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PRINCIPLES OF THE RADIO CONTROL OF PILOTLESS OBJECTS (SELECTED --ETC(U)
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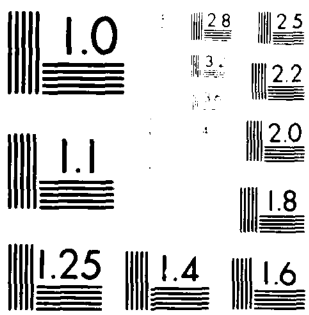

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FOREIGN TECHNOLOGY DIVISION

PRINCIPLES OF THE RADIO CONTROL OF
PILOTLESS OBJECTS

(Selected Chapters)

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FTD-ID(RS)T-0202-82

UNEDITED MACHINE TRANSLATION

FTD-ID(RS)T-0202-82 28 June 1982

MICROFICHE NR: FTD-82-C-000843

PRINCIPLES OF THE RADIO CONTROL OF PILOTLESS OBJECTS (Selected Chapters)

English pages: 250

Source: Printsipy Radioupravleniya Bespilotnymi Ob"yektami, Publishing House "Sovetskoye Radio", Moscow, 1959, pp. 99-225

Country of origin: USSR
This document is a machine translation.
Requester: USAMICOM/DRSMI-YDL
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U. S. BOARD ON GEOGRAPHIC NAMES TRANSLITERATION SYSTEM

Block	Italic	Transliteration	Block	Italic	Transliteration
А а	*А а*	A, a	Р р	*Р р*	R, r
Б б	*Б б*	B, b	С с	*С с*	S, s
В в	*В в*	V, v	Т т	*Т т*	T, t
Г г	*Г г*	G, g	У у	*У у*	U, u
Д д	*Д д*	D, d	Ф ф	*Ф ф*	F, f
Е е	*Е е*	Ye, ye; E, e*	Х х	*Х х*	Kh, kh
Ж ж	*Ж ж*	Zh, zh	Ц ц	*Ц ц*	Ts, ts
З з	*З з*	Z, z	Ч ч	*Ч ч*	Ch, ch
И и	*И и*	I, i	Ш ш	*Ш ш*	Sh, sh
Й й	*Й й*	Y, y	Щ щ	*Щ щ*	Shch, shch
К к	*К к*	K, k	Ъ ъ	*Ъ ъ*	"
Л л	*Л л*	L, l	Ы ы	*Ы ы*	Y, y
М м	*М м*	M, m	Ь ь	*Ь ь*	'
Н н	*Н н*	N, n	Э э	*Э э*	E, e
О о	*О о*	O, o	Ю ю	*Ю ю*	Yu, yu
П п	*П п*	P, p	Я я	*Я я*	Ya, ya

*ye initially, after vowels, and after ъ, ь; e elsewhere.
When written as ё in Russian, transliterate as yё or ё.

RUSSIAN AND ENGLISH TRIGONOMETRIC FUNCTIONS

Russian	English	Russian	English	Russian	English
sin	sin	sh	sinh	arc sh	$\sinh^{-1}$
cos	cos	ch	cosh	arc ch	$\cosh^{-1}$
tg	tan	th	tanh	arc th	$\tanh^{-1}$
ctg	cot	cth	coth	arc cth	$\coth^{-1}$
sec	sec	sch	sech	arc sch	sech^{-1}
cosec	csc	csch	csch	arc csch	csch^{-1}

Russian	English
rot	curl
lg	log

GRAPHICS DISCLAIMER

All figures, graphics, tables, equations, etc. merged into this translation were extracted from the best quality copy available.

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Chapter Five.

HOMING.

5-1. General/common/total relationships/ratios.

In Chapter 1 it was indicated that during the homing on the rocket is established/installed goniometer or radar of target, which is determining direction the rocket - target $\bar{r}$ in certain system of coordinates x_1 y_1 z_1 (see Fig. 1-19). Three-dimensional/space angular coordinate $\bar{\varphi}$ of this direction, and sometimes also range r are used for the rocket control. The system block diagram of homing is given in Fig. 5-1.

The coordinates of target $\bar{\varphi}$ and $\bar{r}$, measured by radar, are supplied to the entry of autopilot. On the basis of these coordinates and given feedback (AOS and EMOS) the autopilot develops necessarily the deflection of controls δ.

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The deflection of controls causes the rotations of missile body around the center of gravity; in this case are changed the lift and the created by it transverse acceleration $\vec{W}$, in consequence of which the missile trajectory is bent in the necessary direction. Deviation of rocket appearing in this case produces change in angle $\bar{\varphi}$. The connection/communication between the angles δ and $\bar{\varphi}$ is represented in Fig. 5-1 in the form of aerodynamic unit. A change in angle $\bar{\varphi}$ is again received by radar and is caused new command/crew to the controls of rocket. Thus, homing system has the exterior of the control circuit, which is closed through the locator of target. Furthermore, in this system are self-feedback: electromechanical feedback and aerodynamic feedback, that have the same value as during preset control.

Autopilot has the same block diagram as during preset control (see Fig. 4-2).

The comparison of block diagrams in Fig. 4-1 and 5-1 shows that the homing system has the following vital differences from self-contained control system:

1. During preset control the rocket is managed by programmer, while with homing - by radar (or goniometer) of target, which discovers the deviations of rocket from the correct flight of the target.

2. During preset control all control circuits are closed within rocket, while during homing one of control circuits is external and is closed through radar (goniometer) of target.

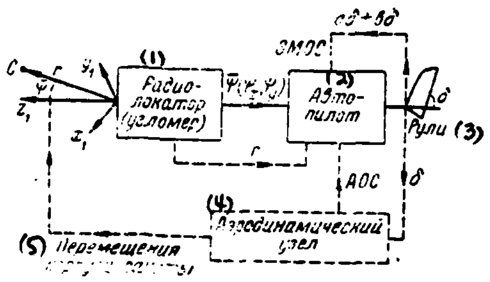

Fig. 5-1.

Key: (1). Radar (goniometer). (2). Autopilot. (3). Controls. (4). Aerodynamic unit. (5). displacement/movement of housing of rocket.

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From these special features/peculiarities escape/ensue the major advantages and deficiencies/lacks in the homing in comparison with preset control:

a). The major advantage - rocket "is managed by target", thanks to which are possible the guidance to the rapidly moving/driving and manuvering targets and greater accuracy of guidance to the fixed targets.

b). Main disadvantage - presence of external control circuit, in consequence of which system is subjected to interference effect.

In Chapter 1 it was indicated that according to the character of the energy, received by radar (or goniometer) of target, the homing systems are divided into the radio engineering, thermal, light (visible light beams) and acoustic, and on the location of the primary energy source - into passive, active and semi-active. Is given below the short comparison of these systems.

5-2. Comparison of passive, active and semi-active systems.

The great advantage of the passive systems over active ones and semi-active ones is the absence of the special irradiation of target from the rocket or the control center. Therefore the system is made simpler, and its action - hidden before the enemy.

Therefore, if target the characteristically pronounced radiation/emission of any form of energy, then this form of energy is most suitable for the homing. Thus, for instance, if target is radar of enemy or any other object, which has continuously or the prolongedly functioning radio transmitter in the range of waves, suitable for the homing (for example, in the range of centimeter or decimeter waves), then is possible the use/application of a passive radio engineering homing system; if target is the intense emitter of heat, then applicably passive heat-seeking guidance, etc.

By the major advantage they are semi-active systems in comparison with the active ones it is that that the powerful/thick primary energy source, which illuminates a target, is placed out of the rocket (usually on KP) and does not perish together with the rocket. Because of this the control mechanism is more simple, has smaller weight and overall sizes. Furthermore, in this case it is possible to carry out greater power and directivity of the action of the irradiating transmitter and to ensure therefore greater range of homing system.

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The major advantage of the active systems over semi-active ones is the complete independence of the process of homing from the control center. This fact can be especially useful, if rocket is produced from the mother aircraft, i.e., with mobile KP. Furthermore, here transmitter puts out to enemy location not of KP, but only rockets.

Finally, active homing can prove to be more appropriate than semi-active, when the process of homing must begin not from the moment/torque of the flight of rocket with KP, but only with the very large removal/distance of rocket from KP and the short distance from the rocket to the target.

Let us assume, for example, the rocket from the moment/torque of flight from KP and to point P (Fig. 5-2) is controlled by means of the remote control and must pass for the homing only at point P. Then during the semi-active homing the irradiating transmitter, arranged/located on KP, must have a power P_1, determined from the following relationships/ratios.

The transmitter (emitter), arranged/located at point O (Fig. 5-2) and which has power P_1 and directive gain η_1 (coefficient of the power gain, created by directivity of radiation/emission), creates in target C the specific power (power per unit of surface)

$$\frac{P_1\eta_1}{4\pi r_{0c}^2}.$$

The specific power reflected from the target at the point of reception/procedure (on the rocket) is equal to:

$$P_{2y}=\frac{P_1\eta_1}{4\pi r_{0c}^2}\cdot\frac{S}{4\pi r_0^2},$$

where S - efficient reflecting cross section of target.

Consequently,

$$P_1=\frac{P_{2y}4\pi r_{0c}^2 4\pi r_0^2}{S\eta_1}. \quad (5\text{-}1)$$

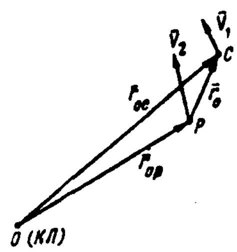

Fig. 5-2.

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With the active homing the transmitter (emitter), which is located on the rocket and which has power P'_1 and directive gain η'_1, creates in the target specific power $\frac{P'_1\eta'_1}{4\pi r_0^2}$. The specific power reflected from the target at the point of reception/procedure (on the rocket) is equal to:

$$P'_{2y} = \frac{P'_1\eta'_1}{4\pi r_0^2}\cdot\frac{S}{4\pi r_0^2},$$

whence

$$P'_1 = \frac{P'_{2y}4\pi r_0^2 4\pi r_0^2}{S\eta'_1}. \qquad (5\text{-}2)$$

Assuming that the receivers of energy, adjusted on the rocket, in both cases are identical, i.e., require the supplies to them of the identical specific power, we will obtain:

$$P'_{2y} = P_{2y}.$$

Therefore from formulas (5-1) and (5-2) we find:

$$\frac{P'_1}{P_1} = \frac{\eta_1}{\eta'_1}\left(\frac{r_0}{r_{0c}}\right)^2. \qquad (5\text{-}3)$$

where P'$_1$ and P$_1$ - required power of transmitter (emitter) with the active and semi-active homing respectively; η_1' and η_1 - directive gains antenna of transmitter with the active and semi-active homing respectively.

Coefficients η_1' and η_1 are greater, the greater the linear dimensions of radiating systems (antennas). Since radiating system, arranged/located on KP, can have considerably larger sizes/dimensions, than radiating system, arranged/located on the rocket, then it is possible to obtain $\eta_1 \gg \eta_1'$.

Thus, for instance, during the use of energy of the radio waves of centimeter band it is possible to obtain:

$$\frac{\eta_1}{\eta_1'} \approx 10^2. \qquad (5\text{-}4)$$

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However, if $r_0 \ll r_{0c}$, then, in spite of this, it can be obtained

$$P_1' \ll P_1.$$

Thus, for instance, when $\frac{r_0}{r_{0c}} \approx 10^{-2}$ from relationships/ratios (5-3) and (5-4) it will be obtained:

$$\frac{P_1'}{P_1} \approx 10^{-2},$$

i.e., in this case with the active homing will be required the considerably smaller power of transmitter, than with the semi-active.

$$\frac{r_{\bullet}}{r_{0c}} > 10^{-2},$$

However, if Λ then active homing will give smaller gain or even loss in energy sense. Furthermore, installation on the rocket even of a comparatively low-power transmitter can prove to be less advantageous than installation on KP of the transmitter of considerably larger power.

From given analysis it follows that active homing can prove to be more advantageous than the semi-active in the energy relationship/ratio only in those comparatively rare cases when from relationship/ratio (5-3) it is obtained

$$\frac{P_1'}{P_1} < 1.$$

However, in the majority of the cases is more profitable in energy sense semi-active homing.

5-3. Comparison of radio engineering, thermal, light and acoustical homing systems.

Main disadvantage in the acoustic systems is the low sound propagation velocity. The majority of the aerial targets has speeds, congruent with the speed of sound in air or even exceeding this speed. Therefore acoustical homing systems were not applied for the damage/defeat of the aerial targets. Fundamental field of application of these systems - damage/defeat of the underwater targets (use of ultrasonic oscillations/vibrations). Possibly also the use/application of acoustic systems for the damage/defeat of some forms of ground targets, which intensely radiate acoustic oscillations.

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The major advantage of the light systems, based on the use of the visible part of the spectrum of electromagnetic waves, is the possibility of applying the passive method of homing, since the majority of targets emits or reflects visible beams. Thus, for instance, majority of the aerial targets reflects solar and moonlight considerably stronger than their surrounding background, and such targets therefore can be isolated against this background with the system of light homing. However, light systems have very essential deficiency/lack - very critical dependence of the range on the meteorological conditions.

Rain, snow, fog and clouds so sharply decrease this range that the homing is made by virtually impossible. Thus, reliable light homing is possible only under good meteorological conditions. But even under good meteorological conditions light homing it will be impossible in those directions in which into the visual angle of the goniometer of system will fall the considerable interfering energy from the sun or the moon.

Considerably greater field of application can have the heat-seeking guidance systems, based on the use of an infrared spectrum of electromagnetic waves. Infrared (thermal) rays/beams have a wave band 0.76-400 μ. However, for purposes of homing in the Second World War were used waves in the range 1-5 μ, since in this range is located the maximum of the thermal radiation of the majority of targets, wave of this range less they attenuate in the atmosphere and are more convenient for the construction of receivers.

The majority of the aerial targets and many ground-based and waterborne targets are the sufficiently strong sources of heat rays. Therefore by the major advantage of thermal systems, as light, is the possibility of applying the passive method of homing. The range of thermal systems to a lesser degree depends on meteorological conditions, than the range of light systems. However, heat rays strongly attenuate in the atmosphere under the poor meteorological conditions (rain, fog, cloud). Furthermore, the action of thermal systems in the daytime strongly prevents thermal solar radiation.

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The smallest dependence on the meteorological conditions have radio engineering homing systems.

At the wavelength on the order of 10 cm or more the work of radio engineering system virtually does not depend on meteorological conditions.

In the range of waves 5-10 cm the action of meteorological conditions also insignificantly and begins to be manifested substantially only with $\lambda<5$ cm. However, even with $\lambda\leqslant3$ cm their action on the radiowave propagation is considerably less than to the propagation of thermal or light waves. Interferences/jammings from the radiation of the Sun and moon also are incomparably less.

Main disadvantage in the radio engineering systems is the impossibility of use/application in the majority of the cases of the passive method of homing. Passive homing is possible only in those special cases when target contains radar or any radio transmitter, which works continuously or sufficiently prolongedly on the wave, close to the wave of homing system. However, it is not always possible to calculate that the conditions, necessary for the passive radio-homing, will be satisfied; therefore in the general case it is necessary to apply the special irradiation ("illumination") of target. This complicates equipment for control, it makes control process that of less hidden and facilitates enemy the possibility of designing of interferences/jammings.

The comparison given above of different homing systems shows that each of these systems has both the serious advantages and serious deficiencies/lacks. Therefore it is difficult to assume that some one of these systems will extrude/exclude subsequently all rest and is occupied monopolistic position. On the contrary, most probable is the development of all described above systems. However, in application to reactive/jet objects the widest use received radio engineering and heat-seeking guidance systems.

Thermal systems, as a rule, are passive, and radio-engineering in the majority of the cases are semi-active or active and only in the special cases passive.

In Chapter 3 are described three methods of guidance of rocket to the target:

1. Method of linear curve.

2. Method PU (method of consecutive preventions/advances).

3. Method of covering of target (guidance on three-point curve).

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The method of the covering of target for the homing is inapplicable, since it requires the continuous determination of direction rocket - KP, but during the homing KP does not participate during the control process.

The first two methods of guidance (linear curve and PU) can be realized without the use of KP and are suitable for the homing. Are examined below consecutively/serially both these of method. In this case the homing on the linear curve is examined only in connection with guidance to the ground-based and waterborne targets, since as noted above (see Chapter 3), for guidance to the rapidly moving/driving aerial targets the method of linear curve little is suitable.

5-4. Homing on the linear curve.

The method of linear curve is based on the maintenance of equality to zero lead angles β, i.e., the angle between vectors of the speed of rocket $\bar{V}_2$ and direction rocket - target $\bar{r}$ (see Fig. 3-1). Consequently, for the guidance on the linear curve equipment for control must measure the angle β and issue commands/crews to the controls in accordance with value and sign of this angle. But β is an angle between vectors $\bar{V}_2$ and $\bar{r}$. Direction $\bar{r}$ is defined during the homing very simply, since homing exactly is based on the use of energy, which goes from the target in this direction. This direction is determined by goniometer or radar, established/installed on the rocket. The sense of the vector of the speed of rocket $\bar{V}_2$ can be measured by different methods. simplest of these methods is based on what vector $\bar{V}_2$ in the first approximation, can be considered coinciding with the axis of rocket z, since angle δ between directions $\bar{V}_2$ and z does not exceed several degrees [see relationship/ratio (1-7)].

Thus, the easiest method of homing on the linear curve is based on the measurement of angle $\bar{\psi}$ between the direction to target $\bar{r}$ and the axis of rocket z.

For this purpose it is sufficiently rigid to establish/install the measuring system of the goniometer of target $x_1y_1z_1$ on the missile body (Fig. 5-3) and to direct axis z_1 of this system along the axis of rocket. Then angle ψ, measured by goniometer will in the first approximation, correspond to lead angle $\bar{\beta}$:

$$\psi=\beta+\delta\approx\beta. \quad (5\text{-}3)$$

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This installation/setting up of the measuring system of coordinates

$x_1y_1z_1$ we will subsequently call installation/setting up on the housing.

During the radio-homing the goniometer (radar) with the measuring system, established/installed on the housing, can be realized by two methods:

a) with the antenna, established/installed on the housing;

b) with that automatic tracking antenna, that is turned relative to housing.

Principle of action of radio-goniometer (radar) with the antenna, established/installed on the housing, it is shown in Fig. 5-3. The measurement of the angular coordinates of target (direction to the target) is made with the help of the equisignal sector with axis z_A, created, for example, by rotation or lobe switching of the directivity of antenna A of radio-goniometer (radar). Antenna is established/installed on the missile body in such a way that the system of coordinates of equisignal sector $x_Ay_Az_A$ proves to be fixed relative to housing, and axis z_A of equisignal sector coincides with the axis of rocket z. Then it is obvious that the system of coordinates of equisignal sector $x_Ay_Az_A$ is the measuring system of coordinates $x_1y_1z_1$, in which is measured direction to target r.

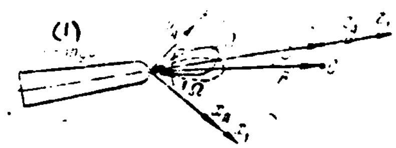

Fig. 5-3.

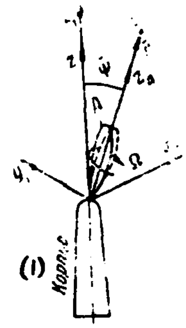

Fig. 5-4.

Fig. 5-3.

Key: (1). Housing.

Fig. 5-4.

Key: (1). Housing.

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The operating principle of radio-goniometer (radar) with the automatic tracking antenna is shown in Fig. 5-4. In this case the antenna of radio-goniometer (radar) is turned relative to missile body with the help of the electric motors in such a way that the axis z_1 of its equisignal sector at each moment of time would be directed exactly to the target (in reality due to the inaccuracy automatic tracking axis z_1 will not coincide precisely with the direction to the target). Assuming that the centerline of antenna sufficiently coincides precisely with its radio-axis (i.e. with the axis of equisignal sector z_1) it is possible to consider that the centerline of the automatic tracking antenna coincides with the direction to target $\bar{r}$. The position of the centerline of antenna relative to missile body is measured with the help of the potentiometric pickups, mounted on the missile body and which form the measuring system of coordinates $x_1y_1z_1$.

The second method of homing in the linear curve is based on the measurement of the velocity vector of rocket with the help of the weathervane (instrument, which is determining air flow direction).

The simplest weathervane, depicted in Fig. 5-5, consists of plate OA, hinged on a pin at point O. Under the action of air flow this plate occupies the position, which coincides with the air flow direction. Therefore the weathervane, hinged attached the missile body and streamlined with undistorted air flow, will indicate the direction of the speed of the rocket relative to this flow, called airspeed $\bar{V}_{a}$. In the absence of wind, i.e., in the absence of the motion of air flow relative to the earth/ground, the airspeed of rocket $\bar{V}_{a}$ coincides with true airspeed of rocket relative to earth/ground $\overline{V}_2$.

In the presence of wind with a speed of $V_в$

[illegible] (1.9)

or

[illegible] (1.10)

i.e., in the presence of wind, weathervane indicates the direction not of true airspeed of rocket $\overline{V}_2$, but to its airspeed V_{air} which differs from true airspeed by the velocity of wind $\overline{V}_в$. However, $V_в \ll V_2$ and in the first approximation, it is possible to consider that the weathervane measures the direction of the speed of rocket $\overline{V}_2$.

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Radio-homing systems, based on the use of a weathervane, can be decomposed into the following:

First form. Systems with the power weathervane.

Second form. Systems with the measuring weathervane and the goniometer with the antenna, rigidly established/installed on the housing.

Third form. Systems with the measuring weathervane and the goniometer with the automatic tracking antenna.

The operating principle of system with the power weathervane is shown in Fig. 5-6.

The antenna system of radio-goniometer (radar) rigidly is connected with the sufficiently powerful/thick, i.e., power, by weathervane in such a way that the weathervane ensures the axis alignment of the equisignal sector of antenna z_A with the direction of the airspeed of rocket $\bar{W}_{2a}$. Therefore radio-goniometer measures the direction to target r in the system of coordinates $x_A y_A z_A$, rigidly connected with the weathervane. Consequently, in this case the measuring system of coordinates $x_1 y_1 z_1$ is the system of coordinates of equisignal sector $x_A y_A z_A$. Angle $\bar{\varphi}_1$ measured in this coordinate system, is used for forming the commands/crews on control vanes of rocket.

The operating principle of system with the measuring weathervane and the radio-goniometer, established/installed motionlessly on the missile body, it is shown in Fig. 5-7. The radio-goniometer, having the antenna, rigidly connected with the missile body, measures angle $\bar{\varphi}_1$, which gives direction to the target in the system of coordinates $x_1 y_1 z_1$, rigidly connected with the missile body. In the same coordinate system is measured angle $\bar{\varphi}_2$, which gives the sense of the vector $\bar{V}_{2a}$, with the help of the measuring, i.e., low-power, weathervane.

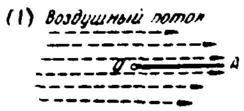

Fig. 5-5.

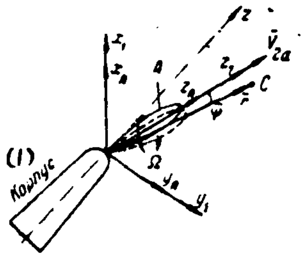

Fig. 5-6.

Fig. 5-5.

Key: (1). air flow.

Fig. 5-6.

Key: (1). housing.

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The differential angle

$$\psi = \psi_1 - \psi_2 \qquad (5-7)$$

is used for forming the commands/crews to the controls. The system block diagram of control, based on this principle, is depicted in Fig. 5-8.

At the output of direction finder is obtained the voltage

$$\dot U_1 = \dot K_1 \bar\varphi_1,$$

where $\dot K_1$ - coefficient of the transfer of direction finder.

At the output of measuring weathervane is obtained the voltage/stress

$$\dot U_2 = -\dot K_2 \bar\varphi_2,$$

where $\dot K_2$ - transmission factor of measuring weathervane.

Voltages/stresses $\dot U_1$ and $\dot U_2$ enter the summing cascade/stage (SK) at output of which is formed the resulting voltage

$$\dot U_p = \dot U_1 - \dot U_2 = \dot K_1 \bar\varphi_1 - \dot K_2 \bar\varphi_2 \qquad (5\text{-}7)$$

This voltage/stress is supplied to the autopilot and is base for forming the commands/crews.

If

$$\dot K_2 = \dot K_1,$$

$$\dot U_p = \dot K_1 (\bar\varphi_1 - \bar\varphi_2) = \dot K_1 \bar\varphi,$$

then

If we construct control system then so that in the trimmed/steady-state mode/conditions there would be $u_p(t) = 0$, then in this case there will be $\varphi_{ycm} = 0$, the velocity vector of rocket $\bar V_{2a}$ will attempt accurately to coincide with the direction to the target. However, in actuality precise equality of the transmission factors of both channels of measurement K_1 and $\dot K_2$ it is not possible to attain and there will occur certain error.

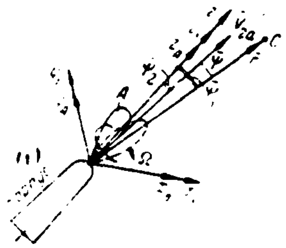

Fig. 5-7.

Key: (1). housing.

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From (5-7') we will obtain:

$$U_D = K_1(\bar{\psi} + \Delta\bar{\psi}).$$

where the error

$$\overline{\Delta\psi} = \frac{K_1 - K_2}{K_1}\bar{\psi}_2 = \frac{\Delta K}{K_1}\bar{\psi}_2. \qquad (5\text{-}7'')$$

In order to decrease this error, it is necessary to ensure the best possible coincidence of transmission factors $\dot{K}_1$ and $\dot{K}_2$.

The operating principle of system with the measuring weathervane and the automatic tracking antenna is shown in Fig. 5-9 (Fig. 5-9 is constructed under the assumption of ideal auto-summation after the target). This method differs from the previous only in terms of the fact that angle $\bar{\psi}_1$ is measured not with the help of the antenna, established/installed on the housing, but with the help of the antenna, automatic tracking after the target. The centerline of this antenna indicates direction to the target in the system of coordinates $x_1y_1z_1$, connected with the missile body.

The methods described above of the realization of homing on the linear curve can be reduced to the following two groups:

First group. Methods, based on the measurement of angle $\bar{\psi}'$ between the direction to the target and axis of rocket, including:

a) with the use/application of an antenna of goniometer, rigidly established/installed on the housing;

b) with the use/application of the automatic tracking antenna.

Second group. Methods, based on the measurement of angle $\bar{\psi}''$ between the direction to the target and vector of airspeed $\bar{V}_{за}$, including:

a) with the power weathervane;

b) with the measuring weathervane and with antenna on the housing;

c) with the measuring weathervane and the automatic tracking antenna.

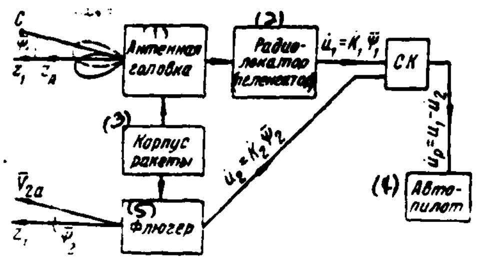

Fig. 5-8.

Key: (1). Antenna head. (2). Radar (direction finder). (3). Housing. (4). Autopilot. (5). Weathervane.

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For the illustration of all given above methods was selected the radio engineering method of homing. But, it is obvious, that these methods can be used also during the thermal or light homing, if we instead of the radio-goniometer use thermal or light goniometers.

Five methods indicated above of the realization of homing on the linear curve by no means exhaust all possibilities, but are sufficiently typical and give representation about the special features/peculiarities of this method of homing.

Given below the short comparison of these methods.

The special feature/peculiarity of methods 1b and 2c in comparison with the rest is the use/application of an antenna, automatic tracking after the target.

By the major advantage, given by the automatic tracking antenna in comparison with the antenna, established/installed on the housing or the weathervane, is the smaller danger of target fade goniometer during the input/introduction of target into the zone of action of goniometer and in the very control process. Therefore the visual angle of goniometer can be undertaken by sharper/more acute. However, the decrease of visual angle raises the range of the system of control (since energy is crowded with more narrow beam), its angular sensitivity and angular resolution. However, the use/application of the automatic tracking antenna complicates the construction/design of the control system and introduces the further source of error - error automatic tracking after the target.

Main disadvantage in the method 2a in comparison with the remaining methods of the second group is the complexity of the combination of the sufficiently simple construction/design of block "antenna - power weathervane" with a sufficient accuracy of readings/indications of weathervane, since for decreasing the error weathervane must be established/installed in this place of the rocket where air flow is not distorted noticeably by housing and rocket fin. Therefore from the measuring weathervane to considerably more easily obtain the required accuracy, than from the power.

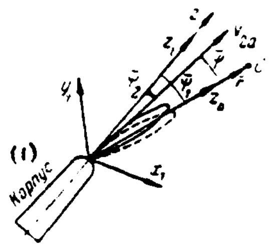

Fig. 5-9.

Key: (1). Housing.

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Let us compare finally the methods of the second group with the methods of the first group.

The special feature/peculiarity of the methods of the second group is the fact that base for forming the commands/crews to the controls is angle $\bar{\psi}''$ between the direction to the target and the airspeed of rocket (Fig. 5-10):

$$\bar{\psi}'' = (\widehat{\bar{r}, \bar{V}_{2a}}). \qquad (5\text{-}8)$$

If wind velocity $\bar{V}_a$ is not measured by any special method and is not introduced into the command/crew to the controls, then the ideal control system will attempt to manage rocket then so that in the trimmed/steady-state mode/conditions would satisfy the condition itself

$$\psi''=0. \qquad (5\text{-}9)$$

In this case, as it follows from Fig. 5-10, lead angle is equal to:

$$\beta=\Delta, \qquad (5\text{-}10)$$

where Δ - angle between vectors $\bar{V}_{2a}$ and $\bar{V}_2$.

Let us explain how will affect the presence of the error Δ, caused by wind, to the accuracy of homing. From condition (5-9) it follows that the rocket will attempt to fly so that its air (but not true) airspeed $\bar{V}_{2a}$ at each moment of time would be directed toward the target. To this it corresponds to Fig. 5-11 (in earth-based coordinate system).

Let us assume during entire time of homing the wind velocity is constant:

$$\bar{V}_e=\text{const}. \qquad (5\text{-}11)$$

Then it is convenient to replace analysis in earth-based coordinate system with analysis in the coordinate system, connected with air flow, i.e., with the wind (Fig. 5-12). In this coordinate system target will have a speed

$$\bar{V}_1'=\bar{V}_1-\bar{V}_e. \qquad (5\text{-}12)$$

and the rocket

$$\bar{V}_2'=\bar{V}_2-\bar{V}_e=\bar{V}_{2a}. \qquad (5\text{-}13)$$

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Since speed $\overline{V}_{2a}$ is always directed toward the target [from the condition (5-9), then in the moving/driving coordinate system rocket will be guided to the target, which has speed V_1, accurately on the linear curve. Consequently, at the constant velocity of wind the action of wind is equivalent so that the rocket, which has speed is induced on the linear curve at the target, which has speed $\overline{V}_{2a}$, $\overline{V}_1' = \overline{V}_1 - \overline{V}_e$, where $\overline{V}_1$ - true target speed; $\overline{V}_e$ - wind velocity.

Since during the guidance on the linear curve the maximum vectoring error $d_{макс}$ caused by path curvature, is determined from formula (3-12), then we obtain:

a) in the absence of the wind

$$d_{макс} = \frac{V_1^2}{2W_{pn}}; \qquad (5\text{-}14)$$

b) in the presence of the wind

$$d'_{макс} = \frac{(V_1')^2}{2W_{pn}}. \qquad (5\text{-}15)$$

Consequently,

$$\frac{d'_{макс}}{d_{макс}} = \left(\frac{V_1'}{V_1}\right)^2. \qquad (5\text{-}16)$$

In the worse case when

$$V_1' = V_{макс} = V_1 + V_e$$

$$d'_{макс} = \frac{(V_1 + V_e)^2}{2W_{pn}}; \qquad (5\text{-}15')$$

$$\frac{d'_{макс}}{d_{макс}} = \left(1 + \frac{V_e}{V_1}\right)^2. \qquad (5\text{-}16')$$

Formulas (5-15') and (5-16') show, to what extent can be increased vectoring error due to the action of wind.

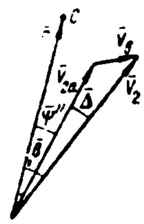

Fig. 5-10.

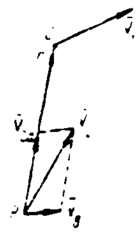

Fig. 5-11.

Fig. 5-12.

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As noted in Chapter 3, the method of linear curve gives satisfactory results only with the guidance to the slow targets - ground-based or marine.

For these purposes

$$V_1 \leqslant 15 \div 30 \text{ m/s.} \qquad (5.17)$$

Wind velocity can reach with the high wind much the same value:

$$V_в \leqslant 15 \div 30 \text{ m/s.} \qquad (5.18)$$

Let us consider for the illustration two numerical examples.

Example 1. $V_1=0$; $V_в=15$ m/s; $W_{рм}=2g$. Then in the absence of wind the error due to the path curvature is equal to zero, and in the presence of wind $d'_{макс}=5,5$ м.

Example 2. V_1=15 m/s; $V_в=15$ m/s; $W_{рм}=2g$. Then $d_{макс}=5,5$ м; $d'_{макс}=22$ м.

From these examples it is evident that high wind can considerably increase vectoring error. However, this error is obtained only in the quite worse case, with the coincidence of a number of unfavorable circumstances, namely:

a) at the moment of the beginning of homing angle α_0 is close to 90° (α_0~~60-120°) (see Fig. 3-6 and 3-7), since is only in this case valid formula (5-14);

b) wind velocity is great, constant and directed in the most dangerous direction (against target speed).

Furthermore, during the derivation of formula (5-16) it was assumed that the appearance of high wind, even contrary, does not vary target speed $\bar{V}_1$ in earth-based coordinate system. In actuality target speed V_1 can somewhat decrease, especially in the case of the waterborne target. Since the probability of coinciding all facts indicated is small, then the error indicated, caused by the curvature of trajectory, in the presence of wind will be in the majority of the cases considerably less than the maximally possible error $d'_{макс}$.

For decreasing the error from the action of wind it is possible to measure wind velocity $\bar{V}_в$ and to introduce the appropriate correction into the steering commands. However, the automatic measurement of speed $\bar{V}_в$, to air-stream velocity relative to the earth/ground, with the help of the equipment, established/installed on the rapidly flying rocket, presents great difficulties.

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To considerably more conveniently conduct the approximate measurement of wind velocity from the control center, arranged/located on the mother aircraft. But in this case would be possible to introduce into the steering commands only the value of wind velocity $\bar{V}_в$, which is correct only for the place of missile takeoff, and could not be taken into account changes in wind velocity in the process of rocket flight which can be very essential, since rocket (torpedo, bomb) in the process of flight passes from the upper layers of the atmosphere into the lower ones and flies sufficiently large distances. Therefore the problem of the introduction of correction for wind by the sufficiently simple and precise method (during guidance on the linear curve) did not obtain during the last war of the practical solution.

Thus, one of the deficiencies/lacks in the methods of homing, based on the use/application of a weathervane, is the presence of the further error, caused by wind.

Let us consider now the special features/peculiarities of the first group of the methods of the homings on the linear curve, in which the weathervane is absent. As noted above, this group of methods was based on the measurement of angle $\bar{\psi}$ between the direction to target $\bar{r}$ and the axis of rocket.

From formula (5-5) it follows that this angle differs from true lead angle β by the value of angle δ between the axis of rocket and its speed $\bar{V}$:

$$\bar{\psi} = \bar{\beta} + \bar{\delta}. \qquad (5\text{-}19)$$

If ψ is the only measured value, introduced into the steering commands, then it is simplest to ensure such regulation with which $\psi \to 0$. Therefore let us assume that the rocket will fly so that continuously would be satisfied the condition

$$\psi = 0, \qquad (5\text{-}20)$$

and let us explain, what can arise vectoring errors because angle $\bar{\psi}$ is not equal to lead angle $\bar{\beta}$.

From formulas (5-19) and (5-20) it follows that in this case

$$\bar{\beta} = = \bar{\delta}, \qquad (5\text{-}21)$$

i.e., rocket will fly not accurately on the linear curve (β=0), but with certain lead angle.

In the absence of wind the angle $\bar{\delta}$ is composed of the angle of attack $\bar{\alpha}_a$ and angle of slip $\bar{\alpha}_c$ (see Fig. 1-14):

$$\left.\begin{aligned} \bar{\delta} = \bar{\delta}_0 = \bar{\alpha}_a + \bar{\alpha}_c; \\ \delta_0 = \sqrt{\alpha_a^2 + \alpha_c^2} \end{aligned}\right\} \qquad (5\text{-}22)$$

Assuming/setting for simplicity, that $\alpha_c^2 \ll \alpha_a^2$, we will obtain:

$$\bar{\delta}_0 \approx \bar{\alpha}_a. \qquad (5\text{-}22')$$

In the presence of the wind

$$\bar{\beta} = \bar{\delta}_0 + \bar{\Delta} \approx \bar{\alpha}_a + \bar{\Delta}. \qquad (5\text{-}23)$$

where $\bar{\Delta}$ - angle between vectors $\bar{V}_{2a}$ and $\bar{V}_2$ (Fig. 5-13). In the worse case when angles $\bar{\delta}_0$ and $\bar{\Delta}$ lie/rest at one plane and coincide in the direction,

$$\beta = \delta_0 + \Delta \approx \alpha_a + \Delta. \qquad (5\text{-}24)$$

Usually

$$\alpha_a \leqslant 10^\circ;\ \Delta_{макс} \approx \frac{V_в}{V_{2a}} \approx \frac{V_в}{V_2}. \qquad (5\text{-}25)$$

Since

$$V_в \leqslant 15 \div 30\ м/сек \text{ и } V_2 \geqslant 300\ m/s,$$

then

$$\Delta_{макс} \leqslant 3 \div 6^\circ; \qquad (5\text{-}26)$$

therefore

$$\beta \leq 15^{\circ}. \qquad (5\text{-}27)$$

Let us explain, to what errors it can give a difference in the lead angle from zero.

For this let us consider the simplest case when target is fixed, but angle β is not changed in the process of flight, i.e.,

$$V_1 = 0; \qquad (5\text{-}28)$$
$$\beta = \text{const.} \qquad (5\text{-}29)$$

The first assumption can occur in actuality, and the second assumption is sufficiently rough idealization. In actuality angle β will be changed in the process of flight and error will be less than when $\beta = \beta_{макс} = \text{const.}$

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Therefore, assuming/setting $\beta = \beta_{макс} = \text{const},$ we find the limiting value of error which simultaneously can serve as the rough estimate of the real value of error. With the assumptions indicated is valid Fig. 5-14.

The fixed target is found in the beginning of the system of coordinates xOy. The velocity vector of rocket $\bar{V}_1$ always coincides with plane xOy. In the absence of the error of prevention/advance, i.e., with $\beta=0$, the missile trajectory would coincide with the straight line P_0O. In the presence of constant error (β=const) the rocket moves over the curve P_0PB.

For the determination of form and curvature of this trajectory let us consider infinitely small the movement of rocket, depicted in Fig. 5-15. For time dt the rocket passes the path

$$dS=V_2 dt.$$

In this case the radius-vector of rocket $\bar{r}$ is turned to angle of $d\varphi$ and is changed to value dr. From Fig. 5-15 it follows:

$$-d\varphi=\frac{\overline{P'B}}{r}=\frac{dS\sin\beta}{r}; \qquad (5\text{-}30)$$

$$-dr\approx dS\cos\beta. \qquad (5\text{-}31)$$

Therefore

$$\frac{dr}{d\varphi}=\frac{r}{\operatorname{tg}\beta};$$

$$\int\frac{dr}{r}=\int\frac{d\varphi}{\operatorname{tg}\beta}+\ln C.$$

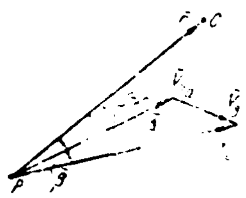

Fig. 5-13.

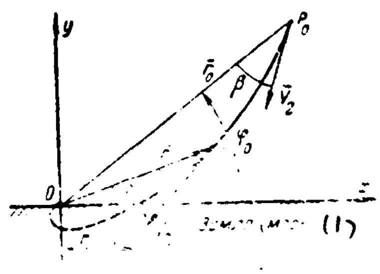

Fig. 5-14.

Fig. 5-14.

Key: (1). Earth (sea).

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i.e.

$$\ln r = \frac{\varphi}{\operatorname{tg} \beta} + \ln C$$

and

$$r = Ce^{\frac{\varphi}{\operatorname{tg} \beta}}.$$

Since at the start of the homing

$$r = r_0 \text{ и } \varphi = \varphi_0,$$

then

$$C = r_0 e^{-\frac{\varphi_0}{\operatorname{tg} \beta}}$$

and

$$r = r_0 e^{\frac{\varphi - \varphi_0}{\operatorname{tg} \beta}}. \quad (5\text{-}32)$$

Consequently, missile trajectory is logarithmic spiral.

Path curvature is expressed, as is known, by the formula

$$K = \frac{d\mu}{dS}, \quad (5\text{-}33)$$

where μ - angle of rotation of tangent to the trajectory with movement along the trajectory by the value dS.

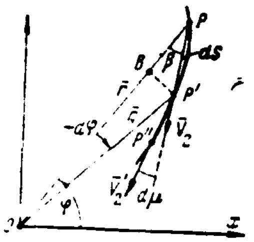

Fig. 5-15.

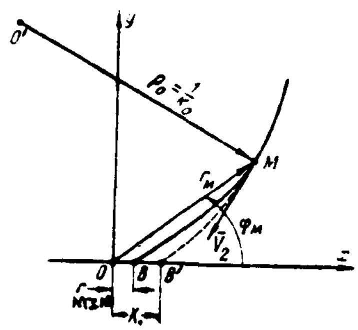

Fig. 5-16.

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But from Fig. 5-15 it follows that

$$(d\mu + \beta) = (-d\varphi + \beta),$$

i.e.

$$d\mu = -d\varphi;$$

therefore

$$K = -\frac{d\varphi}{ds}$$

Taking into account formula (5-30), we get, finally,

$$K = \frac{\sin\mu}{r} \qquad (5-31)$$

As can be seen from this formula, the curvature of the trajectory monotonically grows in proportion to the approximation/approach of rocket to a target.

Let us consider first rocket flight in the vertical plane. In this case x axis is the boundary of terrestrial or water surface (Fig. 5-14), and point B will be the extreme point of spiral trajectory. At this point the curvature is maximum and equal to:

$$K_{max} = \frac{\sin\beta}{r_{min}} \approx \frac{\beta}{r_{min}}, \qquad (5\text{-}35)$$

where

$$r_{min} = r_0 e^{-\frac{\pi}{2}\cot\beta} \approx r_0 e^{-\frac{\pi}{2\beta}} \qquad (5\text{-}36)$$

From formula (2-4) it follows that a maximally possible curvature of the trajectory on which still can follow the rocket.

$$K_0 = \frac{1}{\rho_0} = \frac{\omega_0}{V}$$

Therefore, if for this trajectory $K>K_0$, then rocket descends from this trajectory. Consequently, in the example in question are possible two cases:

The first: $K_B \leq K_0$.

In this case the rocket follows on the spiral curve up to point B and, therefore, appears vectoring error

$$d = r_{min} = r_0 e^{-\frac{\pi}{2\beta}} \qquad (5\text{-}37)$$

The second: $K_B > K_0$.

In this case the rocket descends with the spiral curve at that point M, at which $K_M = K_0$. This case is depicted in Fig. 5-16.

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At point M the rocket descends from spiral MB and it moves further over the circle/circumference (it is depicted as dotted line) of radius $\rho_0 = \frac{1}{K_0}$ up to the rendezvous with the terrestrial or water surface at point B'.

Consequently, in this case the vectoring error

$$d = x_1. \qquad (5\text{-}38)$$

The analysis Fig. 5-16 gives the following expression for error x_1:

$$d = x_1 = \rho_0 \left[\sqrt{1 + \left(\frac{\operatorname{tg} \beta}{\sin \varphi_M} \right)^2} - 1 \right] \sin \varphi_M \cos \beta, \qquad (5\text{-}39)$$

where

$$\varphi_M = \varphi_0 - (\operatorname{tg} \beta) \ln \left(\frac{r_0}{\rho_0 \sin \beta} \right). \qquad (5\text{-}40)$$

If from formula (5-40) is obtained $\varphi_M < 0$, this means that $K_B < K_0$, i.e. occurs not the second, but first case.

If formula (5-40) gives $\varphi_M > \varphi_0$, i.e. $r_0 < \rho_0 \sin \beta$, this means that even in the beginning of homing, i.e., with $r=r_0$, the path curvature is more than permitted and rocket from the very beginning of induction/guidance cannot follow along the spiral trajectory.

In the majority of real cases $\left(\frac{\operatorname{tg}\beta}{\sin\varphi_M}\right)\leqslant 0,5$ and $\beta\leqslant 15°$. In these cases of formula (5-39) and (5-40) they are simplified and take the following form:

$$d=x_1\approx\frac{\rho_0\beta^2}{2\sin\varphi_M}, \qquad (5\text{-}39')$$

where

$$\varphi_M\approx\varphi_0-\beta\ln\left(\frac{r_0}{\rho_0\beta}\right). \qquad (5\text{-}40')$$

From formulas (5-37), (5-39') and (5-40') it follows that error d grows with an increase in values β and r_0 and a decrease of angle φ_0. Let us consider for the illustration several numerical examples.

Example 1. $\varphi_0=30°$; $V_2=300$ m/s; $W=2g$; $r_0=10$ km; $\beta=12°$.

In this case $\rho_0=\frac{V_2^2}{W}=4500$ m, and from formula (5-40) we find:

$$\varphi_M=0,04>0.$$

Consequently, occurs the second case.

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Since in this case $\frac{\text{tg}\,\beta}{\text{tg}\,\varphi_M} > 0.5$, error d should be found by formula (5-39), which gives d~~700 m.

Example 2. φ_0=12°. Remaining data the same as in example 1.

In this case from formula (5-40) is obtained $\varphi_M < 0$, i.e. occurs the first case and error d is determined from formula (5-37) which gives:

$$d = 10\,000\,e^{-\frac{12}{12}} = 3700 \text{ м}. \qquad (5\text{-}41)$$

Example 3. β=1.2°; φ_0=12°; V_2=300 m/s; W=2g; r_0=10 km. In this case $\varphi_M = 0{,}11 > 0$; $\frac{\text{tg}\,\beta}{\text{tg}\,\varphi_M} \approx 0{,}2 < 0{,}5$; therefore error d can be determined from simplified formula (5-39')which gives:

$$d = 8 \text{ м}. \qquad (5\text{-}42)$$

From formulas (5-39') and (5-40') it follows that with a change in the angle β error d is changed very sharply: it is more sharply than according to the quadratic law. This follows also from the comparison of relationships/ratios (5-41) and (5-42): the decrease of angle β from 12 up to 1.2°, i.e. 10 times, it led to the decrease of error from 3700 to 8 m, i.e. 460 times.

Therefore it is possible to count the angular error β of that permitted, if

$$\beta \leq 1. \qquad (5\text{-}43)$$

Until now, we examined rocket flight in the vertical plane. Let us consider now the case when angular error β lies/rests at the horizontal plane and, therefore, it distorts missile trajectory in the horizontal plane. In this case will be again valid Fig. 5-14, with the difference that plane xOy will be now not vertical, but horizontal and because of this x axis will no longer be the boundary of the motion of rocket.

If rocket had infinite maneuverability ($W_{p_m} \to \infty$), it would follow on the spiral further of point B, up to a precise encounter with target at point O (dotted curve in Fig. 5-14). However, as a result of the limited maneuverability rocket will always descend from the spiral at certain point M and, moving further over the circle/circumference, fly wide of the mark. Therefore in this case it is possible to again use Fig. 5-16, and to consider that now the rocket can be circled and after the intersection of x axis. Then we obtain picture, depicted on Fig. 5-17.

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In this case the vectoring error d will, obviously, be equal to minimum distance from the target, i.e., from point O, to the missile trajectory, i.e., to the circle/circumference:

$$d = \overline{ON}. \quad (5\text{-}44)$$

where $\overline{ON}$ - minimum distance from point O to circle/circumference MND.

The analysis Fig. 5-17 gives the following result:

$$d = \overline{ON} = \rho_0 \sqrt{\sin^2\beta - 2(\sqrt{1 - \sin^2\beta} - 1)} \qquad (5\text{-}45)$$

Taking into account the smallness of angle β [see relationship/ratio (5-27)], it is possible with the high accuracy to assume that

$$\sin^2\beta \approx \beta^2 \ll 1.$$

Then formula (5-45) strongly is simplified and takes the form:

$$d = \frac{\rho_0\beta^2}{2}. \qquad (5\text{-}46)$$

Being congruent/equating formulas (5-46) and (5-39'), we are convinced, that with the identical value of the error of prevention/advance β the vectoring error d in the horizontal plane is obtained smaller than in the vertical. This result is completely clear, since during the motion in the horizontal plane x axis is not the boundary of motion and rocket, intersecting this axis, has the capability to approach the target, which is found in the beginning of coordinates.

Let us consider a numerical example, by assuming/setting as in example 3 for the vertical plane, that β=1.2°=0.02; V_2=300 m/s and W=2g. Then ρ_0=4500 m, and according to formula (5-46) we obtain d=0.9 m. At β>4° it is obtained by d>10 m, i.e., the error of guidance proves to be large.

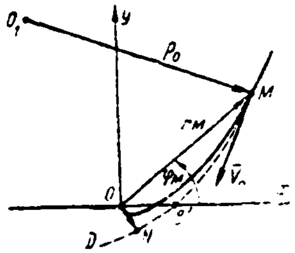

Fig. 5-17.

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The given above approximate error analysis of induction/guidance, caused by a difference in the angle of prevention/advance β from zero, makes it possible to do the following fundamental conclusions:

a) vectoring error d very sharply depends on the error of prevention/advance β (according to the quadratic law or even to the larger degree);

b) most dangerous is the vertical component of angle $\bar{\beta}$;

c) so that the vectoring error would not exceed several meters, must be satisfied the condition

But above it was explained [see formulas (5-24) and (5-27)], that the first group of the methods of induction/guidance gives the error of the prevention/advance

$$\varepsilon \approx \alpha - \Delta \qquad (5.18)$$

where the angle of attack α ... and the angular error, caused by wind, $\Delta \leqslant 3$-$6°$.

Consequently, in the absence of the special correction this group of the methods of induction/guidance will give inadmissibly large errors, even during the induction/guidance to the fixed target.

The angular error Δ, caused by wind, is considerably less dangerous than angular error α caused by the presence of angle of attack, since, first usually $\Delta < \alpha$, and, in the second place, error $\bar{\Delta}$ occurs in essence in the horizontal plane, and error α has the greatest value in the vertical plane (because of the action of the force of gravity of rocket). Therefore for decreasing the vectoring error it is necessary first of all to exclude or to reduce the effect of angle of attack α. For this it is necessary to measure the angle of attack with any direct or indirect method and to introduce it into the steering commands. One of the methods of measurement of angle of attack, i.e., the angle between vectors of aerodynamic speed and axis of rocket, is use measuring weathervane. But the methods, based on the use of a weathervane, are described above and are related to the second group. Therefore if we for some reason or other undesirably apply weathervane, then it is necessary to measure the angle of attack in any other direct or indirect manner.

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5.5. Homing according to the method of PU.

In Chapter 3 it was indicated that during the ideal induction/guidance according to the method of PU at each moment of time must be satisfied the condition for ideal prevention/advance (3-5):

$$\sin\beta=\sin\beta_0=\frac{V_1}{V_4}\sin\alpha, \qquad (5\text{-}49)$$

where β_0 - angle of ideal prevention/advance (Fig. 5-18).

If rocket on any reasons will fly not accurately on trajectory of PU, then will appear the error of prevention/advance $\widetilde{\Delta\beta}$.

Let us consider the first flat/plane case when vector $\bar{V}_2$ remains in plane $\widetilde{V}_{02}$, $\bar{r}$ (Fig. 5-18).

Then the error of the prevention/advance

$$\Delta\beta=\beta-\beta_0. \qquad (5\text{-}50)$$

and angle β satisfies relationship/ratio (3-17):

$$V_2 \sin\beta - V_1 \sin\alpha = r\frac{d\varphi}{dt}.$$

From formulas (5-49) and (3-17) it follows:

$$V_2(\sin\beta - \sin\beta_0) = r\frac{d\varphi}{dt}. \qquad (5\text{-}51)$$

But

$$\sin\beta - \sin\beta_0 = 2\sin\frac{\beta-\beta_0}{2}\cos\frac{\beta+\beta_0}{2} = 2\sin\frac{\Delta\beta}{2}\cos\left(\beta+\frac{\Delta\beta}{2}\right).$$

Since error $\Delta\beta$ must be low, first with the high accuracy it is possible to assume that $\sin\beta - \sin\beta_0 = \Delta\beta\cos\beta$, and equality (5-51) takes the form:

$$\Delta\beta = \frac{r}{V_2\cos\beta}\cdot\frac{d\varphi}{dt} \approx \frac{r}{V_2\cos\beta_0}\cdot\frac{d\varphi}{dt}. \qquad (5\text{-}52)$$

since usually $V_1/V_2 \leqslant 1/2$, then $\sin\beta_0 \leqslant 0.5$ and $\cos\beta_0 \geqslant 0.87$.

Therefore with $V_1/V_2 \leqslant 1/2$ it is obtained

$$\Delta\beta \approx \frac{r}{V_2}\cdot\frac{d\varphi}{dt}, \qquad (5\text{-}52')$$

where φ - angle, formed by vector "rocket - target" $\bar{r}$ in the nonrotating relative to the earth/ground system of coordinates $x_1 z_1$ (Fig. 5-18).

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If rocket moves over the ideal trajectory of PU then $\Delta\beta=0$. Consequently, the problem of control lies in the fact that to discover the error $\Delta\beta$ and to give command to the controls depending on value and sign of this error in such a way that it would be obtained $\Delta\beta \to 0$. The measurement of error $\Delta\beta$ can be produced by direct or indirect method.

Direct method escape/ensues from formulas (5-50) and (5-49):

a). The required lead angle β_0 is automatically computed by computer from formula (5-49) on the basis of continuously measured values V_1, V_2 and α.

b). True lead angle β continuously is measured. β is an angle between $\bar{r}$ and $\bar{V}_2$. Direction $\bar{r}$ is measured by goniometer, and $\bar{V}_2$ is measured approximately by the weathervane: $\bar{V}_2 \approx \bar{V}_{\Sigma}$.

c). Measured angle β continuously is equal with computed angle β_0, and difference in these angles $\Delta\beta$ is used for forming commands/crews to controls.

This method of control has a number of the serious deficiencies/lacks:

1. It is very difficult to ensure the sufficiently precise measurement of values V_1 and α (especially value α, which requires the measurement of the velocity vector of target) with the help of

the small/miniature and cheap equipment, established/installed on the rocket.

2. Weathervane, which measures sense of the vector $\bar{V}_2$, gives error in presence of wind to angle $\Delta=(V_{\omega}, V_2)$. Therefore in the measurement of the direction of speed $\bar{V}_2$ by weathervane wind will cause vectoring error.

The more precise measurement of vector $\bar{V}_2$ with the help of onboard equipment will sharply complicate this equipment.

Consequently, the direct measurement of the error of prevention/advance $\Delta\beta$ presents very great difficulties.

To considerably more simply measure the error $\Delta\beta$ indirectly on the basis of relationship/ratio (5-52').

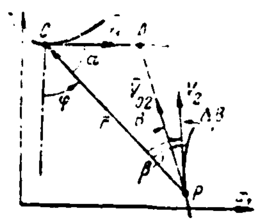

Fig. 5-18.

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From this relationship/ratio it follows that the value and the sign of error $\Delta\beta$ are correctly mapped by value and the sign of derivative $d\varphi/dt$. Therefore instead of the measurement of angle $\Delta\beta$ it is possible to measure the value of the derivative $d\varphi/dt$ and the command/crew to the controls to transmit depending on this derivative (about the effect of change in the time of distance r see Chapter 12);

$$\delta = K_1(p)\frac{d\varphi}{dt}. \qquad (5\text{-}53)$$

where

$$p = \frac{d}{dt}.$$

Since

$$\frac{d\varphi}{dt} = p\varphi,$$

then

$$\delta = K_2(p)\varphi, \qquad (5\text{-}54)$$

where

$$K_2(p) = K_1(p)p.$$

Consequently, depending on the type of regulator, i.e., on the form of the function K(p), it is possible to measure either the derivative of angle $\frac{d\varphi}{dt}$. or angle itself φ. The measurement of any of these values ($d\varphi/dt$ or φ) represents during homing of no fundamental difficulties; for this it suffices to have on the rocket a goniometer (radar), which is determining direction to target $\bar{r}$ in terrestrial or gyro system of coordinates.

Gyroscopes create the coordinate system, connected with world space and, therefore, which is turned relative to earth-based coordinate system. However, the angular velocity is small (0.25° per minute), which in the case of the close-range rockets is unimportant, what system of coordinates - terrestrial or gyroscopic - to take as the base. Since virtually on the rocket it is simpler to create gyroscopic coordinate system, subsequently we will consider that the angle φ (or the angular velocity $d\varphi/dt$) must be measured in the coordinate system stabilized by gyroscopes.

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Above was examined the flat/plane case, i.e., it was assumed that vector $\bar{V}_1$, differing from the direction of the ideal prevention/advance $\overline{V}_{02}$, remains in the plane of ideal prevention/advance ($\overline{r}$, $\overline{V}_{02}$), i.e., in plane z_1x_1 (see Fig. 5-18).

In the general case will occur not the flat/plane, but three-dimensional/space case, i.e., the error of advance $\Delta\overline{\beta}$ and angle $\overline{\varphi}$ will not lie/rest by pillar at plane z_1x_1, but they will have some components, also, in the orthogonal plane z_1y_1 (Fig. 5-19).

Since spatial motion can be decomposed on two plane motion in two orthogonal planes, then instead of the three-dimensio-al/space (Fig. 5-19) it is possible to examine two flat/plane figures (5-20a and 5-20b), being to projection Fig. 5-19 on plane z_1x_1 and z_1y_1 respectively.

For each of the planes (z_1x_1 and z_1y_1) are valid relationships/ratios (5-52) and (5-52), derived above for the flat/plane the case, i.e.,

$$\left.\begin{aligned}\Delta\beta_x &\approx \frac{r_x}{V_{2x}}\cdot\frac{d\varphi_x}{dt},\\ \Delta\beta_y &\approx \frac{r_y}{V_{2y}}\cdot\frac{d\varphi_y}{dt}.\end{aligned}\right\} \quad (5\text{-}55)$$

For guaranteeing the ideal prevention/advance must be satisfied the conditions:

$$\Delta\beta_x=0;\ \Delta\beta_y=0 \quad (5\text{-}56)$$

or

$$\frac{d\varphi_x}{dt}=0;\ \frac{d\varphi_y}{dt}=0. \quad (5\text{-}5$$

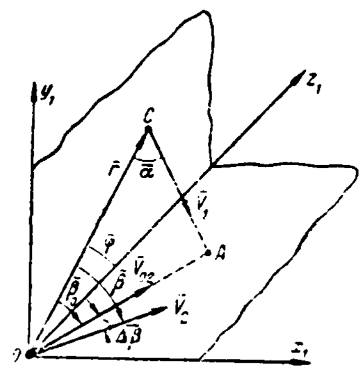

Fig. 5-19.

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Therefore during the symmetrical Cartesian steering of command/crew to rudders and height/altitude they must by analogy with formulas (5-53) and (5-54) be expressed by the following relationships/ratios:

$$\left.\begin{aligned}\delta_x &= K_1(p)\frac{d\varphi_x}{dt};\\ \delta_y &= K_1(p)\frac{d\varphi_y}{dt}\end{aligned}\right\} \qquad (5\text{-}58)$$

or

$$\left.\begin{aligned}\delta_x &= K_2(p)\varphi_x;\\ \delta_y &= K_2(p)\varphi_y.\end{aligned}\right\} \qquad (5\text{-}59)$$

where

$$K_2(p) = K_1(p)\,p.$$

Consequently, the goniometer (radar), adjusted on the rocket, must ensure measurement in the stabilized system of coordinates $x_1y_1z_1$ component φ_x and φ_y of solid angle φ (Fig. 5-20 and 5-21) or their derivatives $\frac{d\varphi_x}{dt}$ and $\frac{d\varphi_y}{dt}$.

Control system [form of the function $K_1(p)$ or $K_1(p)$] must be constructed so that in the trimmed/steady-state mode/conditions would be satisfied conditions (5-57), i.e., the derivatives of angular coordinates on the time vanished:

$$\left|\frac{d\varphi_x}{dt}\right|_{ycm} \to 0;\quad \left|\frac{d\varphi_y}{dt}\right|_{ycm} \to 0,$$

i.e.

$$\varphi_{x\,ycm} \to \text{const};\ \varphi_{y\,ycm} \to \text{const}. \qquad (5\text{-}60)$$

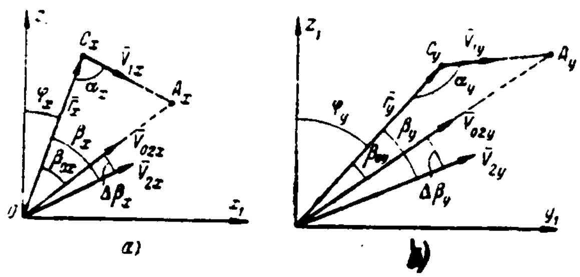

Fig. 5-20.

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Conditions (5-60) are the necessary and sufficient conditions of moving the rocket along ideal trajectory of PU.

If instead of conditions (5-60) will be implemented the more rigorous conditions

$$\varphi_{z\,\text{ycm}} \to 0;\ \varphi_{y\,\text{ycm}} \to 0, \qquad (5\text{-}61)$$

being a special case of conditions (5-60), then rocket will also move over the ideal trajectory of PU. Consequently, conditions (5-61) are sufficient, but not necessary conditions of moving the rocket along the ideal trajectory of PU.

In accordance with conditions (5-57), (5-60) and (5-61) it is possible to subdivide the possible methods of regulation, which ensure induction/guidance according to the method of PU, into the following two forms:

1. The method of the constant angle φ (method of zero derivative), which accomplishes conditions (5-60) or conditions (5-57) corresponding to them.

2. Method of zero angle φ, which ensures satisfaction of conditions (5-61).

Advantages and deficiencies/lacks in both methods of regulation will be explained in the process of further analysis.

Let us consider some most obvious methods of measurement of components φ_x and φ_y or their derivatives $\dot{\varphi}_x$ and $\dot{\varphi}_y$ (obviously, if angles φ_x and φ_y are measrued that their derivatives can be found, generally speaking, with electrical or any other method of differentiation.

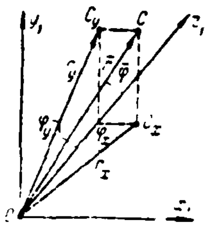

Fig. 5-21.

1. Method, based on the use/application of the stabilized antenna head (method of antihunting antenna).

The operating principle of the homing system, which has the stabilized antenna head, it is shown in Fig. 5-22a and b. The antenna head of airborne radar (direction finder) is stabilized in the space with the help of the gyroscopes. This stabilization can be realized, for example, by the installation/setting up of antenna head on GSP (gyroscope-stabilized platform).

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Before beginning homing the axis of antenna head is induced by any method at the target with the help of the motors of the initial installation/setting up (MNU), which turn head relative to GSP.

In the process of homing electric motors of MNU are fixed and antenna head is stabilized in the space. Consequently, proves to be that stabilized in the space and the system of coordinates $x_A y_A z_A$, formed by antenna head. Therefore the angle between the axis of antenna z_A and the direction to the target, measured by radar, and is the unknown angle $\tilde{\omega}$, measured in the stabilized coordinate system.

From the output of radar the error signal $A\tilde{\omega}$, which is angular measure $\tilde{\omega}$, is supplied to the entry of autopilot.

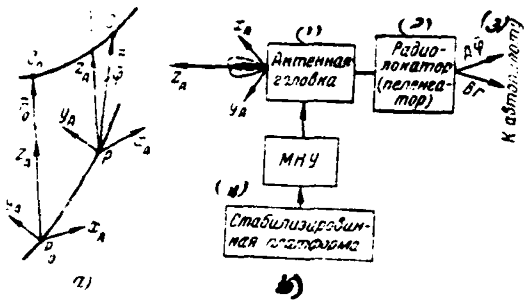

Fig. 5.22. Key: (1). Antenna barrel finishing. (2). Radar (direction finder). (3). To autopilot. (4). Stabilized platform.

2. Method, based on the use/application of the automatic tracking antenna, established/installed on GSP.

The operating principle of the system, based on this method, it is shown in Fig. 5-23a and b.

The antenna head of airborne radar is established/installed on GSP and is furnished with the servo electric motors which turn antenna head relative to GSP in such a way that its axis z_A at each moment of time would be directed toward the target. This auto-tracking of antenna head the target is ensured by feeding to the servo electric motors of the error signal Δ_e of radar (Fig 5-23b). The same servo electric motors are used for the initial focusing/induction of axis z_A of antenna head to the target.

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If we disregard/neglect error in automatic tracking, then it is possible to consider that the centerline of antenna z_A at each moment of time coincides with the vector of target $\bar{r}$. Therefore the angle of rotation of the axis of antenna z_A relative to GSP is in this case the unknown angle $\bar{\varphi}$. This angle can be measured with the help of two potentiometers, arranged/located on the rotational axes of antenna relative to GSP, and is given to the entry of autopilot.

In reality the centerline of antenna z_A does not coincide accurately with the direction to target $\bar{r}$, but continuously oscillates around this direction due to the presence of error automatic tracking. Because of this appears the corresponding error also in the measurement of angle $\bar{\varphi}$ and its derivative. It is possible to compensate this error. However, the presence of error automatic tracking is nevertheless a deficiency/lack in the automatic tracking antenna in comparison with the stabilized antenna.

The major advantage of the automatic tracking antenna in comparison with that stabilized are the smaller danger of target fade and the possibility of applying the narrower visual angle of antenna head. This advantage most is substantial with the regulation according to the method of constant angle φ and somewhat less substantially with the regulation according to the method of zero angle φ.

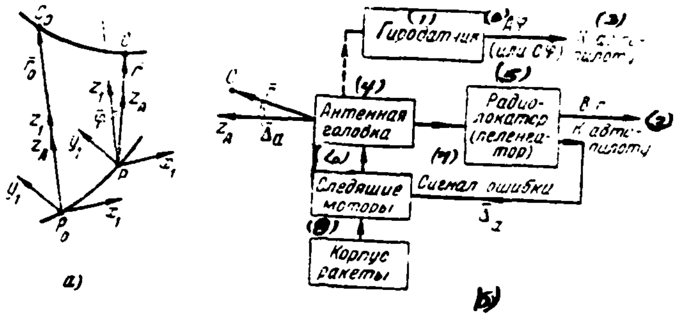

Fig. 5-23. Key: (1). Gyrosensor. (2). or. (3). To autopilot. (4). Antenna barrel finishing. (5). Radar (direction finder). (6). Servo motors. (7). Error signal. (8). Missile body.

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Actually/really, during the induction/guidance according to the method of constant angle φ in the trimmed/steady-state mode/conditions the angle φ can considerably differ from zero. Therefore during the use/application of an antihunting antenna, which measures this angle, can be required the very wide visual angle of airborne radar.

3. Method, based on the use/application of the automatic tracking antenna, established/installed on the housing, and measuring gyroscopes (method of the automatic tracking antenna, established/installed on the housing).

The operating principle of the system, based on this method, it is shown in Fig. 5-24a and b. This method differs from previous in terms of the fact that the antenna head, automatic tracking after the target, is turned by the servo electric motors not relative to GSP but relative to missile body. Therefore here is not required the uses/applications of GSP. If we disregard/neglect error automatic tracking, then again it is possible to consider that the centerline of antenna z_A at each moment of time coincides with the radius-vector of target $\bar{r}$. Therefore for measuring of angle $\bar{\varphi}$ or its derivative $\dot{\bar{\varphi}}$ it is necessary to measure the angle of rotation of the centerline of antenna or the derivative this angle in the stabilized coordinate system.

Applying for the measurement the free gyroscopes, it is possible to measure angles φ_x and φ_y. Derivatives $\dot{\varphi}_x$ and $\dot{\varphi}_y$ can be in this case obtained, for example, by electrical differentiation of the voltages/stresses, which correspond to angles φ_x and φ_y.

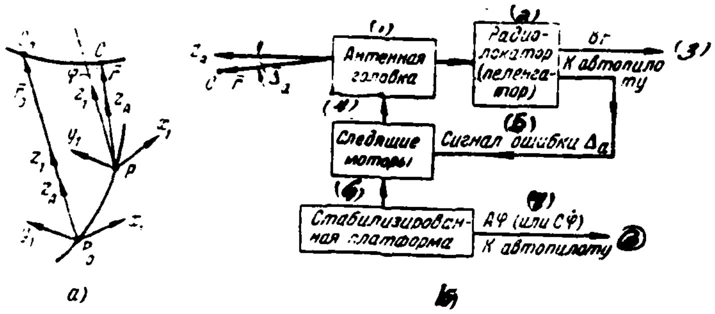

Fig. 5-24. Key: (1). Antenna barrel finishing. (2). Radar (direction finder). (3). To autopilot. (4). Servo motors. (5). It drove off errors. (6). Stabilized platform. (7). or.

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Applying for the measurement precessional gyroscopes, it is possible to directly measure the angular speeds of the rotations of the housing (axis) of antenna, i.e., derivatives $\dot{\varphi}$ and $\dot{\psi}$

Therefore during the use of the automatic tracking antenna installed on the rocket body measurement data ($\dot{\psi}$ or $\dot{\varphi}$) they come to the entry of autopilot from the gyrosensors (free of precessional measuring gyroscopes) connected with the antenna head (Fig. 5-24b).

The major advantage of the installation/setting up of the automatic tracking antenna on the housing instead of its installation/setting up on GSP is considerable simplification in the construction/design of an antenna-gyroscopic block. However, this method has very serious deficiency/lack, which consists in the appearance in the control system of parasitic feedback through the missile body.

Actually/really, if antenna head is established/installed on the housing, then rapid rotation emergent on any reason of missile body relative to the earth/ground will cause the rotation of antenna head relative to the earth/ground (since follower motors cannot instantly turn antenna head relative to housing then so that this head would preserve constant/invariable position relative to the earth/ground). The rotation of antenna head will cause the impulse/momentum/pulse of output potential of the gyrosensor which in turn, will cause a change in the surface position. However, a change in the surface position will lead to the new rotation of missile body, and thus, the circuit of parasitic feedback proves to be locked. The presence of parasitic feedback through the housing and the antenna head can considerably decrease the accuracy of the guidance of rocket to the target, and in the worse case lead even to the loss of stability of control.

For weakening of the effect of spurious coupling it is possible to increase the operating speed of follower motors, i.e., to increase the passband of system automatic tracking, but in this case will increase the action of interferences/jammings.

The given above three methods of measurement of angle φ (or $\dot{\varphi}$) by no means exhaust all possible methods of measurement, but nevertheless they give representation about the possible paths of the realization of homing according to the method of PU and difficulties appearing in this case.

Let us compare the now two methods of the regulations, used during the homing according to the method of PU: the method of constant angle φ and the method of zero angle φ. Let us consider first the method of constant angle φ.

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In this case in the trimmed/steady-state mode/conditions must be satisfied conditions (5-57). We will for simplicity examine by flat/plane the case.

Then conditions (5-57) are reduced to one condition:

i.e.

$$\left.\left|\frac{d\varphi}{dt}\right|\right|_{ycm}=0, \qquad \varphi_{ycm}=\text{const.} \qquad (5\text{-}62)$$

Let us take the further following simplifying assumptions:

a). Target flies rectilinearly and V_1/V_2=const. This means that in the trimmed/steady-state mode/conditions and the rocket must fly rectilinearly.

b). For guaranteeing s raight flight deflection of control δ must be equal to zero.

c). Control system is ideal.

Then satisfaction of condition (5-62) is provided, if control will deviate according to the law, depicted in Fig. 5-25.

Actually/really, if condition (5-62) is not satisfied, i.e., $\frac{d\varphi}{dt}\neq 0$, to the controls will be given command X, which calls the deflection of control and flight path curvature of rocket in the necessary direction. Flight path curvature will cease only $d\varphi/dt=0$, i.e., with satisfaction of condition (5-62), which ensures a precise flight into set forward point.

Let us consider now the process of setting the mode/conditions of a precise flight into set forward point for the case when at the moment of the beginning of homing the velocity vector of rocket $\tilde{V}_2$ is sharply deflected from the direction of ideal prevention/advance and, therefore, there is a strong initial disturbance. This process is depicted in Fig. 5-26 and 5-27.

Target flies with a constant velocity of $\overline{V}_1$. At the moment of the beginning of homing (t=0) target is located at point C_0, and the ro.·et at point P_0. At this moment the velocity vector of rocket is directed not accurately to set forward point A, but with certain error of prevention/advance $\Delta\beta=\Delta\beta_0$. Since is examined the flat/plane case, then angle $\Delta\beta$ lies/rests at the plane $(\overline{r}_0, \overline{V}_{02})$.

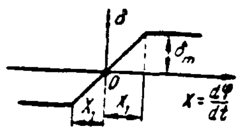

Fig. 5.25.

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During correct control the rocket must fly so that would be established/installed the mode/conditions in which $\Delta\beta=0$.

At moment/torque t=0 there is a large error of prevention/advance $\Delta\beta=\Delta\beta_0$ (Fig. 5-26).

Since $\Delta\beta\approx\frac{r}{V_1}\cdot\frac{d\varphi}{dt}$, that at this moment the derivative $d\varphi/dt$ is also great.

Since in the trimmed/steady-state mode/conditions must be $d\varphi/dt=0$, this means that at the moment of the beginning of homing the mode/conditions is distant from that being steady ($|x|>x_1$) and in the controls is supplied maximum command/crew.

Therefore rocket will be circled minimum radius of curvature ρ_0 to certain point P_1 and the derivative $d\varphi/dt$ will be reduced according to the law, depicted in Fig. 5-27a.

At certain moment/torque t_1 disturbance/perturbation $d\varphi/dt$ will become such low that further decrease of this disturbance/perturbation will cause the gradual decrease of the deflection of the control (see Fig. 5-25).

At point P_2 (Fig. 5-26) it will be established/installed $d\varphi/dt=0$, the command/crew to controls will vanish and, therefore, rocket will fly subsequently on the straight line P_2A' into the *collision point* A'.

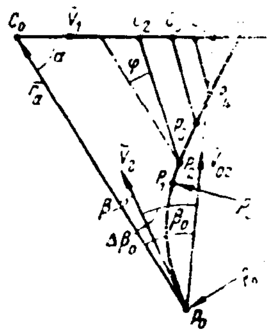

Fig. 5-26.

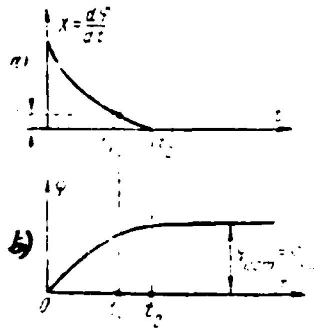

Fig. 5-27.

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Since in the process of rocket flight the derivative $d\varphi/dt$ is changed on the curve, depicted in Fig. 5-27a, then angle φ must in this case be changed according to the law, depicted in Fig. 5-27b (φ - the angle between the current direction $\bar{r}$ and the initial direction $\bar{r}_0$ fixed in the space with the help of the gyroscopes). Consequently, up to moment/torque t_2, when rocket proves to be at point P_2, is established/installed certain maximum value of angle $\varphi_{макс}=\varphi_{н.м.}$ which subsequently remains constant/invariable.

Thus, during the control according to the method of constant angle φ transient mode/conditions is finished at point P_2, where is satisfied condition (5-62). Beginning from this point, rocket flies along the ideal trajectory of PU (in this case - on the straight line P_2A') and is encountered with the target at point A'.

Let us consider now how will occur transition/transfer to the ideal trajectory of PU (i.e. transition/transfer to the trimmed/steady-state mode/conditions) during the control according to the method of zero angle φ.

In this case in the trimmed/steady-state mode/conditions must be satisfied the condition

$$\varphi_{\text{уст}} = 0, \qquad (5\text{-}63)$$

and, consequently, the condition

$$\left|\frac{d\varphi}{dt}\right|_{\text{уст}} = 0. \qquad (5\text{-}64)$$

Retaining the same assumptions which were done in the examination of the method of constant angle φ, it is not difficult to ascertain that satisfaction of conditions (5-63) and (5-64) will be provided, if command/crew to controls x is formed according to the law

$$x = a\frac{d\varphi}{dt} + b\varphi, \qquad (5\text{-}65)$$

while the deflection of control δ depends on command/crew x according to the law, depicted in Fig. 5-28, i.e., according to the same law, as in the preceding case.

The process of setting straight flight with the regulation according to this method is depicted in Fig. 5-29 and 5-30, carried out for convenience in the comparison on the same scale as Fig. 5-26 and 5-27.

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To point P_1, which corresponds to moment/torque t_1, rocket moves accurately over the same law as in the preceding case, i.e., in the circle/circumference of a radius ρ_0. This is explained by the fact that in this section the derivative $d\varphi/dt$ is great, mode/conditions distant of that being steady and command/crew to the controls is maximum. In section P_1P_2 the derivative $d\varphi/dt$, as in the preceding case, it is reduced up to zero. Therefore during the control using the method of constant φ at point P_2 would already take place steady-state conditions. But during the control using the method of bullet φ it is required not only so that the derivative $d\varphi/dt$ would become equal to zero, but so that and the angle φ would be made equal to zero $(\varphi_{уст}=0)$, and at point P_2 (moment/torque t_2) this angle was maximum. Consequently, during the control using the second method at point P_2 (moment/torque t_2) mode/conditions will be still distant from that being steady; to the controls will continue to be supplied large command/crew and rocket will continue to turn up in the

circle/circumference to the same side. As a result of this motion of rocket the angle φ will begin to be reduced and, therefore, the derivative $d\varphi/dt$ will become negative. Since subsequently must be established/installed the mode/conditions during which $\varphi_{ycm} = 0$ and, consequently, $\frac{d\varphi}{dt}_{ycm} = 0$ that values $d\varphi/dt$ and φ will be with the ideal regulation changed according to the laws, depicted in Fig. 5-30a and b.

The trimmed/steady-state mode/conditions will be now achieved/reached not at point P_2 (at moment/torque t_2), but at certain point P'_2 (at moment/torque t'_2) beginning with which will be satisfied condition φ=const=0. Beginning from this point, rocket will move over the ideal trajectory of PU, in this case - on the straight line P'_2A', and it will meet target at point A'.

From the comparison of Fig. 5-29 and 5-26 it seems at first glance that during the induction/guidance according to the method of zero angle φ the process of the output of rocket to the ideal trajectory always occupies more time, than during the induction/guidance to to the method of constant angle φ. However, it is necessary to keep in mind that the examination given above carries only qualitative character, since it does not consider the inertness of control system.

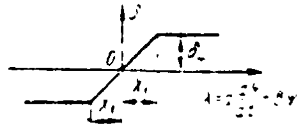

Fig. 5-28.

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But if we take into account the properties of real control system, then it can seem that moment/torque t_2 (see Fig. 5-27 and 5-30) in the system, which has $\varphi_{ycm} = 0$, will occur considerably earlier than in the system, which has $\varphi_{ycm} = \text{const}$. In this case the complete duration of transient process in the system with the zero angle φ can prove to be much the same or even somewhat smaller than in the system of constant angle φ. Therefore judgment about duration and character of transient processes can be obtained only on the base of detailed analysis of the dynamics of rocket flight taking into account the properties of real control system.

During the use/application for measuring the angle φ radio direction finder (radar) with that stabilized by antenna the advantage of the method of zero angle φ is the possibility of employing the narrower visual angle of the antenna system of direction

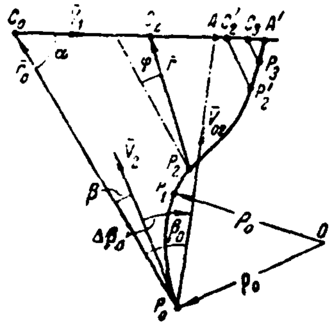

Fig. 5-29.

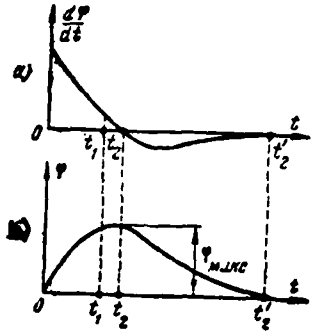

Fig. 5-30.

5-6. Required minimum range of homing.

At the moment of transition/transfer for the homing the velocity vector of rocket $\bar{V}_2$ composes with the direction to the target certain lead angle β (Fig. 5-31).

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This angle can differ from the required lead angle β_0 by certain value

$$\Delta\beta = \beta_0 - \beta \quad (5\text{-}66)$$

(during guidance on the linear curve $\beta_0=0$).

If for the flight time to the target rocket does not have time to eliminate this error of prevention/advance $\Delta\beta$, then it will fly wide of the mark.

Consequently, the minimally necessary range of homing r_0 must be determined from the condition so that within the time of homing rocket would have time to correct its trajectory and to reduce to to zero errors of prevention/advance $\Delta\beta$. The greater there will be the initial error $\Delta\beta$, the greater will be required the range r_0.

Is given below the calculation of required minimum range r_0 for two fundamental cases of the homing: guidance according to the method PU and according to the method of linear curve.

Let us consider first guidance according to the method PU.

In this case the process of reducing of the error $\Delta\beta$ to zero, i.e., transition rocket to the ideal trajectory PU, is illustrated by Fig. 5-32. It is analogous to Fig. examined above 5-26 and corresponds to the method of constant angle φ.

Fig. 5-32 differs from Fig. 5-26 only in terms of the fact that for simplification in the analysis it is accepted that the rocket flies in the circle/circumference not to point P_1 (see Fig. 5-26), but to point P_2, i.e., before the transition/transfer to the ideal trajectory PU.

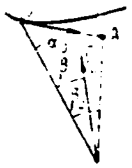

Fig. 5-31.

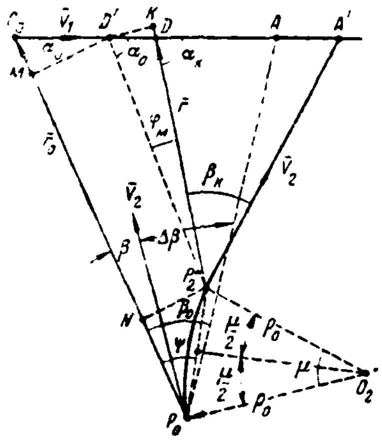

Fig. 5-32.

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This idealization indicates the assumption that the command/crew to the controls is maximum, if $\frac{d\varphi}{dt}\neq 0$. is equal to zero, if $\frac{d\varphi}{dt}=0$

At the moment of the beginning of homing the target and rocket are located at point_ C_0 and P_0 respectively. At this moment the error of prevention/advance is equal to $\Delta\beta=\beta_0=\beta$.

At certain moment/torque t_1, when target proves to be at point D, and rocket - at point P_1, the error of prevention/advance is made by equal to zero. From point P_0 to point P_1 the rocket flies in the circle/circumference of a radius ρ_0, while from point P_2 and to the encounter with target - on the tangent P_2A'. Target flies with a constant velocity of $\bar{V}_1$. At point P_2 the angle of rotation φ reaches its maximum value $\varphi_м$ (since at this point $\frac{d\varphi}{dt}=0$) The value of this angle is especially important during the use of an antihunting antenna, since in this case an increase of angle $\varphi_м$ will require corresponding increase of the visual angle of radio-goniometer.

The straight line $\overline{P_2D'}$ is carried out from point P_2 in parallel to vector $\bar{r}_0$ up to the intersection with the trajectory of target at point D'. Straight lines $\overline{D'M}$ and $\overline{D'K}$ - the perpendiculars, omitted from point D' to the straight lines $\bar{r}_0$ and $\bar{r}$ respectively. The straight line $\overline{P_2N}$ is perpendicular to vector $\bar{r}_0$.

Remaining designations in Fig. 5-32 do not require explanations. Since at point P_2 the error $\Delta\beta=0$, then at this point is satisfied the condition for ideal prevention/advance, i.e.,

$$\sin\beta_\kappa = x\sin\alpha_\kappa, \tag{5-67}$$

where

$$\alpha_\kappa = \alpha_0 + \varphi_\kappa. \tag{5-68}$$

For the time Δt, during which the rocket flies in circle/circumference P_0P_2, its velocity vector $\bar{V}_2$ is turned to the angle

$$\mu = \frac{V_2}{\rho_0}\Delta t. \tag{5-69}$$

From Fig. 5-32 it follows that

$$\alpha_0 + \beta = \beta_r + \varphi_M. \qquad (5\text{-}70)$$

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Furthermore, from the same figure escape/ensue the following relationships/ratios:

$$\left.\begin{aligned} &\overline{D'K} = \overline{DD'}\sin(\alpha_0 + \varphi_M) = \overline{P_2D'}\sin\varphi_M;\\ &\overline{P_2D'} = r_0 - \overline{C_0D'}\cos\alpha_0 - \overline{P_0P_2}\cos\psi;\\ &\psi = \beta - \frac{\mu}{2}, \text{ or } \psi = -\frac{\mu}{2} + \beta;\\ &\overline{P_0P_2} = 2\rho_0\sin\frac{\mu}{2};\\ &\overline{C_0D'} = \overline{C_0D} \quad \overline{DD'} = V_1\Delta t \quad \overline{DD'} \end{aligned}\right\} \qquad (5\text{-}71)$$

From relationships/ratios (5-71) is obtained the following equation relative to value $\overline{DD'}$:

$$\overline{DD'} = \frac{r_0 - V_1\Delta t\cos\alpha_0 - 2\rho_0\sin\frac{\mu}{2}\cos\left(\frac{\mu}{2} - \beta\right)}{\frac{\sin(\alpha_0 + \varphi_M)}{\sin\varphi_M} - \cos\alpha_0}. \qquad (5\text{-}72)$$

Since angle φ_M is low (will be proved below that $\varphi_M < 10°$), that with the high accuracy it is possible to assume that

$$\frac{\sin(\alpha_0 + \varphi_M)}{\sin\varphi_M} = \frac{\sin\alpha_0 + \varphi_M\cos\alpha_0}{\varphi_M} = \frac{\sin\alpha_0}{\varphi_M} + \cos\alpha_0.$$

Then equation (5-72) is simplified and takes the form:

$$\overline{DD'} = \frac{\varphi_M}{\sin\alpha_0}\left[r_0 - V_1\Delta t\cos\alpha_0 - 2\rho_0\sin\frac{\mu}{2}\cos\left(\frac{\mu}{2} - \beta\right)\right]. \qquad (5\text{-}73)$$

From Fig. 5-32 we obtain the following further relationships/ratios:

$$\overline{MD'} = \overline{C_0D'}\sin\alpha_0 = (V_1\Delta t - \overline{DD'})\sin\alpha_0$$

$$\overline{NP_2} = \overline{P_0P_2}\sin\psi = 2\rho_0\sin\frac{\mu}{2}\sin\left(\frac{\mu}{2}+\beta\right)$$

Since $\overline{MD'} = \overline{NP_2}$, then

$$(V_1\Delta t - \overline{DD'})\sin\alpha_0 = 2\rho_0\sin\frac{\mu}{2}\sin\left(\frac{\mu}{2}+\beta\right). \qquad (5\text{-}74)$$

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Solving equations (5-74) and (5-73) together and excluding of them value $\overline{DD'}$, we obtain:

$$V_1\Delta t(\sin\alpha_0 + \varphi_\kappa\cos\alpha_0) = 2\rho_0\sin\frac{\mu}{2}\sin\left(\frac{\mu}{2}+\beta\right) + \varphi_\kappa r_0 - 2\varphi_\kappa\rho_0\sin\frac{\mu}{2}\cos\left(\frac{\mu}{2}+\beta\right). \qquad (5\text{-}75)$$

Equations (5-67)-(5-70) and (5-75) are system of five equations with five unknowns (α_κ, β_κ, Δt, φ_κ and μ). After exception/elimination of three of these unknowns are obtained the following equations with two unknowns (μ and φ_κ):

$$\varphi_\kappa = \frac{\varepsilon\mu\sin\alpha_0 - 2\sin\frac{\mu}{2}\sin\left(\frac{\mu}{2}+\beta\right)}{\frac{r_0}{\rho_0} - 2\sin\frac{\mu}{2}\cos\left(\frac{\mu}{2}+\beta\right) - \varepsilon\mu\cos\alpha_0} = \varphi_{1\kappa}. \qquad (5\text{-}76)$$

$$\varphi_\kappa = \frac{\sin(\mu+\beta) - \varepsilon\sin\alpha_0}{\cos(\mu-\beta) + \varepsilon\cos\alpha_0} = \varphi_{2\kappa}. \qquad (5\text{-}77)$$

Assuming/setting $\varphi_{1\kappa} = \varphi_{2\kappa} = \varphi_\kappa$, it is possible from equations (5-76) and (5-77) to obtain one equation with one unknown (μ):

$$\frac{\varepsilon\mu\sin\alpha_0 - 2\sin\frac{\mu}{2}\sin\left(\frac{\mu}{2}+\beta\right)}{\frac{r_0}{\rho_0} - 2\sin\frac{\mu}{2}\cos\left(\frac{\mu}{2}+\beta\right) - \varepsilon\mu\cos\alpha_0} = \frac{\sin(\mu+\beta) - \varepsilon\sin\alpha_0}{\cos(\mu-\beta) + \varepsilon\cos\alpha_0}. \qquad (5\text{-}78)$$

However, this equation is transcendental and in general form analytically accurately is not solved. Therefore in general form it is possible to find only approximate solution of this equation, taking into account that in the cases $\mu \leqslant 0.5$ interesting us.

Actually/really, if at the moment of transition/transfer for the homing error $\Delta\beta$ is equal to zero, i.e., rocket immediately flies along the ideal trajectory, then $\mu=0$. The greater the error $\Delta\beta$, the larger angle μ. As we shall see further on analysis, at the real values of errors ($\Delta\beta \leqslant 30°$) the angle μ does not exceed 20-30°, i.e., $\mu \leqslant 0.3-0.5$.

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Therefore in the first approximation, it is possible to expand functions $\sin\mu$, $\cos\mu$, $\sin\mu/2$ and $\cos\mu/2$ in the series/rows according to degrees μ and, after substituting these series/rows into equation (5-78), to disregard/neglect the members of the third and the higher orders of smallness (i.e. by terms, which contain μ^3, μ^4 and so forth). Then instead of precise equation (5-78) we obtain the following equation of first approximation:

$$A\mu^2 - 2B\mu + C = 0. \qquad (5\text{-}79)$$

where

$$A = 3\Delta\beta' \sin\beta - \cos^2\beta - \varepsilon\cos\alpha_0\cos\beta - \frac{r_0}{\rho_0}\sin\beta; \quad B = \frac{r_0}{\rho_0}\cos\beta; \quad C = 2\Delta\beta'\frac{r_0}{\rho_0} \qquad (5\text{-}80)$$

$$\Delta\beta' = \sin\beta_0 - \sin\beta \qquad (5\text{-}81)$$

It is obvious,

$$\Delta\beta' \approx \Delta\beta \qquad (5\text{-}82)$$

Therefore approximate value of angle μ is found by the formula

$$\mu = \frac{B \pm \sqrt{B^2 - 4AC}}{2A} \qquad (5\text{-}83)$$

Substituting approximate value μ into precise formulas (5-76) and (5-77), we find approximate values $\varphi_{1м}$ and $\varphi_{2м}$ angle of rotation $\varphi_м$. If the obtained value μ was sufficiently precise, then it will be obtained $\varphi_{2м} \approx \varphi_м$. If between $\varphi_{1м}$ and $\varphi_{2м}$ proves to be the difference of more than 10o/o, then should be slightly changed value μ then, so as to obtain $\varphi_{2м} \approx \varphi_{1м}$. [Virtually this refinement of value μ it is obtained very simply, since $\varphi_{1м}$ barely is changed with the small changes μ, and $\varphi_{2м}$ is changed very sharply, also, to the same side, as value μ (i.e. for increase $\varphi_{2м}$ it is necessary slightly to increase μ]. However, with $\mu \leqslant 0.4$ the first approximation, found from formula (5-83), proves to be already sufficiently to precise ones and it is possible to assume/set $\varphi_м = \varphi_{1м}$ without producing further refinement.

From formula (5-83) it follows that for the existence of the actual value of angle μ must be satisfied the condition

$$B^2 > 4AC. \qquad (5\text{-}84)$$

If this condition is not satisfied, then actual value μ there does not exist, i.e., transition/transfer to the ideal trajectory PU does not manage to occur, and rocket flies wide of the mark.

Consequently, condition (5-84) is the necessary condition of the feasibility of transition/transfer to the trajectory PU.

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In the revealed form after the substitution of values A, B and C from relationships/ratios (5-80) is obtained the following condition of the feasibility of transition/transfer to the ideal trajectory PU:

$$\frac{r_0}{\rho_0} \geqslant 2\Delta\beta' \frac{\cos^2\beta + \epsilon\cos\beta\cos\alpha_0 + 3\Delta\beta'\sin\beta}{\cos^2\beta - 2\Delta\beta'\sin\beta}. \quad (5\text{-}85)$$

Therefore the critical value of distance r_{0kp} is determined from the formula

$$\frac{r_{0kp}}{\rho_0} = 2\Delta\beta' \frac{\cos^2\beta + \epsilon\cos\beta\cos\alpha_0 + 3\Delta\beta'\sin\beta}{\cos^2\beta - 2\Delta\beta'\sin\beta}. \quad (5\text{-}86)$$

When $r_0 < r_{0.kp}$ the rocket does not manage to pass to the ideal trajectory PU. Consequently, distance r_0 always must be chosen from the condition

$$r_0 > r_{kp}. \quad (5\text{-}87)$$

Using relationships/ratios (5-86) and (5-80), it is possible to reduce formula (5-83) to the following form:

$$\mu = \frac{2\Delta\beta' \frac{r_0}{r_{\kappa p}} \cos\beta}{\cos^2\beta - 2\Delta\beta' \left(\frac{r_0}{r_{\kappa p}} - 1\right) \sin\beta} \cdot \left[1 - \sqrt{\left(1 - \frac{r_{\kappa p}}{r_0}\right)\left(1 - 2\Delta\beta' \frac{\operatorname{tg}\beta}{\cos\beta}\right)}\right]. \quad (5\text{-}88)$$

The obtained relationships/ratios in the general case are bulky. Therefore let us consider the separately most interesting cases.

First case - case of the low errors of prevention/advance ($\Delta\beta \leqslant 10°$).

With $\Delta\beta' \leqslant 0.2$ formulas (5-86) and (5-88) strongly are simplified and take the form:

$$r_{\kappa p} \approx \rho_0 2\Delta\beta' \left(1 + \frac{\varepsilon \cos\alpha_0}{\cos\beta_0}\right); \quad (5\text{-}89)$$

$$\mu \approx \frac{2\Delta\beta' \frac{r_0}{r_{\kappa p}}}{\cos\beta_0} \left(1 - \sqrt{1 - \frac{r_{\kappa p}}{r_0}}\right). \quad (5\text{-}90)$$

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When $r_0 \geqslant 2r_{\kappa p}$ it is obtained

$$\mu \approx \frac{\Delta\beta'}{\cos\beta_0}, \quad (5\text{-}90')$$

where

$$\Delta\beta' \approx \Delta\beta.$$

Since $\sin\beta \leqslant \sin\beta_0 = \varepsilon \sin\alpha_0$, then when $\varepsilon < \frac{1}{2}$ is obtained by $\beta \leqslant 30°$ $\cos\beta \geqslant 0.87$.

Therefore when $\varepsilon \leqslant \frac{1}{2}$ we have:

$$r_{\kappa p} \approx \rho_0 2\Delta\beta'(1+\varepsilon\cos\alpha_0); \qquad (5\text{-}89')$$

$$\mu \approx \Delta\beta' \qquad (5\text{-}90')$$

In the worse case when $\alpha_0=0$,

$$r_{\kappa p} \approx \rho_0 2\Delta\beta'(1+\varepsilon). \qquad (5\text{-}91)$$

Since in the case $\Delta\beta'\leqslant 0.2$ in question, then $\mu\leqslant 0.2$. This confirms the correctness of the assumption made above that $\mu\leqslant 0.5$.

With the same assumptions ($\Delta\beta'\leqslant 0.2$; $r_0\geqslant 2r_{\kappa p}$) we obtain from formula (5-76) approximate value of angle φ_{κ}:

$$\varphi_{\kappa} \approx \frac{\rho_0}{2r_0}(\Delta\beta')^2 \approx \frac{1}{4}\Delta\beta'\frac{r_{\kappa p}}{r_0}. \qquad (5\text{-}92)$$

Consequently, with $\Delta\beta'\leqslant 0.2$ and $r_0\geqslant 2r_{\kappa p}$ will be $\varphi_{\kappa}\leqslant 1.5°$.

Second case - case of the large errors of prevention/advance ($\Delta\beta>10°$). In this case simplest formulas are obtained, if one assumes that $\Delta\beta=\beta_0$, i.e., $\beta=0$. Under this assumption of formula (5-86) and (5-88) they take the following form:

$$\frac{r_{o\kappa p}}{\rho_0} = 2\Delta\beta'(1+\varepsilon\cos\alpha_0); \qquad (5\text{-}93)$$

$$\mu = 2\Delta\beta'\frac{r_0}{r_{o\kappa p}}\left(1-\sqrt{1-\frac{r_{o\kappa p}}{r_0}}\right), \qquad (5\text{-}94)$$

where

$$\Delta\beta' \approx \sin\beta_0 = \varepsilon\sin\alpha_0. \qquad (5\text{-}95)$$

Being congruent/equating formulas (5-93) and (5-94) with (5-89) and (5-90) respectively, it is not difficult to ascertain that they approximately/exemplarily coincide, since $\cos B_0 \sim\sim 1$.

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From formulas (5-93) and (5-95) we obtain:

$$\frac{r_{\text{окр}}}{\rho_0} = 2\varepsilon(1 + \varepsilon\cos\alpha_0)\sin\alpha_0. \tag{5-96}$$

This expression has a maximum when $\alpha_0 = \alpha_{\text{окр}}$, where $\alpha_{\text{окр}}$ is determined from the relationship/ratio

$$\cos\alpha_{\text{окр}} = \frac{\sqrt{1 + 8\varepsilon^2} - 1}{4\varepsilon}. \tag{5-97}$$

In this case

$$\left(\frac{r_{\text{окр}}}{\rho_0}\right)_{\text{макс}} = \frac{(3 - \sqrt{1 + 8\varepsilon^2})\sqrt{\sqrt{1 + 8\varepsilon^2} + 4\varepsilon^2 - 1}}{4\sqrt{2}}. \tag{5-98}$$

From formula (5-97) it follows that when $\varepsilon \leq \frac{1}{2}$ there will be $\cos\alpha_{\text{окр}} \geq 68°$. Taking into account this relationship/ratio, and also the low dependence of value $r_{\text{окр}}$ on α_0 at α_0=68-90°), it is possible to consider that the worse case occurs at $\alpha_0 \sim\sim 90°$, and to assume/set

$$\frac{r_{\text{окр}}}{\rho_0} \approx 2\varepsilon = 2\Delta\beta'. \tag{5-99}$$

The analysis of formula (5-76) shows that the greatest value of angle $\vartheta_{\text{м}}$ is obtained also at $\alpha_0 \sim\sim 90°$. (This analysis is not given here due to its unwieldiness). Therefore in formula (5-76) it is possible to assume/set α_0=90°. Then when $r_0 \geq 2r_{\text{окр}}$ it will be:

$$\vartheta_{\text{м}} \approx \frac{1}{4}\varepsilon\frac{r_{\text{окр}}}{r_0} = \frac{1}{4}\Delta\beta'\frac{r_{\text{окр}}}{r_0}. \tag{5-100}$$

This formula also coincides with formula (5-91), derived for the case of the low errors of prevention/advance $\Delta\beta$.

When $r_0 < 2r_{\kappa p}$ formula (5-100) is imprecise and calculation φ_κ must be produced according to precise formulas (5-76) and (5-77) by method indicated above. On the basis of these precise calculations is constructed the curve, represented in Fig. 5-33 (upper curve). It corresponds to following data: $\alpha_0 = 90°$; $\varepsilon = 1/2$; $\Delta\beta = 30°$ $(\beta = 0)$. In the same figure is plotted the curve, which corresponds to $\Delta\beta$=15°; α_0=90° and ε=0.5. Since in this case $\beta \neq 0$, the calculation is done directly according to precise formulas.

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The analysis given above from the very beginning was based on the following assumptions:

1. Target speed $\vec{V}_1$ is constant, i.e., target does not implement maneuver.

2. Control system is inertia-free, i.e., time lag of control τ is equal to zero.

The analysis of the effect of the maneuvers of target on values

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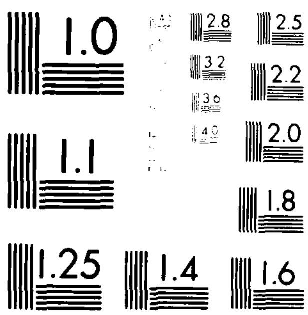
1.0
2.8
2.5
3.2
2.2
3.6
2.0
1.1
4.0
1.8
1.25
1.4
1.6

$r_{кр}$ and φ showed that while most dangerous maneuverings the results are approximately/exemplarily the same as in the most dangerous case of rectilinear motion of target ($\alpha_0 \sim 90°$; $\beta=0$). Control lag, which does not exceed 0.5-1 s, also does not worsen/impair the substantially obtained above results.

Therefore to admissibly take into account all disregarded during the conclusion/output factors by introduction to final formulas for $r_{..}$ and $\varphi_{..}$ the reserve to 30-50o/o.

For the illustration let us consider several numerical examples.

Example 1. $V_1 \leqslant 300$ m/s; $V_2=600$ m/s; $W_{рн}=10g$; $\Delta\beta \leqslant 10°$.

Then $\rho_0 = \frac{V_2^2}{W_{рн}} = 3,6$ км; $\varepsilon = \frac{V_1}{V_2} \leqslant \frac{1}{2}$.

Since $\Delta\beta \leqslant 10°$, then the error of prevention/advance can be considered low. Then according to the formula (5-89") we have:

$$r_{окр} \approx \rho_0 2\Delta\beta (1-\varepsilon) \approx \rho_0 2\Delta\beta (1-\varepsilon).$$

In the worse case when $\Delta\beta = \Delta\beta_{макс} = 10°$ and $\varepsilon = \varepsilon_{макс} = 0,5$, is obtained $r_{окр} \approx 1,8$ km. Therefore, taking small reserve, it is possible to require satisfaction of the following condition:

$$r_з \geqslant 2,5 \text{ км}. \quad (5\text{-}..)$$

For the evaluation of a maximally possible angle of rotation $\varphi_{..}$ we find relation $\frac{r_0}{r_{0кр}}$.

When $\frac{r_0}{r_{0кр}} \geqslant 2$ angle $\varphi_м$ will be determined according to formula (5-91):

$$\varphi_м = \frac{1}{4} \Delta\beta \frac{r_{0кр}}{r_0}.$$

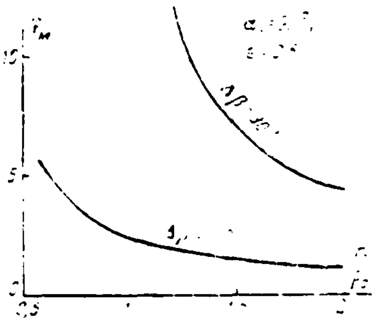

Fig. 5-33.

In the worse case when r_0 is minimal, i.e., with r_0=2.5 km, is obtained $\frac{r_0}{r_{\partial \kappa p}}<2$, and formula (5-91) is wrong. In this case rough estimate can be produced on the curves in Fig. 5-33.

Since in this case of r_0/ρ_0=0.7, then at $\Delta\beta$~~15° it would be obtained $\varphi_м=4°$. Taking into account that with the decrease of $\Delta\beta$ angle $\varphi_м$ sharply is reduced, one should consider that at $\Delta\beta\leqslant 10°$, even taking into account reserve, will be satisfied ondition

$$\varphi_м<4°. \qquad (5- \quad)$$

Example 2. $\Delta\beta=\beta_{0\,макс}=30°$; remaining data - the same as in example 1.

In this case β=0 are valid formulas (5-99) and (5-100), from which are obtained following data:

$$r_{окр} \approx 2\Delta\beta\rho_0 \approx \rho_0 = 3{,}6 \text{ км}$$

or with the small reserve

$$r_0 \geqslant 5 \text{ км}. \qquad (5\text{-}103)$$

When $r_0 \geqslant 2r_{окр} = 7{,}2$ км it will be:

$$\varphi_м = \frac{1}{4}\Delta\beta\frac{r_{окр}}{r_0} \leqslant 7^\circ. \qquad (5\text{-}104)$$

In the worse case (with r_0=5 km) r_0/ρ_0=1.4, and through the curve of Fig. 5-33 we find: $\varphi_м = 8^\circ$.

Therefore, taking small reserve, we assume/set

$$\varphi_м \leqslant 10^\circ. \qquad (5\text{-}104')$$

One should recall that all obtained results relate to the case of control according to the method of constant angle φ. During the control according to the method of zero angle φ the required range of homing can somewhat be changed as a result of a change in the form of the transfer trajectory (see Fig. 5-27 and 5-30). The maximum value of angle of rotation $\varphi_м$ remains the same as with the method of constant angle φ (Fig. 5-27 and 5-30).

Above were examined requirements for the range of homing during the induction/guidance according to the method PU. Let us consider now requirements for the range of homing during the induction/guidance according to the method of linear curve. It was above indicated that the method of linear curve gives low vectoring error only when $\varepsilon \ll 1$.

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Therefore during the determination of the required range it is possible in the first approximation, to assume/set $\varepsilon = 0$, i.e. to consider the target fixed. Then minimum distance r_{okp}, necessary for the incidence/impingement of rocket into the target, when the initial error of prevention/advance is present, $\Delta\beta$ is determined from Fig. 5-34.

At the moment of the loading of homing the rocket is located at point P_0, and target - at point C_0. Since up to the incidence/impingement of rocket into the target there is certain error of prevention/advance $\Delta\beta$, then, assuming/setting control system by ideal, it is possible to consider that the rocket moves always with maximally deflected control surfaces, i.e., over the circle/circumference of a minimum radius ρ_0.

From Fig. 5-34 it follows:

$$r_{okp} = 2\overline{P_0B} = 2\rho_0 \sin\frac{\Delta\beta}{2} \cos\frac{\Delta\beta}{2} = \Delta\beta,$$

therefore

$$r_{okp} = 2\rho_0 \sin\Delta\beta. \qquad (5-105)$$

Since usually $\Delta\beta \leqslant 30°$, then $\sin\Delta\beta \sim\sim \Delta\beta$ and

$$r_{кр} \approx 2\rho_0 \Delta\beta. \qquad (5\text{-}105')$$

This formula approximately/exemplarily coincides with formulas (5-89) and (5-99), obtained above for guidance according to the method PU.

Moving small reserve to the inertness of control and target, it is possible to assume that the range of homing on the linear curve must satisfy the condition

$$r_0 \geqslant (2,5 \div 3)\,\rho_0 \Delta\beta. \qquad (5\text{-}106)$$

where

$$\rho_0 = \frac{V_2^2}{W_{рм}}.$$

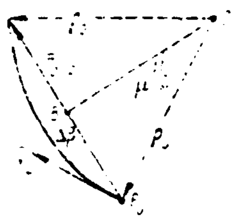

Fig. 5-34.

5-7. Fundamental sources of the errors of homing.

The errors of homing rocket to the target can be decomposed into the following basic groups depending on their character and the origins:

1. Vectoring errors caused by the inertness of control, when the maneuvers of target are present,.

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2. Vectoring errors, caused by limited maneuverability of rocket, with large curvature of required trajectory, caused by selected guidance method.

3. Vectoring errors, caused by limited maneuverability of

rocket, with large path curvature, caused by action of instrument errors.

4. Vectoring errors, caused by instrument errors as a result of presence of dead zone of control.

Let us consider in somewhat more detail each group of errors.

The errors of the first group are characteristic not only to homing, but also to remote control, and they were examined in § 2-2. The character of these errors is clearly illustrated by relationship/ratio (2-19'):

$$d_{max} \approx W_{cn} \tau^2,$$

where d_{max} - maximum value of the vectoring error, caused by the maneuver of target with acceleration W_{cn}, when in system the control of delay is present, τ. From relationship/ratio (2-19') it is evident that this error is caused by the joint action of two factors - maneuver of target and delay τ. In the absence of any of these factors (i.e. when $W_{cn}=0$ or $\tau=0$) the error of this form vanishes.

The errors of the second group are also characteristic not only to homing, but also to remote control they were examined above (see Chapter 2 and 3). In Chapter 3 it was shown that the greatest path curvature is required with the guidance of rocket to the target according to the method of linear curve. In this case due to the limited maneuverability of rocket appears the error whose maximum value is determined by relationship/ratio (3-12):

$$d_{макс}=\frac{V_1^2}{2W_{pм}},$$

where V_1 - target speed;

$W_{pм}$ - maximally possible acceleration of rocket.

From formula (3-12) evident that this error is reduced with an increase in the maneuverability of rocket (i.e. with an increase in acceleration $W_{pм}$) and by a decrease in the velocity of target it is made by equal to zero in the case of the fixed target.

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Since there are such methods of guidance which do not require the large path curvature (for example, method PU), then the errors of the second group are characteristic not to all homing systems, but only those of them, which are based on the unfavorable (from the point of view of the form of trajectory) guidance methods, for example to homing systems on the linear curve.

The vectoring errors of the first and second groups occur even in the systems, deprived of the instrument errors (here and throughout by instrument errors are understood all errors, caused by the imperfection of the action of equipment for control, with exception of the errors, caused by the inertness of the controls which are carried out into the separate class of errors).

The presence of instrument errors proves to be in the appearance of the third and fourth groups of vectoring errors.

The vectoring errors of the third group appear as a result of the fact that the presence of instrument errors can so bend the missile trajectory, required for the hit, that the rocket in the form of its limited maneuverability will not be able to follow along this trajectory and will fly wide of the mark.

As an example of this error can serve the error of homing on the linear curve, which appears when in the system is not considered angle of attack. This error was examined in § 4 of this chapter. In this paragraph was examined the case of homing on the linear curve to the fixed target, which is located at point O (Fig. 5-14).

In the absence of instrument errors (i.e. with $\beta=0$) the required missile trajectory would be rectilinear (straight line P_0O in Fig. 5-14) and rocket accurately would hit the target.

The presence of instrument error, i.e., the deviation of angle β from zero, leads to flight path curvature of rocket. With β=const this trajectory is converted into the logarithmic spiral (Fig. 5-17). If it was rocket it possessed the unlimited maneuverability, then, moving over this spiral, it hit was accurately the target, i.e., the presence of instrument error β would not cause vectoring error to the target.

However, due to the limitedness of maneuverability rocket at certain point M differs from the required spiral and flies wide of the mark (Fig. 5-17).

In § 5-4 it was shown that flight path curvature, caused by instrument errors, can cause vectoring error also as a result of the premature collision of rocket with the earth's surface or sea (Fig. 5-14).

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Thus, instrument error affects vectoring error not directly, but indirectly, i.e., only due to the presence of further unfavorable factors (limited maneuverability of rocket or the collision of rocket with the earth's surface or sea).

Let us consider finally the errors of the fourth group, caused by instrument errors as a result of the presence of the so-called dead zone of control.

Let $\Delta\beta$ - angular error in direction of the velocity vector of rocket (error of prevention/advance); then the linear error corresponding to it is equal to:

$$\Delta x \approx r\Delta\beta,$$

where r - distance from the rocket to the target.

If the control system functioned normally up to the encounter with target, then with $r \to 0$ value Δx also was approached was zero. However, in actuality not certain minimum distance $r_{мин}$ of the target the control system virtually ceases to function and error $\Delta x_{мин} \approx r_{мин}\Delta\beta$, which occurs at this moment of time, subsequently is not corrected and can even somewhat increase. Therefore rocket flies wide of the mark at a distance

$$d \geqslant \Delta x_{мин} = r_{мин}\Delta\beta.$$

[A more precise formula for calculating the error d it is given below). The minimum distance $r_{мин}$, on which the control system ceases normally to function, it is called the dead zone of control.

The dead zone of control can be caused by the following fundamental reasons:

1. Pulse radar, entering the control system, has the dead zone, caused by the impossibility to separate/liberate the impulse/momentum/pulse, reflected from the target, from the powerful/thick sounding pulse, if distance from the rocket to the target is reduced to certain value r_1, determined by the known relationship/ratio

$$r_1 = \frac{\Delta t c}{2}. \quad (5\text{-}107)$$

where Δt - minimally descernible by radar time interval between the straight/direct and echo pulses;

c - speed of light.

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Guarantee in the rocket equipment of the interval Δt, considerably less than 0.3-0.5 μs, can present difficulty. This causes dead zone r_1 on the order of 50-70 m.

2. Control system as a result of its inertness gives certain equivalent delay τ in execution of command/crew appearing at its entry. This means that all deviations of rocket from the correct flight of the target, discovered by radar for a time less than τ, to

the encounter of rocket with target, will not have time to cause the appropriate correction of the flight trajectory of rocket to the encounter of rocket with target. Thus, the inertness of control creates the dead zone

$$r_z = V_r \tau, \qquad (5\text{-}108)$$

where τ - equivalent delay of the system of control (see Chapter 1);

V_r - rate of closure of rocket with the target, determined by the relationship/ratio

$$V_r = \frac{dr}{dt}. \qquad (5\text{-}109)$$

If rocket and target have speeds $\bar{V}_2$ and $\bar{V}_1$ respectively (Fig. 5-35), then it is possible to write:

$$V_r = V_1 \cos\alpha + V_2 \cos\beta. \qquad (5\text{-}110)$$

In the worse case, with the firing accurately towards, the rate of closure is maximum and equal to:

$$V_{r\,макс} = V_1 + V_2. \qquad (5\text{-}110')$$

Therefore the maximum value of dead zone is equal to:

$$r_{z\,макс} \approx (V_1 + V_2)\tau. \qquad (5\text{-}111)$$

For example

$$(V_1 + V_2)_{макс} \approx 1\,000 \overset{(1)}{м/сек}. \qquad (5\text{-}112)$$

Key: (1). m/s.

Assuming that the delay τ does not exceed 0.3-0.5 s, we will obtain:

$$r_{z\,макс} \approx 300 \div 500 \text{ м}. \qquad (5\text{-}113)$$

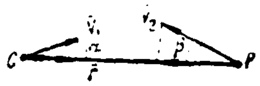

Fig. 5-35.

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3. With very low distances of target linear dimension of target becomes congruent with distance of r, it exceeds the limits useful visual angle of radio direction finder and work of latter sharply deteriorates. It is obvious, this phenomenon will occur with the following distance of the target:

$$r = r_з = \frac{l_m}{\varphi_n},$$

where l_m - maximum linear size of target;

φ_n - complete useful visual angle of radio direction finder.

If we take $l_m = 15 \div 30$ м and $\varphi_n = 10^\circ$, then it will be obtained by $r_з$=75-150 m.

4. With low distances of target voltage on input of radio direction finder grows in time so/such rapidly, that automatic gain control can not manage to compensate this change. Therefore in the absence of further special programmed gain control the normal

operation of radio direction finder can be broken.

5. In the absence of special measures sharp increase of sensitivity of radio direction finder for linear displacements of rocket which is obtained with $1/r \to \infty$, can lead to loss of dynamic stability of regulation.

From the aforesaid it is obvious that the homing system has certain dead zone $r_{мин}$, which for the known types of rockets can reach 300-500 m and is caused in essence by the inertness of control (delay). Let us explain, what vectoring error can be caused by the presence of dead zone.

For the approximate estimate of this error let us consider Fig. 5-36.

At the moment of reaching/achievement by the rocket of distance $r_{мин}$, equal to dead zone, the target it is located at point C_1, and rocket - at point P_1. In this case the velocity vector of rocket $\overline{V}_2$ has a lead angle β, different in the general case from the desired value β_0 ($\overline{V}_{02}$ - direction of ideal prevention/advance), i.e., occurs certain error of prevention/advance $\Delta\beta$.

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Since rocket and target have the limited transverse accelerations, then it is possible in the first approximation, to count those segments of their trajectories which they fly for the short time Δt, rectilinear, and speed V_1 and V_2 - by constants. Therefore it is possible to consider that the target moves over the straight line C_1A', and rocket - on the straight line P_1A', as it is indicated in Fig. 5-36. At the arbitrary moment of time the distance from the rocket to the target is equal to r.

With the help of Fig. 5-36 it is possible to demonstrate, that the minimum value of this distance, determining vectoring error, is equal to:

$$d=\frac{r_{мин}\Delta\beta}{\sqrt{1+\varepsilon^2-2\varepsilon\cos(\alpha-\beta)}}. \qquad (5\text{-}114)$$

where $\varepsilon=\frac{V_1}{V_2}$.

If dead zone $r_{мин}$ is defined by the inertness of control, then in accordance with formulas (5-108) and (5-110) we will obtain:

$$r_{мин}=V_2\tau(\cos\beta+\varepsilon\cos\alpha) \qquad (5\text{-}115)$$

and

$$d=V_2\tau\Delta\beta\frac{\cos\beta+\varepsilon\cos\alpha}{\sqrt{1+\varepsilon^2+2\varepsilon\cos(\alpha+\beta)}}. \qquad (5\text{-}116)$$

The greatest value of this expression is equal to:

$$d_{макс}=V_2\tau\Delta\beta. \qquad (5\text{-}117)$$

Let V_2=700 m/s; τ=0.4 s; $\Delta\beta$=5.7°. Then it is obtained

$$d_{макс} \approx 30 \text{ м}.$$

At the same values of V_2 and τ, but at $\Delta\beta$=1° it will be obtained $d_{макс} \approx 5$ м.

These examples show that it is very important to provide smallest possible values of the instrument error of prevention/advance $\Delta\beta$ and dead zone $r_{мин}$.

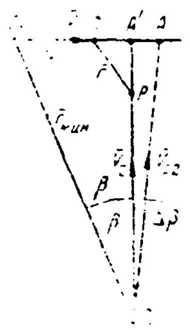

Fig. 5-36.

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Chapter Six.

REMOTE CONTROL.

6-1. General/common/total characteristic of remote-control systems.

In Chapter 1 was indicated that the remote-control systems can be subdivided into two forms:

the remote control of the first form - with the direct control/check of target (Fig. 1-20) and the remote control of the second form - with the control/check of target by means of the onboard equipment (Fig. 1-21).

Both forms of remote control can be automatic or be realized by an operator.

Each remote-control system contains the channels of control and direction (see Fig. 1-20 and 1-21).

For the control/check of target (or rocket) is used any form of the energy, emitted or by the reflected target (rocket). Depending on the form of the utilized energy the control/check can be:

a) visual (during the use of the visible light beams);

b) thermal [during the use of thermal infrared ones) of rays/beams]:

c) radio engineering (during the use of radio waves).

The major advantages and deficiencies/lacks in each of these forms of energy were described above (see § 5-2c).

Depending on the location of the primary source of energy of the system of the control/check of target as homing system, it is possible to subdivide into:

a) passive - during the use of its own radiation of the controllable/controlled/inspected object (target) or reflected radiation/emission of natural sources (sun, the Moon);

b) active - during the arrangement/position of primary source on the rocket;

c) semi-active - during the arrangement/position of primary source on KP.

The major advantages and deficiencies/lacks in these methods of control/check - the same as during the homing (see § 5-2 b).

The methods of control/check of the position of rocket differ somewhat from the methods of control/check of the target, since target is enemy object, and in the rocket it is possible to place special devices, which facilitate control/check of its position.

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Depending on type and the locations of the source of energy of the system of the inspection of rocket (bomb) can be:

a) passive;

b) semi-active;

c) active.

Passive are called the systems, which use natural radiation/emission of rocket (for example, heat or light/world, the emitted by engine rockets) or reflected rocket energy of natural sources (Sun, the Moon).

Active are called the systems, based on the use of energy of the emitter, installed on the rocket (bomb) for facilitating the inspection of it. During visual monitoring such emitters were usually the tracers, adjusted in the tail of rocket and which radiate light/world with force of several million candles. During the radio engineering inspection such emitter is the radio transmitter, installed on the rocket and oriented from the control center.

Semi-active are called the systems, in which primary source of energy is placed on KP and irradiates rocket, and the position of rocket is monitored on the response signals, which come from rocket on KP. Semi-active systems can be:

a) without the responder;

b) with the passive responder;

c) with the active responder.

In the systems without the responder is used natural reflection by the rocket of energy of its irradiating source, i.e., reflection from the missile body. In the systems with the passive responder on the rocket (bomb) is installed the special passive reflector, which considerably increases the efficiency of the reflection of energy of primary source. In the systems with the active responder on the rocket (bomb) is installed the transceiver, excited by energy of the incident waves of primary source and which sends reciprocal energy of considerably larger power.

During visual monitoring of the rocket in the last war were applied passive or active systems. Active systems give the long range of inspection, but they strongly decamouflage rocket and surrounding locality. During the heat control of rocket are applied passive systems, since jet engine is a comparatively powerful/thick source of thermal energy.

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During the radio engineering control/check are applied usually active systems and semi-active systems with the responder, since the sizes/dimensions of rocket usually are considerably less than the sizes/dimensions of target, and the low natural reflection of radio waves by missile body impedes the realization of control/check for long range.

The data about the mutual location of rocket and the targets, obtained along the channel of control/check, are used for mining the corresponding steering commands. Steering commands are developed by operator or appropriate automatic device/equipment and enter control channel. The typical block diagram of control channel is depicted in Fig. 6-1. The steering commands, manufactured by the sensor of commands/crews, are coded and are transmitted by the line of communications with KP to the rocket. By the rocket the signals accepted are decoded (they are deciphered) and they are transmitted to the actuating unit, which puts to the action of control surface of rocket.

Coding and decoding corresponding to it have twofold designation/purpose:

a) the classifying of steering commands and an increase in the freedom from interference of control channel;

b) the guarantee of distribution (selection) of different commands/crews (to the left, to the right, upward, down and the like).

The line of communications, entering the control channel, can be radio engineering, wire, thermal or light depending on the form of the energy, utilized for the transfer of commands/crews from KP to the rocket. The major advantage of the radio engineering principle of connection/communication is the possibility of the very long range of action.

The major advantage of connection/communication on the wires/conductors is the greatest freedom from interference. However, the guarantee of a reliable (without the breaks) connection/communication on the wires/conductors with the rapidly flying rocket presents great structural/design difficulties, even with the short distance of connection/communication. The thermal and light lines of communications, possessing no essential advantages over the radio link, have considerably smaller range.

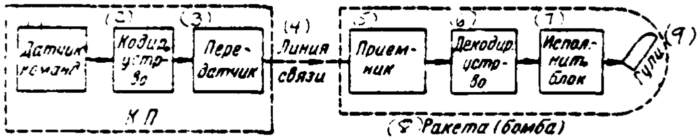

Fig. 6-1.

Key: (1). Sensor of commands/crews. (2). Coder. (3). Transmitter. (4). Line of communications. (5). Receiver. (6). Decoder. (7). Will carry out. block. (8). Rocket (bomb). (9). Controls.

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Therefore in the control channel in practice are applied the almost exclusively radio engineering lines of communications.

For the same reasons the line of communications, used with the remote control of the second form for the data transmission of control from the rocket to KP (see Fig. 1-21), also almost always is constructed on the radio engineering principle.

Thus, the channels of control and connections/communications, used in the remote-control systems, are constructed, as a rule, according to the radio engineering principle; whereas control of the rocket and the target can be both the radio engineering and visual or thermal.

The more detailed comparison of different methods of control, given below, shows that widest application it must have radio engineering control. However, the visual and thermal methods of control have such advantages that in a number of cases becomes appropriate their use/application.

6-2. Remote control of the second form.

The remote control of the second form has much in common with homing. Of this it is possible to be convinced from the comparison of the system block diagrams of homing and remote control of the second form, depicted in Fig. 6-2a and b respectively.

During the homing on the rocket is established/installed the equipment, which controls target position relative to rocket. Data of this control are converted within the rocket into the appropriate commands/crews to the controls.

With the remote control of the second view of rocket also is established/installed the equipment, which controls target position relative to rocket. Data of this control are transmitted by the line of communications on KP, are converted there into the appropriate commands/crews and are returned to the rocket along the control channel.

Thus, a vital difference in the remote control of the second form from the homing consists only of the fact that data of the control of target are converted into the commands/crews to the controls not within rocket itself, but on KP.

This further "journey" of data of control with the rocket to KP and their return after the appropriate conversion into the commands/crews to the rocket they substantially complicate equipment for control as a whole and onboard equipment they in particular and worsen/impair the freedom from interference of system due to the appearance of two further radio channels (communication channel and a control channel).

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Therefore it at first glance seems that the remote control of the second form must be considerably worse than the homing and the use/application of this principle of control is generally inexpedient. However, in actuality this conclusion/output is valid only with respect to the automatic remote control of the second form and it is not entirely valid in such a case, when the remote control of the second form realizes an operator.

Actually/really, if the remote control of the second form is made automatic, then it there will have no essential advantages over the homing, which is also automatic, but it will have a number of very essential deficiencies/lacks (greater complexity, less freedom from interference, the further distortion of data of control and steering commands in the channels of communication and control).

But if the remote control of the second form realizes an operator, then it acquires new quality in comparison with the homing - participation of man (operator) during the control process of rocket. This new quality imply the series/row of further deficiencies/lacks and advantages.

Main disadvantage in the control with the help of the operator in comparison with the automatic is the effect of the subjective properties of operator on quality and accuracy of control. Reliability and accuracy of this control depend not only on the qualification of operator, but also on his physical and morale state. Entire control process lasts usually several minutes, and sometimes even several seconds. The decisive time interval when rocket approaches a target, it is measured by the tenths of second. Therefore even very short-term quivering or inaccurate motion of the hand of operator can cause the inadmissible vectoring error. Large deficiency/lack in this control are also complexity and high cost/value of the preparation of the qualified operators.

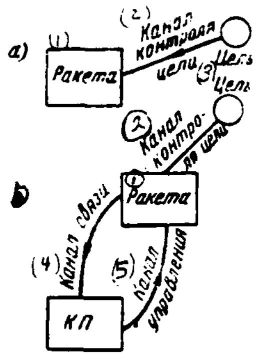

Fig. 6-2.

Key: (1). Rocket. (2). Channel of control of target. (3). Target. (4). Channel of communication. (5). Channel of control.

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Aging/training operators, conducted on the models, cannot completely replace agings/trainings by the real objects, under the actual conditions. But the aging/training, accompanied by real rocket launchings, is bypassed very dearly.

However, control of operator, besides deficiencies/lacks, has very essential advantage - participation during the control process of the conscious will of man, which ensures the flexibility of control and in a number of cases greater freedom from interference.

For the explanation of the special features/peculiarities of this control let us compare the block diagrams of closed-cycle control system with both forms of the controls, depicted in Fig. 6-3a and b respectively.

During the automatic control (Fig. 6-3b) measuring device discovers deviation of u_1 of the motion of rocket from the correct flight of the target. This is deviation u_1, passing through the control system with the transmission factor K_a, it acts on the actuating mechanism, which turns the controls of rocket. The deflection of control δ changes missile trajectory so that the deviation u_1 would decrease.

Transmission factor K_a is complex quantity. The dynamic-response characteristics of block K_a is selected in such a way that the quality of the regulation of system as a whole would be best, i.e., so that the rocket would approach the required trajectory as rapidly as possible, also, with the largest possible accuracy.

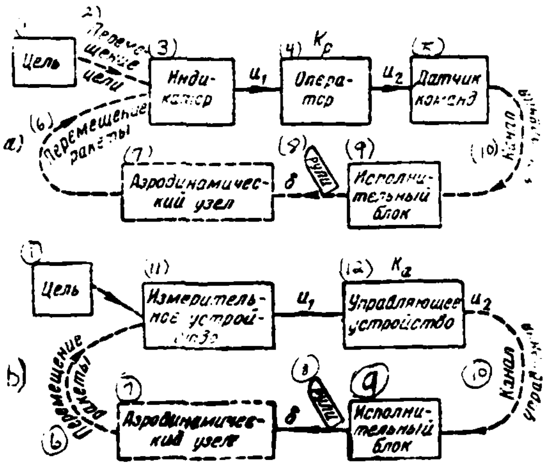

Fig. 6-3.

Key: (1). Target. (2). Shifting of target. (3). Indicator. (4). Operator. (5). Sensor of commands/crews. (6). Displacement/movement of rocket. (7). Aerodynamic unit. (8). Controls. (9). Actuating unit. (10). Control channel. (11). Measuring device. (12). Control system.

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However, it is very complicated to ensure the optimum selection of value K_a for all possible conditions: all copies of rocket, all possible flight conditions, all possible levels and forms of interferences/jammings, etc. Therefore with each such concrete/specific/actual shot when the flight conditions and control considerably differ from averages, coefficient K_a can noticeably differ from optimum, and because of this the accuracy of the guidance of rocket to the target can substantially deteriorate.

Control of operator (Fig. 6-3a) differs from automatic by the fact that instead of the control system with the transmission factor K_a functions the operator with certain equivalent transmission factor K_p, in terms of the equal to the ratio of the displacement/movement of control handle u_2 to deviation of u_1, observed on the indicator.

The value of this transmission factor K_p depends on the subjective properties of operator, but any qualified operator instinctively attempts to manage handle so that in each specific case deviation u_1 would be minimum. It is obvious that this flexibility (adaptability to the specific conditions), characteristic to operator, can prove to be highly useful when the flight conditions and control differ significantly from averages.

However, the flexibility of the control, realized by an operator, can consist not only in its adaptability to the specific conditions, but also in the fact that man is capable to control target in some relations "more thinly", is better to react to these or other the special features/peculiarities of target, than any of the contemporary automatic machines. For the illustration let us consider missile targeting with the help of the television head, described briefly in Chapter 1 (see Fig. 1-23). In this case the operator sees on the shield not the conditional image of target in the form of spot or point, but its complete picture. Thus, for instance, if rocket is induced aboard the ships of enemy, then operator can see these ships approximately/exemplarily with such as if he looked on them, being located on the rocket. At the same time the known automatic control systems usually can distinguish targets only by the intensity of the total energy going from them, but their area and speed.

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This special feature/peculiarity facilitates the selection of necessary target during the incidence/impingement into the visual angle of the system of control of immediately several targets, and also during the creation by the enemy of confusing reflectors.

The comparison given above of the remote control of the second type with the homing leads to the following conclusions:

1. The remote control of the second form must, as a rule, be realized by an operator.

2. Advantages, given by remote control of second form, are greater, the more in detail image of target (targets), reproduced on shield of operator.

The detail of the image of target first of all depends on the method of the control of target. If we for the control of the target use goniometer or radar, similar to the volume which is applied during the homing, then target will be reproduced on the screen of cathode-ray tube in the form of light spot. Detail of the image of target in this case will be minimum. If we for the control of the target use the television head, described above, then on the screen there will be the evidently approximately/exemplarily such image of target (if we do not consider distortions), as during the direct visual observation of target from the rocket. The detail of image in this case will be maximally possible. (Detail would be still larger only in such a case, when operator was located on the rocket and controlled target directly, i.e., during visual monitoring. But rocket is pilotless object, and this possibility is excluded).

If we instead of the television use "thermal vision" or "radiovision", i.e., the reproduction of the complete image of target with the help of the thermal ones or the radio waves, then detail will be obtained smaller than with the television, and distinguish targets will be more difficultly, since thermal and especially radio-image targets differ significantly from its visible image (target sectors, most intensely reflecting visible rays/beams, they can prove to be most intensely reflecting radios or thermal waves, and vice versa). Furthermore, the practical implementation of the systems of thermal and radiovision, suitable for the installation/setting up to the rocket, presents great difficulties.

Thus, it is virtually most simple to carry out remote control of second type in the following versions

1. System with the television head.

2. System with goniometer (radar), established/installed on rocket and similar to goniometer (radar) of homing system.

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Comparing these remote-control systems with the homing, it is possible to do the following fundamental conclusions:

1. System with the television head makes it possible to the greatest extent to use the advantages, given by the remote control of

the second form (connected with the advantages control with the help of the operator), since it gives the most detailed image of target. However, to this system are characteristic to the greatest degree both deficiencies/lacks in the remote control of the second form - greater complexity and possibility of the further effect of interferences/jammings on the channels of control and communications.

2. Remote-control system of second form with goniometer (radar) is considerably simpler than system with television head. Furthermore, it makes it possible to use radio engineering control and, therefore, to considerably raise range and reliability of the action of system. However, this system gives the image of target in the form of spot and does not make it possible to use this major advantage, as the possibility of the more detailed analysis of the image of target by man. At the same time this system remains more complicated than homing system, and it retains all deficiencies/lacks, inherent in control of operator.

From the aforesaid it follows that the fundamental method of the remote control of the second form is the control with the help of the television head. This method of control found practical use/application even in the last war. Therefore subsequently, speaking about the remote control of the second form, we will have in mind first of all control with the help of the television head.

What methods of guidance to the target are applicable with the remote control of the second form?

The guidance of rocket to the target with the remote control of the second form occurs in the same manner as for during the homing: with the help of the goniometer or the television camera, established/installed on the rocket, is discovered deviation from the direction the "rocket - target" in the measuring coordinate system, formed by goniometer or television camera. On the basis of value and sign of this deviation are developed the corresponding commands/crews to the controls..The fact that these commands/crews are developed by operator, but not by automatic machine, it does not vary substantially the character of missile trajectory.

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Therefore with the remote control of the second form it is possible to apply the same methods of guidance to the target, as during the homing, for example, guidance according to the method of linear curve or according to the method PU.

For the realization of guidance according to the linear curve or the method PU are suitable many of those methods which are applied during the homing. Thus, for instance, for the guidance of the rocket, furnished with television head, on the linear curve it is possible to establish/install television camera on the power weathervane. In this case the optical axis of television camera is rigidly connected with the direction of the airspeed of rockets $\bar{V}_{1в}$. Then, disregarding as during the homing, wind effect, it is possible to consider that the optical axis of television camera z_1 coincides with the direction of the speed of rocket $\bar{V}_1$ (Fig. 6-4). If direction to target $\bar{r}$ will coincide with the axis of television camera, then the glowing spot C' on the shield will coincide with the center of screen O. If direction $\bar{r}$ will compose with vector $\bar{V}_1$ certain angle $\bar{\varphi}$, then spot C' will be displaced relative to center of the screen to certain value $\bar{\rho}$, proportional to angle $\bar{\varphi}$. Therefore, if operator, who is found on KP, will manage the handle of the sensor of commands/crews by such form, in order that spot C' would remain approximately/exemplarily combined with the center of screen O, then in this case $\varphi=0$ and, therefore, rocket will move over the linear curve.

For the guidance according to the method PU it is necessary as during the homing, to stabilize the measuring system of coordinates $x_1y_1z_1$.

For this it is possible to hang up television camera in the gimbal suspension and to stabilize it from the rotations by gyroscopic device (Fig. 6-5). In this case the deviation $\bar{\varphi}$ will be measured in the stabilized coordinate system.

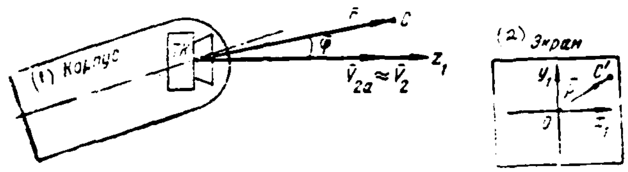

Fig. 6-4.

Key: (1). housing. (2). Screen.

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During guidance according to the method of constant angle φ the operator must manage rocket so that the deviation ρ of image (spot), which corresponds to angle $\bar{\varphi}$, would remain constant both in the value and in the direction. This means that during correct rocket flight along the trajectory PU the glowing spot C' on the shield must retain approximately/exemplarily constant/invariable position. Hence it follows that the operator must approach not for the decrease of the amount of deflection of spot, but a decrease in the velocity of the motion of spot. It is obvious, control according to this principle for the operator is considerably more complicated than is simple reducing of spot to center of the screen. Therefore control with the help of the television head is produced not according to the method of constant angle φ, but according to the method of zero angle φ. In this case the operator must approach only the coincidence of image (spot) with the center of the screen.

The analysis given above of the remote control of the second form makes it possible to do the following fundamental conclusions:

1. The automatic remote control of the second form, without giving essential advantages over homing, has at the same time very serious deficiencies/lacks (greater complexity and smaller freedom from interference).

2. Remote control of second form, realized by operator, has over homing only the advantage which is connected with replacement of automatic human control: greater flexibility of control, i.e., best adaptability of control to specific conditions of firing. This advantage most vividly is exhibited during the use/application for the remote control of the second type of television head.

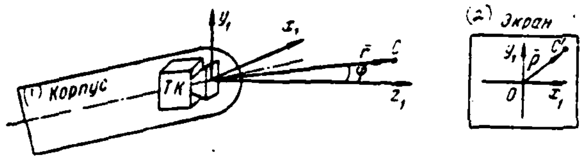

Fig. 6-5.

Key: (1). Housing. (2). Screen.

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3. Remote control of second form with the help of operator has in comparison with homing number of very serious deficiencies/lacks:

a) greater complexity of equipment for control (including onboard apparatuses), especially during use of television head;

b) smaller freedom from interference, caused by presence of two further radio channels (communication channel and control channel);

c) deficiencies/lacks, inherent in manual control in comparison with automatic;

d) deficiencies/lacks, connected with use of visible rays/beams during use/application of television head (smaller range and greater dependence on meteorological conditions, than during use of radio waves).

4. Since in majority of practical cases deficiencies/lacks, inherent in remote control of second form, exceed advantages characteristic to it, then this form of control obtained considerably smaller use/application, than homing (for example, see [10], [11], [12], etc.).

Let us now move on to the examination of the remote control of the first form.

6-3. Remote control of the first form with the help of the operator.

The block diagram of remote control with the help of the operator in the general case takes the form, depicted in Fig. 6-6. On the control center is located the measuring device, which controls rocket flight to the target. The deviation of rocket from the required trajectory, which corresponds to the selected method of guidance to the target, is mapped onto shield by the appropriate deviation of the glowing spot C from the center of the screen O. Operator controls shield and manages control handle (RU), issuing steering commands of rocket in such a way as to combine the glowing spot with the center of the screen. The coincidence of spot with the center of the screen corresponds to the coincidence of the actual trajectory of rocket flight with the required trajectory and thereby ensures the correct guidance of rocket to the target.

As noted above, control channel is constructed always according to the radio engineering principle. However, the inspection of target and rocket can be both the radio engineering and visual or thermal.

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Let us explain now, what methods of guidance to the target are applicable with the remote control with the help of the operator.

Simplest of that described above (Chapter 3) is the method of the covering of target. In this case the problem of control is reduced to retain the rocket on the line of sighting of target, i.e., in the direction KP - target.

During visual monitoring of rocket and target this problem is solved very simply with the help of telescopic sight. The operating principle of this tube is clarified by Fig. 6-7. Telescopic sight A can be directed for any point of path space of rotation around two orthogonal axes: I and II. One operator manages telescopic sight in such a way that its optical axis z_1 continuously would be directed to target C. In this case the image of target coincides with the center O against the cross lines of the axes of sight x_1y_1. After the rocket, guided to the target, comes into view of sight, the second operator manages handle RU in such a way as to combine the image of rocket with center O, and also, therefore, with the image of target.

Because of this rocket control continuously is retained on the line of the sighting of target. During the radio engineering control the role of sight implements radar or radio direction finder on shield of which is mapped angle $\bar{\psi}$ between the directions to the target and the rocket (see Fig. 1-22). This device/equipment is called sometimes radio sight, since it has the same designation/purpose, as optical sight. According to the analogous principle can be constructed the thermal sight, which uses an infrared spectrum, emitted by target and rocket.

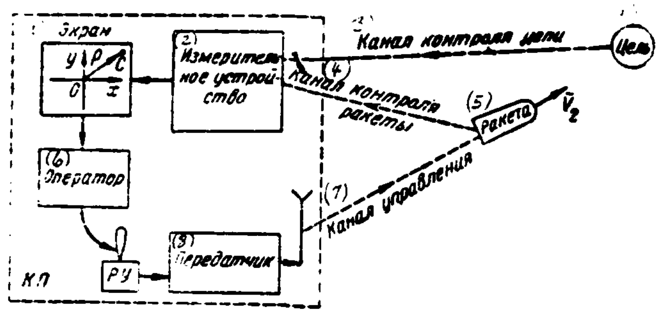

Fig. 6-6.

Key: (1). Shield. (2). Measuring device. (3). Channel of control of circuit. (4). Channel of inspection of rocket. (5). Rocket. (6). Operator. (7). Control channel. (8). Transmitter. (9). Target.

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Optical sight has the following major advantages:

1) greatest simplicity;

2) the greatest accuracy;

3) the greatest angular resolution (in the naked eye - 0.5-1'=0.15-0.3 the thousandth of distance).

Main disadvantage in the optical sight - short distance and the strong dependence of this range on the meteorological conditions. The limitedness of the action of visual monitoring is so serious a deficiency/lack, that it cannot serve as the fundamental method of control. The considerably greater propagation obtained the radio engineering control, which ensures many times greater ranges with the retention/preservation/maintaining very nearly the same accuracy.

In Chapter 4 it was indicated that to the fundamental methods the guidance of rocket to the target relates, besides the method of the covering of target, the method of linear curved and method PU. However, with the remote control of the first form the use/application of a method of linear curve does not make sense.

For guidance on the linear curve are required the continuous determination of direction rocket - target, $\bar{r}$. With the remote control of the first form when measuring equipment is found only on KP, the determination of direction $\bar{r}$ requires considerably more complicated equipment, than the determination of directions to the rocket and the target, required for the guidance from the method of the covering of target. At the same time, as it was shown in Chapter 3, guidance on the linear curve gives greater path curvature, than induction/guidance according to the method of the covering of target. Consequently, with the remote control of the first form guidance on the linear curve is more complicated and gives worse results than guidance according to the method of hitting of target. Therefore with the remote control of the first form this guidance method is not applied.

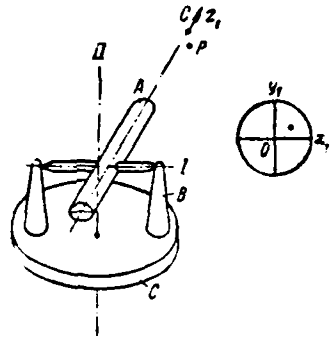

Fig. 6-7.

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Guidance according to the method PU with the remote control also requires more complicated equipment for control, than induction/guidance according to the method of the covering of target, but it gives more straight path of rocket. Therefore the fundamental field of application of remote control, based on the method PU, are the cases when induction/guidance according to the method of the covering of target requires very curved path, namely with the large relation of target speeds and rocket V_1/V_2 or the low target ranges (see § 3-5). However, remote control according to the method PU, as it will be explained below, requires the complicated computers and its realization proves to be more simply during the automatic control, but not during the control with the help of the operator.

Therefore remote control according to the method PU is described below in connection with automatic control.

6-4. Automatic remote control of the first form.

Automatic control differs from manual in terms of the absence of the operator, which issues steering commands.

When operator is one of the components/links of closed-cycle control system (see Fig. 6-3a), control circuit must be closed through KP, where is located operator. During the automatic control when operator is substituted by automatic machine, control circuit can be closed both through KP and out of KP. Therefore the systems of automatic remote control can be two types:

a) with the closing a circuit of regulation through KP;

b) with the closing a circuit of regulation out of KP.

We will for brevity call these systems respectively systems of the type a and the type b.

It is obvious, a system of the type a can be obtained from the remote-control system with the help of the operator by the simple replacement of blocks shield - operator - control handle (see Fig. 6-6) of the automatic control system, which converts the voltages/stresses, developed by measuring device, into the appropriate commands/crews, which modulate transmitter. All remaining units of equipment for control, arranged/located on KP and rocket, can remain constant.

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Consequently, with respect to equipment for control the automatic control system of the type a comparatively differs little from the remote-control system with the operator, and with the wish to easily carry out the universal control system, which ensures rapid switching from the control of operator to the automatic and back.

The system of automatic remote control of the type b differs from control system by operator to the considerably larger degree. The fundamental element of this control system is lead beam (radio beam, radio-path).

The operating principle of lead beam is shown in Fig. 6-8. Radio transmitter P creates the equisignal sector, along axis z_1 of which must fly the rocket R. This equisignal zone is lead beam. On the rocket is located the radio receiver, which receives the signals of transmitter. This receiver is the measuring device, which are determining value and direction of deviation ψ of rocket from the axis of equisignal sector (axis of ray/beam) in the system of coordinates $x_1y_1z_1$, connected with this zone. From the output of receiver the voltage/stress enters the autopilot, which manages the controls of rocket (Fig. 6-9). Upon the appearance of deviation ψ the autopilot turns controls, and is created the transverse acceleration, which returns rocket to axis z_1 of equisignal sector. Therefore rocket always flies along the axis of equisignal sector. Thus, rocket is retained on the axis of equisignal sector (ray/beam) automatically with the help of the equipment, established/installed on board the rocket. The control circuit, depicted in Fig. 6-9, is closed out of the control center - through the radio beam and the control mechanism.

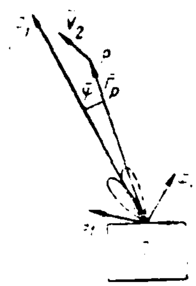

Fig. 6-8.

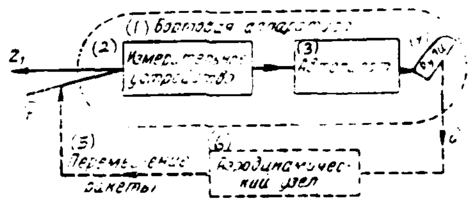

Fig. 6-9.

Fig. 6.9. Key: (1). Onboard equipment. (2). Measuring device. (3). Autopilot. (4). Controls. (5). Displacement/movement of rocket. (6). Aerodynamic unit.

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If KP and target are fixed (for example, during the rocket control of long-range), then the axis of ray/beam is directed to target and remains subsequently of fixed; rocket flies along the fixed ray/beam. In the general case when target moves over the arbitrary trajectory, the ray/beam, which manages rocket, must also be moved for the realization of the corresponding correction of missile trajectory. With the help of the lead beam it is possible to realize missile targeting both according to the method of the covering of target and according to different methods with the prevention/advance.

During guidance according to the method of the covering of target the axis of lead beam must continuously be directed to target i.e. be combined with the line of the sighting of target. Then the rocket, retained on the axis of ray/beam, will thereby be retained on the line of the sighting of target and, therefore, move over the three-point curve. The simplest and precise method of the coincidence of the axis of lead beam with the direction to the target consists in the use as this ray/beam of an equisignal sector of the radar, which works in the mode/conditions automatic tracking after the target in the direction.

Actually/really, let us assume radar of target LTs works in the mode/conditions ideally precise the automatic tracking after the target (Fig. 6-10); then at each moment of time axis z_1 of its equisignal sector is directed accurately toward the target. This equisignal sector can be used also for the rocket control, i.e., the ray/beam of radar can serve as the lead beam. Therefore, if we introduce in any manner rocket inside the ray/beam of radar, then it subsequently will be automatically retained on the axis of this ray/beam, and consequently, on the line of the sighting of target.

During missile targeting on any of the methods with the prevention/advance, the rocket at each moment of time must be found not on the line of the sighting of target $\bar{r}_c$, and in certain direction of prevention/advance $\bar{r}'_c$ (Fig. 6-11), that forms in earth-based coordinate system x_0 y_0 z_0 certain angle $\bar{\varphi}_3$. The value of this angle depends on the required lead angle β and must be computed by the special computers on the basis of the data about the motion of target and rocket. Therefore the system block diagram of control, which ensures missile targeting into lead point with the help of the lead beam, must take the form, depicted in Fig. 6.12. On the control center are arranged/located the radar of target LTs, radar of the rocket LR and computer SU.

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Radars LTs and LR put out to computer the data about the motion of target and rocket respectively. The ray/beam of radar LR is used simultaneously as the ray/beam, which manages rocket, i.e., rocket automatically is retained on the axis of the equisignal sector of this radar, z_1. On the basis of data, obtained from radars LTs and LR, computer develops angle $\bar{\varphi}_4$ and ensures installation/setting up at this angle of the axis of the antenna of radar LR.

Selecting the appropriate law of the work of computer, is possible, generally speaking, to provide guidance of rocket both according to the method PU and according to any method with the prevention/advance. General/common/total for all these methods will be the fact that with rectilinear motion of target and constant proportions V_1/V_2 the missile trajectory will be rectilinear and lead beam z_2 will be fixed. The displacements/movements of lead beam (with the ideal work of the system of control and absence of interferences/jammings) will appear only while maneuverings of target or as a result of the inconstancy of the speed of rocket. This special feature/peculiarity is the major advantage of predicted point guidance in comparison with the guidance according to the method of the covering of target.

However, predicted point guidance has a number of the very serious deficiencies/lacks which are obvious even during the surface comparison of block diagrams in Fig. 6-10 and 6-12.

Fig. 6-10.

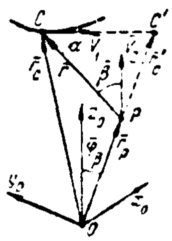

Fig. 6.11.

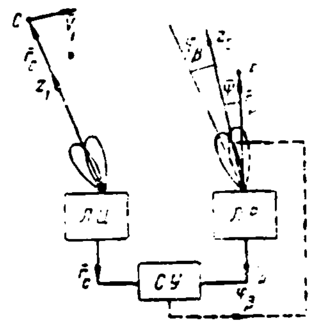

Fig. 6.12.

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Main disadvantages are:

a) considerably greater complexity of equipment for control;

b) the smaller accuracy of control.

Large complexity is caused by the fact that for predicted point guidance are necessary the further devices/equipment - radar LR and the complicated computer.

Smaller accuracy is caused by the presence of the further sources of errors. Such sources of errors include:

a) computer SU;

b) radar LR;

c) the presence of two separate rays/beams (z_1 and z_2) instead of one.

These deficiencies/lacks are so/such serious, that with the remote control of the first form fundamental proved to be the nonlead methods, and the method of the covering of the target (for example, see [9], [10], [11], [12], etc.).

One should note also that with the remote control of the first form, realized from mobile KP, for example from the mother aircraft, it is possible by guaranteeing the corresponding law of motion of KP (mother aircraft) to provide the very low path curvature of rocket, even during the rocket control according to the method of the covering of target.

6-5. Fundamental sources of the errors of remote control.

With the remote control of the second form, as during the homing, measuring device is located on the rocket (for example, television head). Therefore to the remote control of the second form are characteristic the same four basic groups of errors, that also homing, (see p. 151-157). Since, as with homing, the instrument errors have here not direct, but indirect effect on guidance miss distance, in consequence of which the vectoring error d proves to be not depending on the range of the control system.

With the remote control of the first form measuring device (goniometer, radar) is found on KP, and rocket in the process of guidance to the target continuously recedes from this measuring device. Because of this instrument errors have a direct effect on vectoring error.

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Actually/really, let the remote control of the first form follow method itself of the covering of target and due to the presence of instrument errors angle Θ between directions $\bar{r}_c$ and $\bar{r}_p$ is not equal to zero (Fig. 6-13). Then rocket flies wide of the mark at a distance.

$$d = r_c\Theta. \quad (6\text{-}1)$$

Since instrument error Θ is the value of random, then it is possible to register:

$$d_{cp} = r_c\Theta_{cp}. \quad (6\text{-}2)$$

where $d_{эф}$ and $\theta_{эф}$ - efficient (rms) values of errors d and θ.

From relationship/ratio (6-2) it follows that the vectoring error $d_{эф}$ caused by the direct action of instrument error $\theta_{эф}$ increases with an increase in distance to the target r_c, i.e. with an increase in the range of the control system, and with the long range of action can be very great.

Thus, for instance, when $\theta_{эф}=1$ so forth =3.6' and $r_c=100$ км, $d_{эф}=100$ м. This error in the majority of the cases can exceed the allowed value many times, and a reduction/descent in instrument error $\theta_{эф}$ up to the required value (for example, to 0.1 so forth) can prove to be very difficult. From this example it is evident that at the long range the fundamental source of vectoring error is the direct action of instrument error. With comparatively small ranges, on the contrary, can prove to be most essential other sources of errors, namely:

1) the vectoring errors, caused by the inertness of control, when the maneuvers of target are present,;

2) the vectoring errors, caused by the rocket limited by maneuverability, with the large curvature of the required trajectory,

caused by the selected guidance method.

The errors of the second form (due to the path curvature) with the remote control of the first form have the smaller value than during the remote control of the second form or the homing, since the guidance method most unfavorable with respect to the form of trajectory - method of linear curve - with the remote control of the first form is not applied.

Fig. 6-13.

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Thus, with the remote control of the first form by the fundamental sources of errors are:

1) the vectoring errors, caused by the inertness of control when the maneuvers of target are present ;

2) the vectoring errors, caused by the direct action of instrument errors [relationship/ratio (6-2)].

At the short distances predominate the errors of the first group, while at the long range - errors of the second group.

6-6. Comparison of remote control with the homing.

Since the comparison of the remote control of the second form with the homing was already done above, it remains to compare with the homing the remote control of the first form. The major advantages of the latter in comparison with the homing are:

1) greater range;

2) the smaller complexity of control mechanism.

For the explanation of the first advantage let us turn to Fig. 6-14, in which is shown the remote-control system of the first form (Fig. 6--14a). This system contains three channels - channel of the control of target, channels of the inspection of rocket and control channels. It is obvious, the range of system as a whole is limited by the range of the channel of the control of target, since the control of target is realized via the reception of the very weak signals, reflected from the target, and remaining two channels work by forward signals. Furthermore, the channel of the control of target is more subjected to the action of interferences/jammings from the side of the enemy, since reception/procedure occurs on the same wave, as the irradiation of target.

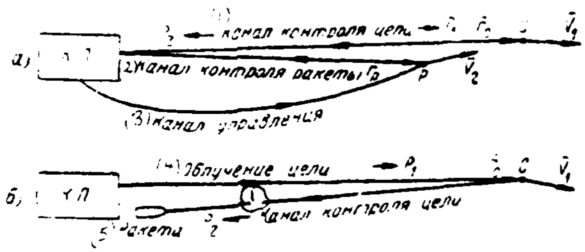

Fig. 6-14.

Key: (1). Channel of the control of target. (2). Channel of inspection of rocket. (3). Channel of control. (4). Irradiation of target. (5). Rocket.

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Fig. 6-14b shows the system of the semi-active homing: target is irradiated by the transmitter, arranged/located on KP, and the signals echo from the target are oriented by onboard receiver. (In the comparison of the remote control of the first form with the active homing there is no need, since in Chapter 5 it was shown that the range of active homing is less than the range of semi-active homing). Since there is here only one channel - channel of the control of target, then the range of this system is estimated distance of the control of target. Smallest signal strength, taken by onboard direction finder, will occur with the flight of rocket with

KP. In this case the reflected from the target energy P_2 passes to the rocket the same path r_0 as the falling/incident to the target energy P_1.

Thus, for the comparison of the attainable range of the remote-control systems of the first form and homing, depicted in Fig. 6-14, it suffices to compare the range of the channels of the inspection of the target of these systems; in this case in the case of homing rocket should be considered being located near KP.

Since the transmitters, which illuminate a target, in both systems are arranged/located on KP, then in principle they can be identical. The path of the falling/incident and reflected from the target energy in both systems also proves to be identical. Consequently, the difference in the range will be obtained only because in the remote-control system of the first form the receiver is arranged/located on KP, and in the homing system it is located on the rocket. Since for the receiver, arranged/located on KP, receiving antenna can be done with the considerably greater amplification (due to larger permissible overall sizes of reflector), and receiver itself - those more advanced than receiver on the rocket, then it is obvious that also the range of the remote-control system of the first form can be greater than the range of homing system.

The second advantage of the remote control of the first form (smaller complexity of onboard equipment) is obvious, since in this case measuring device (goniometer, radar) is placed on KP, and in the case of homing - on the rocket.

Main disadvantages in the remote control of the first form in comparison with the homing they are:

1) smaller accuracy for the long range;

2) the need for the continuous participation of KP during the control process.

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The first deficiency/lack is obvious, since with the remote control of the first form vectoring error grows with an increase in the range [see formula (6-2)], but during the homing vectoring error virtually does not depend on range. Therefore with the sufficiently long range the error of remote control is done more than the error of homing.

The second deficiency/lack is connected with the fact that with the remote control of the first form KP must realize continuous precise inspection of rocket and target during entire flight time of rocket from KP to the target. This requirement is especially unpleasant in the case of location of KP on the mother aircraft (i.e. during the control of bombs, by torpedoes and by aircraft rockets), since strongly impedes the maneuverability of mother aircraft and subjects this aircraft to the larger danger of bombardment on the part of enemy.

With active homing the KP does not participate at all during the control process of rocket flight. Therefore mother aircraft, that let out rocket, can complete any maneuvers. During the semi-active homing the only function of KP in the process of rocket flight is the irradiation of target, and displacements/movements of KP affect the accuracy of rocket control considerably less than with the remote control of the first form.

The comparison given above of the remote control of the first form and homing shows that the major advantage of the remote control of the first form is the long range of action; but exactly with the long range the accuracy of induction/guidance it can prove to be insufficient. Homing, on the contrary, it possesses that most valuable property, that its accuracy does not depend on range, but attainable range during the homing proves to be smaller than with the remote control of the first form. The permission/resolution of these contradictions, i.e., the guarantee of combination of long range with the high accuracy, can be obtained during the combined control.

Chapter Seven.

COMBINED CONTROL.

7-1. Generalities.

The combined control, i.e., the combination of several methods of control, is applied for the more complete utilization of advantages of different ones the method of control.

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Most widely used are the following combinations of the fundamental methods of the control:

1. Combination of the remote control of the first form with the homing.

2. Combination of remote control of first form with remote control of second form.

In both combinations during the first stage of rocket flight is applied the remote control of the first form, and transition/transfer to homing or remote control of the second form is produced in the second stage of the flight when rocket (bomb) sufficiently approaches the target. In both combinations indicated above is obtained the combination of the long range of action, inherent in the remote control of the first form, with the high accuracy, inherent in homing and to the remote control of the second form. This combination of the long range of action with the high accuracy of induction/guidance is the major advantage of the combined control in comparison with the simple (uncombined) control. However, during the combined control equipment for control, especially onboard, it is obtained considerably more complicated, it is more expensive and it is more by the overall sizes and the weight. Therefore the use/application of the combined control is expedient only when simple methods of control do not provide the satisfactory solution of problem.

Of two given above combinations that of more spread is the first, i.e., the combination of the remote control of the first form with the homing. This is explained by the fact that the homing received larger propagation than the remote control of the second form (see the comparison given above of these methods of control, and also literature [9], [10], [11], [12], etc.).

However, if for one reason or the other is applied the remote control of the second form, then the combination of this method of control with the remote control of the first form can prove to be very appropriate, especially if one considers that the control channel, necessary for the remote control of the second form, can be used also for the remote control of the first form. Subsequently for the larger brevity and concreteness the presentation is led in connection with the combination of remote control (first form) with the homing, but the significant part of the conclusion/output can be used also to the combination of the remote control of the first form with the remote control of the second form.

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During coupling (combination) of two methods of control it is necessary to solve the following fundamental problems:

a) coupling the trajectories, which correspond to both methods of control;

b) guidance of onboard direction finder (radar) to the target, i.e., the guarantee of "capture/grip" of target with the onboard direction finder (radar) in the direction (and range);

c) the guarantee of switching the autopilot of rocket from the remote control to the homing;

d) the combined use of units of equipment for control for the purpose of the creation of the largest possible quantity of units, utilized in both stages of control (i.e. both with the remote control and during the homing).

Equipment of the system of combined control must consist of the following basic building blocks, utilized:

1) only with the remote control;

2) only during the homing;

3) for both forms of control;

4) for guidance of onboard direction finder (radar) to the target and the switching from the remote control to the homing.

7-2. Coupling of trajectories.

It was above indicated that during the homing is applied missile targeting on the linear curve or according to the method PU, and with the remote control (first form) - induction/guidance according to the method of the covering of target or any predicted method. To each guidance method are characteristic special shape of missile trajectory and special importance of lead angle (see Chapter 3):

1) при наведении по погонной кривой $\beta = 0$;
2) при наведении по методу накрытия цели с неподвижного $K\Pi$

$$\sin\beta = \frac{r_0}{r_c}\frac{V_1}{V_2}\sin\alpha;$$

3) при наведении по методу ПУ

$$\sin\beta = \frac{V_1}{V_2}\sin\alpha. \qquad (7\text{-}1)$$

Key: 1) during the guidance on the linear curve β=0. 2) during the guidance according to the method of the covering of target from fixed KP. 3) during induction/guidance according to the method PU.

Therefore if transition/transfer from the remote control to the homing is accompanied by transition/transfer from one guidance method to another (for example, from one method of the covering of target to the next PU), then at the moment of transition/transfer will occur an abrupt change in the required lead angle β; consequently, will appear the fracture of the required (ideal) missile trajectory.

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If during the remote control and the homing is applied identical method of guidance (for example, method PU), then ideal (required) trajectory will not have fracture. However, virtually also in this case at the moment of transition/transfer for the homing can occur the considerable error of prevention/advance $\Delta\beta$ due to an inaccuracy in the remote control, i.e., due to the deviation of missile operating trajectory from the ideal. For the illustration of these positions let us consider in more detail two typical cases of the combined control.

First case. Missile targeting during the remote control and the homing is realized by an identical method - the method PU.

Second case. Remote control is produced according to the method of the covering of target (from fixed KP), and homing - according to the method PU.

First case. Missile targeting during the remote control and the homing is realized according to the method PU.

The required lead angle in the process of entire control is determined by the constant/invariable formula

$$\sin\beta = \frac{V_{ц}}{V_{р}}\sin\alpha.$$

Therefore with the ideally precise remote control at the moment of transition/transfer for the homing lead angle will be equal to the required angle and the error of prevention/advance is equal to zero:

$$\Delta\beta = 0.$$

However, with the real remote control as a result of an inaccuracy in the control occurs certain error of prevention/advance $\Delta\beta$. Since the accuracy of remote control is reduced with an increase in the range of remote-control system, then with long range $r_{p\,макс}$ the error $\Delta\beta$, which occurs at the moment of transition/transfer for the homing, can reach the significant magnitude.

In § 5-6 it is shown that the minimum required range of homing r_0 is proportional to the error of prevention/advance $\Delta\beta$, which occurs at the moment of transition/transfer for the homing:

$$r_0 \geqslant 2{,}5 \div 3\rho_0\Delta\beta.$$

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Hence it follows that the less the accuracy of remote control, the longer range of the action of homing system is required. Thus, for instance, with ρ_0=5 km and $\Delta\beta$=10° are required, which was $r_0 \geqslant 2.7$ km. With ρ_0=5 km and $\Delta\beta$=20° must be $r_0 \geqslant 5.4$ km.

Second case. Remote control is produced from fixed KP according to the method of the covering of target, and homing - according to the method PU.

It is obvious, this case is typical for the induction/guidance of surface-to-air missiles.

Let us suppose first that remote control - ideally precise. Then rocket will with the remote control move over the ideal three-point curve with the lead angle

$$\sin\beta = \frac{r_p}{r_c}\varepsilon\sin\alpha. \qquad (7\text{-}2)$$

At the moment of transition/transfer for the homing

$$\sin\beta = \frac{r_{0p}}{r_{0c}}\varepsilon\sin\alpha_0. \qquad (7\text{-}3)$$

where r_{0p}, r_{0c} and α_0 - values of values r_p, r_c and α at the moment of transition/transfer for the homing (Fig. 7-1).

The required lead angle at the moment of transition/transfer for the homing is determined by the relationship/ratio

$$\sin\beta_0 = \varepsilon\sin\alpha_0. \qquad (7\text{-}4)$$

Consequently, at the moment of transition/transfer for the homing occurs the error of the prevention/advance

$$\Delta\beta = \beta_0 - \beta. \qquad (7\text{-}5)$$

From expressions (7-3) and (7-4) we have:

$$\Delta\beta' = \sin\beta_0 - \sin\beta = \varepsilon\left(1 - \frac{r_{0p}}{r_{0c}}\right)\sin\alpha_0 = \varepsilon\frac{r_0}{r_{0c}}\sin\alpha_0,$$

i.e.

$$\Delta\beta' = \sin\beta - \sin\beta_0 = \varepsilon\frac{r_0}{r_{0c}}\sin\alpha_0. \qquad (7\text{-}6)$$

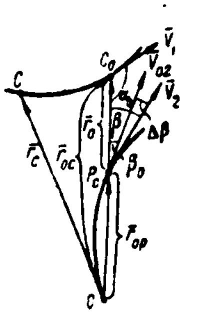

Fig. 7-1.

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Since usually $\varepsilon < \frac{1}{2}$, then $\sin\beta_0 \leq 0.5$ and $\sin\beta \leq 0.5$. Therefore in the first approximation, it is possible to assume/set:

$$\Delta\beta \approx \Delta\beta' = \varepsilon \frac{r_0}{r_{0c}} \sin\alpha_0 \qquad (7\text{-}7)$$

It is obvious, with the assumption $\varepsilon \leq \frac{1}{2}$ indicated it will be:

$$\Delta\beta < 30^\circ$$

The less is relation $\frac{r_0}{r_c}$, the less the error of prevention/advance $\Delta\beta$.

But the minimally permissible range of homing r_0 is determined from the condition

$$r_0 \approx 2.5 - 3\rho_0\Delta\beta$$

Therefore, taking small reserve, we will obtain the following condition:

$$r_0 \geq 3\rho_0\Delta\beta$$

Substituting here value $\Delta\beta$ from formula (7-7), we will obtain the condition:

$$r_{0c} \geqslant 3\xi\rho_0 \sin\alpha_0 \qquad (7\text{-}8)$$

Condition (7-8) is the necessary condition of the feasibility of transition/transfer from the three-point trajectory to the trajectory PU. With the nonfulfillment of this condition rocket will not manage to pass to the ideal trajectory PU and, therefore, it will fly wide of the mark.

Since $\xi \leqslant 1/2$ and $\sin\alpha_0 \leqslant 1$, the relationship/ratio (7-8) takes the form:

$$r_{0c} \geqslant 1.5\rho_0. \qquad (7\text{-}8')$$

With V_2=600 m/s $W_{pm} = 8g$ is obtained by ρ_0=4.5 km, and the condition of the feasibility of transition/transfer to the ideal trajectory PU takes the form:

$$r_{0c} \geqslant 7 \text{ км}.$$

i.e. the distance from KP to the target at the moment of transition/transfer for the homing must be not less than 7 km.

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Distance r_0 does not enter into formulas (7-8) and (7-8'). This means that with satisfaction of condition (7-8) the transition/transfer to the ideal trajectory PU is feasible with how convenient to the short distance of homing r_0. This result was obtained only because we took remote control ideally precise.

Let us explain now, will be such the conditions of the feasibility of transition/transfer to the ideal trajectory PU with the real remote control.

In this case the missile trajectory differs from ideal three-point curve. In Fig. 7-2 they are depicted ideal three-point curved (by dotted line) and missile operating trajectory (by solid line). For the ideal curved lead angle β is determined by formula (7-3).

For the real curve lead angle is equal to:

$$\beta' = \beta + \delta, \qquad (7\text{-}9)$$

where δ - angle between vectors of the speed of rocket and tangent to ideal three-point curve (Fig. 7-2). We will for the brevity call angle δ the error of remote control; β - lead angle of ideal three-point curve, determined by formula (7-3).

Therefore the error of prevention/advance is equal to:

$$\Delta\beta = \beta_0 - \beta' = \beta_0 - \beta - \delta.$$

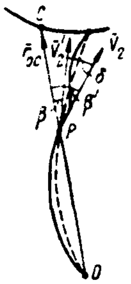

Fig. 7-2.

Fig. 7-3.

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From the comparison of expressions (7-3) and (7-4) it follows that always $\beta<\beta_0$. Therefore the most dangerous will be such case, with which the angle δ will be opposite on the angle β_0. In this case

$$\Delta\beta=\beta_0+\delta-\beta. \qquad (7-10)$$

At the moment of transition/transfer for the homing the angle δ can have the most diverse values. Only most dangerous will be such case when at the moment of transition/transfer for the homing angle δ is maximum on the value and the the opposite on the angle β_0 (Fig. 7-3).

In this case

$$\Delta\beta=\beta_0+\delta_м-\beta. \qquad (7-11)$$

Value $\delta_м$ depends on the accuracy of remote control. The rough estimate of the greatest value of error δ_M can be done as follows.

Error δ will be maximum, if before the transition/transfer for the homing rocket turned up with the smallest possible radius of curvature ρ_0 to the side, opposite to that required. Therefore the worse case can be evaluated on the basis Fig. 7-4.

In this figure ideal three-point curve is substituted by the broken circular arc of a radius ρ_0, and real curve - by continuous circular arc of the same radius. Transition/transfer for the homing occurs at the most unfavorable moment/torque - when rocket is located at point P_0. At this moment the angle δ is equal to δ_M.

The maximum distance between trajectories $d_M = \overline{ab}$ is, obviously, the maximum linear error of remote control.

The analysis Fig. 7-4 gives the following result:

$$\operatorname{tg}\frac{\delta_M}{2} = \sqrt{\frac{1}{\left(1 - \frac{d_M}{2\rho_0}\right)^2} - 1.}$$

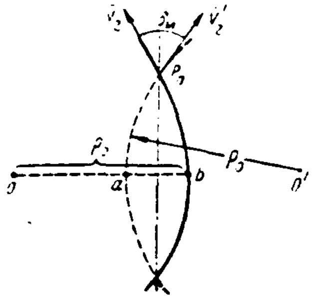

Fig. 7-4.

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Since $\frac{d_м}{2\rho_0} \ll 1$ and $\frac{\delta_м}{2} < 0.5$, that with the sufficient accuracy

$$\delta_м \approx 2\sqrt{\frac{d_м}{\rho_0}}. \qquad (7\text{-}12)$$

If $d_м \leqslant 200$ м and $\rho_0 \geqslant 5$ км,

then

$$\delta_м \leqslant 25°. \qquad (7\text{-}12')$$

It should be noted that formulas (7-12) and (7-12') give the exaggerated value of error $\delta_м$, since they are based on the assumption of the coincidence of the series/row of unfavorable factors. In the majority of the real cases one should expect that

$$\delta_м \leqslant 10 \div 15°. \qquad (7\text{-}12'')$$

From formula (7-11) we have:

$$\Delta\beta=(\beta_0-\beta)+\delta_м.$$

Above it was proved that in the first approximation, error $(\beta_0-\beta)$ is expressed by formula (7-7):

$$\beta_0-\beta\approx\varepsilon\frac{\rho_0}{r_{0c}}\sin\alpha_0,$$

therefore

$$\Delta\beta\approx\varepsilon\frac{\rho_0}{r_{0c}}\sin\alpha_0+\delta_м. \qquad (7\text{-}13)$$

Proposing as before that the minimally permissible range of homing is determined by the relationship/ratio

$$r_0\geqslant 3\rho_0\Delta\beta,$$

we obtain from (7-13) the following condition of the feasibility of transition/transfer to the ideal trajectory PU:

$$r_0\geqslant\frac{3\rho_0\delta_м}{1-3\varepsilon\frac{\rho_0}{r_{0c}}\sin\alpha_0}. \qquad (7\text{-}14)$$

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From Fig. 7-1 it is obvious that always must be

$$r_{0c}>r_0 \qquad (7\text{-}15)$$

Therefore from formulas (7-14) and (7-15) we obtain the further condition:

$$r_{0c}>3\varepsilon\rho_0\sin\alpha_0+3\delta_м\rho_0 \qquad (7\text{-}16)$$

When $\delta_м=0$ this condition coincides with condition (7-8). Thus, if with the ideal remote control is imposed requirement only for the minimum target range r_{0c} [condition (7-8)], then with the real remote control is required not only sufficiently large target range [condition (7-16)], but also the sufficiently long range of homing r_0 [condition (7-14)]. In the worse case when $\sin\alpha_0 \approx 1$, conditions (7-14) and (7-15) take the following form:

$$r_0 \geqslant \frac{3\delta_м \rho_0}{1 - 3\varepsilon \frac{\rho_0}{r_{0c}}} \qquad (7\text{-}14')$$

$$r_{0c} \geqslant 3(\varepsilon + \delta_м)\rho_0 \qquad (7\text{-}16')$$

Let us consider two examples.

Example 1. $\rho_0 = 5$ км; $\varepsilon = 0{,}5$; $\delta_м = 10^\circ = 0{,}18$

From formula (7-16') we obtain the condition:

$$r_{0c} > 10{,}2 \text{ км}.$$

We take $r_{0c} = 12$ км.

Then from formula (7-14') is obtained the condition:

$$r_0 \geqslant 7{,}2 \text{ км}.$$

With $r_{0c} = 15$ km will be obtained the lighter requirement:

$$r_0 \geqslant 5{,}4 \text{ км}.$$

Finally, with $r_{0c} = 30$ km we will obtain:

$$r_0 \geqslant 3{,}6 \text{ км}.$$

Example 2. $\rho_0 = 5$ км; $\varepsilon = 0{,}5$; $\delta_м = 20^\circ = 0{,}35$.

We obtain condition $r_{0c} > 13$ km.

With $r_{0c}=15$ km is required $r_0 \geq 10.5$ km, while with $r_{0c}=30$ km is required

$$r_0 \geq 7.2 \text{ km}.$$

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From these examples it is evident that the combination of remote control according to the method of the covering of target with the homing according to the method PU gives better results, the greater the target range r_{0c}. Satisfactory results are obtained already with $r_{0c} \geqslant 10 \div 15$ km.

Since with the smaller target ranges the use/application of the more complicated and more expensive combined control is generally little expedient, the requirement of target range 10-15 km is not excessive.

7-3. Guidance of onboard direction finder to the target.

The onboard direction finder (radar) of homing system must have an antenna with a comparatively narrow visual angle 2θ (Fig. 7-5). Contraction of visual angle gives the following advantages:

1) an increase in the range (due to the use/application of a narrower antenna radiation pattern);

2) an increase in the angular sensitivity;

3) an increase in the angular resolution;

4) an increase in the freedom from interference of homing system.

Limit to the contraction of visual angle usually place the following fundamental factors:

1. The contraction of visual angle requires increase in the overall sizes of antenna head and shortening of the working wavelength of direction finder. The use/application of waves of shorter than 2-3 cm sharply raises the dependence of the range on the meteorological conditions. The transverse overall size of antenna head did not usually exceed 0.3-0.5 mm. During such limitations of overall sizes and wavelength it is obtained:

$$2\theta_{m} \geqslant 60 \div 20^{\circ}. \qquad (7\text{-}17)$$

2. At too small a visual angle in process of homing is feasible output of target beyond limits of this visual angle while maneuvering of target, especially with low distance of target r.

3. At small visual angle target, which has finite dimensions, cannot be considered already point and at small distances r can even fall outside the limits visual angle, which worsens/impairs work of direction finder.

4. At small visual angle hinders guidance of onboard direction finder to target, i.e., lockon by visual angle of direction finder.

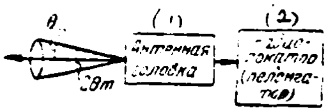

Fig. 7-5.

Key: (1). Antenna head. (2). Radar (direction finder).

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For all these reasons there is certain optimum value of visual angle, which for given above data is found within the following limits:

$$2\Theta_m \approx 10 \div 30^\circ. \qquad (7\text{-}18)$$

Consequently, for the provision for transfer from the remote control to the homing it is necessary that at the moment of the transition/transfer of rocket for the homing target would prove to be within the limits of a comparatively narrow visual angle of onboard direction finder. Is given below the short characteristic of some obvious ones the method of the solution of this problem.

Method 1. Use/application of the antenna, automatic tracking after the target, guided to the target even to the production of rocket with KP.

This method can be used, if homing begins immediately or almost immediately after missile takeoff, but it cannot be applied during the combined control. Actually/really, the fundamental sense of the combined control consists in reception/procedure by the onboard direction finder of the signals echo from the target being begun not from the moment/torque of missile takeoff, but only with a sufficient approximation/approach of rocket to a target, since in this case are raised the range and the freedom from interference of control.

Method 2. Induction/guidance of antenna head to the target prior to the start and its rigid attachment in this position on the missile body.

Prior to the missile takeoff when rocket is found even on KP, it is possible to determine with the help of the radar on KP (out of the rocket) direction to the target, to establish/install in accordance with these data the antenna head of direction finder, which is located on board the rocket, and to fasten it in this direction it is motionless relative to missile body.

The use/application of this method, obviously, possibly only when there is a confidence, what in takeoff and subsequent remote control vector $\overline{r}$ (direction rocket - target) cannot turn itself relative to missile body to the angle, greater than θ_m. However, in the majority of the cases this condition can be broken.

Let us consider for the illustration of this position the case when remote control is produced according to the method of the covering of target.

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Let us assume at the arbitrary moment of time the rocket is located at certain point P (Fig. 7-6) and velocity vector composes with the direction to the target lead angle β, moreover

$$\sin\beta = \frac{r_p}{r_c}\varepsilon\sin\alpha.$$

Axis of rocket z_2 composes with the velocity vector V_2 the angle $\bar{\delta}$, equal approximately/exemplarily to angle of attack $\bar{\alpha}_a$.

Consequently, the angle between the direction to the target and the axis of rocket is equal to:

$$\psi = \beta + \delta \approx \frac{r_p}{r_c}\varepsilon\sin\alpha + \delta.$$

In the beginning of the remote control when $r_p \ll r_c$, is obtained

$$\psi = \psi_1 \approx \delta_1,$$

while at the moment of transition/transfer for the homing when $\frac{r_p}{r_c} \approx 1$, it will be:

$$\psi = \psi_2 \approx \varepsilon\sin\alpha = \delta_2.$$

Consequently, in the process of remote control is possible a change of angle ψ between the housing and the target to value

$$\Delta\psi = \psi_2 - \psi_1 \approx \varepsilon \sin\alpha + (\delta_2 - \delta_1).$$

In the worse case

$$\Delta\psi_{макс} \approx \varepsilon + 2\alpha_{a\ макс}. \qquad (7\text{-}19)$$

where $\alpha_{a\ макс}$ - maximum angle of attack.

When

$\varepsilon = 0{,}5$ and $\alpha_{a\ макс} = 10°$ will be $\Delta\psi_{макс} = 50°$. (7-20)

Since one-half angle of view is equal to $\theta_m < 5 \div 15°$, then it is obvious that abrupt changes in angle ψ, described by formulas (7-19) and (7-20), can lead to the target fade.

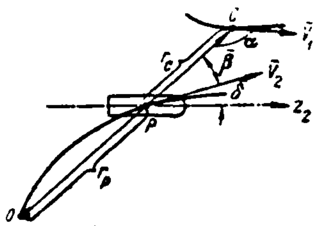

Fig. 7-6.

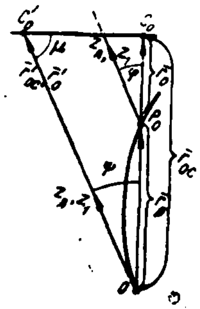

Fig. 7-7.

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Method 3. Guidance of antenna head to the target prior to the start and its stabilization in this position with the help of the gyroscopes.

This method can be applied during the use for homing the antihunting antenna. However, the use/application of this method is possible only when there is a confidence, that in takeoff (in the section of the unguided flight or flight according to the program) and subsequent remote control the line of fire $\bar{r}$ cannot be turned relative to that stabilized by the onboard gyroscopes of the coordinate system to the angle, greater than ψ_m.

Let us consider two typical examples.

Example 1. The remote control of antiaircraft missile is produced according to the method of the covering of target (Fig. 7-7).

Let us assume at the moment/torque, which preceded start, the rocket is found at point O (on KP), and target - at point C'_0. Axis z_1 of antenna head is combined with the direction to target $\bar{r}'_0$ and is stabilized in this direction.

Up to the moment/torque of transition/transfer for the homing the rocket is located at point P_0, and target - at point C_0. If we consider the stabilization of antenna ideally precise, then direction to target $\bar{r}$ is verified relative to the axis of antenna z_4 (axis of visual angle) to angle $\varphi = \varepsilon$, where ε - angle of rotation of a radius of the vector of target relative to KP - $\bar{r}_c$. Angle ε will be maximum, if target flies straight angle of 90° to $\bar{r}_{0c}$.

In this case $\sin \varphi_{max} = \frac{\overline{C'_0 C_0}}{\overline{OC_0}} = \frac{V_c \Delta t}{r_0 - r_0}$,

where Δt - flight time of target and rocket from the moment/torque of missile takeoff to the transition/transfer for the homing.

Obviously,

$$r_p = V_2 \Delta t,$$

where V_2 - the average speed of rocket to point P_0.

Therefore

$$\sin \psi_{макс} = \frac{\varepsilon}{1 + \frac{r_0}{r_p}};$$

since $\varepsilon \leqslant 0{,}5$, then

$$\psi_{макс} \approx \sin \psi_{макс} = \frac{\varepsilon}{1 + \frac{r_0}{r_p}}.$$

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Until now, we counted the stabilization of ideal. In actuality for the time remote control Δt the stabilized z_1 axis, and also, therefore, axis z_A can turn itself to certain angle $\Delta_{гир}$. Therefore a maximally possible angle of rotation of direction to the target relative to the axis of antenna will be equal to:

$$\varphi_{макс} = \psi_{макс} + \Delta_{гир} = \frac{\varepsilon}{1 + \frac{r_0}{r_p}} + \Delta_{гир} \qquad (7\text{-}21)$$

where r_p - range of remote control;

r_0 - range of homing.

Since for the surface-to-air missile the time of remote control Δt does not exceed usually one or several minutes, then without the special labor/work it is possible to obtain:

$$\Delta_{сно} \leqslant 1 \div 2^{\circ}. \quad (7\text{-}22)$$

Therefore

$$\varphi_{макс} \approx \frac{\varepsilon}{1 + \frac{r_0}{r_p}}. \quad (7\text{-}21')$$

When $\varepsilon_{макс} = 0{,}5$ and $\frac{r_0}{r_p} = 0{,}2 \div 0{,}5$ it is obtained:

$$\varphi_{макс} = 20 \div 25^{\circ}. \quad (7\text{-}23)$$

These relationships/ratios show that this method of the guidance of antenna to the target gives the best results, than during the installation/setting up of antenna on the housing; however, in this case is required too great a visual angle of the onboard direction finder:

$$\theta_m \geqslant 25^{\circ}.$$

Example 2. Remote control by a surface-to-air missile is realized according to the method PU (Fig. 7-8).

During the ideal induction/guidance according to the method PU the line of fire $\bar{r}$ retains in the space constant/invariable direction. However, in actuality from the moment/torque of missile takeoff (point P'_0) to the moment/torque of transition/transfer for the homing (point P_0) line of fire can turn itself to certain angle $\varphi_1+\varphi_2$. Rotation to the angle φ_1 occurs in the unguided phase of flight of rocket $P'_0P''_0$, while rotation φ_2 occurs due to an inaccuracy in remote control and presence of the transient process, caused by the presence of certain error of prevention/advance $\Delta\beta$ at the moment of the beginning of remote control. Angle φ_1 is small, since the unguided flight lasts usually a total of several seconds. Furthermore, this angle can be approximately calculated prior to the start and the axis of antenna z_A can be stabilized not in direction $\bar{r}'_0$, but in the calculated predicted direction φ_1.

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Therefore the effect of angle φ_1 can be disregarded/neglected. By the effect of inaccuracy (drift) in stabilized axis z_A also it is possible in the first approximation, to disregard [see relationship/ratio (7-22)]. Consequently, the fundamental source of the rotation of line of fire relative to the stabilized axis of antenna z_A (axis of visual angle) will be angle φ_2. The value of this angle is less, the less the error of prevention/advance $\Delta\beta$ at the moment of the beginning of remote control, i.e., the more precise the rocket it will be derived to the trajectory required for the remote control and the more precise will occur very process of remote control, i.e., the less there will be the value of the derivative $|d\varphi/dt.|$

So that in the process of remote control the target not fall outside the limits visual angle, must be satisfied the condition:

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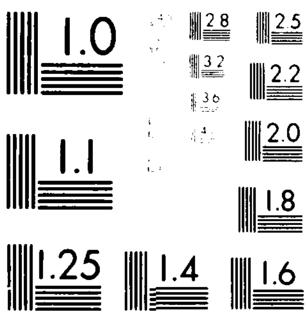

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$$\varepsilon_2 - \int_0^t \frac{d\varepsilon}{dt} \cdot dt \leq 2\Theta_m. \quad (7\text{-}24)$$

Method 4. Variable visual angle.

From the examination of the methods given above it is clear that it is possible without the special difficulties to ensure the incidence/impingement of target within the limits of wide visual angle $2\Theta_m' \approx 80 \div 100°$.

At the same time for obtaining the satisfactory quality of homing is necessary considerably narrower visual angle $2\Theta_m \approx 10 \div 30°$.

At first glance seems appropriate the use/application of a variable visual angle: in the process of remote control and to the lockon to have wide visual angle $2\Theta_m'$, and after lockon to narrow this visual angle to $2\Theta_m$.

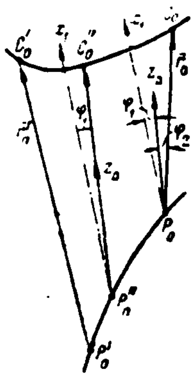

Fig. 7-8.

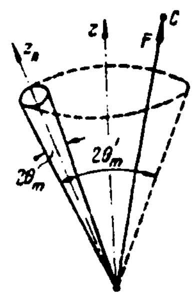

Fig. 7-9.

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However, this method is little suitable, since precisely before the moment/torque of transition/transfer for the homing when target range is maximum, it is very important to have small visual angle for guaranteeing of a long range, resolutions and freedom from interference.

Method 5. Use/application of an onboard antenna with the automatic target search (Fig. 7-9).

It is possible to provide the capture of target C, which is located within the limits of cone with the wide visual angle $2\theta_m'$, via the automatic scanning of this space with the antenna head, which has narrow visual angle $2\theta_m$. However, this method has a number of the serious deficiencies/lacks:

a) The need for the survey of wide visual angle worsens/impairs freedom from interference and angular resolution of the control system.

b) Target search must be completed very rapidly (for several seconds), since otherwise rocket for the retrieval time uselessly will fly large distance.

Realization is sufficient reliable automatic rapid search and rapid cessation of search by the onboard direction finder which must be simple, cheap and interference-free, it is complex problem.

Method 6. Remote/distance focusing/induction of antenna.

With this method the instruments, established/ins[illegible] on KP, they automatically determine, in what direction must be directed the axis of onboard antenna, and transmit the required angular coordinates onboard the rocket by the radio. Onboard instruments receive these coordinates and is set to them onboard antenna. It is obvious, for guaranteeing this remote/distance installation/setting up it is necessary to have on KP and board of rocket the matched coordinate systems. The agreement of coordinates can be achieved/reached by use/application on the rocket of the coordinate system, stabilized by gyroscopes.

The comparison of the given above six methods of the guidance of onboard antenna to the target during the combined rocket control they show that each of these methods has serious deficiencies/lacks. Therefore the development of the simple and reliable guidance system of onboard antenna to the target is one of the fundamental problems of the combined control.

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Chapter Eight.

AGREEMENT OF THE COORDINATE SYSTEM.

8-1. Formation/education of commands/crews with the Cartesian and polar forms of steering.

Fig. 8-1 depicts the block diagram, which illustrates the general/common/total principle of steering of rocket, identical for the different methods of control (homing, remote control, the combined control, the control, realized by an operator, automatic control).

This principle consists of the following.

Measuring device (IU) measures the three-dimensional/space disturbance/perturbation $\bar{\Theta}$ the characteristic deviation of rocket from the correct flight of the target. This disturbance/perturbation $\bar{\Theta}$ is measured in certain system of coordinates $x_1y_1z_1$, connected with the measuring device and the called subsequently measuring coordinate system.

Let Θ_x and Θ_y - two the mutually perpendicular components of angle $\bar{\Theta}$($\bar{\Theta}$, $\bar{\Theta}_x$ and $\bar{\Theta}_y$ correspond to angles $\bar{\varphi}$, $\bar{\varphi}_x$ and $\bar{\varphi}_y$ in Fig. 5-21, but in this chapter for us it is more convenient to designate angle by letter Θ, but not φ).

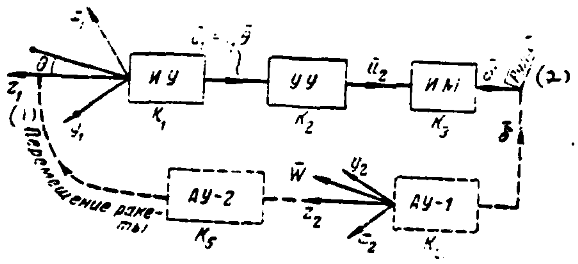

Fig. 8-1.

Key: (1). Displacement/movement of rocket. (2). Controls.

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The results of measuring the components Θ_x and Θ_y are equal to u_{1x} and u_{1y} respectively, where

$$\left.\begin{aligned} u_{1x}(t) &= K_{1x}(p)\Theta_x(t); \\ u_{1y}(t) &= K_{1y}(p)\Theta_y(t); \end{aligned}\right\} \quad (8\text{-}1)$$

$K_{1x}(p)$ and $K_{1y}(p)$ - operational transmission factors of the channels of measurement Θ_x and $\dot{\Theta}_y$ respectively (p=d/dt - differential operator).

In the case of identity of both the channels

$$K_{1y}(p)=K_{1x}(p)=K_1(p).$$

Here and in the following presentation we propose for simplicity that for low deviations Θ_x and Θ_y the system in Fig. 8-1 is linear and, therefore, all conversions of values Θ_x and Θ_y are linear. With the operations with such conversions it is convenient, as is known, to use matrices/dies. Expression (8-1) in the matrix form can be registered as follows:

$$\bar{u}_1 = K_1 \bar{\Theta},$$

where K_1 - diagonal matrix/die of the form:

$$K_1 = \begin{Vmatrix} K_{1x}(p), & 0 \\ 0, & K_{1y}(p) \end{Vmatrix} \quad (8\text{-}1')$$

We will for the brevity call value K_1 the matrix transmission factor of measuring device.

Voltage/stress $\bar{u}_1$ enters further into the control system UU, which converts the result of measurement into the command/crew for the actuating mechanism IM (control actuator):

$$\bar{u}_2 = K_2 \bar{u}_1, \quad (8\text{-}2)$$

where K_2 - matrix transmission factor of the control system.

Actuating mechanism converts command/crew $\bar{u}_2$ into the deflection of the controls:

$$\bar{\delta} = K_3 \bar{u}_2, \quad (8\text{-}3)$$

where K_3 - matrix transmission factor of actuating mechanism.

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The deflection of controls communicates to rocket by means of the aerodynamic medium the transverse acceleration

$$\overline{W}=K_4\overline{\delta} \qquad (8\text{-}4)$$

where K_4 - matrix transmission factor of aerodynamic unit (AU-1), equal to ratio $\overline{W}/\overline{\delta}$.

Acceleration $\overline{W}$ creates the displacement/movement of rocket in the space.

The connection/communication between this displacement/movement and acceleration is represented in Fig. 8-1 by the aerodynamic unit AU-2.

Acceleration $\overline{W}$ bends missile trajectory in this direction so as to decrease to the minimum the disturbance/perturbation $\overline{\Theta}$. (in the ideal system of control $\Theta \rightarrow 0$).

Thus, the control system is the actually locked servo system, which ensures the tracking of rocket the target. (In Fig. 8-1 for simplicity it is shown only external feedback loop, and internal circuits of electromechanical and aerodynamic feedback are omitted).

Acceleration $\overline{W}$, created by controls, is mastered in certain system of coordinates $x_2y_2z_2$, connected with the axis of rocket and the called subsequently executive coordinate system.

From Fig. 8-1 it follows that between the G-vector $\overline{W}$ and the vector of disturbance/perturbation $\overline{\Theta}$ there is a following connection/communication:

$$\left.\begin{array}{r}\overline{W}=Q\overline{\Theta},\\ Q=K_1K_2K_3K_4\end{array}\right\}. \qquad (8\text{-}5)$$

where

Since we assumed that matrices/dies K_1, K_2, K_3 and so forth are diagonal, then matrix/die Q will be also diagonal, and relationships/ratios (8-5) in the expanded/scanned form will be registered as follows:

$$\left.\begin{array}{l}W_x=Q_1(p)\Theta_x;\\ W_y=Q_2(p)\Theta_y;\end{array}\right\} \qquad (8\text{-}6)$$

where

$$\left.\begin{array}{l}Q_1(p)=K_{1x}(p)K_{2x}(p)K_{3x}(p)K_{4x}(p);\\ Q_2(p)=K_{1y}(p)K_{2y}(p)K_{3y}(p)K_{4y}(p).\end{array}\right\}$$

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From equations (8-6) it follows that W_x does not depend on Θ_y, and W_y does not depend on Θ_x. This complete mutual independence (decoupling) of two control channels actually/really occurs only in the absence of the torsion of the system of coordinates $x_2y_2z_2$ relative to system $x_1y_1z_1$ (see below) and absence of the specially introduced or spurious coupling between the channels. However, in the general case instead of relationships/ratios (8-6) occur the following relationships/ratios:

$$\left.\begin{aligned} W_x &= Q_{11}(p)\Theta_x + Q_{12}(p)\Theta_y; \\ W_y &= Q_{21}(p)\Theta_x + Q_{22}(p)\Theta_y. \end{aligned}\right\} \quad (8\text{-}6'')$$

These relationships/ratios in the matrix form also can be expressed by formulas (8-5), but in this case matrix/die Q and all or some of the matrices/dies K_n will not be already diagonal.

In § 1-8 it was indicated that during the Cartesian steering vector $\bar{W}$ is formed from components $\bar{W}_x$ and $\bar{W}_y$, created by rudders and height/altitude respectively (see Fig. 1-7). In the first approximation, it is possible to consider vectors $\bar{W}_x$ and $\bar{W}_y$ as those coinciding with the transverse axes of rocket (in view of the smallness of angle δ between the velocity vector of rocket and its longitudinal axis).

Therefore during the Cartesian steering axes $x_2y_2z_2$ of the executive coordinate system are rigidly connected with the missile body and are arranged/located as follows (Fig. 8-2).

Axis z_2 coincides with the axis of rocket.

Axes x_2 and y_2 coincide with the transverse axes of rocket.

During the polar steering vector $\bar{W}$ is created by the elevators and bank (see § 1-8).

Aileron causes the rapid rotations of missile body around its longitudinal axis, which reach to 90° (during the control with the economy of bank) or even exceeding 90° (during the simple control). Consequently, during the polar control it is inexpedient to choose as the executive system of coordinates $x_2y_2z_2$ the system whose all axes are rigidly connected with the missile body.

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During the polar steering of axis $x_2y_2z_2$ it is possible to choose as follows (Fig. 8-3): axis z_2 coincides with the axis of rocket; axis x_2 is placed in the plane, from which in this control system is counted off the attitude of roll of rocket φ_κ. If for example, the equipment, established/installed on the rocket, counts off roll attitude from the horizontal plane, then axis x_2 must be directed horizontally. However, in the majority of the control systems the plane, from which is counted off the roll attitude, in the general case is not horizontal; the position of this plane depends on the selected method of the stabilization of the onboard coordinate system, intended for measuring the attitude of roll of rocket φ_κ. Therefore in the general case by axis x_2 should be understood that of the axes of the onboard coordinate system stabilized by gyroscopes, from which is counted off the attitude of roll of rocket φ_κ.

Thus, with both methods of steering (Cartesian and polar) disturbance/perturbation $\overline{\Theta}$ is measured in the measuring system of coordinates $x_1y_1z_1$, and the final adjustment (use) of commands/crews is produced in the executive system of coordinates $x_2y_2z_2$.

Let us consider the special features/peculiarities of the formation/education of commands/crews during the Cartesian and polar steering.

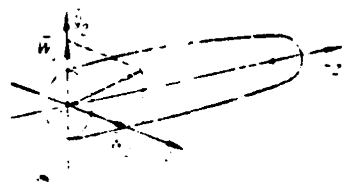

Fig. 8-2.

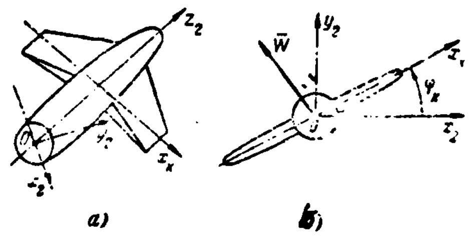

Fig. 8-3.

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During the Cartesian steering vector equation (8-5) falls into two given above scalar equations (8-6):

$$\left.\begin{aligned} W_x &= Q_1(p)\Theta_x; \\ W_y &= Q_2(p)\Theta_y. \end{aligned}\right\}$$

[For simplification in the analysis we we will proceed from the relationships/ratios (8-6), but not of the more general/more common/more total relationships/ratios (8-6")].

During the symmetrical Cartesian control (see § 1-8)

$$Q_2(p) = Q_1(p) = Q(p). \qquad (8\text{-}7)$$

However, in the general case there can be $Q_2(p) \neq Q_1(p)$.

From formula (8-6) it follows that during the Cartesian steering the disturbance/perturbation $\bar{\Theta}$ must be measured also in the Cartesian coordinates, and general/common/total block diagram (Fig. 8-1) consists of two channels: the channel of rudder and elevator channel (Fig. 8-4).

During the polar steering the formation/education of commands/crews can be produced by one of the following two methods.

Method 1. It is based on the measurement of disturbance/perturbation $\bar{\Theta}$ in the polar coordinates.

The block diagram, which corresponds to this method, is given in Fig. 8-5.

Disturbance/perturbation $\bar{\Theta}$ is measured in the polar coordinates. This means that the error signal $\bar{u}_1$ is put out in the form of the sine voltage

$$u_1(t) = u_{m\Theta} \sin(\omega t + \varphi_\Theta), \qquad (8\text{-}8)$$

amplitude of which $u_{m\Theta}$ is proportional to the value of disturbance/perturbation Θ, and phase φ_Θ characterizes the direction (vectorial angle) of vector $\bar{\Theta}$ in the system of coordinates $x_1 y_1 z_1$.

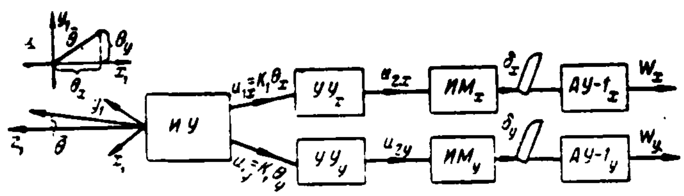

Fig. 8-4.

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The control system UU realizes above voltage/stress $\bar{u}_1$ the vector operation $K_2(p)$ (vector amplification, vector differentiation, etc.), converting voltage/stress $u_1(t)$ into certain voltage/stress

$$u_2(t)=u_{m\rho}\sin(\omega t+\varphi_p), \qquad (8\text{-}9)$$

amplitude of which $u_{m\rho}$ it is used for forming the command/crew to the elevator, and phase φ_p - for forming the command/crew to aileron:

$$\left.\begin{aligned} u_\gamma &= K_a u_{m\rho}, \\ u_\varphi &= K_b(\varphi_p-\varphi_\kappa). \end{aligned}\right\} \qquad (8\text{-}10)$$

where φ_p - required attitude of roll of rocket; φ_κ - true angle of bank, measured by onboard gyrosensor; K_a and K_b - constant coefficients.

Isolation/liberation from vector $\bar{u}_2$ of amplitude $u_{m\rho}$ and phase difference $(\varphi_p-\varphi_\kappa)$ is realized in the unit of the distribution of commands/crews RK (isolation/liberation of amplitude can be realized by a usual amplitude detector, and the determination of a phase difference $(\varphi_p - \varphi_k)$ - by the phase discriminator.

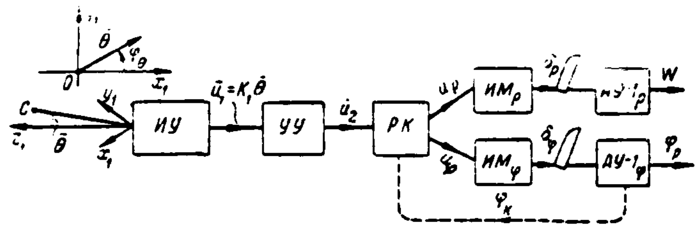

Fig. 8-5.

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Method 2. It is based on the measurement of disturbance/perturbation $\bar{\Theta}$ in the Cartesian coordinates.

The block diagram, which corresponds to this method, is given in Fig. 8-6.

Voltages/stresses u_{1x} and u_{1y} on the output of measuring device correspond to the Cartesian coordinates of disturbance/perturbation (Θ_x and Θ_y) and take the form of the DC voltages, which vary into the stroke/cycle with changes in components Θ_x and Θ_y.

Voltages/stresses u_{1x} and u_{1y} enter further into control systems YY_x and YY_y, accomplishing above these voltages/stresses scalar operations $K_{2x}(p)$ and $K_{2y}(p)$ (amplification, differentiation, integration, etc.) and converting them into the DC voltages u_{2x} and u_{2y}.

Coordinate converter (PK) converts Cartesian coordinates into the polar ones, i.e., forms from the DC voltages u_{2x} and u_{2y} the alternating voltage

where

$$\left.\begin{aligned} u_2(t) &= u_{m\varphi}\sin(\omega t+\varphi_p), \\ u_{m\varphi} &= \sqrt{u_{2x}^2+u_{2y}^2};\ \operatorname{tg}\varphi_p=\frac{u_{2y}}{u_{2x}}. \end{aligned}\right\} \quad (8\text{-}11)$$

So expression (8-11) coincides with expression (8-9), then further part of the block diagram in Fig. 8-6 coincides with the counterpart of the block diagram in Fig. 8-5.

The transformation of Cartesian coordinates into the polar ones can be realized, for example, with the help of the coordinate converter PK whose block diagram is depicted in Fig. 8-7a.

DC voltages u_x and u_y, corresponding to Cartesian coordinates θ_x and θ_y, are supplied to the input of the balanced modulators BM_1 and BM_2. To the same balanced modulators proceed the voltages/stresses of low frequency ω: $u_0 \cos \omega t$ and $u_0 \sin \omega t$ respectively.

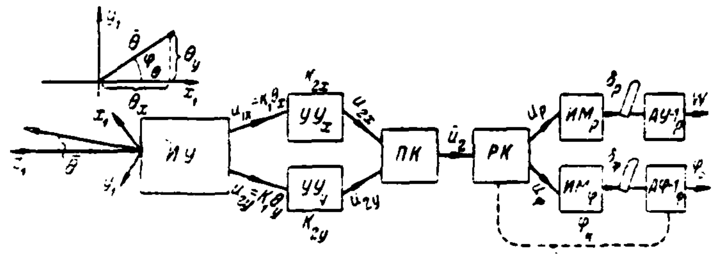

Fig. 8-6.

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The balanced modulators BM_1 and BM_2 develop voltages/stresses $K_1 u_x \cos\omega t$ and $K_1 u_y \sin\omega t$ respectively. In the summing cascade/stage SK these voltages/stresses store/add up, forming voltages/stresses u_2:

$$u_2 = K_1 u_x \cos\omega t + K_1 u_y \sin\omega t = K_1 u_{m0} \sin(\omega t + \varphi_0),$$

where

$$u_{m0} = \sqrt{u_x^2 + u_y^2};\ \mathrm{tg}\,\varphi_0 = \frac{u_x}{u_y}.$$

Schematic diagram of one of the balanced modulators (BM_1) is shown in Fig. 8-7b.

Comparison of both methods of polar steering leads to the following conclusions:

1. A deficiency/lack in the second method is the need for having

two control systems instead of one and applying transformation of coordinates.

2. Advantage of second method lies in the fact that "scalar" control systems yy_x and yy_y (Fig. 8-6) can be considerably simpler and more stable in work, than "vector" control system UU (Fig. 8-5), intended for realization of vector operation $K_2(p)\overline{u}_1$. This is advantage is more considerable, the more complicated the form of the operator function $K_2(p)$.

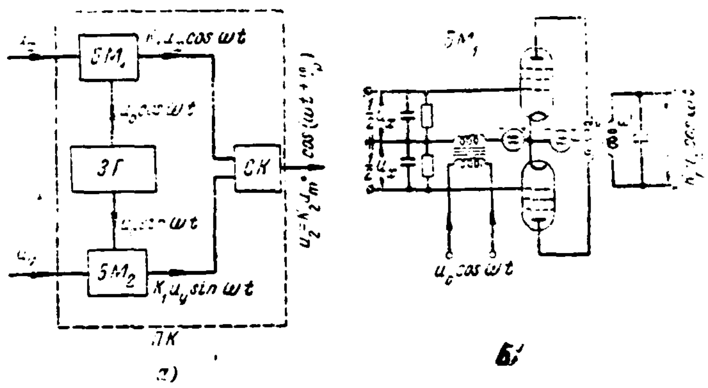

Fig. 8-7.

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3. From paragraphs 1 and 2 it follows that with simple form of required operation $K_2(p)$ can prove to be more convenient first method of forming commands/crews, and with complex form - second method.

8-2. Agreement of the systems of coordinates $x_1y_1z_1$ and $x_2y_2z_2$.

Let us assume first that the axes of the executive system of coordinates $x_2y_2z_2$ are parallel to the appropriate axes of the measuring system of coordinates $x_1y_1z_1$ (Fig. 8-8).

The deviation of rocket from the correct flight of the target $\bar{\Theta}$, measured in system $x_1y_1z_1$, is characterized by polar coordinates Θ

and φ or Cartesian coordinates Θ_x and Θ_y.

Disturbance/perturbation $\bar{\Theta}$ causes the deflection of controls, which creates acceleration $\bar{W}$, which is characterized in the polar coordinates by modulus/module W and vectorial angle φ_p, and in the Cartesian coordinates - by components W_x and W_y.

According to the equations of control (8-6) the components of acceleration, W_x and W_y are connected with the components of disturbance/perturbation, Θ_x and Θ_y, with the following equations:

$$\left.\begin{aligned} W_x &= Q_1(p)\Theta_x; \\ W_y &= Q_2(p)\Theta_y. \end{aligned}\right\} \qquad (8\text{-}6')$$

During the polar steering respectively are mastered values

$$\left.\begin{aligned} &W = \sqrt{W_y^2 + W_y^2} \\ &(1)\text{ и } \varphi_p, \\ &(2)\text{ где } \operatorname{tg}\varphi_p = \frac{W_y}{W_x}. \end{aligned}\right\} \qquad (8\text{-}12)$$

Key: (1). and. (2). where.

If transmission factors $Q_1(p)$ and $Q_2(p)$ are selected correctly, then under the effect of the steering commands (8-6') or (8-12) is created such acceleration $\bar{W}$, which returns rocket to the required trajectory (i.e. reduces disturbance/perturbation $\bar{\Theta}$) on the shortest path and in the minimum time. (In actuality it is not always possible to ensure ideal control one only of selection of the linear operators $Q_1(p)$ and $Q_2(p)$, but this fact does not deprive the validity of the presented in this paragraph qualitative analysis of the effect of the torsion of the coordinate systems).

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Let us assume now that in the process of flight the system of coordinates $x_2y_2x_2$ turned itself relative to the system of coordinates $x_1y_1z_1$ so that axis z_2 remained parallel to axis z_1, and axes x_2 and y_2 turned themselves relative to axes x_1 and y_1 to certain angle τ and took position x_2, y_2 (Fig. 8-9).

Then, if will not be taken any special measures, controls create instead of the required acceleration $\overline{W}$ acceleration $\overline{W}'$, bored relative to the required direction also to the angle τ.

For the illustration of this position let us consider the case when torsion angle τ is caused by rotation in the space only of the executive system of coordinates (system $x_1y_1z_1$ is not turned). Since the system of coordinates $x_1y_1z_1$ is not turned, then disturbances/perturbations Θ_x and Θ_y remained constant. Functions $Q_1(p)$ and $Q_2(p)$ also were not changed. Consequently, the components of acceleration W_x and W_y will also remain constant [see formulas

(8-5')] in the value, but they will be now depleted not along axes x_2 and y_2, but along axes x_2' and y_2'. Hence it follows that acceleration $\overline{W}'$ will be also arranged/located relative to axes x_2', y_2', as acceleration $\overline{W}$ is arranged/located relative to axes x_2, y_2.

Since axes x_2', y_2' are turned relative to axes x_2, y_2 to the angle τ, then acceleration $\overline{W}'$ will prove to be turned relative to acceleration $\overline{W}$ to the angle τ. It is not difficult to ascertain that the same result will be obtained also when torsion τ is caused by the rotation of system $x_1y_1z_1$ in the absence of the rotations of system $x_2y_2z_2$ and in the general case when angle τ is caused by rotations of both coordinate systems.

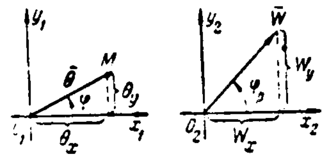

Fig. 8-8.

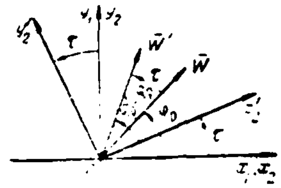

Fig. 8-9.

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Thus, the torsion of the system of coordinates $x_2y_2z_2$ relative to system of coordinates $x_1y_1z_1$ to the angle τ leads to the fact that acceleration $\bar{W}'$, created by controls, differs from the required direction in the terrestrial space $\bar{W}$ to the same angle τ ($W'=W$).

Fig. 8-10 depicts four characteristic cases.

Case 1. $\tau<<90°$. In this case torsion will bring only to the fact that rocket will be returned to required trajectory on somewhat extended path and somewhat larger period, than with $\tau=0$.

Case of 2. $\tau=180°$. In this case rocket acquires acceleration in direction, opposite that required, and therefore it will bend its trajectory in direction, directly opposite to that required.

It is obvious that in this case the control will be completely broken.

Case of 3. $90° < \tau < 180°$ (Fig. 8-10c). In this case rocket acquires greatest acceleration in direction, opposite to that required, and is obtained no acceleration in required direction. it is obvious, this torsion τ also completely breaks rocket control.

Case of 4. $45° < \tau < 90°$ (Fig. 8-10d). In this case fundamental component of acceleration $\overline{W}'$ is directed perpendicularly to required direction and only smaller component of this acceleration (W_Y') functions in required direction. It is obvious, this torsion will lead to a sharp deterioration in the quality of control, and in the unfavorable cases it is possible to bring also to the total loss of the stability of control. Actually/really,, until now, we assumed that the torsion is the sole reason, which deviates the direction of transverse acceleration from that required.

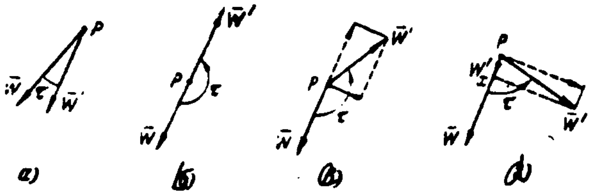

Fig. 8-10.

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In actuality, even with $\tau=0$, G-vector $\overline{W}'$ can at each given moment of time differ from that required in the value and the direction due to the inertness of the system of control, erroneousness of the measurement of disturbance/perturbation $\overline{\Theta}$ and inaccuracy in the formation/education of commands/crews. Therefore in the real control system torsion even on 50-70° can lead to the total loss of the stability of control.

The analysis given above makes it possible to make the following conclusions about the effect of the torsion of the coordinate systems on the quality of the control:

1. Small torsion angles τ only a little increase the inertness of control and worsen/impair the quality of control. Large torsion

angles sharply worsen/impair the quality of control or they lead even to the total loss of the stability of control.

2. Permissible value of torsion angle τ depends first of all on inertness of control system: the greater inertness of control system and the less factor of its stability, the less permissible torsion angle τ. Precise values of value τ can be determined only as a result of experimental research of the action of closed-cycle control system in flight or, in the first approximation,, on the model of the control system.

The approximate qualitative analysis, given above shows that, on one hand, the torsion angles, which exceed 45°, are knowingly not admitted, but on the other hand - the torsion angles, which do not exceed 5-10°, are not knowingly dangerous. Therefore with the rough estimate it can assume/set

$$\tau_{доп} \leq 10^\circ. \qquad (8\text{-}13)$$

8-3. Origin of the torsion of the coordinate systems.

In the general case all axes of the executive system of coordinates $x_2y_2z_2$ can prove to be those turned relative to the appropriate axes of the measuring system of coordinates $x_1y_1z_1$.

Let us explain first the reasons, which call misalignment z_2 and

z_1, i.e., the appearance of angle δ_{12} between the axes (Fig. 8-11). We will subsequently call this angle the angle of the fracture of axes. Disregarding, as earlier, the angle between the axis of rocket and the velocity vector of rocket $\overline{V}_2$, it is possible to consider that axis z_2 coincides with the velocity vector $\overline{V}_2$.

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The direction of axis z_1 in the majority to the control system approximately/exemplarily or coincide precisely with the direction rocket - target, $\overline{r}$.

Actually/really, with homing and remote control of the second form the measuring coordinate system is arranged on the rocket, and for the measurement is used the energy, which goes from the target to the rocket in the direction $\overline{r}$. Therefore most convenient it proves to be to combine axis z_1 with direction $\overline{r}$.

With the remote control of the first form according to the method of covering of target the rocket is retained in the direction KP - target, $\overline{r_c}$ - (Fig. 8-12). Therefore with the error, which does not exceed several degrees, it is possible to consider direction $\overline{r}$ as that coinciding with direction $\overline{r_c}$.

Measuring device in this case is placed on KP (point 0 in Fig. 8-12), and measurement is conducted with the help of the energy, which goes from the target and the rocket in the direction of KP. Therefore axis z_1 most is most conveniently combined with direction $\bar{r}$. Consequently, and in this case it is possible to consider that axis z_1 approximately/exemplarily coincides with direction $\bar{r}$.

Thus, approximately it is possible to consider that axis z_1 coincides with vector $\bar{r}$, and axis z_2 - with the vector $\bar{V}_2$. But the angle between vectors $\bar{V}_2$ and $\bar{r}$ is a lead angle β. Therefore it is possible to assume/set:

$$\varphi_{12} \approx \beta = (\bar{r}, \bar{V}_2). \qquad (8\text{-}14)$$

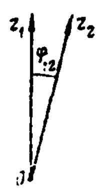
In chapter 3 it was shown that the lead angle is determined by the following relationships/ratios:

1. During guidance on the linear curve

$$\beta = 0. \qquad (8\text{-}15a)$$

2. During guidance according to method PU

$$\sin \beta = \frac{V_1}{V_2} \sin \alpha. \qquad (8\text{-}15b)$$



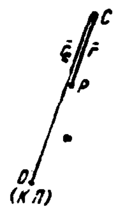


Fig. 8-11. Fig. 8-12.

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3. During guidance according to method of covering of target:

a) in the case of fixed KP

$$\sin\beta = \frac{r_2}{r} \cdot \frac{V_1}{V_2} \sin\alpha; \qquad (8\text{-}15c)$$

b) in the case of KP, which moves with a speed of $\bar{V}_K$, and slow target $(V_1 \ll V_K)$ is obtained

$$\sin\beta = \frac{r_K}{r} \cdot \frac{V_K}{V_2} \sin\alpha_K. \qquad (8\text{-}15d)$$

In these formulas

$$\alpha = (\widehat{\bar{r}, \bar{V}_1}); \quad \alpha_K = (\widehat{\bar{r}, \bar{V}_K});$$

r_2 - distance KP - target; V_1 - target speed.

Since usually $V_1/V_2 \leqslant 0.5$ and [illegible] 0.5, that from formulas (8-15a)-(8-15d) is obtained

[illegible] (8-15e)

From formulas (8-14) and (8-15) it follows that during guidance on linear curve the fracture of axes φ_{12} is approximately/exemplarily equal to zero, but with other methods of guidance it does not exceed 30-40°:

$$\varphi_{12} \leq (30-40^{\circ}). \qquad (8\text{-}16)$$

Let us consider now the rotation of axes x_2, y_2 relative to axes x_1, y_1. In this case we will for simplicity disregard the fracture of axes z_1 and z_2, i.e., consider axes z_1 and z_2 parallel. Then it is obvious that before the production of rocket it is possible to establish/install axes x_2 and y_2 by the parallel to axes x_1 and y_1 respectively, i.e., to make a torsion angle τ equal to zero. However, subsequently in the process of the guidance of rocket to the target can in the general case appear certain torsion angle τ due to the dissimilarity of the laws of the rotation of the systems of coordinates $x_2y_2z_2$ and $x_1y_1z_1$.

The smallest torsion angle will occur during the homing (or remote control of the second form) according to the method of linear curve. Actually/really, in this case the fracture of axes z_1 and z_2 is absent during entire flight time of rocket. Furthermore, in this case the systems of coordinates $x_1y_1z_1$ and $x_2y_2z_2$ are located in one place - on the rocket.

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Because of the presence of these properties it is possible to ensure the immobility of system $x_1y_1z_1$ relative to system $x_2y_2z_2$, and consequently, the absence of torsion. During the homing (or TU-2) according to the method PU the fracture of axes in principle cannot be made equal to zero [see formulas (8-11) and (8-15b)]. Therefore in the process of rocket flight system $x_2y_2z_2$ will be turned relative to system $x_1y_1z_1$ (in this case system $x_1y_1z_1$ is stabilized in the space by gyroscopes) and in the general case can occur certain torsion τ.

The greatest values torsion angle can reach with the remote control of the first form. Therefore this case is taken apart below in more detail.

As an example let us consider remote control with the help of the lead beam.

This method was described above (see Chapter 6) and it consists in the fact that the rocket automatically is retained near the axis of the controlling ray/beam, created, for example, with the rotation of radiation pattern A around axis z_1 (Fig. 8-13). In this case the measuring coordinate system is rigidly connected with the lead beam: axis z_1 coincides with the axis of ray/beam (by axis of equisignal sector), and axes x_1 and y_1 are perpendicular to axis z_1 and it is rigidly connected with the antenna system, which creates lead beam.

During missile targeting to the moving target the controlling ray/beam is moved in the space: during guidance by the method of the covering of target the axis of ray/beam at each moment of time is directed to the target ("follows" the target), while during guidance by the methods of prevention/advance this axis at each moment of time it is directed for certain set forward point, which corresponds to this moment/torque of the time (it follows set forward point).

Let us assume for the concreteness that the control center is located on the earth/ground and the displacements/movements of ray/beam in the space are achieved by the rotations of this ray/beam on azimuth and angle of elevation, as shown in Fig. 8-14.

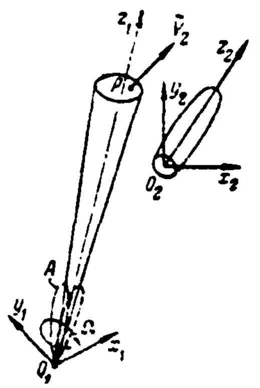

Fig. 8-13.

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The antenna system A, which creates ray/beam $x_1y_1z_1$ is turned in bearings bc around the horizontal axis MN, i.e., on the angle of elevation γ, and in step bearing a - around the vertical axis BC, i.e., but to azimuth σ.

Consequently, angular velocities $\dot{\gamma}$ and $\dot{\sigma}$ the rotations of ray/beam along the angle of elevation and the azimuth are directed along axes MN and BC respectively. Thus, in the process of displacing the ray/beam the measuring system of coordinates $x_1y_1z_1$, connected with the ray/beam, is turned along the angle of elevation and the azimuth with the angular velocities $\dot{\gamma}$ and $\dot{\sigma}$ respectively.

The executive system of coordinates $x_2y_2z_2$ is arranged/located on the rocket and is connected with its longitudinal axis z_2 (Fig. 8-13).

Let us assume that this coordinate system is stabilized by onboard stabilizer from the rotation around longitudinal axis z_2.

Let us assume further that prior to the missile takeoff the axes of system $x_2y_2z_2$ are combined in direction with the appropriate axes of system $x_1y_1z_1$. Then in the process of rocket flight the position of the system of coordinates $x_2y_2z_2$ will be determined by those conditions that axis z_2 coincides with the axis of rocket, but axes x_2 and y_2 cannot be turned around axis z_2 as a result of the presence of gyrostabilization.

In the process of rocket flight the system of coordinates $x_1y_1z_1$, connected with the ray/beam, also is moved as a result of the rotations of the axis of ray/beam z_1 along the angle of elevation and the azimuth (axis of ray/beam must track a target or set forward point).

Consequently, in the process of rocket flight both systems of coordinates ($x_1y_1z_1$ and $x_2y_2z_2$) are displaced in space besides differently. Therefore the system of coordinates $x_2y_2z_2$ is turned (it is twisted) relative to the system of coordinates $x_1y_1z_1$. In the general case the torsion occurs along all three axes, i.e., none of the axes of system $x_2y_2z_2$ will already coincide in the direction with the appropriate axis of system $x_1y_1z_1$ (Fig. 8-15). We will call this torsion three-dimensional/space.

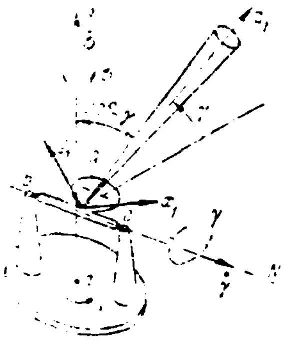

Fig. 8-14.

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However, in the first approximation, it is possible for simplicity to consider that axis z_2 remains parallel to axis z_1 and are twisted only axes x_2 and y_2 relative to axes z_1 and y_1 (Fig. 8-16). This approximation/approach is based on what axis z_2 cannot compose with axis z_1 the angle, greater 30-40° [see (8-16)]. We will call this torsion flat/plane. Let us compute torsion angle τ.

Since axes z_2 and z_1 are parallel, then

$$\tau = \tau_1 - \tau_2 \qquad (8\text{-}17)$$

where τ_1 - rotation of the system of coordinates $x_1y_1z_1$ around axis z_1, and τ_2 - rotation of the system of coordinates $x_2y_2z_2$ around axis z_2, and also, therefore, around axis z_1. The reading of angles τ_1 and

τ_2 is conducted from the moment/torque of missile takeoff, i.e., is assumed that at the moment of the start

$$\tau_2 = \tau_1 = 0.$$

The stabilization system selected by us of the onboard system of coordinates $x_2y_2z_2$ provides the absence of the rotations of this coordinate system around axis z_2. Consequently, during entire flight time of rocket it will be:

$$\tau_2 = 0 \text{ and } \tau = \tau_1. \qquad (8\text{-}17')$$

Angle τ_1 is an angle of rotation of the system of coordinates $x_1y_1z_1$ around axis z_1. If system $x_1y_1z_1$ is turned around axis z_1, then must exist angular velocity $\omega_{\tau_1} = d\tau_1/dt$, directed along this axis. Therefore for calculating the angle τ_1 should be found angular velocity along axis z_1.

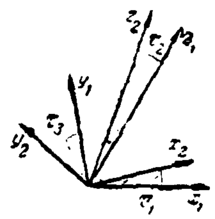

Fig. 8-15.

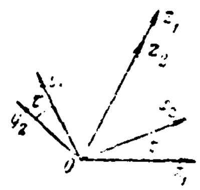

Fig. 8-16.

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Above it was shown (see Fig. 8-14), that lead beam $x_1y_1z_1$ revolves with the angular velocities $\dot{\gamma}$ and $\dot{\sigma}$. Consequently,

$$\omega_{z_1} = \dot{\gamma}_{z_1} + \dot{\sigma}_{z_1}, \quad (8.15)$$

where $\dot{\gamma}_{z_1}$ and $\dot{\sigma}_{z_1}$ - projection of angular velocities $\dot{\gamma}$ and $\dot{\sigma}$ to axis z_1. But from Fig. 8-14 it follows that the vector $\dot{\gamma}$ is always perpendicular axis z_1. Therefore

$$\dot{\gamma}_{z_1} = 0 \text{ and } \omega_{z_1} = \dot{\sigma}_{z_1}.$$

From the same figure it follows that

$$\dot{\sigma}_{z_1} = \dot{\sigma}\cos(90^\circ - \gamma) = \dot{\sigma}\sin\gamma,$$

where γ - angle of elevation for axis z_1 (angle γ is counted off from the horizontal plane).

Consequently,

[illegible]

and

$$\tau_1 = \int \frac{d\sigma}{dt} \sin\gamma \cdot dt.$$

Then from relationship/ratio (8-17') is obtained the following expression for the torsion angle:

$$\tau = \int_0^t \frac{d\sigma}{dt} \sin\gamma \cdot dt = \int_{\sigma_1}^{\sigma} \sin\gamma \cdot d\sigma, \qquad (8\text{-}19)$$

where σ_1 - azimuth of the axis of ray/beam at the moment of missile takeoff (t=0), and σ - azimuth of the axis of ray/beam at the arbitrary moment t of rocket flight.

From formula (8-19) it follows that the greatest torsion will be obtained at the large angles of elevation γ.

Let us assume that during entire process of guidance the angle of elevation remains sufficiently large, namely

$$\gamma \geqslant 60^\circ. \qquad (8\text{-}20)$$

Then we obtain:

$$0{,}87 \leqslant \sin\gamma \leqslant 1.$$

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Therefore with satisfaction of condition (8-20) it is possible to assume/set:

$$\sin\gamma \approx \text{const} = 1,$$

formula (8-19) gives:

$$\tau = \sigma - \sigma_1 = \Delta\sigma, \qquad (8\text{-}21a)$$

where $\Delta\sigma$ - change in the azimuth of ray/beam for the time, which passed from the moment/torque of start to the moment of time in question.

At the end of the induction/guidance when $\sigma=\sigma_2$, it is obtained .

$$\tau = \sigma_2 - \sigma_1 = \Delta\sigma. \qquad (8\text{-}21b)$$

Let us explain, what greatest values of angle τ can occur with the induction/guidance according to the method of the covering of target and the predicted point guidance.

Let us consider first induction/guidance according to the method of the covering of target. Let the target move rectilinearly over horizontal line C_0C_κ. and KP is located at point O (Fig. 8-17). Let us lead through the straight/direct C_0C_κ horizontal plane M. Let O' - projection of point O on this plane, and p - perpendicular from point O' to straight line C_0C_κ. i.e. the parameter of target.

Let at the moment of the production of rocket the target be located at point C_0, and at the moment of rendezvous with the rocket - at point C_κ. Then the axis of ray/beam z_1 at the moment of missile takeoff is directed into point C_0, and at the moment of the encounter of rocket with target - into point C_κ.

Consequently, in the process of the guidance of rocket to the target axis z_1 is moved from direction OC_0 to direction OC_κ and, therefore, a change in the azimuth of this axis is equal to:

$$\Delta\sigma = \angle C_0O'C_\kappa.$$

Since during entire time of induction/guidance angle of elevation γ is sufficiently great [it satisfies condition (8-20)], the is correct relationship/ratio (8-21b), and torsion angle is equal to:

$$\tau = \Delta\sigma = \angle C_0O'C_\kappa. \qquad (8\text{-}22)$$

From Fig. 8-17 it is evident that the less the parameter of circuit P, the more $\angle C_0O'C_\kappa$; in the low parameter it is close to 180°.

Consequently, with rectilinear motion of target angle τ can reach 180°.

With the curvilinear motion of target can occur the case, depicted in Fig. 8-18.

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In this case the target is circled of radius located in the horizontal plane M. Center O' of this circle/circumference is arranged/located exactly above control center O. Therefore angle of elevation for axis z_1 remains always large and, therefore, is correct relationship/ratio (8-21b), namely:

$$\tau = \Delta\beta.$$

It is obvious, in this case the angle τ can, generally speaking, reach 360° and more, if target for the flight time of rocket has time to complete more than one circle.

Angular velocity of the motion of the target

$$\omega_c = \frac{W_c}{V_1},$$

where V_1 and W_c - speed and the acceleration of target respectively. Let, for example, $W_c = 3g = 30$ m/s² and V_1=200 m/s. Then $\omega_c = 9°/$s.

Let the height/altitude of target is h=15 km the speed of rocket V_1=600 m/s. Then the time of flight of rocket to the target will be about 30 s. For this time the azimuth of target will have time it will be changed to value

$$\Delta\beta = \omega_c \Delta t = 270°.$$

From this example it is evident that when the maneuvers of target are present, the torsion angle can exceed 180°.

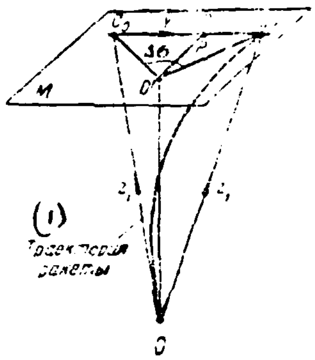

Fig. 8-17.

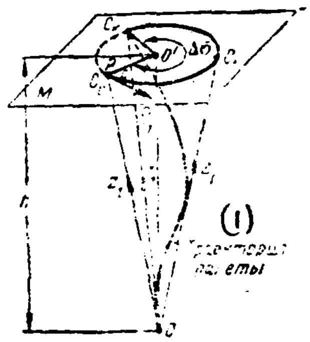

Fig. 8-18.

Fig. 8-17.

Key: (1). Missile trajectory.

Fig. 8-18.

Key: (1). Missile trajectory.

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Let us consider now predicted point guidance. Let the target travel at constant speed $\overline{V}_1$ (Fig. 8-19), and rocket has the constant in the velocity V_2. Then will exist constant set forward point A, at which the rocket will meet target in flight on the straight line OA. Consequently, the axis of lead beam z_1, along which flies the rocket, must be always directed to the constant/invariable point A. Thus, in the case in question lead beam is fixed during entire time of guidance. Therefore occur the relationships/ratios

$$\Delta\sigma = 0 \quad \text{and} \quad \tau = 0,$$

i.e. the torsion of coordinates is absent.

Let us consider now the case of the curvilinear motion of target (Fig. 8-20).

Let us assume at the moment of missile takeoff (t=0) target speed $\overline{V}_1$ is directed along horizontal line C_0A_1, and then target completes maneuver in the horizontal plane M, moving with lengthwise curved C_0A_k. Then at moment/torque t=0 set forward point A_1 will be found on the straight line OA_1 and, therefore, the axis of ray/beam z_1 will be directed to point A_1. At the end of the induction/guidance the rocket must meet target. Therefore end position of set forward point will be located to the trajectory of target at certain point A_k and axis z_1 at the end of the induction/guidance will be directed to point A_k.

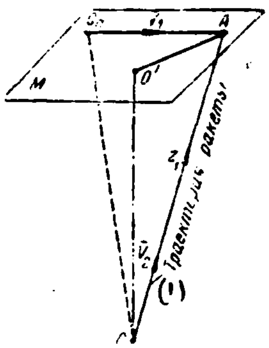

Fig. 8-19.

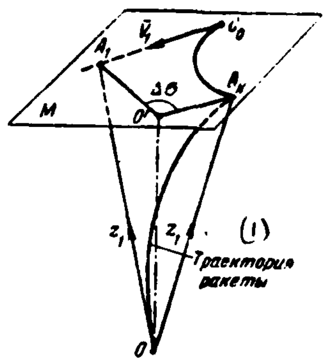

Fig. 8-20.

Fig. 8-19.

Key: (1). Missile trajectory.

Fig. 8-20.

Key: (1). Missile trajectory.

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Therefore a change of the azimuth of axis z_1 in the process of induction/guidance will be equal to:

$$\Delta\sigma = \angle A_1 O' A_k$$

where O' - projection of point O (location of KP) on plane M. Since the angle of elevation of ray/beam remains always large, the is correct relationship/ratio (8-20b) and torsion angle is equal to:

$$\tau = \Delta\sigma - \angle A_1 O' A_{k}.$$

From Fig. 8-19 it is evident that the angle $\Delta\sigma$ and, consequently, also τ can be on the order of 180°, and in the most unfavorable cases even exceed 180°.

In the analysis given above we disregarded the fracture of axes z_2 and z_1, assuming/setting $\varphi_{12}=0$. In actuality the curvilinearity of missile trajectory it will be unavoidable to cause the fracture of axes. Taking into account this fracture the torsion angle is equal to:

$$\tau' = \tau + \Delta\tau, \qquad (8\text{-}21)$$

where τ - torsion angle, calculated under the assumption of the parallelism of axes z_2 and z_1; $\Delta\tau$ - correction, which considers the effect of the fracture of axes z_2 and z_1.

However, the analysis conducted showed that with the remote control of the first form this correction is relatively small and in the very worst cases does not exceed 20-30°, but in the majority of the cases compose a total of several degrees. On the other hand, the calculation of this correction proves to be very complicated, and final formula for value $\Delta\tau$ is considerably more complicated than formula (8-19), which is determining value τ. Therefore in the first approximation, it is possible to determine the value of torsion according to formula (8-19).

From examples examined above it is evident that with the remote control of the first form the torsion angle can reach inadmissibly high values and, if we do not take special measures, control it will be impossible.

8-4. Ways of decreasing the effect of the torsion of the coordinate systems.

Are possible two ways of decreasing the effect of the torsion of the coordinate systems:

1. Elimination of the reasons, which call torsion.

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2. Measurement of torsion angle and introduction of corresponding corrections to steering commands.

Let us consider first the possibilities of eliminating the reasons, which call torsion. As noted above, torsion is generated by a difference in the laws of the motion (rotation) of the measuring and executive coordinate systems. Consequently, it is necessary to strive as far as possible to decrease these differences in the laws of motion.

With some guidance methods this difference can be reduced to zero. Thus, for instance, it was above indicated that during the homing on the linear curve with the help of the antenna, established/installed on the power weathervane, the measuring and executive coordinate systems they coincide and torsion is absent.

However, in a number of cases this complete coincidence of the coordinate systems is in principle impossible. Thus, for instance, in case examined above of the remote control of the first form according to the method of covering of target unavoidably appears the fracture of axes z_2 and z_1, since with curved path of rocket axis of rocket and its velocity vector compose certain angle with axis z_1 of lead beam. Consequently, in this case in principle it is not possible to reduce the effect, caused by the fracture of axes. However, in case examined above of remote control of the first form (Fig. 8-14), when lead beam z_1 is moved in the path space of rotations along the angle of elevation and the azimuth, the fundamental effect of torsion is caused not by the fracture of axes, namely by the selected method of displacing the ray/beam in the space.

Actually/really, above it was shown that the torsion angle, caused by the rotations of antenna along the angle of elevation and the azimuth, can reach 180° and more, whereas the correction $\Delta\tau$, caused by the fracture of axes, in the majority of the cases does not exceed several degrees [see formula (8-23)]. Therefore, if we approach the law of the rotation of ray/beam (i.e. the measuring coordinate system) the law of the rotation of executive system of coordinates $x_2y_2z_2$, then it is possible to sharply decrease the torsion. In example examined above system $x_2y_2z_2$ is stabilized with the help of the gyroscopes in such a way that is removed its rotation around axis z_2; therefore, if we hang up the antenna system of radar, which creates lead beam, in the gimbal suspension and to also stabilize it with the help of the gyroscopes from the rotations around axis z_1, then torsion will be called only the fracture of axes z_2 and z_1 and, therefore, sharply decreases.

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However, this method of rotating the antenna of locator in a number of cases can prove to be unacceptable according to the design considerations. Then it is possible to use any another method, that ensures the approximation/approach of the law of the rotation of the system of coordinates $x_1y_1z_1$ to a law of the rotation of system $x_2y_2z_2$.

In method examined above of rotating the antenna along the angle of elevation and the azimuth (see Fig. 8-14) the greatest torsion was obtained at the large angles and smallest - at the small angles of elevation [see formula (8-19)]. It is obvious, if we select other rotational axes of antenna, then it is possible to be obtained that torsion would be smallest not with small, but at the large angles of elevation.

Thus, in a number of cases it is possible to decrease the torsion angle τ via the appropriate selection of the methods of rotating the coordinate systems.

When on any reasons (usually structural/design) to decrease the torsion angle τ to the permissible value proves to be impossible, should be used the alternate path, which consists in the measurement of angle τ and the introduction of the corresponding corrections to commands/crews for the controls. Let us consider this method in somewhat more detail. Let us assume the value of torsion angle τ are measured by any method. Let us explain, how should be introduced this value into the commands/crews to the controls. For this let us consider Fig. 8-21, by which are accepted the following designations: $x_2y_2z_2$ - executive coordinate system; $x_1y_1z_1$ - measuring coordinate system, in which are measured the disturbances/perturbations Θ, the characteristic deviation of rocket from the correct flight of the target; x_1'y_1'z_1' - position of the measuring coordinate system, which would correspond to agreement with the executive coordinate system.

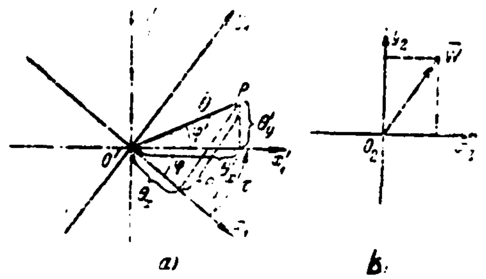

Fig. 8-21.

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From these determinations it follows that θ_x' and θ_y' - true (correct) components of disturbance/perturbation $\bar{\Theta}$, and θ_x and θ_y - measured components of the same disturbance/perturbation.

From Fig. 8-21a escape/ensues the following connection/communication between values θ_x' and θ_y' and values θ_x and θ_y.

$$\theta_x' = \theta_x \cos\tau + \theta_y \sin\tau; \qquad (8\text{-}24a)$$

$$\theta_y' = \theta_y \cos\tau - \theta_x \sin\tau. \qquad (8\text{-}24b)$$

Since values θ_x, θ_y and τ are known from the measurements, then the true values of disturbances/perturbations θ_x' and θ_y', which must be used for forming the commands/crews to the controls, they can be determined from relationships/ratios (8-24).

For the realization of the automatic transformation, which corresponds to these relationships/ratios, can serve the system, depicted in Fig. 8-22. It consists of four potentiometers: two sine (realizing transformation according to the law $u_{вых}=u_{вх}\sin\tau$) and two cosine (realizing transformation according to the law $u_{вых}=u_{вх}\cos\tau$). The rotors of all potentiometers sit on one axis and are turned to the identical angle τ.

To the entry of system are supplied measured values θ_x and θ_y (in the form of DC voltages) and τ (in the form of the mechanical rotation of axis).

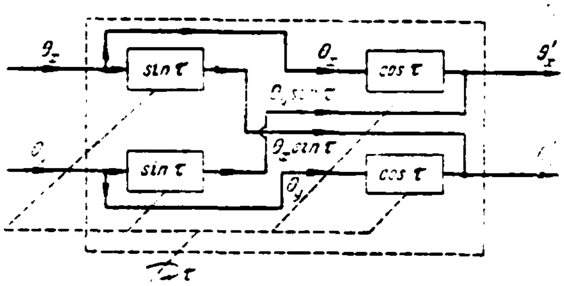

Fig. 8-22.

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Voltages/stresses Θ_x' and Θ_y' obtained on the output systems, are used further for forming the commands/crews.

Equations (8-24) and system (Fig. 8-22) corresponding to them relate to the case when the measurement of disturbance/perturbation $\bar{\Theta}$ is made in the Cartesian coordinates (see Fig. 8-4).

If the measurement of disturbance/perturbation $\bar{\Theta}$ is made in the polar coordinates, then the result of measurement is obtained in the form of the voltage/stress of alternating current (error signal):

$$e_\theta = u_\theta \cos(\Omega t + \varphi). \qquad (8\text{-}25)$$

where

$$u_\theta = K\Theta.$$

i.e.

$$e_{\Omega} = K\Theta \cos(\Omega t + \varphi). \qquad (8\text{-}25a)$$

Here K - proportionality factor; phase φ is equal to the vectorial angle of vector $\overline{\Theta}$ in the system of coordinates $x_1 y_1 z_1$ (Fig. 8-21). The reference point of phase (zero time reference) is assigned by the reference voltage

$$e_{on} = u_{on} \cos \Omega t. \qquad (8\text{-}25b)$$

From Fig. 8-21 it follows that in the presence of torsion to the angle τ the true vectorial angle of vector $\overline{\Theta}$ is equal not to φ, but

$$\varphi' = \varphi - \tau. \qquad (8\text{-}26)$$

Therefore the true error signal is equal not to e_{Ω}, but e_{Ω}', where

$$e_{\Omega}' = K\Theta \cos(\Omega t + \varphi') = K\Theta \cos(\Omega t + \varphi - \tau). \qquad (8\text{-}27)$$

From the comparison of relationships/ratios (8-27) and (8-25a) it follows that the true error signal e_{Ω}' can be obtained from measured signal e_{Ω} by its shift/shear on the phase to the angle τ, with the retention/preservation/maintaining of the phase of the reference voltage of constant/invariable.

This operation can be performed automatically by phase inverter FV, depicted in Fig. 8-23.

It is obvious, it is possible to obtain the same result, if we instead of the shift/shear to angle $(-\tau)$ of the error signal e_{Ω} use shift/shear to the angle τ of reference voltage e_{on}.

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In this case the error signal e_0 remains constant/invariable, and reference voltage is passed through the phase inverter FV (Fig. 8-24) and takes the form:

$$e'_{on} = E_{on} \cos(\Omega t + \tau). \qquad (8\text{-}28)$$

From the comparison of relationships/ratios (8.25a) and (8.28) it follows that after this correction the error signal anticipates/leads reference voltage on angle $(\varphi-\tau)$, which also is required for the correct formation/education of commands/crews.

In the analysis given above it was assumed that the value of angle τ was known from the measurements. In the case of homing and remote control of the second form when the measuring and executive coordinate systems are located in one place (on the rocket), this measurement does not present fundamental difficulties. In the case of the remote control of the first form with the help of the ray/beam (Fig. 8-13) approximate value of angle τ can be found on the basis of formula (8-19):

$$\tau \approx \int_0^t \frac{d\alpha}{dt} \sin \gamma \cdot dt.$$

(This formula does not consider the effect of the fracture of axes).

The principle of the measurement of angle τ and introduction of the corresponding correction for a similar control system is shown for Fig. 8-25. Rocket flies in the ray/beam of the radar, automatic tracking after the target. The ray/beam of radar in the process automatic tracking is moved along the angle of elevation γ and the azimuth σ. The data about the angle of elevation and the azimuth (γ and σ) enter the input of the computer SU, which automatically computes angle τ according to formula (8-19) and is realized rotation to this angle of the axis of phase inverter FV. At the input of phase inverter enters also the reference voltage whose phase is rigidly connected with the phase of the rotation of radiation pattern A around the axis of ray/beam z_1.

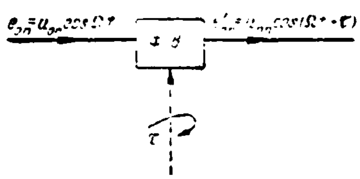

Fig. 8-23.

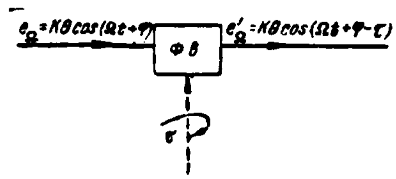

Fig. 8-24.

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Phase inverter realizes a shift/shear of reference voltage on the phase on the angle τ. Reference voltage corrected thus is transmitted by the radio to the rocket (via modulation of the signals of radar or along the separate channel).

On the rocket is established/installed the receiver, which with the deviation of the rocket of the axis of the axis of ray/beam selects the error signal $e_{\Omega} = K\Theta \cos(\Omega t+\varphi)$ and compares it with reference voltage e'_{on}. As a result of this comparison on the rocket are formed the voltages/stresses, utilized then for forming the commands/crews on the controls (in accordance with the diagram in Fig. 8-4, 8-5 or 8-6).

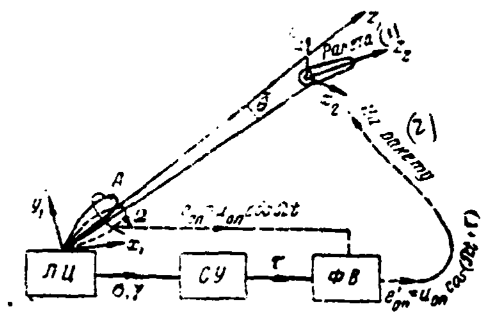

Fig. 8-25.

Key: (1). Rocket. (2). To rocket.

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