out}) signals when taking the frequency characteristic.



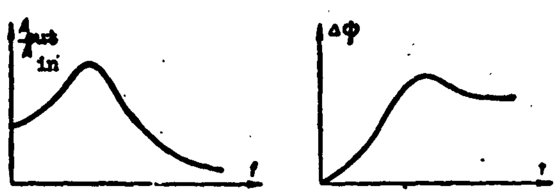


Fig. 5. Amplitude-frequency and phase-frequency characteristics.

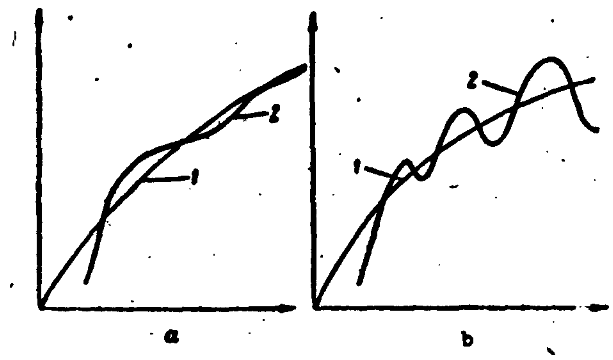


Fig. 6. Stability (a) and instability (b) of motion of a missile: 1) trajectories of assigned motion; 2) actual trajectories of missile.

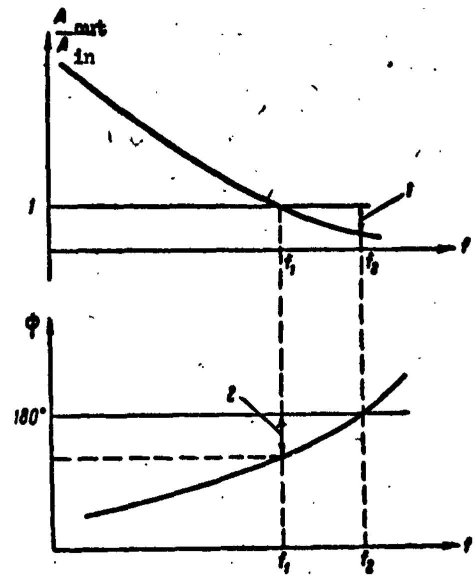


Fig. 7. Frequency characteristics of an open system: 1) amplitude stability margin; 2) phase stability margin.

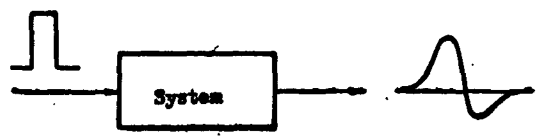


Fig. 8. Action of the system on a pulse.

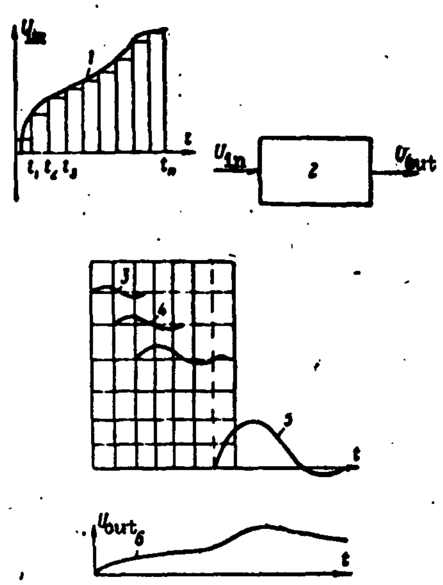


Fig. 9. Finding the output signal with an input signal of arbitrary shape: 1) input signal; 2) system under study; 3) action of system on first pulse; 4) action os system on second pulse; 5) action of system on n-th pulse; 6) output signal, found as sum of all elementary signals.

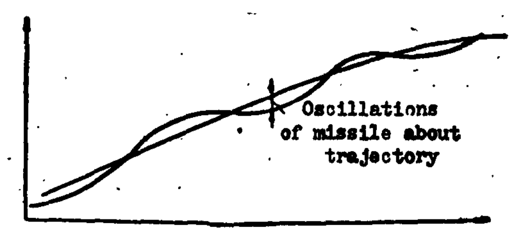


Fig. 10. Motion of missile in the presence of fluctuating disturbances

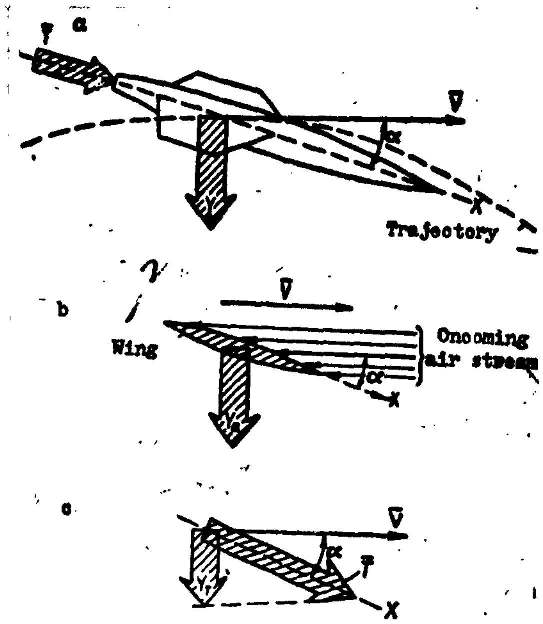


Fig. 11. Lift force diagrams: a) over-all diagram; b) aerodynamic lift; c) lift due to thrust; V is the velocity vector; X the longitudianl axis of the missile; the angle of attack; T the thrust; Y the total lift force; Y_a the aerodynamic lift; Y_T the lift component due to thrust.

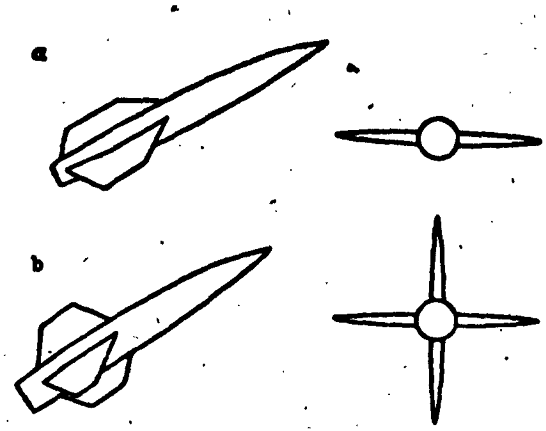


Fig. 12. Two configurations for aerodynamic surfaces: a) single-plane aerodynamic plan; b) two-plane symmetrical aerodynamic plan.

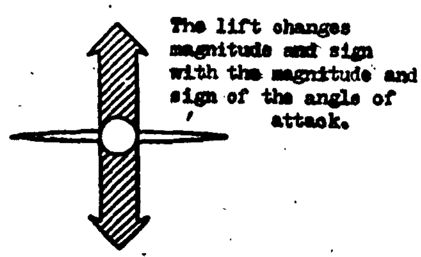


Fig. 13. Direction of lift in a single-plane plan.

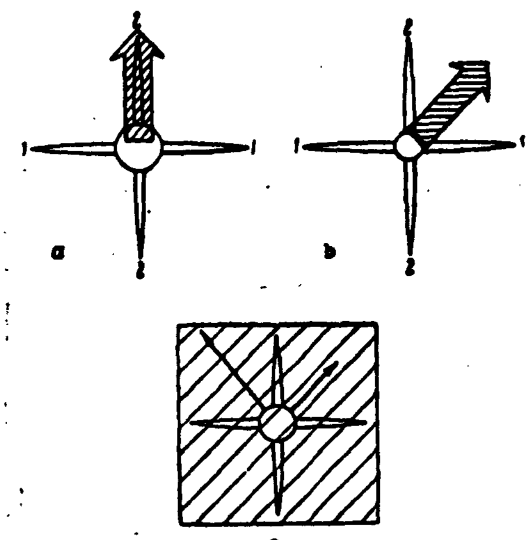


Fig. 14. Lift force in a two-plane aerodynamic plan: a) angle of attack in plane 2--2; b) angle of attack in planes 1--1 and 2--2; c) region of possible lift forces for various combinations of angles of attack.

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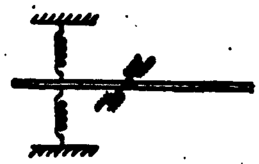

Fig. 15. Mechanical analog of a stable missile.

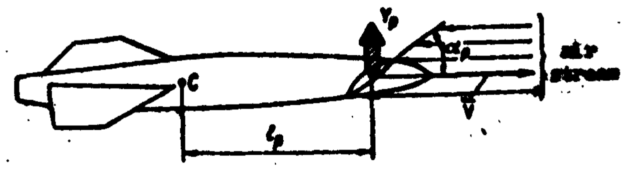

Fig. 16. Missile guidance using an aerodynamic rudder: a_p is the angle of attack of the rudder; Y_p the aerodynamic lift force of the rudder; V the direction of the velocity of the missile; C the center of gravity of the missile; and l_p the force arm of the rudder.

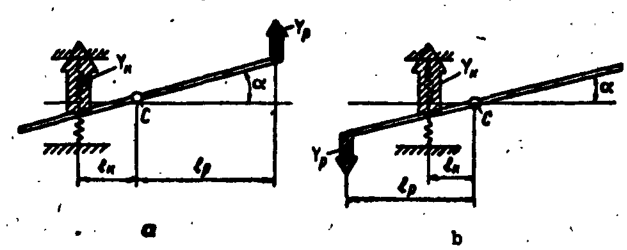

Fig. 17. Mechanical analogs of two control plans: a) "canard" plan; b) normal plan; α is the angle of attack; Y_k the lift force of the wing; Y_p the lift force of the rudder; l_k the force arm of the wing; l_p the force arm of the rudder; C the center of gravity.

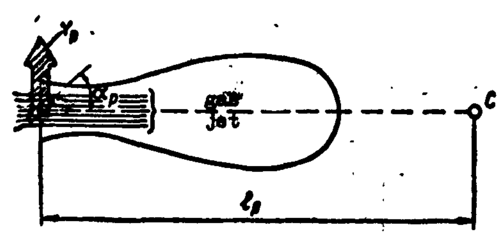

Fig. 18. Diagram of action of gas rudder: Y_p is the gas dynamic loft force of the rudder; α_p the angle of attack of the rudder; C the center of gravity of the missile; and l_p the force arm of the gas rudder.

CHAPTER 2

FLIGHT STABILIZATION OF A MISSILE

1. The Principles of Stabilization

For normal operation of a guidance system, it is necessary that a guided missile possess a number of properties which will ensure stability of its flight characteristics. These properties are the following.

1. A missile must have a definite flight stability.

2. The axes of the missile must have a definite orientation in space.

3. The parameters of the missile (lift force, turning speed of its axes, etc.) must remain constant when the commands are the same.

In the previous chapter it was noted that a missile usually possesses an inherent flight stability. This means that if any forces take it out of dynamic equilibrium, after they cease to act it will return to stable flight.

However, inherent stability can be insufficient, and then the missile must be equipped with stabilizers which act on the rudders and which oppose the action of oscillations of the missile and stabilize its motion relative to the mean position of the axes.

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Besides this stabilization, which protects the missile from harmful oscillations, there must also be on the missile a device which ensures a stable mean position of its axes. The absence of such stabilization in autonomous guidance systems can make guidance of the missile impossible. In command methods of guidance, even if the missile is continuously controlled from the ground, the absence of a definite orientation of the transverse axes can lead to incorrect execution of commands, i.e., to disturbance of the functioning of the guidance system.

Special flight stabilizing apparatus is installed in a missile, in order to ensure stable flight.

Let us examine the widely used methods of flight stabilization using gyroscopic instruments.

2. FLIGHT STABILIZATION USING GYROSCOPIC INSTRUMENTS

The concept of gyroscopes. Everyone has had an opportunity to observe an ordinary top. Its fundamental property is the ability to hold its axis of turning fixed in space.

This property of a top is used in the so-called three-degree or position gyroscope.

Note that only ideal gyroscopes can hold the position of their axis absolutely fixed in space. Real gyroscopes have the property of precession, in other words, a drifting of the axis. The quality of a gyroscopic system is chiefly determined by the magnitude of its drift with respect to time.

Let us consider another property of a top. We attempt to push a spinning top, and we discover that it starts to deviate not in the direction of the imparted force, but in a direction perpendicular to

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it. This property of the direction of precession is used in the second basic type of gyroscope: the two-degree or damping gyroscope.

Stabilization of the axes of a missile by using a three-degree gyroscope. When the missile turns, in order to keep the position of the axis of turning of the gyroscope fixed in space (this axis is called the principal axis), the gyroscope is mounted in a special gimbal, the operation of which is explained in Fig. 19. The rotor of the gyroscope turns in the inner frame about the OX axis of the gimbal. The inner frame turns freely in the outer frame about the OY axis, and, finally, the outer frame turns relative to the body of the missile. A three-degree-of-freedom support allows the gyroscope to preserve its position in space with any turnings of the missile.

In fact, if the missile turns relative to the OZ axis, the inner frame remains fixed relative to the OZ axis. The same thing happens with the inner frame, if the missile turns relative to the OY axis. If the turning is relative to the OX axis, the gyroscope will not detect this turning, inasmuch as it itself turns relative to this axis with a high angular velocity, and an insignificant turning of its bearings is in no way reflected in the operation of the gyroscope.

This gyroscope support makes it completely fixed relative to the missile, and, therefore, in such a system the gyroscope is called free.

For the location of axes shown in Fig. 19, the gyroscope can measure turning of the missile relative to the OZ and OY axes. The angle of turning of the missile is measured by a special potentiometer, which is rigidly connected to the body of the missile, and its slide is rigidly connected to the turning axis of the gyroscope frame. (Fig. 19 shows a potentiometer which measures the turning of the missile about the OZ axis).

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Any turning of the body of the missile about the OZ axis corresponds to turning of the potentiometer slide relative to a conditional zero, owing to which the voltage U_p from the potentiometer varies in proportion to the angle of turning of the missile relative to some set position. The same thing happens if the potentiometer slide is connected to the OY axis, except that in this case the turning of the rocket relative to the OY axis is measured.

The measured value of the signal, proportional to the turning angle of the missile relative to the required position, is fed through amplifiers to the rudder devices. By acting on them this signal forces the missile to return to the original position.

Thus this three-degree-of-freedom gyroscope can measure the angle of turning of the missile relative to fixed axes of space and a stabilizer of its axes.

<u>Stabilization of the speed of turning of a missile by using a two-degree-of-freedom gyroscope</u>. Now let us examine the use of a gyroscope as a measurer-stabilizer of the turning speed of the axis of a missile.

The need for such measurers arises when the rate of turning of the missile must be strictly fixed or when its flight must be stabilized.

The problems in both cases are sufficiently compatible, and a gyroscope, which is a stabilizer of the angular velocity can be a device for stabilizing the missile at the same time. For this, a <u>two-degree</u> or <u>velocity</u> gyroscope is used.

As distinguished from a three-degree, free gyroscope, a two-degree gyroscope has only one frame in the gimbal (Fig. 20).

We have already spoken of the property of precession of a top. Precession is also a property of gyroscopes: if an external perturbing moment is applied, the gyroscope begins to turn on another plane, and, conversely, if a gyroscope with a gimbal is turned about the OZ axis, as shown in Fig. 20, then a moment which is perpendicular to the plane of rotation acts on the gyroscope, i.e., about the OY axis.

If this moment is inhibited by a spring, the potentiometer slide will be deflected by an angle which is proportional to the angular velocity relative to the OZ axis.

Thus the voltage U_p read from the potentiometer will be proportional to the angular velocity of the missile.

If some command is sent to the rudder device, the missile will turn with a definite angular velocity. Inasmuch as its flight regime can vary substantially for the reasons examined above, then various angular velocities can result from the same command.

The measurer of angular velocity eliminates this to a considerable extent.

The output signal from the velocity gyroscope goes through a feed-back loop to an amplifier and is compared with the command signal. As a result, a signal for the mismatch between the given command and the velocity of rotation is generated. When the system has a high Q-factor, owing to the presence of amplifiers, even when the error signal is weak, the required turning speed of the missile, proportional to the given command, is established.

Thus, by using automatic control methods, an accurate agreement between the angular velocity and the given command is ensured.

A velocity gyroscope can also ensure the required flight stability. In this case the velocity gyroscope is a correlating element of an automatic control system which stabilizes a missile relative to center of gravity. If there is no stabilization, the missile can oscillate about its center of gravity. With these oscillations, the angular velocity becomes highest when the missile passes through the zero position of

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the angle of attack. Passing through this neutral position, the missile, because of inertia, will be deflected in the other direction, despite the fact that when flying in an air stream natural damping is created due to the resistance of the external medium which is proportional to the angular velocity of the missile, this damping can be insufficient and the amplitude of the oscillations can exceed the allowable values. In this case g-forces which are dangerous to the structure of the missile arise, the operation of all on-board apparatus is complicated and the flight can be disrupted due to excessively large angles of attack. In order to avoid this, as indicated in Chapter 1, artificial damping must be introduced by feeding into the control channel a preventive, braking command. This command, the magnitude of which is proportional to the turning speed of the missile axis, is generated by the velocity gyroscope.

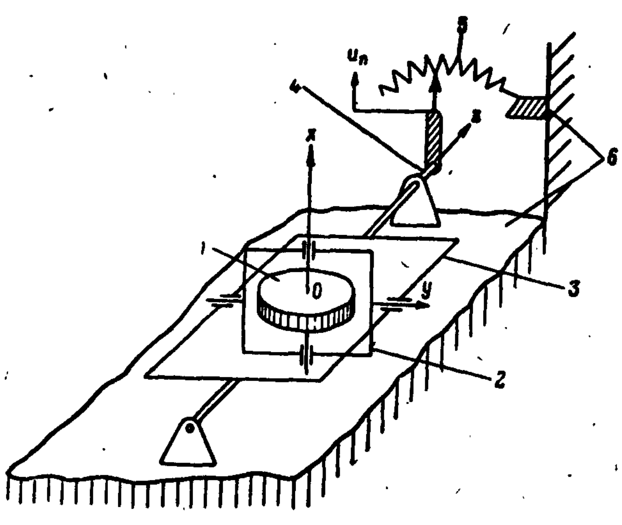

Fig. 19. Diagram of three-degree-of-freedom gyroscope: 1--rotor; 2--inner frame of gimbal; 3--outer frame of gimbal; 4--measuring axis; 5--measuring potentiometer; 6--body of missile.

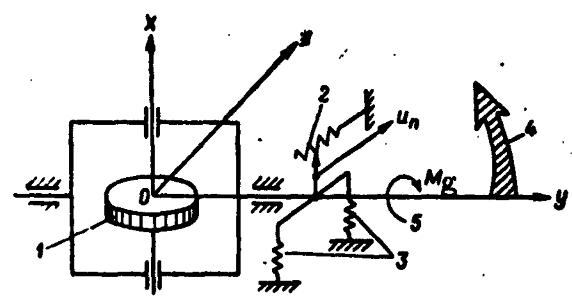

Fig. 20. Diagram of a two-degree gyroscope: 1--rotor; 2--potentiometer; 3--spring; 4--direction of turning of missile; 5--momentum of gyroscope M_g.

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Chapter 3

INTERNAL GUIDANCE SYSTEMS

1. Definition of Internal Guidance Systems

In internal guidance systems the on-board apparatus generates control commands in accordance with a previously assigned program for the law of motion.

This law can be motion with a constant course or motion at some angle above the horizon.

Internal systems can control the altitude, speed, etc. of the flight.

Astronavigation systems are also internal systems. The distinguishing feature of these is the possibility of continuous automatic adjustment of the location of missile with respect to stars by using on-board optical means; this allows the program of its motion to be corrected.

Internal systems, because of their tactical and technical characteristics, can be used only in surface-to-surface (ship-to-ground) missiles, and sometimes in air-to-ground missiles. This limitation is unavoidable, since an internal guidance system acts according to a previously developed program, which cannot take into account the

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possible variations of target motion; these types of guided missiles are designed chiefly for stationary targets.

Internal systems can be used for guidance of ballistic missiles and missiles with an aerodynamic configuration. In the first case guidance is accomplished in the initial phase, and in the second case, during the entire flight of the missile up to the point of impact.

2. INTERNAL GUIDANCE SYSTEMS FOR "V-2" TYPE BALLISTIC MISSILES

General Principles of Fire Control

Let us examine one possible version of a ballistic-missile guidance system.

Before firing, a ballistic missile is set up vertically on a launching pad with strict maintenance of orientation of its axes relative to the countries of the world. Then, taking all corrections into account, the firing plane of the missile is set up (Fig. 21). For reasons which will be mentioned below, the preliminary direction of the flight plane of the missile will differ somewhat from the direction to the target.

After vertical launching and declination at a given altitude, the missile is put on course. With any deviation of the missile from a given course, the three-degree gyroscope gives out an error signal. A rudder command is generated by the guidance system which is maintained until the missile is back on course.

However, under the influence of various external disturbances (wind, aerodynamic asymmetry of the missile, etc.), parallel drift of the missile is possible. Inasmuch as in this case the axis of the missile will maintain the required direction, the error signal from the course gyroscope will be zero, since gyroscopic systems react only to turning and do not react to linear displacement of the missile. In order to keep the missile in the firing plane, on-board devices are used which measure these linear deflections; the guidance system reduces them to zero and thus the missile is kept in a given plane of fire.

Now let us see how firing range is controlled.

After declination, the missile continues to fly at a constant angle above the horizon, close to 45° (Fig. 21). This trajectory is maintained in the vertical plane by using a special measuring device which determines the declination of the axis in the pitch (vertical) plane from the required direction. This device operates in the same way as the course control device.

According to the error signal received, a command is generated which acts on the rudders, causing the missile to keep on a straight line lying in the vertical plane at an angle of about 45° above the horizon.

But the range of a missile is not determined by this angle alone. It is also necessary that the missile have a definite velocity at a given point in the trajectory (at the instant of engine cutoff). For this the engine must have strictly fixed thrust, which also allows a fully determined programed velocity to be obtained at each point in the trajectory.

Inasmuch as, at a given angle of firing, the range is chiefly determined by the velocity of the missile, the range is controlled by an internal system for measuring the velocity, which continuously determines the velocity of the missile by using a special measuring device.

At the moment the given velocity is reached, the engine is cut off, i.e., its operation ceases instantly. The device for cutting off

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the engine is also one of the instruments of automatic guidance of a ballistic missile.

After the engine is shut off, the missile continues to fly in a ballistic trajectory, subjected only to the forces of gravity and air resistance (in the lower layers of the atomsphere).

Besides the devices which control the course and range, there is also a device for automatic stabilization of the banking of the missile.

Axis Stabilization and Control of Declination

The fundamental system for stabilization of course, pitch and bank is a precision gyroscopic system. The principles of stabilizing missile axes were examined in Chapter 2.

Pitch control is accomplished by using a program device.

The voltage from the gyroscope potentiometers, which measures the angle of pitch, corresponds to the angle between the plane of the horizon and the longitudinal axis of the missile. This voltage is fed to a computer, where it is compared with the voltage from the program device.

When these voltages do not match, an error signal is generated and a pitch command is formulated which forces the axis of the missile to take the required position. Thus the axis of the missile follows the program signal.

There can be other solutions for the program declination of the missile. For example, in the German V-2 rocket, declination was accomplished by program turning of the potentiometer stator, which was mounted on the body of the rocket. Striving to keep near zero, the potentiometer slide, which is connected to the pitch gyroscope, the missile follows the turning of the potentiometer.

The lateral-correction system. Measurement of lateral drift in a lateral-correction system is accomplished by a pickup-measurer of lateral acceleration.

The simplest layout of a pickup-measurer of acceleration (accelerometer) is shown in the block diagram of the course control (Fig. 22). An inertial body (b), which can move along the directrices (c), is mounted on springs (a) in a special housing. The deviation of this body from the neutral position is proportional to the acceleration acting along the directrices, and is measured by potnetiometer (d). By knowing the acceleration at each time instant, it is possible to find the velocity by adding the accelerations.

Then there is successive summation of the velocities, which makes it possible to measure the transverse linear displacement of the missile. These operations are performed by integrators.

An integrator is a device which continuously calculates the sum of very small increments. This operation can be performed by any counter (watt-hour meter, gas meter, water meter, odometer, etc.). For example, an odometer turns at a rate which is directly proportional to the speed of the automobile. When the speed of the automobile changes, there is a corresponding change in the turning rate of the odometer. Thus an odometer keeps a running count of the distance covered.

The counter-integrators in the computers of missile guidance systems operate similarly. If, for example, a varying voltage U is fed to an electric motor and the speed of the motor varies in strict proportion to this voltage, then the number of revolutions measured by a counter connected to the axle of an electric motor will be the result of the continuous summation or integration of the input voltage U.

Besides these errors, which are caused by instrument errors in the guidance system, the strike accuracy is also a function of the accuracy in calculating the program.

One of the important factors affecting the firing accuracy is the earth's rotation and the deviation of the missile from the firing plane caused by it.

Therefore, as was mentioned above, the initial position of the firing plane must differ somewhat from the direction to the target.

In order to explain the reason for this difference, let us consider the following example.

Let us launch a missile absolutely vertically and assume that there is nothing to change its flight trajectory. Returning to earth, the missile does not land at the point from which it was launched.

In order that it return to the launching point A (Fig. 26), it is necessary that the missile at all times maintain a radial direction relative to the center of the earth. Even before launching, the missile has a certain velocity V_0 due to the earth's rotation, equal to the rotational velocity of point A. After leaving the earth, this velocity component remains unchanged. If the missile climbs to a great altitude (point B), then, in order that it not lag behind the motion of radius OB, it must have a horizontal velocity which is somewhat greater than V_0, since the path BB' is greater than AA'. Owing to this lagging behind, the missile falls to earth somewhat closer to the point from which it was launched. This can be calculated mathematically by using the formula for Coriolis acceleration.

In long-range firing, the error caused by this factor can reach tens of kilometers. Therefore, this error must be taken into account.

From the output of the integrator, a signal which is proportional to the linear lateral motion is fed to the course control instruments, and the missile starts to move in a direction opposite to that of the drift. This motion will continue until the correction is completed, and the error signal equals zero.

Range Control and Firing Errors

In order to ensure the required firing range, it is necessary to determine accurately the moment for cutting off the engine; this is accomplished by a special velocity meter.

The block diagram of this part of the system is shown in Fig. 23. Velocity measurement is made indirectly, as the lateral displacement is determined.

The initial parameter for finding the velocity is the longitudinal acceleration of the missile, which is determined by the accelerometer; by successive summation of values of acceleration, the axial velocity of the missile is determined. The output signal of the velocity meter is sent to the comparison unit.

When the program velocity is reached, a command for shutting off the engine is sent.

The accuracy of range firing ΔX(Fig. 24) is a function of the error with which the required angle of pitch ($\Delta\theta$) is maintained at the moment of engine cutoff; of the error is measuring the velocity (ΔV); of the deviation of the point at which the engine is cut off from the set altitude (ΔH) and range (ΔX).

The lateral deviation (ΔZ) (Fig. 25) of the missile at the moment of cutting off the engine is determined by the course error ($\Delta\psi$) at the moment of shutting off the engine and by the parallel drift ($\Delta 1$).

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3. Inertial Systems Of Internal Guidance

Inertial systems of internal guidance operate by measurement of the linear distances covered by the missile.

As distinct from the inertial pickups examined above for lateral correction, where these devices pick up only the error in the main guidance system, in inertial systems the pickups measure the total distance covered by the missile.

Let us examine the operating principle of an inertial guidance system. On board a guided missile there is a special gyro-stabilized platform on which the accelerometer is mounted. The gyro-stabilized platform can be made in the form of an inner framework of a three-degree-of-freedom gimbal (Fig. 27), which makes it fully independent of the position of the body of the missile.

Three free gyroscopes are mounted on the platform. These gyroscopes can measure any deviations of the platform from the required position and can stabilize the platform in space.

Thus, owing to the gyroscopes, the platform is stabilized in space relative to fixed axes.

The accelerometers are mounted on the platform, therefore, they are stabilized in space and measure the accelerations along fixed axes. The accelerometer signals are sent to integrators, where the linear displacement of the missile relative to fixed axes is measured.

As a result, there is continuous measurement of the position of the missile in an absolute (world) coordinate system, or, as is sometimes said, the course of the missile from the starting point is read.

Simultaneously, in this same coordinate system, the required (program) trajectory of the missile is sent to a computer. If the missile deviates from the required trajectory, this computer forms

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an error signal which acts on the rudders of the missile, forcing it to return to the assigned trajectory.

In an absolute coordinate system, the point on the earth's surface which the missile must hit is also moving, due to the earth's rotation. Therefore, to calculate the position of this point relative to fixed axes, it is necessary to introduce the correct time into the computer, which is given by a special on-board timing device.

In practice, it is more advantageous to calculate the distance covered by the missile not in a fixed, absolute coordinate system, but in the plane of the horizon (i.e., in the plane tangent to the earth's surface at the point where the missile is located). For this purpose, it is possible to correct continuously the position of the gyro-stabilized platform so that it is constantly parallel to this plane. If in this case the missile maintains a constant altitude by means of an internal guidance system, then the course can be read by using two accelerometers.

Such, in general form, is the operating principle of an inertial system. The most difficult problem for inertial systems is providing accurate operation of the gyro-stabilized platform and the accelerometers. Errors in gyro-stabilization lead, in the final analysis, to shifting of the firing plane of the missile, and the systematic instrumental error in the accelerometer as a result of the storage effect in the integrator is the greatest source of error in reading the course.

Besides these systems of internal guidance, there are a number of other internal guidance system which are not examined in this book.

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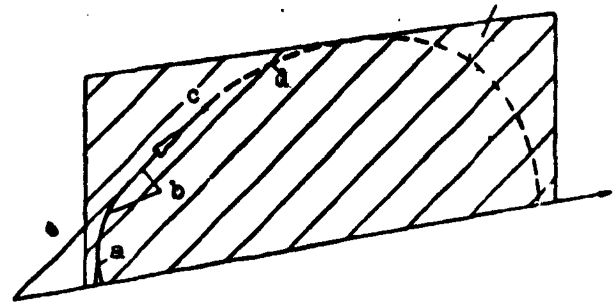

Fig. 21. Trajectory of a ballistic missile; a) vertical phase of flight; b) declination phase; c) engine-cutoff point; d) ballistic trajectory.

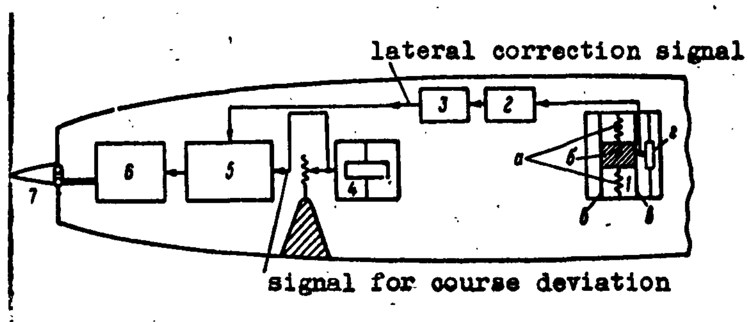

Fig. 22. Block diagram of course control; 1) lateral-acceleration pickup; 2 and 3) integrators; 4) gyroscopic device; 5) command-formation unit; 6) rudder drive; 7) rudder.

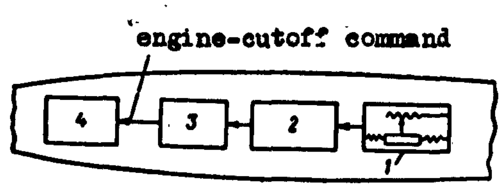

Fig. 23. Block diagram of engine-cutoff device: 1) axial-acceleration pickup; 2) integrator-determiner of velocity; 3) comparison unit; 4) engine-cutoff control.

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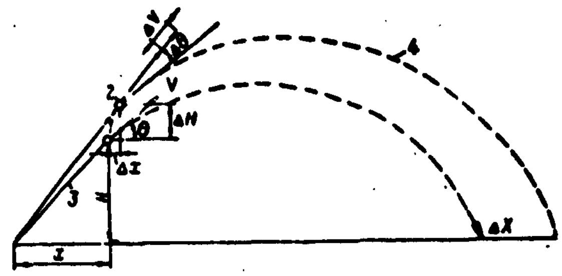

Fig. 24. The components of range error: 1) calculated point of engine cutoff; 2) actual point of engine cutoff; 3) guidance phase; 4) ballistic trajectory phase; X, H, θ, V are the calculated values of the range, altitude, deflection angle and the velocity at the moment of engine cutoff; ΔX, ΔH, Δθ, ΔV are the errors in range, altitude, deflection angle and velocity at the actual point of engine cutoff; and ΔX is the over-all range error.

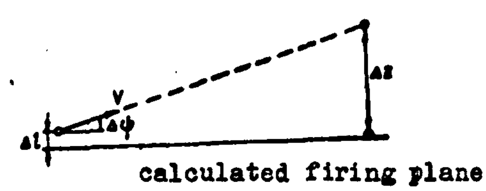

Fig. 25. The components of course error: Δl is the lateral deviation of the point of engine cutoff from the calculated firing plane; Δψ the deviation of the velocity vector at the moment of engine cutoff; and ΔZ the over-all lateral firing error.

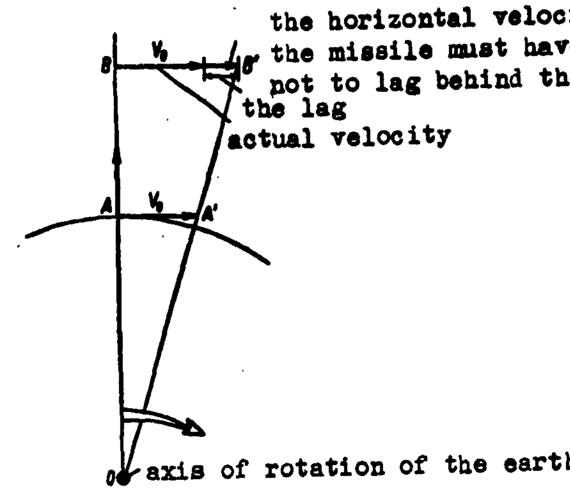

Fig. 26. Relative displacement of missile due to the earth's rotation.

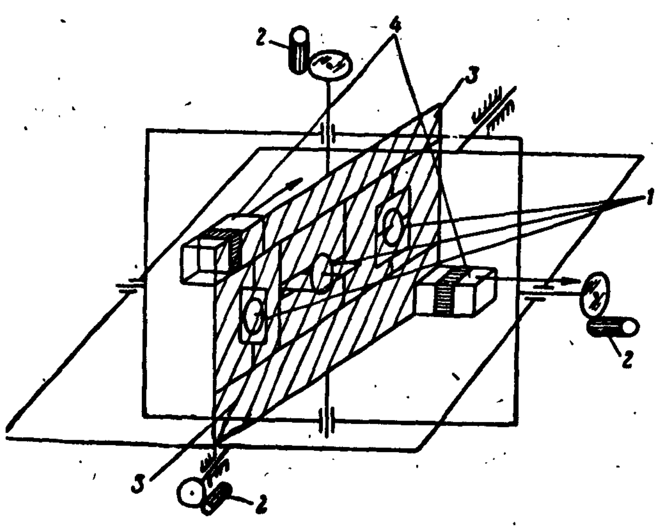

Fig. 27. Block diagram of gyrostabilized platform:
1) stabilizing gyroscopes; 2) stabilization gears; 3) gyro-stabilized platform; 4) acceleration pickup.

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Chapter 4

COMMAND METHODS OF GUIDANCE

1. Systems of Command Guidance

Command methods of missile guidance (remote control) are those methods in which the law of motion of the missile is determined at a control point outside of the missile itself.

The control point can be on the ground (for surface-to-air and surface-to-surface missiles), in an airplane (for air-to-air and air-to-ground missiles), on a ship, or some other place. Command methods of guidance are used for all types of missiles, including in the initial phase of ballistic missiles.

Command guidance of missiles can be used alone or preliminarily with transition to homing, when it is necessary to increase the accuracy of guidance in the terminal phase of engaging the target.

Depending upon the means and place of generation of the error signal, command guidance systems may be divided into the following three types: radio sighting, beam guidance and guidance by two radar tracking stations.

<u>The radio sighting method.</u> After launching, the missile is

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"fired" into a radar sighting beam and is kept in this beam by commands generated at the control point and transmitted to the missile.

The radar beam is directed to the required impact point. In some cases the radar beam can coincide with the target, but in the general case of a moving target, this will be a predicted point.

The radar antenna is made so that the axis of the radiation pattern does not coincide with the geometric axis of the antenna. In this case the tapered scanning beam forms an <u>equal-signal</u> zone (RSZ) in space, its axis (which coincides with the axis of the antenna) is called the <u>equal-signal-zone axis</u> (Fig. 28 a).

The operation of many modern radar devices is based upon the principle of the equal-signal zone. The presence of an equal-signal zone makes it possible to determine the direction and magnitude of the angular deflection of a radiator from the RSZ axis.

The radiator can be a target exposed to radiation from radar or a transmitter on board a guided missile.

An equal-signal zone can be formed not only by a scanning beam, but also by several fixed beams shifted relative to one another. There must be two such beams in guidance relative only to azimuth, and in guidance relative to azimuth and angle of elevation simultaneously, there must be four.

A scanning beam creates a signal of variable strength from a radiator which does not lie on the RSZ axis. The received signal will fluctuate with the rotation rate of the beam (Fig. 28 b).

Up to definite limits, the amplitude of the signal (ΔU) is proportional to the angular deflection of the missile from the RSZ axis (Fig. 28 c).

The variable part of the signal with amplitude ΔU is the error signal of the guidance system. Depending upon the amplitude and phase of the error signal, a command is formulated for guiding the missile.

Thus in radio sighting, the radar does not track the missile, but only creates a sighting line. The missile is sighted in the equal-signal zone, i.e., its deviation from the axis of this zone is determined (hence the name "sighting," by analogy with an optical sight, in which deviation from the crosshairs is determined).

Let us see how the predicted direction of a radar beam is found.

The prediction is made in order to decrease the curvature of the trajectory.

The principle of firing with prediction can be explained by the example of antiaircraft artillery.

As is well-known, in antiaircraft artillery the angular position of the predicted point is chosen so that during the flight of the missile to the point of impact, the target moves with respect to the prediction angle. The prediction angle is chosen before firing and cannot change during the flight. Therefore, if the target makes an anti-antiaircraft maneuver or the velocity of the missile is incorrect, it will fly past the target.

The main characteristic of guided-missile flight is the possibility of continuous correction of the position of the predicted beam, which allows a change in the course of the target and the velocity of the missile to be taken into account.

In order to determine continuously the required prediction angular position of the beam, it is necessary that information on the coordinates of the target and the range be supplied continuously to the computer.

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If, for example, the prediction point is chosen as it is done in antiaircraft artillery, then the missile will fly to the impact point along a trajectory which is close to ballistic. In this case the missile will maneuver only as a function of deviations in the laws of motion of the target and the missile.

If we let ϵ_t and β_t be the angle of elevation and azimuth of the target (Fig. 29), then the predicted position of the radio-sighting axis of the missile will be

$$\epsilon_r = \epsilon_t + \Delta\epsilon$$
$$\beta_t = \beta_t + \Delta\beta$$

The values of the running angular prediction $\Delta\epsilon$ and $\Delta\beta$ must be decreased continuously as the missile approaches the target (times t_1 and t_2).

The condition for an accurate hit is that the angular prediction becomes zero, when the ranges to the target (D_t) and the missile (D_r) are equal (i.e., at the moment of impact); otherwise, the missile will pass by the target.

Let us examine the block diagram of a radio-sighting guidance system (Fig. 30).

The coordinates of the target (ϵ_t, β_t and D_t) from the radar tracking of the missile are sent to a computer, where the predicted angle of elevation and azimuth of the missile $\Delta\epsilon_r$ and $\Delta\beta_r$ are formulated on the basis of these data. In order to make the prediction angle more precise, information about the range to the missile is used, which is determined in radio sighting by special range-finder systems.

The radio sight is forced to turn according to the data from the computer. A signal δ, proportional to the angular deviation

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of the missile from the axis of the radio-signal zone (RSZ) of the radio sight, is sent to the command-formation unit and the commands are sent on to the missile.

Remember that in ordinary radar tracking the signal δ controls the turning of the radar antenna, i.e., it is the error signal of the tracking system. In this case, however, the signal serves as the basis for command formation for missile guidance, i.e., it is the error signal of the guidance system.

The guidance command is sent to the missile and acts on the rudder control circuits.

A receiver-responder operating from pulses of the radio sight should be placed on board the missile, for increased reliability of operation of the sighting channel.

The system examined, with a radio sight having a narrow radiation patter, allows only one missile to be guided to the predicted point, although in principle a radio sight can sight several missiles simultaneously. This happens because for missiles at various ranges the law of prediction will be different and the moment when the angle of prediction must become zero (i.e., when the RSZ axis must "look at" the target) will also be different for them.

The situation changes when the missile is guided by the three-point method.

In guidance by the three-point method, the direction of the beam to the missile must always coincide with the direction to the target. Therefore, the angles of elevation and azimuths of all the missiles, independent of their range, will be equal to one another and equal to the angle of elevation and azimuth of the target at a

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given time instant. Thus simultaneous guidance of several missiles to a single target is possible, in principle, using one radio sight.

Another feature of guidance by the three-point method (also called the coincidence method) should be mentioned: in this type of guidance the radio sight can perform the function of locator of the target. This coincidence circuit is structurally very simple. However, guidance by the three-point method is limited by the great curvature of the trajectory, which requires highly maneuverable missiles.

Let us examine the block diagram of a combined radio sight (Fig. 31). Such a radio sight must have two channels: a clear radar channel for tracking the target and a channel for radio sighting of the missile (or missiles). The signals of the separate channels can differ in frequency from the signal to be received (frequency selection) due to the difference in range (range selection or according to some other command. Each method of selection has its own advantages and disadvantages. Let us examine the operation of a system with frequency selection.

The channel for tracking the target operates as an ordinary radar channel: a signal reflected from a target and modulated by the frequency of the scanning beam enters the receiving channel and, acting on the drive of the radar antenna, forces it to track the target.

The signals arriving from the missiles have different frequencies: those of the on-board responders.

As a result, the signals from the missiles go along other channels of the radio-sight receiver. Separation of the sighting signals takes place in these channels, i.e., of signals which are

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proportional to the angular deviation of the missiles from the RSZ axis (δ_1, δ_2). On the basis of these signals, commands are formulated which are sent to one or another of the missiles along the appropriate transmission line.

Guidance by a radio beam differs from radio sighting in that the error signals of the guidance system δ_1 and δ_2 are formed not in the radio sight, but on board the missile itself, and, therefore, there is no need for command lines. As in the case examined above, when firing at a predicted point only one missile at a time can be guided by the beam, while in guidance by the three-point method, several missiles can be guided simultaneously. In this case the radar which creates the radio beam can track the target simultaneously.

A special receiver, tuned to the radio tracking frequency is installed on the missile. As in guidance by radio sighting, a radio track can be created by a narrowly directed scanning beam or several beams which are fixed relative to one another. In this case each beam must have its own "color": a distinct frequency, specially frequency modulated, etc.

Let us examine the principle of error-signal formation for the case of the scanning beam in Fig. 28.

A revolving beam creates an alternating field strength at all points not lying on the axis of the equal-signal zone. The signal received by the missile will oscillate with the scanning rate (see Fig. 28 b). The amplitude of the signal (ΔU), up to definite limits of deviation of the missile from the RSZ axis, will be proportional to this deviation (see Fig. 28 c). If the angular deviation of the missile from the RSZ axis exceeds the level of the linear part of the modulation characteristic, then the normal functioning of the

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guidance can be disrupted and it will cease to operate as a guidance system. The scanning beam makes it possible to determine only the deviation of the missile from the RSZ axis, and does not determine its direction.

In order to determine the direction of the deviation, special reference signals must be sent to the missile which are synchronized with the scanning beam. Let us assume that when the scanning beam is to the left of the RSZ axis, a reference signal of one form is sent to the missile (for example, a single pulse), when it is to the right, a signal of another form is sent (two pulses), and when it is above or below, other special signals are sent. Further, let us assume that the missile is to the left of the RSZ axis. Then the maximum value of the received signal will coincide with the reception of a single-pulse reference signal. Thus the coincidence of a maximum signal with the reception of a single-pulse reference signal attests to the fact that the missile is to the left of the RSZ axis. If the maximum signal coincides with reception of a two-pulse reference signal, it is to the right, etc. The reference signal can be sent in the form of a smoothly varying curve, and the error signal isolated by a special phase detector, where the magnitude and direction of the deviation of the missile from the RSZ axis are determined immediately.

If there are several antennas with various "colors" for each beam, a reference signal is not needed, inasmuch as the direction of the deviation is determined by the "color" of the signal.

<u>Double radar tracking.</u> In this guidance method, one radar set tracks the target, while the other tracks the missile (Fig. 32). Each radar set determines all three coordinates of the target and missile, which are sent to the computer.

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On the basis of these data, and error signal and the corresponding commands are formed and sent to the missile by radio. These commands are picked up by command receivers and go to the rudder control elements.

There is also a receiver responder in the on-board apparatus, which receives pulses from the radar station, amplifies them, and retransmits them back to earth. This increases the stability and range of the automatic escort of the missile.

Let us analyze some special features of this method of guidance, as compared with the radio-sighting and radio-track methods.

When using radio-sighting or radio-tracking methods, guidance is accomplished by keeping the missile within the area of some beam.

The geometric picture of guidance (or, as it is called, the kinematic diagram) is as follows. A beam is directed towards the target at some predicted point in space. While in motion, the missile is continuously held on this beam. Deviation of the center of gravity of the missile from the beam is recorded by the guidance system as an error signal, and the system attempts to the keep the error signal at zero.

Such a system may be called beam guidance. This system is obviously the only possible one for these methods.

In guidance with double radar tracking, not only beam homing possible, but also course guidance. In beam homing by the double radar tracking method, the apparatus differs little from that shown in Fig. 30. The device for generating the predicted direction of the beam ε_r and β_r is based on the same principle. The difference lies

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in the fact that the predicted values of the angular coordinates generated by the computer do not guide the beam, but are compared with the true angular attitudes of the missile. An error signal is generated on the basis of this comparison.

Owing to radar tracking of the missile, it is possible to set up its trajectory and compare it with the required trajectory by using the computer. An error signal is generated on the basis of this comparison.

In the most simple case, the required course is the direction to a target. An error signal in this case is a signal for the deviation of the velocity vector of the missile from the direction to the target. This is called homing by the pursuit method. However, this method has, as does the three-point method, the disadvantage that the trajectories for homing on a moving target have great curvature, which requires highly maneuverable missiles. In addition, the total duration of its flight can be considerably greater than that of a straight line to the point of impact. Therefore, the use of this method is limited to firing on fixed and low-speed targets.

In general, the required course of a missile is directed to a predicted point, i.e., the missile flies for interception of a moving target. Such a method is, for example, the method of parallel engagement, in which the target-to-missile line remains parallel to itself at all times. This method, as will be shown below, provides considerable straightening of the trajectory.

Let us examine the block diagram of the course method of control shown in Fig. 33.

From the known coordinates of the target entering the computer, it is possible to construct the course of the target and to calculate

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its velocity V_t (Fig. 34). The missile coordinates from the radar station allow its velocity V_r to be determined.

By knowing the mutual position of the target and missile and their velocities, it is not difficult to find the predicted point of impact C by extension of the missile course.

Ordinary trigonometric equations can be used to solve this problem. For this, it is necessary to solve triangle ABC (Fig. 34).

Side AB and angle A are determined in the computer by the coordinates of the target and missile and the course of the target. The position of the predicted point, which lies on the target trajectory, is determined by a condition according to which the flight times of the target and missile up to this point are equal. The flight time of the target is found by the formula $t_t = \frac{AC}{V_t}$, and the flight time of the missile by the formula $t_r = \frac{BC}{V_r}$.

Inasmuch as $t_t = t_r$, then $AC = BC \cdot \frac{V_t}{V_r}$.

Bearing this expression in mind, it is not difficult to solve triangle ABC by trigonometric formulas and find the angle σ, which is the required prediction angle, and then the required course θ_r.

Further, it is necessary compare the required course with the true course of the missile. From the known coordinates of the missile, its true course θ_{true}, which is compared with the required course θ_r, is formed in the computer. From this comparison, an error signal $\Delta\theta_r$ is formed, which controls further the homing system by the commands received on board.

Control commands, as in any automatic control system, must be formed taking into account ensuring the required stability of motion.

For simplicity, let us examine the problem of course control in one plane. The problem is more complicated in the case of

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spatial guidance. However, the essence principles of control remain the same.

In this case, two angles must be determined in the computer: the course and pitch of the missile, and, correspondingly, two error signals for formation of the course and pitch control commands.

2. ELEMENTS OF COMMAND-FORMATION DEVICES

The error-signal formation unit. For the double-radar method of command guidance, the difference between the angular coordinates $\Delta\beta$ and $\Delta\epsilon$ is not difficult to formulate and can be done with potentiometers and amplifiers, for example. With potentiometer circuits, the coordinates of the required position of the missile ϵ_r and β_r are given out by the computer as a d-c voltage. Information about the true position of the missile is taken from the potentiometers, which are mechanically connected to the radar antenna tracking the missile. Comparison is carried out in a difference amplifier. At the output of this amplifier, a voltage is obtained which is proportional to the difference between the angular coordinates $\Delta\beta$ and $\Delta\epsilon$; this difference is the error signal.

It is more difficult to form error signals in the other two systems examined.

In these systems, when the missile deviates from the RSZ axis, the signal at the output of the receiver is a sum of two voltages: a d-c voltage V_0 and an a-c voltage ΔU with frequency equal to the scanning rate Ω (Fig. 35).

In this case the modulation factor $m = \frac{\Delta U}{U_0}$ in some scale determines the magnitude of the angular deviation of the missile from the RSZ axis.

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The signal obtained can be expressed as follows :

$$U = U_0 \, [1 + m \cos \, (\Omega t + \Omega)],$$

where Ω is the scanning rate, and ψ the phase (angular position) determining the position of the missile in the pictorial plane (the plane perpendicular to the RSZ axis in Fig. 36).

It is necessary to determine the modulation factor m and phase ψ from this signal. The fluctuating part of the signal, which can be isolated by a filter, is $U = U_0 m \cos \, (\Omega t + \psi)$, i.e., proportional not only to the modulation factor m, but also to the signal U_0 is a function of the signal strength, of the stability of the receiver, etc.

Therefore, the first task of the error-signal separation unit is to obtain the alternating signal in the form

$$U = m \cos \, (\Omega t + \psi).$$

This can be accomplished by using an amplifier which has an amplification factor which is inversely proportional to U_0.

The task also consists of determining the phase ψ of the alternating signal or the components of the factor m which correspond to the angular deviation of the missile with respect to the Y and Z axes (Fig. 36). This operation is performed in a special phase detector, where the phase of the reference is compared with the phase of the error signal. The comparison is made in two switching circuits. The phase of the reference signal is always strictly constant and tied to the position of the scanning beam. Therefore, as a result of comparing the phases of the reference signal and error signal, the position of the missile in the pictorial plane is determined, i.e., the components of the deviation of the missile with respect to the Y and Z axes are determined.

The unit for converting angular error into linear. When the angular error (error signal) is constant, the linear deviation will increase as the range of missile from the control point.

If a command is formulated on the basis of an angular error, then a single command will correspond to various linear deviations of the missile from its trajectory. Therefore, at long ranges the missile will approach the trajectory for a longer period of time Such guidance conditions are unsuitable, and it is advantageous to use the linear error for command formation.

For example, in the Oerlikon-54 system, the linear error l is determined by multiplying the angular error $\Delta\varphi$ by a function which is proportional to the range to the missile R:

$$l = R \cdot \Delta\varphi.$$

Multiplication is performed by using a potentiometer multiplier. In this R is introduced approximately by using a time mechanism (Fig. 37).

Units which take coordinate skew into account. The phenomenon of coordinate skew arises in spatial missile guidance. The essence of this phenomenon is that in flight the position of the axes of the missile relative to the axes of the measuring system in which deviations are determined is changed.

The magnitude of this skew is described by a certain angle τ, which can be called the angle of skew or the angle of banking of the axes of the missile relative to the coordinate system of the measuring device.

As has already been noted, it is necessary for missile guidance that its axes be definitely stabilized in space. In particular, starting from the condition of roll stabilization, it is possible

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to force the missile to keep its transverse axis parallel to some fixed plane in space. This can be seen by the example of an ordinary airplane. The roll stabilization system in an airplane operates so that the plane of the wing is always held in a horizontal position, i.e., it does not turn relative to the longitudinal axis.

In the general case, the axes of a missile can be stabilized in space arbitrarily.

Let us examine the formation of the skew angle for missile homing by radio sighting (Fig. 38).

At the moment of lauching the direction of the transverse axes of the missile coincide with the direction of the reference axes of the radar installation. The flight direction coincides with the RSZ axis. Let us assume that the missile is fired in the plane P_c an an angle ϵ_0 above the horizon (P_h).

Let us call this the firing plane of the missile. For simplicity, let us assume that the missile does not leave this plane during homing and that automatic roll stabilization does not allow the missile to turn relative to its longitudinal axis. Then the transverse axis Z_r will always lie in the firing plane P_c.

The missile can accomplish any maneuvers during homing. Let us assume that it has turned 90° as a result of a complicated maneuver, flown to point 2 and will fly further in a radial direction, i.e., the X_r axis is directed along the RSZ axis. It is not diffiuclt to see that there will be an angular skew ϵ_0 at this point.

In fact, according to the cnndition, the Z_r axis is in the firing plane and, as is apparent from the figure, it makes an angle ϵ_0 with the plane of the horizon. Thus an observer at point 0 will see the missile axes turned an an angle ϵ_0. The reference axis of

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the radio sight Z_1 remains horizontal at all times. Therefore, the skew angle is ϵ_0.

Let us examine a simplified model in the presence of skew (Fig. 39).

Let the missile be on the Y_1 axis of the reference system with deviation Δ_{1y} from the RSZ axis as some moment of time (point 1).

In this case the automatic control system generates a command which is proportional to Δ_{1y} and which forces the missile to move to the center of the equal-signal zone.

However, inasmuch as the axes of the missile are turned relative to the Y_1 and Z_1 axes, the command, which creates a lift force along the transverse axis Y_r, will force the missile to move to point 2, which lies outside of the vertical axis Y_1.

At point 2, the command will be redistributed along the axes of the missile in accordance with the legs Δ_{2z} and Δ_{2y}. Motion proportional to Δ_{2z} and Δ_{2y} will take place in the axes of the missile and it will move to point 3, etc. By successive construction of the curve of the center of gravity of the missile in the axes of the reference system, it is not difficult to see that it approaches the center of the equal-signal zone not in a straight line, as it should, but along some spiral, i.e., along a longer route. It is obvious that this is undesirable, since it introduces additional delays in command fulfillment, i.e., it is as if the guidance system takes on additional inertia, and its sensitivity can be worsened considerably.

In order to prevent this, it is necessary to send a command to the missile so that the total lift force is directed towards the center.

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This is accomplished by a special device. Let us assume that the missile is deflected along the Y_1 axis by the value ΔY (Fig. 40a). If the deviation ΔY is divided into two command components $K'_y = \Delta Y \cos \tau$ and $K'_z = \Delta Y \sin \tau$ and these are sent to the missile, then it is not difficult to see that the missile will move precisely toward the center of the RSZ.

Similarly, it can be shown that, if the deviation is only along the Z_1 axis (Fig. 40 b), then the commands must be $K''_z = \Delta Z \cos \tau$ and $K''_y = - \Delta Z \sin \tau$. It is easily seen that in the presence of simultaneous deviations with respect to the Y_1 and Z_1 axes, the command must be the sums:

$$K_z = K'_z + K''_z \text{ and } K_y = K'_y + K''_y.$$

We have examined only the simplest case of coordinate skew and the simplest method for its compensation. This phenomenon is considerably more complicated in spatial guidance.

3. TRAJECTORIES

<u>The phases of trajectories.</u> In command systems, the trajectory can be divided into two basic parts.

The first part is usually called the launching phase; and the second part is called the guidance phase.

In the first phase, the law of motion can not be set exactly beforehand. The trajectory in this phase is determined by the position of the missile at the moment of the beginning of its guidance. This position can be random and is determined by many factors, e.g.: accuracy of firing the missile into the beam of the guidance station, the position of the missile at launching, deviation of the flight direction from that assigned , the attitude of the carrier aircraft at the moment of launching, etc.

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The guidance phase begins when these initial conditions are completely compensated. From this time on, we may speak of the accuracy of guidance, since deviations can be sufficiently great in the first phase (Fig. 41).

An independent initial launching phase can also be isolated. In this phase the missile is unguided either because it has not yet reached the required velocity, or because the guidance system has not "taken over" (for example, when firing a missile into a radio beam).

The characteristics of the end of this phase also determine the initial conditions for placing the missile in the guidance trajectory.

Missiles can be guided with the engine either on or off.

The flight phase when the engine is on is called the power phase, and the part when the engine is off is called the free-flight phase.

A property of the free-flight phase is that here the missile loses part of its lift force due to cessation of the action of gas-dynamic forces. The loss of lift force manifests itself strongly at high altitudes because, in this case, the aerodynamic lift force is decreased.

A second property of this phase is an additional bending of the trajectory, due to the fact that the missile is decelerated by any resistance in this phase.

In Fig. 42, let us trace the appearance of this bending of the trajectory. Let the missile move uniformly in the power phase, i.e., let the force of any resistance be compensated by the engine thrust. A missile guided by the command method will always remain on some beam, which moves with a constant angular velocity. Then over equal

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time intervals ΔT, this beam will move from point 1 to points 2, 3, 4, etc.; the angular shifts will be unchanged.

Accordingly, a missile moves the distance $\Delta S = V \cdot \Delta T$ from point O_1 for the time ΔT and hits point O_2. If the velocity of the missile V is constant, with sufficiently careful construction of the trajectory it can be seen that the angle of turning of the missile $\Delta\theta$ during time ΔT will be constant, i.e., the missile carries out the same maneuver at all times.

If from a certain time (point O_4) the velocity drops due to engine cutoff, the path ΔS_1 will be shorter for time ΔT and, as is easily seen from the figure, the turning angle $\Delta\theta_3$ will be increased. In this case the missile must, in order to fulfill the conditions of guidance, make another maneuver: it must move about a trajectory of greater curvature. This effect can be especially strong in guidance by the three-point method, where the trajectory has considerable curvature even without this.

Various types of guidance trajectories. One of the basic requirements for trajectories is maximum linearity. This is due to the limited maneuverability of guided missiles, because it may happen that, when the curvature of the trajectory is great, the missile will not be in a position to carry out all required maneuvers, and it will not reach the target.

On the other hand, the most simple method, as already stated, is guidance by combination (three-point method), which requires considerable g-forces. This method makes guidance possible when the angular coordinates of the target are known and ensures, within limits, simultaneous firing of several missiles upon a single target.

As already been mentioned, beam guidance is characteristic of

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command methods, wherein the trajectory can be considered the locus of points lying on beams issuing from a single control point.

During guidance, a missile must be held on this beam at all times.

In the case when the beam follows the target continuously, we have guidance by the three-point method. When the beam is directed to a point ahead of a moving target, we have a prediction method of guidance.

If the law motion of the beam and the velocity of the missile V_r are known, it is not difficult to determine the trajectory.

As an example, let us construct the trajectory of a surface-to-air missile for guidance by the three-point method (Fig. 43).

Let us assume that the target moves uniformly. Let us divide the trajectory into equal segments. Then the intervals between the points t_1, t_2, $t_3, \ldots,$ t_n will be the same and equal to Δt. Let us connect point 0 (control point) to these points by beams.

If at time t_1 the missile is at point C_1 on beam 1, then at time t_2, having flown the distance $C_1C_2 = V_r \cdot \Delta t$, it will be at point C_2 on beam 2. To find point 2 we place a compass point at point C_1 and intersect beam 2 with a radius $V_r \cdot \Delta t$. Having done this, we find the points C_3, C_4, etc.

Having connected all points $C_1, C_2, C_3, \ldots, C_n$ with a smooth curve, we obtain a target trajectory on which it is easy to determine the moment of impact of the missile with the target. It is not difficult to see that, as the velocity of the target increases, the curvature of the trajectory will increase all the more. At some target velocities the maneuverability of the missile will be

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insufficient.

The flight trajectory obtained here is an ideal one, i.e., it is determined by a given law of motion for a guidance system. In reality, as in any automatic control system, there are deviations in the motion of the object under control.

A missile, moving close to this ideal (kinematic) trajectory, describes a certain real trajectory, the magnitude of the deviation of which is determined by the quality of guidance, i.e., it is a function of the error in the guidance system.

The form of the trajectory varies according to the guidance system used.

For surface-to-air missiles, it is possible to use combined (three-point) as well as prediction methods of guidance.

For air-to-ground missiles, which are designed to destroy fixed or low-speed targets, three-point methods are applicable, inasmuch as the trajectory has low curvature in this case.

These types of missiles can be guided according to the principle of the equal-signal zone. Range control of ballistic missiles with limited power phase is accomplished by velocity control. In this case the trajectory in the vertical plane is ballistic.

Winged missiles can fly at a constant altitude, and the moment of diving toward the target is determined either by internal means or by command from the control point.

The characteristic feature of air-to-ground missiles is that the control point moves continuously, together with the carrier aircraft.

Figure 44 shows an example trajectory for guidance in this case; it is constructed by the same method as was the trajectory of the surface-to-air missile (Fig. 43).

The fact that for air-to-air systems the control point is in motion leads to substantial changes in trajectory. In particular, in guidance by the three-point method, when a definite course is maintained by the carrier aircraft, the missile trajectory can be completely linear.

For example, the course of the carrier aircraft can be chosen such that the line from the control point to the target is mutually parallel at all times (Fig. 45). In this case the three-point method coincides with the method of parallel engagement.

If it is assumed that the velocity of the target is uniformly linear, while the velocity of the missile is held constant, then the trajectory will be a straight line.

Since the missile velocity is assumed to be constant, the segments $C_1C_2 = C_2C_3 = C_3C_4 = \ldots = C_{n-1}C_n$. It is apparent from this that the shaded right triangles are equal. Therefore, the angle formed by the trajectory and the line HT (Fig. 45) will remain the same at all times, i.e., the missile holds a constant course and flies linearly.

We have examined trajectories for beam methods of guidance. For course guidance the trajectory of parallel engagement is characteristic.

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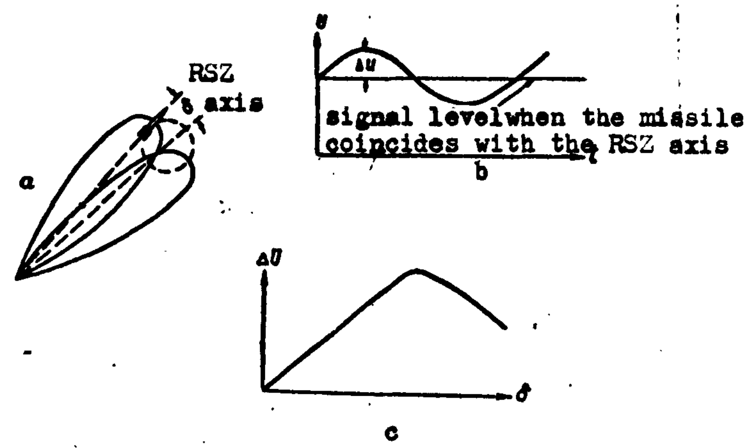

Fig. 28. Formation of an error signal with a scanning beam: a) scanning beam; b) fluctuation of signal at receiving point; c) amplitude ΔU as a function of deviation δ.

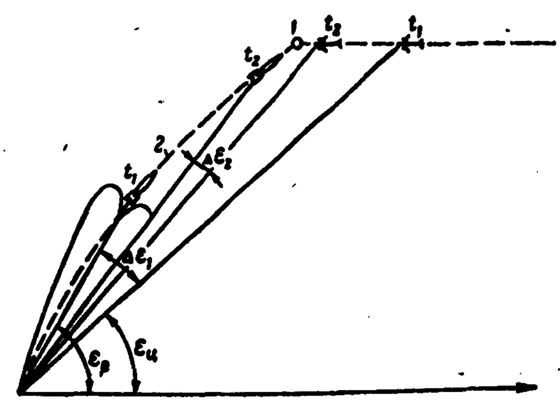

Fig. 29. The positions of the target and missile in guidance to a predicted point: 1) predicted point of impact; 2) trajectory of missile; $\Delta\epsilon_1$ and $\Delta\epsilon_2$ are values of the angular prediction at times t_1 and t_2.

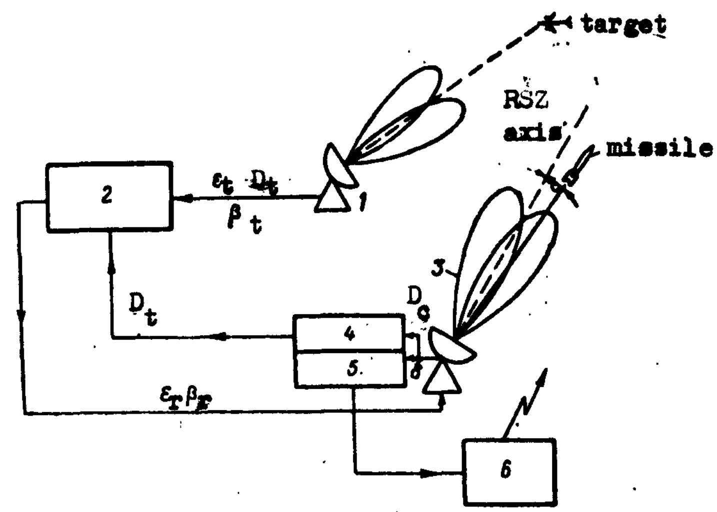

Fig. 30. Block diagram of ground complex for guidance by radio sighting: 1) radar tracking of target ; 2) computer for predicted values of azimuth β_r and angle of elevation δ_r of missile; 3) radio sighter; 4) missile range finder; 5) command-formation unit; 6) command transmission line.

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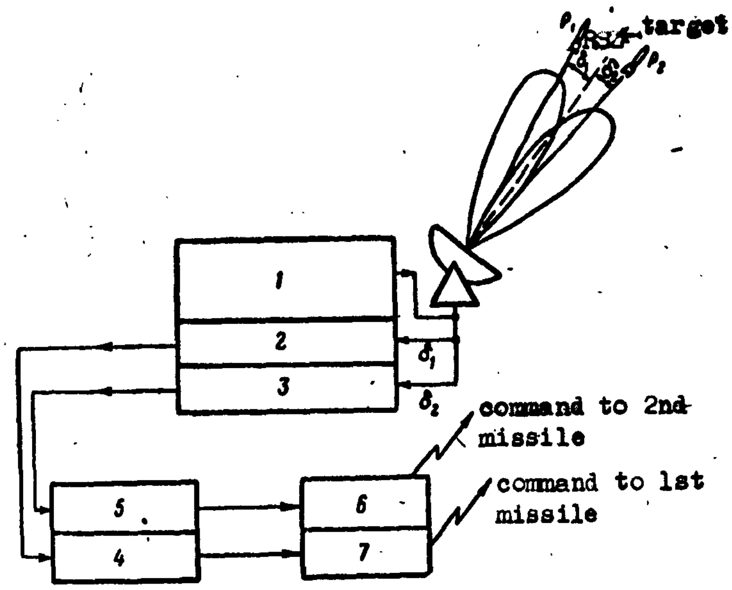

Fig. 31. Block diagram of a radio sight combined with radar target tracking: 1) radar tracking; 2), 3) sighting units for the first and second missiles; 4), 5) command-formation units for the first and second missiles; 6), 7) command-transmission lines.

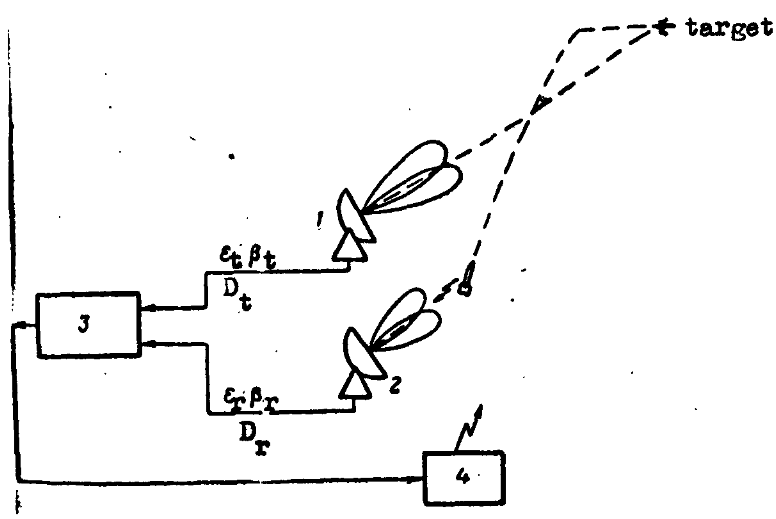

Fig. 32. Block diagram of ground complex for guidance by two radar signals: 1) radar tracking of target; 2) radar tracking of missile; 3) command-forming computer; 4) transmission line for **commands**.

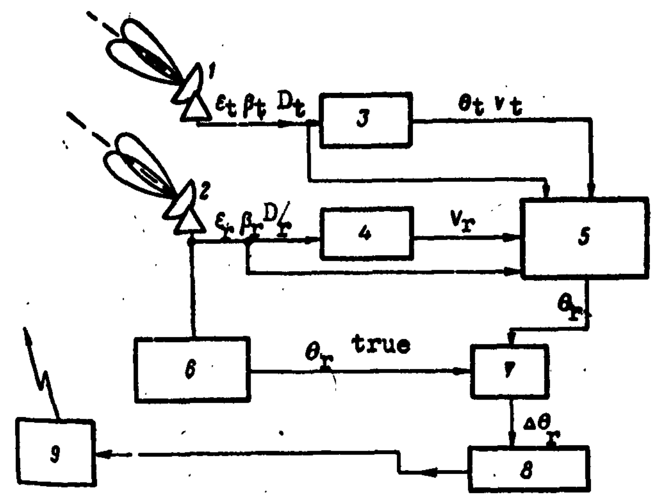

Fig. 33. Block diagram of course control method. 1) radar tracking of target; 2) radar tracking of missile; 3) device for determining the course and velocity of target; 4) device for determining the velocity of the missile; 5) device for solving the predicted triangle and forming the required course θ_r; 6) device for determining the true course of the missile $\theta_{r\ true}$; 7) comparison unit and unit for formation of error signal $\Delta\theta_r$; 8) command-formation unit; 9) transmission line for commands.

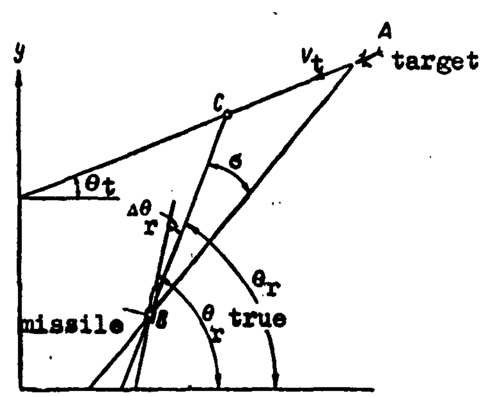

Fig. 34 Homing with course guidance: θ_t- course of target; C - predicted point of impact; σ - prediction angle; θ_r- required **course of missile; $\theta_{r\ true}$- true course of missile; $\Delta\theta_r$- error signal.**

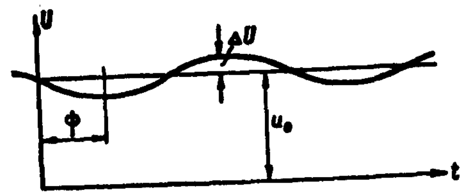

Fig. 35. Output signal of receiver in the case of a scanning beam.

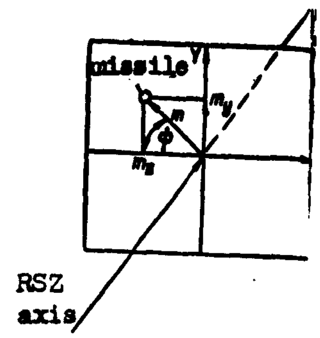

Fig. 36. Deviation of missile is pictorial plane.

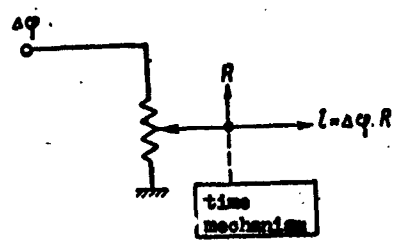

Fig. 37. Block diagram of converter of angular error to linear used in the Oerlikon-54 system.

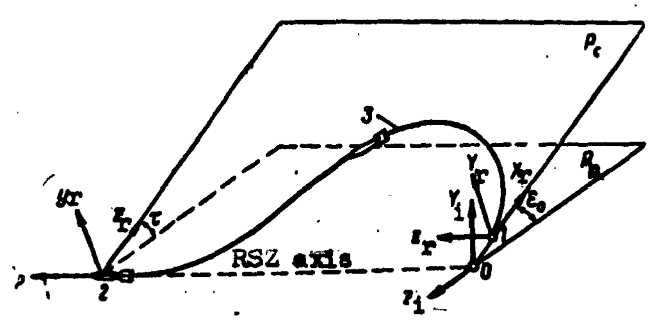

Fig. 38. Formation of skew angle τ: P_h is the plane of the horizon; P_c the firing plane; 1) the initial position of the axes of the missile; 2) the final position of the missile; 3) trajectory; Y_1, Z_1 is the final position of measuring the radio-sight axes.

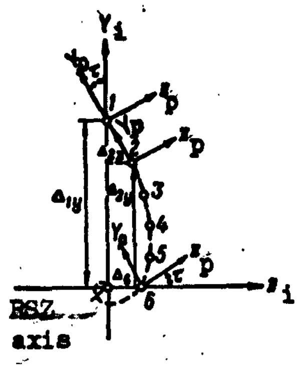

Fig. 39. Initial deviation Δ_{1y} in the presence of skew.

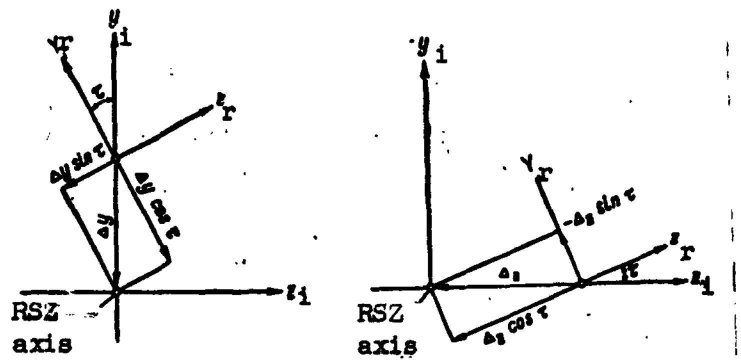

Fig. 40. Redistribution of commands taking the skew angle into account.

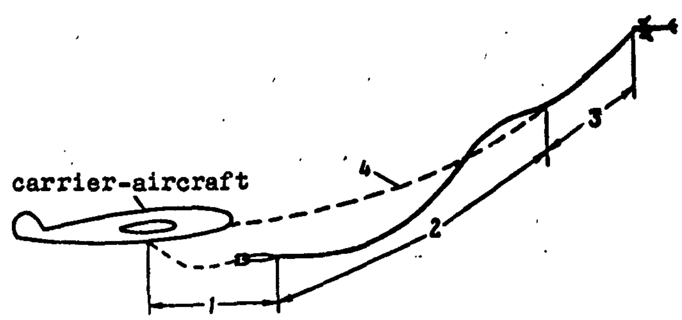

Fig. 41. Phase of the trajectory of a guided missile: 1) launching phase; 2) entry into trajectory; 3) guidance phase; 4) required trajectory.

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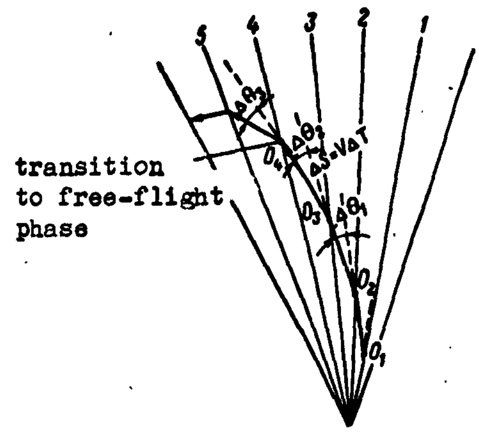

Fig. 42. Additional bending of trajectory during transition to the free-flight phase.

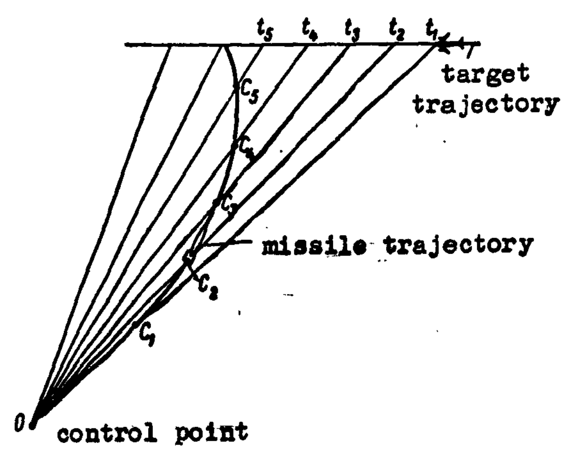

Fig. 43. Trajectory of surface-to-air missile in three-point guidance.

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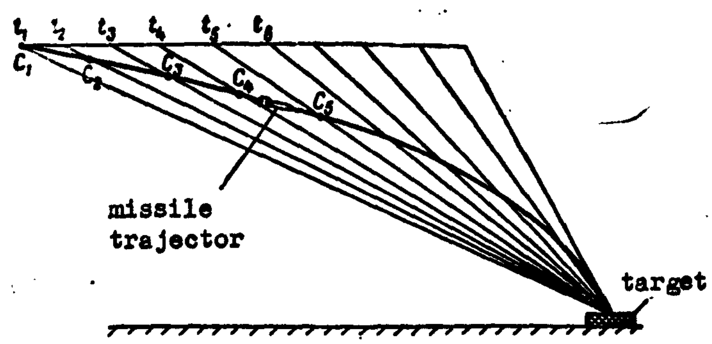

Fig. 44. Trajectory of air-to-ground missile with three-point guidance.

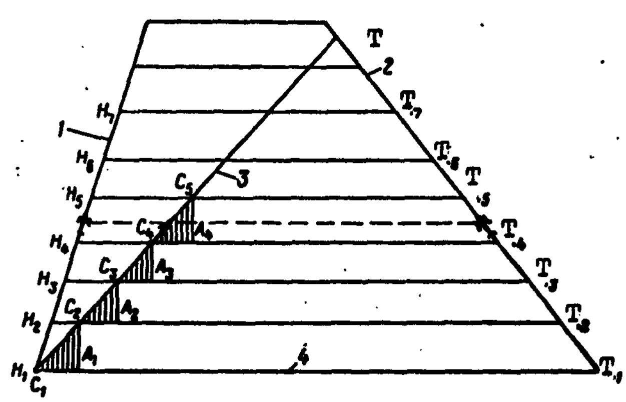

Fig. 45. Trajectory of air-to-air missile with three-point guidance: 1) carrier aircraft (control point) trajectory; 2) target trajectory; 3) missile trajectory; 4) guidance line.

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Chapter 5

HOMING METHODS AND COMBINED METHODS OF GUIDANCE

1. The Operating Principles of Homing Systems

A classification of homing methods. Homing methods for missiles are those in which reception and processing of target information and generation of command signals are carried out by equipment on board the missile.

Homing methods can be used in any type of missile. Abroad, they are most widely used in surface-to-air and air-to-air missiles.

Homing can be the sole means of guiding a missile, or it can be used only in the last stage of guidance.

Inasmuch as in homing, target information is generated in the missile, the guidance accuracy is not a function of the flight range of the missile. This is the main advantage of homing systems over methods of command guidance.

The principle instrument in homing systems is the homing head (target coordinator), which can operate electronically or by reception of heat (infrared) radiation*.

* Sound and visual-light radiation of targets can also be used for homing.

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The action of the homing head is based on the operating principle of the ordinary direction finder or radar installation. The difference lies only in the dimensions and operating conditions, which are complicated by vibrations of the missile, g-forces and effects due to refraction and absorption of the signal from the target in the cowling.

Heating of the cowling and the increase in the level of thermal noise connected with it substantially affect the operation of heat-seeking heads.

In order to receive a signal with which a homing device can operate, the target must be a source of primary or secondary radiation.

With primary radiation the target itself is a source of thermal or radio signals. An operating engine, a nose section heated by aerodynamic resistance, etc. can be sources of thermal radiation. Sources of radio signals can be: radar, radio beacons, radio interference stations, etc.

Systems which use target radiation for guidance are called passive homing systems.

In secondary radiation the target is exposed to a narrowly directed energy source.

If the energy source radiating the target is in the missile itself, then it is an active homing system.

When the radiation source is not in the missile, but at the control point, the system is called semi-active.

Active homing has the advantage that it does not require any equipment outside of the missile. The missile becomes autonomous the moment it enters its trajectory, and the entire external

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guidance system can be used for another missile.

However, in active homing the equipment in the missile is complicated and bulky, since it includes a comparatively powerful transmitter.

In semi-active homing, the radiation seeker must operate from the moment of launching to the moment of impact.

The radiation device is a complex radio engineering installation. Radar tracking of the target can be used for this. The signal reflected from the target is used simultaneously by the radar set and the homing head.

An advantage of semi-active homing is that the target can be subjected to considerably more intense radiation than in active homing, since an external radar station can be more powerful and have larger antennas.

Homing devices can be divided into two basic groups.

The first group contains homing heads which measure the positions of their axes relative to the axes of the missile. They are called connected homing systems.

The second group contains devices in which the homing heads measure the position of their axes relative to fixed axes in space which are set by a gyroplatform. They can be called <u>disconnected homing systems with a reference gyroplatform</u>.

Both homing systems uniquely determine all possible missile trajectories in the homing stage.

<u>Connected homing systems</u>. When the axes of the antenna of the homing head are connected with the missile axes, homing can be accomplished by a direct-pursuit trajectory.

The simplest plan of a connected homing system can be

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constructed as follows (Fig. 46).

The homing-head antenna is located in the nose section of the missile, behind the cowling.

When the longitudinal axis of the missile and, correspondingly, the antenna axis are precisely on target, the error signal equals zero. When the missile axis deviates from the direction to the target, an error signal δ is formed, which is amplified and sent to the rudder drive. This configuration (by analogy with remote control systems) can be called a system of target sighting by a homing head.

Another version of this system can be one in which the head-locator tracks the target, turning relative to the missile axes in a cardan suspension (Fig. 47).

In this case the angular deviation of the axis of the locator from the missile axes is recorded by a potentiometer.

The magnitude of this deviation is the error signal, and the error signal is used for generating a control command.

Common to both systems is the fact that the angular error of the missile axis relative to the direction to the target is used as the control signal.

When the required law of motion is accurately adhered to, the axis of the missile will be directed toward the target at all times during the flight.

The trajectory along which the missile will move will be a curve which is close to a pursuit curve.

The latter requires some explanation. In order to obtain a pursuit curve, it is necessary that the direction of motion of the missile, i.e., the direction of its velocity vector, exactly coincide with the direction to the target. This condition, strictly speaking, is not satisfied in the example in question. This is because the direction of the missile axis differs from the direction of the velocity vector by the angle of attack. When necessary, the angle of attack can be corrected by using an ordinary deflector mounted in the nose section or on the wings of the missile. By changing the angle of attack, it is possible to turn the missile axis so that the direction of the velocity vector coincides with the direction to the target.

At first glance, it seems that accurate direction of the missile to the target is an obligatory condition for engagement. But this is not so. The obligatory condition for engagement is continuous reduction of the distance between the target and the missile. In other words, it is necessary that the velocity of the missile relative to the target be greater than zero at all times.

This is a mathematical condition. It is actually necessary that the engagement time not exceed the maximum flying time for the missile, and that the curvature of the trajectory not exceed certain limits.

A basic disadvantage of connected systems is that the trajectories have low curvature.

When firing at a moving target when the condition of direct pursuit (pursuit curve) is fulfilled, where the velocity vector is at all times directed towards the target exactly, the trajectory of the missile close to the point of impact is greatly bent and it can happen that a similar maneuver cannot be performed by the missile. In this case, starting from some time t_n (Fig. 48), the missile will move with maximum curvature, continuously falling behind the required trajectory. This leads to an additional error in guidance.

It is assumed that this drawback limits the use of such homing methods. It is not difficult to show, by graphical methods, that as the velocity of the target decreases the curvature of the trajectory decreases. Therefore, connected methods of homing can be used only for low-speed targets.

In homing systems with a reference gyroplatform, a gyrostabilized platform, which operates as the one described when examining inertial internal systems, is installed in the missile. The axes of this platform are fixed in space and create a reference coordinate system.

The position of the antenna axes are measured, in this case, not relative to the missile axes, but relative to absolute space axes.

The locator head is mounted in the nose section behind a radio-transparent cowl (Fig. 49). The base of the antenna is mounted directly on the missile. The antenna can be turned about two axes. The slides of potentiometers the housings of which are mounted on the gyroplatform are connected to the antenna axes. These potentiometers record any deviations of the antenna axes relative to the platform (or relative to the absolute axes).

There are several versions of guidance with a disconnected homing head. All these versions give trajectories with less curvature than those with connected heads. This is especially valuable when firing on high-speed targets.

One of the best known methods of guidance which can be used in such systems is the method of parallel engagement. Sometimes this method is called the predicted-point method.

As already noted, the essence of this method is the fact that the sighting line from the missile to the target always remains parallel to itself. In the case of uniform linear motion of the target, a missile having a constant velocity must also move linearly. This distinguishes the method of parallel engagement from the pursuit method.

Parallel engagement when using a disconnected head is carried out as follows (Fig. 50).

The locator head continuously tracks the target. The magnitude of the voltage from the potentiometers records in a fixed coordinate system the deviation δ of the homing head (i.e., the direction from the missile to the target) from the initial position (the angle φ_0).

The magnitude of this deviation is the error signal of the guidance system. A command will be sent to the rudder of the missile until the direction of the missile to the target is parallel to the initial direction.

The error signal can also be the angular velocity of the motion of the locator beam.

These versions are the same in principle. Under ideal conditions for the second method the angular velocity of the target-to-missile line will be zero, and this is equivalent to the condition that the sighting beam be parallel, which is satisfied in the first case. The difference lies only in the fact that, if there is a change in the direction of the missile to the target (for example, because of limited maneuverability) and a new direction is established, then, in the second case, guidance will be relative to the new direction. In the first case, the initial direction is maintained exactly. If an angular error appears, it will be eliminated sooner or later.

2. COMBINED METHODS OF GUIDANCE

Since a homing head installed in a missile has comparatively small dimensions and at long target ranges it cannot provide sure reception of signals reflected from the target at the moment of launching the missile, in a number of systems homing is used only in the last stage of guidance. A missile with a homing head first of all approaches the target, and only after having "caught" it does homing begin.

In this case guidance takes place as follows.

After launching, in the first phase of guidance, the missile is guided either by internal or command systems. Command methods are used when firing on moving targets. Internal systems can be used when firing on fixed and low-speed targets.

Let us examine how the homing head catches the target.

In command guidance the coordinates of the target and missile must be known at the control point. If the missile has a gyro-stabilized platform creating a reference coordinate system, then the direction of the homing-head antenna axis can be set exactly. By knowing the position of the target and the missile and the direction of the homing-head axis, the required target designation can be calculated at the command point and the necessary commands for turning the homing head in the direction of the target can be sent to the missile.

If there is no gyrostabilized platform on the missile, then reading of the angle of turning of the homing-head is made relative to the axes of the missile. The position of these axes can be calculated approximately from the trajectory of the missile.

In internal guidance the command for target search must be sent to the homing-head at a previously calculated time by a special program mechanism.

Let us examine what the conditions at the point of target capture must be in order that a missile with a definite maneuverability be able to capture the target.

For simplicity, let us assume that the target is fixed (Fig. 51). At the moment of capture t_c the target is at a distance ΔS from the missile; the angular error between the direction of the velocity vector of the missile $\overline{V}_r$ and the target is $\Delta\varphi_0$. The magnitude $\Delta\varphi_0$ describes the accuracy of the system (command or internal). In ideal operation of this system, $\Delta\varphi_0$ must be equal to zero.

The capture range ΔS is a function of the power of the homing-head transmitter, the sensitivity of its receivers, the radiation pattern of the antenna and the accuracy of the target designation (of the duration of target search).

The maneuverability of the missile is determined by the maximum lateral acceleration W which it can develop.

The angular velocity ω of turning of the velocity vector of the missile is proportional to W and inversely proportional to the velocity of the missile : $\omega = \frac{W}{V_r}$.

For remaining time of engagement, approximately equal to $\Delta t = \frac{\Delta S}{V_r}$, the velocity vector of the missile can be turned by the angle

$$\Delta\varphi_r = \omega \cdot \Delta t = \frac{W}{V_r} \cdot \frac{\Delta S}{V_r} = \frac{W \cdot \Delta S}{V_r^2} .$$

In order that a missile which maneuvers with a constant g-force and a constant velocity reach the target it is necessary that the velocity vector turn by an angle which is not less than $2\Delta\varphi_0$. In fact, before the intermediate point 1, the missile has turned by

$\Delta\varphi_0$. Before the point T, the missile must have turned by $\Delta\varphi_0$ again.

Therefore, in order to ensure capture, it is necessary that the turning angle of the missile during the required approach time Δt be not less than $2\Delta\varphi_0$. This condition can be written as

$$\Delta\varphi_r \geqslant 2\Delta\varphi_0 .$$

From this inequality and the ratio for $\Delta\varphi_r$ come the requirements for maneuverability when the guidance errors $\Delta\varphi_0$ and the capture range ΔS are set beforehand. The required maneuverability of the missile must be greater the greater the predicted error. With an increase in the capture range ΔS, the maneuverability requirements are decreased:

$$W \geqslant \frac{2\Delta\varphi_0 \cdot V_r^2}{\Delta S} .$$

This relation makes it possible to determine the capture range requirements ΔS or the accuracy requirements $\Delta\varphi_0$ when the maneuverability W of the missile is given.

For the more complicated case when the target is moving, all the above reasoning will be about the same. In this case, however, the parameters ΔS and $\Delta\varphi_0$ in the formula for W must be calculated with respect to the predicted point of impact.

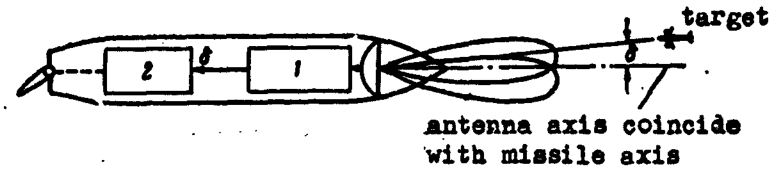

Fig. 46. Homing with target sighting (the homing-head antenna is rigidly mounted on the body of the missile): δ- is the error signal; 1) unit for error signal separation and command formation; 2) rudder drive.

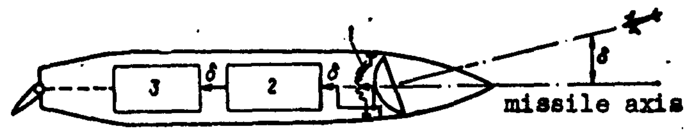

Fig. 47. Guidance with the head tracking the target: δ - error signal; 1) potentiometer for recording error signal; 2) command unit; 3) rudder drive.

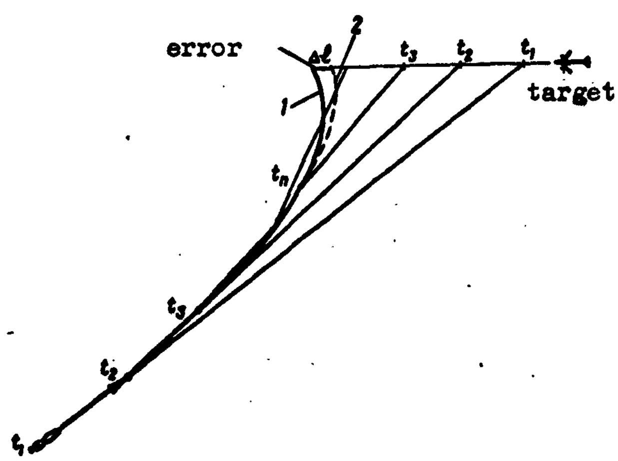

Fig. 48. Required (1) and real (2) trajectories in guidance on a pursuit curve.

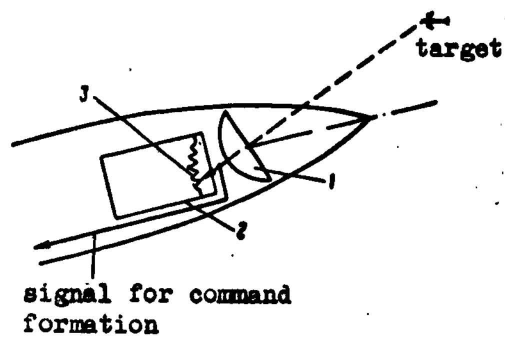

Fig. 49. Diagram of homing-head with gyroplatform: 1) homing-head; 2) gyroplatform; 3) potentiometer for measuring the position of the antenna axis relative to the gyroplatform.

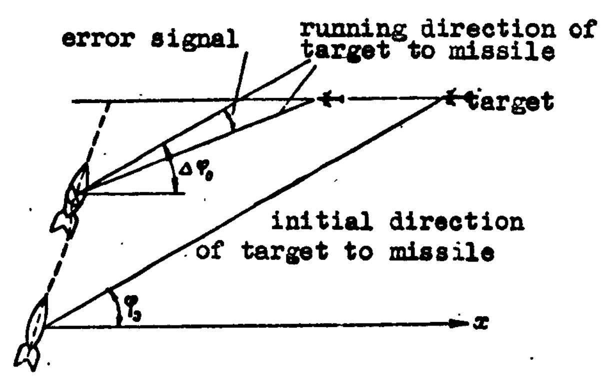

Fig. 50. The error signal in the method of parallel engagement.

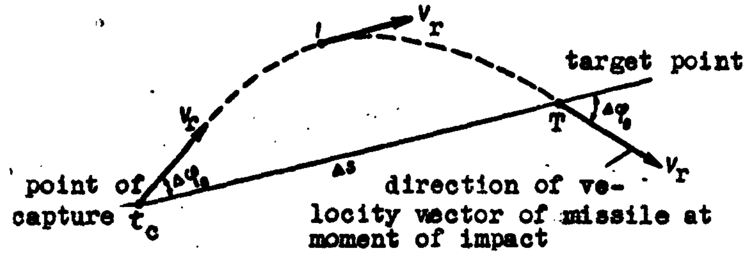

Fig. 51. Limiting trajectory which ensures guidance to the target.

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