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HIGH-PRECISION AIRCRAFT GUIDANCE SYSTEM WITH AXIAL ACCELERATION SELF-TUNING

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Abstract

Creation of aircraft control systems that provide high quality guidance is an urgent task associated with increasing efficiency of the modern missile systems. Classical autonomous guidance systems without integration with any other external correction systems are making it impossible to ensure high-precision target engagement. An approach based on the use of adaptive guidance system is proposed. Besides, an approach to synthesis and analysis of the self-tuning aircraft control system that implements terminal homing is illustrated. A technique for forming the control system structure is presented ensuring the control quality constant level in all operating modes due to self-tuning of the constituent elements variable coefficients. Structural schemes and control equations were determined to correct the aircraft flight. Implementation of the developed technique on board the aircraft is proposed determining relationship between the aircraft control system parameters and the apparent acceleration. Adaptive system operation is demonstrated on a typical model of the aircraft flying in the atmosphere with guidance at the stationary target. Results of numerical simulation are presented, and high efficiency of the developed technique is revealed

Keywords

Adaptive guidance system, homing, stabilization system, aircraft, mathematical simulation

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Introduction. A rather complex technical task of the target engagement ensuring high accuracy is assigned to modern aircraft control systems. Wide range of studies in this area is a clear evidence [1–13].

More and more attention is being paid to practical issues of building adaptive control systems. The main part of literature is occupied by sources of the adaptive and optimal control theory [2–5, 7–10].

Various approaches are used in implementation of the adaptive control systems. The first type is based on the self-tuning systems adapted only by changing the control device parameters [7, 14–16]. If adaptation is achieved by changing the control device structure or the control algorithm, such systems are the self-organizing [14, 16].

If, constant control quality corresponding to a given value is achieved with changing the control parameter values, such systems are the search systems. The synthesis of such systems is presented in [14]. Systems are searchless [3, 7, 14, 16, 17], if the desired parameter values are established on the basis of analytical analysis of the conditions, under which restrictions in regard to adaptation quality would be lifted.

Despite the abundance of literature on the general theory of adaptive systems, methods for analyzing such systems require further research.

Problem statement. It is required in synthesizing the adaptive control systems to achieve the required quality indicators having certain information about the control object and the system operating conditions. However, properties of the control object could change during such systems operation for various reasons, so it is not possible to reliably predict the nature of changes in advance. For example, the aircraft dynamic properties depend on the constantly changing speed and flight altitude, mass and moments of inertia of the aircraft in flight. In addition, aerodynamic characteristics and elements of the control system are characterized by significant non-linearity. Moreover, the disturbing influences are non-stationary. All these alterations occurring in large ranges could lead to the fact that the synthesized system designed and oriented towards certain initial information, would no longer ensure in the new operating conditions the quality indicators compliance with the existing limitations.

The problem of synthesizing the adaptive homing systems (HS), which would make it possible by self-tuning the variable coefficients of its constituent elements, to compensate for the influence of the system non-linearity and non-stationarity is currently extremely relevant. Such non-linear control provides significant additional opportunities in improving the control processes.

Priority objective of the described study is to synthesize an adaptive HS using the non-identification approach [14].

The essence of this approach lies in ensuring a constant or optimal level of the control quality indicator in all the operating modes by maintaining constancy of the control system parameters using self-tuning of the constituent elements variable coefficients.

The following tasks were set to achieve the objective.

1. Construct block diagrams to analyze and synthesize the stabilization system (SS) and the HS in general along the control channels.

2. Form the structure of control equations for the control channels.

3. Form the control parameters equations relative to the basic HS (as a base system, such a system was chosen, where the proportional approach method was implemented known for ensuring its high operation accuracy against both the fixed and the high-speed moving targets [1, 16]).

4. Determine functional dependencies in the variable gains of the HS and SS constituent elements.

5. Evaluate the developed methodology efficiency on the example of a hypothetical aircraft using the simulation software.

Flight mathematical model. Controlled spatial motion of the aircraft at the homing stage is determined by integrating the system of motion differential equations.

Detachable nose (warhead) in the shape of a cone is considered as the control object. Aerodynamic coefficients are assumed to be known.

Spatial motion equations of the aircraft center of mass are written down in projections on the $O_0X_gY_gZ_g$ axes of the normal terrestrial coordinate system. These equations have the following form in the matrix form:

$$m\frac{d^2}{dt^2}\begin{bmatrix} X_g \\ Y_g \\ Z_g \end{bmatrix} = \mathbf{A}\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} + \begin{bmatrix} G_x \\ G_y \\ G_z \end{bmatrix}, \tag{1}$$

where **A** is the transition matrix from the bound coordinate system to the normal Earth coordinate system[1].

In this case, the equations of the center of mass translational movement have the following form:

$$\dot{V}_{Xg} = \frac{1}{m}(F_{Xg} + G_{Xg}); \tag{2}$$

$$\dot{V}_{Yg} = \frac{1}{m}(F_{Yg} + G_{Yg}); \tag{3}$$

$$\dot{V}_{Zg} = \frac{1}{m}(F_{Zg} + G_{Zg}), \tag{4}$$

where V_{Xg}, V_{Yg}, V_{Zg} are projections of the aircraft velocity on the axes of the normal Earth coordinate system; X_g, Y_g, Z_g are projections of the radius vector connecting the O_0 point with the aircraft on the axes of the normal Earth

[1] GOST 20058–80. Aircraft dynamics in the atmosphere. Terms, definitions and designations. Moscow, Standardy Publ., 1981.

coordinate system; F_{Xg}, F_{Yg}, F_{Zg} are projections of the total aerodynamic force on the axes of the normal Earth coordinate system; G_{Xg}, G_{Yg}, G_{Zg} are projections of gravity on the axes of the normal Earth coordinate system.

Equations of the aircraft rotational motion relative to its center of mass written in projections on the $OXYZ$ axes of the associated coordinate system [1], provided that the inertia centrifugal moments are equal to zero, have the following form:

$$\dot{\omega}_x = \frac{M_x}{I_x} - \frac{I_z - I_y}{I_x}\omega_y\omega_z; \quad (5)$$

$$\dot{\omega}_y = \frac{M_y}{I_y} - \frac{I_x - I_z}{I_y}\omega_x\omega_z; \quad (6)$$

$$\dot{\omega}_z = \frac{M_z}{I_z} - \frac{I_y - I_x}{I_z}\omega_z\omega_y, \quad (7)$$

where I_x, I_y, I_z are the axial moments of the aircraft inertia.

In order for the system of equations (1)–(7) that describes the aircraft spatial motion to become closed, it is necessary to set the controls deviations as functions of time. In this case, the control equations in the general form are taking the following form:

$$F(\delta_\vartheta, \delta_\psi, \delta_\gamma, X_g, Y_g, Z_g, \vartheta, \psi, \gamma, \ldots) = 0, \quad (8)$$

where δ_ϑ, δ_ψ, δ_γ are the equivalent controls rotation angles reduced to the pitch, roll and yaw channels, respectively.

Methods formatting the HS and the mathematical model of its operation. Equation (8) contains both the low-frequency components and the high-frequency components reflecting the processes occurring in the aircraft SS and HS. Therefore, their numerical integration with system (1)–(7) generally in the digital simulation requires significant amount of time due to the extremely small integration step. However, the noted high-frequency components of the control signals in the real flight conditions are being filtered due to the aircraft significant inertia. Thus, the principle of low-frequency limitation of the spectrum of the considered technical devices of the aircraft angular stabilization loop and of the guidance loop was applied in formatting the block diagrams. Therefore, all elements of the onboard flight control system could be assumed the ideal inertialess links in calculating the aircraft flight trajectories.

This paper considers a classical HS scheme consisting of three orthogonal control channels, i.e., the pitch and yaw channels that control longitudinal motion and the autonomous roll channel that keeps the roll angle equal to zero. Due to the fact that differential equations describing spatial motion along the pitch and yaw channels are identical, transfer functions and structural diagrams for these channels would be the same, except for the corresponding indexation (ϑ index characterizes the pitch channel, and ψ index characterizes the yaw channel).

The principle of coefficient freezing is used to describe the aircraft longitudinal perturbed motion characteristics in the theory of automatic flight control [16]. Since one of the work tasks is formation of the stabilization system adaptation algorithm, which would be implemented in the onboard computer (OBC) and used continuously, then the aircraft parameters could be considered unchanged and frozen within the discreteness interval of the machine computing process during self-tuning. This makes it possible to use the following block diagrams to analyze and synthesize the SS and HS controls as a whole (Fig. 1). The transfer functions apparatus is used to simplify representation of the processes.

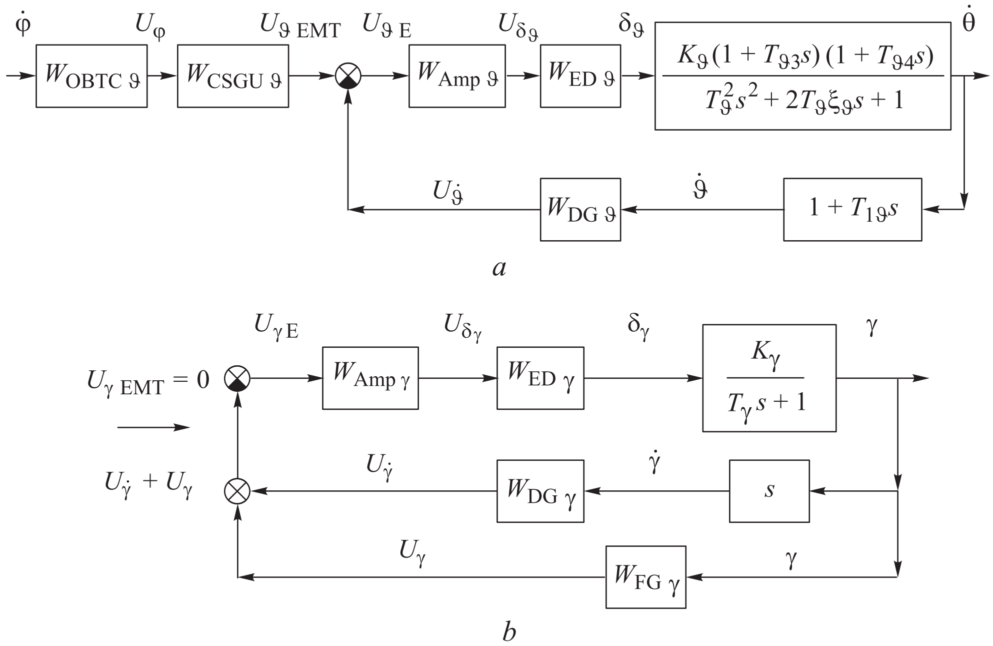

Fig. 1. Block diagram of the guidance loop:
a is along the pitch channel, *b* is along the roll channel

In accordance with the assumptions made, transfer functions of the onboard control system elements provided in the design block diagrams (see Fig. 1) have the following approximate expressions:

$$W_{\text{OBTC}\,\vartheta} = \frac{K_{\dot{\varphi}}}{s} \quad (9)$$

are transfer functions that determine the onboard target coordinator (OBTC) properties; transfer functions that determine the autopilot elements properties: $W_{\text{CSGU}\,\vartheta} = K_{\text{C}\,\vartheta}$ is control signal generation unit (CSGU); $W_{\text{Amp}\,\vartheta} = K_{\text{Amp}\,\vartheta}$, $W_{\text{Amp}\,\gamma} = K_{\text{Amp}\,\gamma}$ are matching amplifiers; $W_{\text{ED}\,\vartheta} = K_{\text{ED}\,\vartheta}$, $W_{\text{ED}\,\gamma} = K_{\text{ED}\,\gamma}$ are executive devices; $W_{\text{DG}\,\vartheta} = K_{\text{DG}\,\vartheta}$, $W_{\text{DG}\,\gamma} = K_{\text{DG}\,\gamma}$ are damping gyroscopes; $W_{\text{FG}\,\gamma} = K_{\text{FG}\,\gamma}$ is free roll channel gyroscope.

Basic HS control equations. The task of forming the control equations is to establish functional relationships between the aircraft trajectory parameters measured by the corresponding sensors of the controls rotation angles through the corresponding channels. To solve this problem, it is necessary to determine first the general structure of controls.

Taking into account the assumptions made about ideality and missing inertia of the control system elements in accordance with the previously mentioned calculated block diagrams of the dedicated channels, the control equations in general form could be written as follows (EMT index is reference model):

$$U_{\vartheta\,\text{E}} = U_{\vartheta\,\text{EMT}} + U_{\vartheta\,\text{SS}} \quad (10)$$

— for pitch channel;

$$U_{\gamma\,\text{E}} = U_{\gamma\,\text{SS}} \quad (11)$$

— for roll channel.

Here $U_{\vartheta\,\text{EMT}}$ is the control signal that directly implements the selected homing method, and $U_{\vartheta\,\text{SS}}$, $U_{\gamma\,\text{SS}}$ are the control signals ensuring the aircraft stabilization. The type of the latter is not depending on the homing method, but is determined only by the SS structure.

In accordance with the diagrams (see Fig. 1), expressions (10) and (11) could be expanded and reduced to the following form:

$$U_{\vartheta\,\text{EMT}} = K_{\dot{\varphi}} K_{\text{C}\,\vartheta}\dot{\varphi}; \quad (12)$$

$$U_{\vartheta\,\text{SS}} = -U_{\dot{\vartheta}} = -K_{\text{Amp}\,\vartheta} K_{\text{DG}\,\vartheta}\dot{\vartheta}; \quad (13)$$

$$U_{\gamma\,\text{SS}} = -U_{\dot{\gamma}} - U_{\gamma} = -K_{\text{Amp}\,\gamma} K_{\text{DG}\,\gamma}\dot{\gamma} - K_{\text{Amp}\,\gamma} K_{\text{FG}\,\gamma}\gamma, \quad (14)$$

where $K_{\dot{\varphi}}$ is the coefficient that determines the target coordinator properties; $K_{\text{C}\,\vartheta}$ is the coefficient that determines the CSGU properties; $K_{\text{Amp}\,\vartheta}$, $K_{\text{Amp}\,\gamma}$ are the coefficients that determine the matching amplifiers properties; $K_{\text{DG}\,\vartheta}$,

$K_{DG\,\gamma}$ are the coefficients that determine the SS damping gyroscopes properties in the pitch and roll channels; $K_{FG\,\gamma}$ is the coefficient that determines the free gyroscope properties in the SS roll channel.

Therefore, substituting expressions (12)–(14) into equations (10)–(11), the complete equations for the basic HS control signals are obtained:

$$U_{\vartheta\,E} = K_{\dot\varphi} K_{C\,\vartheta} \dot\varphi - K_{Amp\,\vartheta} K_{DG\,\vartheta} \dot\vartheta; \qquad (15)$$

$$U_{\gamma\,E} = -K_{Amp\,\gamma} K_{DG\,\gamma} \dot\gamma - K_{Amp\,\gamma} K_{FG\,\gamma} \gamma. \qquad (16)$$

In general, equations of the controls rotation parameters are directly similar to the control equations and have the following form:

$$\delta_\vartheta = \delta_{\vartheta\,EMT} + \delta_{\vartheta\,SS} \qquad (17)$$

— for the pitch channel;

$$\delta_\gamma = \delta_{\gamma\,SS} \qquad (18)$$

— for the roll channel, where $\delta_{\vartheta\,EMT}$ is the reduced angle that determines the controls deviation necessary to implement the reference homing technique $\delta_{\vartheta\,SS}$, $\delta_{\gamma\,SS}$ are the reduced angles determined by the aircraft SS. The method for establishing these angles also does not depend on the type of homing and is determined only by the SS structure.

In accordance with the diagrams in Fig. 1, let us write:

$$\delta_\vartheta = U_{\vartheta\,E} K_{Amp\,\vartheta} K_{ED\,\vartheta} = (U_{\vartheta\,EMT} - U_{\vartheta\,SS}) K_{Amp\,\vartheta} K_{ED\,\vartheta} =$$
$$= K_{Amp\,\vartheta} K_{ED\,\vartheta} K_{\dot\varphi} K_{C\,\vartheta} \dot\varphi - K_{Amp\,\vartheta} K_{ED\,\vartheta} K_{DG\,\vartheta} \dot\vartheta =$$
$$= \delta_{\vartheta\,EMT} + \delta_{\vartheta\,SS} = K_{1\,\vartheta} \dot\varphi - K_{2\,\vartheta} \dot\vartheta;$$
$$\delta_\gamma = U_{\gamma\,E} K_{Amp\,\gamma} K_{ED\,\gamma} = -U_{\gamma\,SS} K_{Amp\,\gamma} K_{ED\,\gamma} =$$
$$= -K_{Amp\,\gamma} K_{ED\,\gamma} K_{DG\,\gamma} \dot\gamma - K_{Amp\,\gamma} K_{ED\,\gamma} K_{FG\,\gamma} \gamma =$$
$$= \delta_{\gamma\,SS} = -K_{1\,\gamma} \dot\gamma - K_{2\,\gamma} \gamma,$$

where $K_{1\,\vartheta} = K_{Amp\,\vartheta} K_{ED\,\vartheta} K_{\dot\varphi} K_{C\,\vartheta}$; $K_{2\,\vartheta} = K_{Amp\,\vartheta} K_{ED\,\vartheta} K_{DG\,\vartheta}$; $K_{1\gamma} = K_{Amp\,\gamma} K_{ED\,\gamma} K_{DG\,\gamma}$; $K_{2\,\gamma} = K_{Amp\,\gamma} K_{ED\,\gamma} K_{FG\,\gamma}$.

Due to the fact that during a controlled flight many parameters that determine the aircraft state are changing intensively, control coefficients describing the SS properties should also be variable to ensure high-quality operation of the SS.

In the general case, dependences of the SS elements coefficients alteration on the parameters of an aircraft being a control object have the following form: $K_{1\,\vartheta,\,2\,\vartheta,\,1\,\gamma,\,2\,\gamma} = K_{1\,\vartheta,\,2\,\vartheta,\,1\,\gamma,\,2\,\gamma}(t,\, q,\, V,\, F,\, ...)$.

The nature of alterations in certain parameters of the aircraft state and coefficients describing the SS elements properties (see Fig. 1) for a hypothetical aircraft under conditions of engagement in the homing final section and in the case of a fixed target is illustrated by graphs in Figs. 2, 3. Calculation was carried out for a test homing trajectory (the target was at the distance of 50 000 m along the OX_g axis from the point, where the aircraft started to stabilize, and was located on the OZ_g axis).

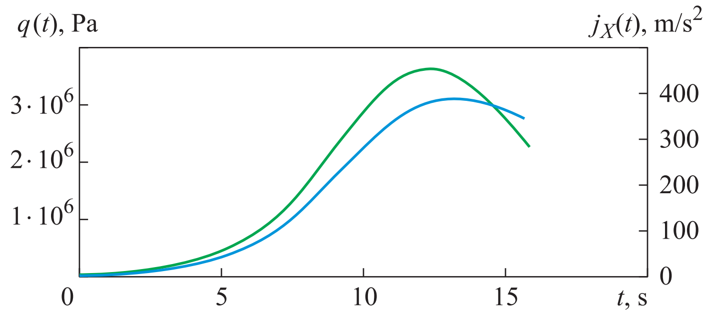

Fig. 2. Alterations in impact air pressure $q(t)$ (green curve) and axial $j_X(t)$ (blue curve) acceleration (hypothetical aircraft)

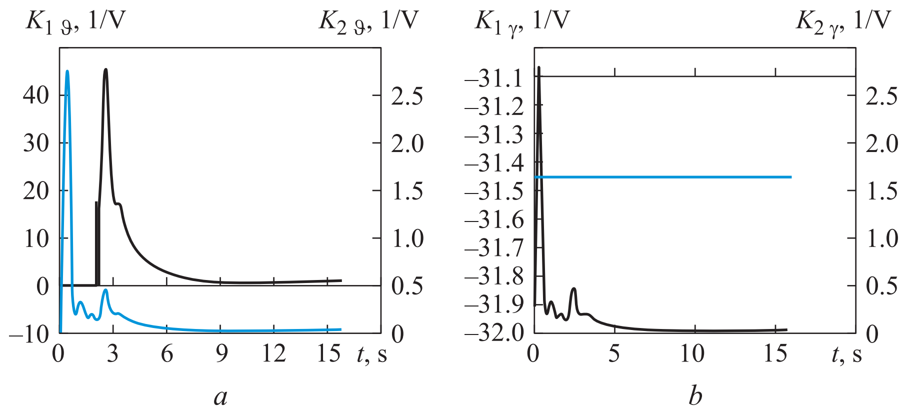

Fig. 3. Alterations in the control coefficients for a hypothetical aircraft:
a is pitch channel $K_{1\vartheta}(t)$ (black curve), $K_{2\vartheta}(t)$ (blue curve);
b is roll channel $K_{1\gamma}(t)$ (black curve), $K_{2\gamma}(t)$ (blue curve)

CSGU and OBTC amplification factors are constant throughout the entire flight, and the constant elements variable coefficients should be self-tuned only in the aircraft SS and HS.

Schemes in Fig. 1 could be transformed as follows (Fig. 4).

Block diagrams (see Fig. 4) were obtained under the assumption that all control system elements are ideal inertialess links, and the aircraft transfer

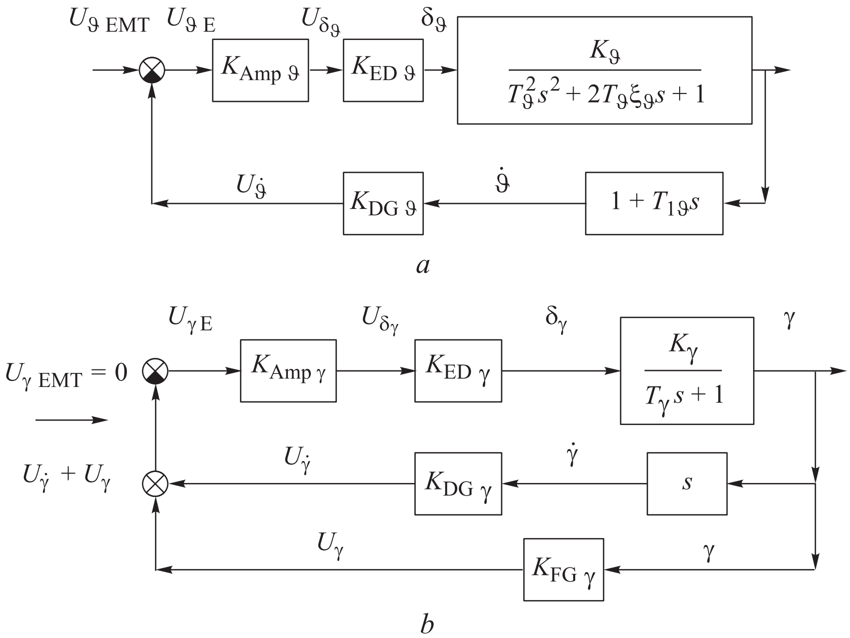

Fig. 4. Diagram transformation (see Fig. 1) along the pitch (*a*) and roll (*b*) channels

functions through the pitch and yaw channels become simplified under the assumption that the normal force created by the controls is negligible [16, 17].

In view of these limitations, block diagrams in Fig. 4 take the form of oscillatory links of the second order:

$$W_\vartheta = \frac{K_{\mathrm{Amp}\,\vartheta} K_{\mathrm{ED}\,\vartheta} K_\vartheta}{T_\vartheta^2 s^2 + 2T_\vartheta \xi_\vartheta s + 1}; \quad W_{\mathrm{fl}\vartheta} = K_{\mathrm{DG}\,\vartheta}(1 + T_{1\vartheta} s);$$

$$W_{\mathrm{eq}\,\vartheta} = \frac{W_\vartheta}{1 + W_\vartheta W_{\mathrm{fl}\vartheta}} = \frac{\dfrac{K_{\mathrm{Amp}\,\vartheta} K_{\mathrm{ED}\,\vartheta} K_\vartheta}{T_\vartheta^2 s^2 + 2T_\vartheta \xi_\vartheta s + 1}}{1 + \dfrac{K_{\mathrm{Amp}\,\vartheta} K_{\mathrm{ED}\,\vartheta} K_\vartheta K_{\mathrm{DG}\,\vartheta}(1 + T_{1\vartheta} s)}{T_\vartheta^2 s^2 + 2T_\vartheta \xi_\vartheta s + 1}} =$$

$$= \frac{K_{\mathrm{Amp}\,\vartheta} K_{\mathrm{ED}\,\vartheta} K_\vartheta}{K_{\mathrm{Amp}\,\vartheta} K_{\mathrm{ED}\,\vartheta} K_\vartheta K_{\mathrm{DG}\,\vartheta}(1 + T_{1\vartheta} s) + T_\vartheta^2 s^2 + 2T_\vartheta \xi_\vartheta s + 1} =$$

$$= \frac{K_{\mathrm{SS}\,\vartheta}}{T_{\mathrm{SS}\,\vartheta}^2 s^2 + 2T_{\mathrm{SS}\,\vartheta} \xi_{\mathrm{SS}\,\vartheta} s + 1}$$

— for pitch channel;

$$W_\gamma = \frac{K_{\mathrm{Amp}\,\gamma} K_{\mathrm{ED}\,\gamma} K_\gamma}{s(T_\gamma s + 1)}; \quad W_{\mathrm{fl}\gamma} = K_{\mathrm{DG}\,\gamma} s + K_{\mathrm{FG}\,\gamma};$$

$$W_{\text{eq}\gamma} = \frac{W_\gamma}{1 + W_\gamma W_{\text{fl}\gamma}} = \frac{\dfrac{K_{\text{Amp}\,\gamma} K_{\text{ED}\,\gamma} K_\gamma}{s(T_\gamma s + 1)}}{1 + \dfrac{K_{\text{Amp}\,\gamma} K_{\text{ED}\,\gamma} K_\gamma (K_{\text{DG}\,\gamma} s + K_{\text{FG}\,\gamma})}{s(T_\gamma s + 1)}} =$$

$$= \frac{K_{\text{Amp}\,\gamma} K_{\text{ED}\,\gamma} K_\gamma}{K_{\text{Amp}\,\gamma} K_{\text{ED}\,\gamma} K_\gamma (K_{\text{DG}\,\gamma} s + K_{\text{FC}\,\gamma}) + s(T_\gamma s + 1)} =$$

$$= \frac{K_{\text{SS}\,\gamma}}{T_{\text{SS}\,\gamma}^2 s^2 + 2T_{\text{SS}\,\gamma} \xi_{\text{SS}\,\gamma} s + 1}$$

— for roll channel.

Here

$$K_{\text{SS}\,\vartheta} = \frac{K_{\text{Amp}\,\vartheta} K_{\text{ED}\,\vartheta} K_\vartheta}{1 + K_{\text{Amp}\,\vartheta} K_{\text{ED}\,\vartheta} K_\vartheta K_{\text{DG}\,\vartheta}}, \quad (19)$$

$$K_{\text{SS}\,\gamma} = \frac{1}{K_{\text{FG}\,\gamma}} \quad (20)$$

are the amplification factors of the oscillatory links in the respective control channels;

$$T_{\text{SS}\,\vartheta} = \frac{T_\vartheta}{\sqrt{1 + K_{\text{Amp}\,\vartheta} K_{\text{ED}\,\vartheta} K_\vartheta K_{\text{DG}\,\vartheta}}}, \quad (21)$$

$$T_{\text{SS}\,\gamma} = \sqrt{\frac{T_\gamma}{K_{\text{Amp}\,\gamma} K_{\text{ED}\,\gamma} K_\gamma K_{\text{DG}\,\gamma}}} \quad (22)$$

are the time constants of oscillatory links in the respective control channels;

$$\xi_{\text{SS}\,\vartheta} = \frac{2\xi_\vartheta T_\vartheta + K_{\text{Amp}\,\vartheta} K_{\text{ED}\,\vartheta} K_\vartheta K_{\text{DG}\,\vartheta} T_{1\vartheta}}{2T_\vartheta \sqrt{1 + K_{\text{Amp}\,\vartheta} K_{\text{ED}\,\vartheta} K_\vartheta K_{\text{DG}\,\vartheta}}}, \quad (23)$$

$$\xi_{\text{SS}\,\gamma} = \frac{1 + K_{\text{Amp}\,\gamma} K_{\text{ED}\,\gamma} K_\gamma K_{\text{DG}\,\gamma}}{2\sqrt{T_\gamma K_{\text{Amp}\,\gamma} K_{\text{ED}\,\gamma} K_\gamma K_{\text{FG}\,\gamma}}} \quad (24)$$

are the damping parameters of oscillatory links in the respective control channels.

These parameters should remain constant throughout the entire flight to ensure the required HS quality. To do this, it is necessary to develop an adaptive system that should tune the SS and HS parameters to ensure this requirement.

For the aircraft considered in this work, the following parameters are fixed with the following values: $\xi_{SS\,\vartheta} = 0.35$; $K_{SS\,\vartheta} = 0.95$ 1/(V · s); $T_{SS\,\gamma} = 0.01$ s; $\xi_{SS\,\gamma} = 0.35$.

Based on the accepted fixed parameters, the system of equations (19), (21), (23) is closed with respect to three unknowns $K_{\text{Amp}\,\vartheta}$, $K_{\text{ED}\,\vartheta}$, $K_{\text{DG}\,\vartheta}$; the system of equations (20), (22), (24) is closed with respect to the unknowns $K_{\text{Amp}\,\gamma}$, $K_{\text{ED}\,\gamma}$, $K_{\text{FG}\,\gamma}$, $K_{\text{DG}\,\gamma}$.

Based on the fact that the controls deviation equations (17), (18) include variables determined earlier, let us write the final expressions to calculate the coefficients:

$$K_{1\,\vartheta} = \frac{K_{SS\,\vartheta}(1 + K_{2\,\vartheta}K_{\vartheta})}{K_{\vartheta}} K_{\phi} K_{C\,\vartheta}; \tag{25}$$

$$K_{2\,\vartheta} = \\ = \frac{-2T_{\vartheta}\left(\xi_{\vartheta}T_{1\,\vartheta} - \xi^{2}_{SS\,\vartheta}T_{\vartheta} - \sqrt{\xi^{4}_{SS\,\vartheta}T^{2}_{\vartheta} - 2\xi_{\vartheta}\,\xi^{2}_{SS\,\vartheta}T_{\vartheta}T_{1\,\vartheta} + T^{2}_{1\,\vartheta}\xi^{2}_{SS\,\vartheta}}\right.}{K_{\vartheta}T^{2}_{1\,\vartheta}} \tag{26}$$

and roll

$$K_{1\,\gamma} = \frac{2\xi_{SS\,\vartheta}T_{\vartheta} - T_{SS\,\vartheta}}{K_{\gamma}T_{SS\,\vartheta}}; \tag{27}$$

$$K_{2\,\gamma} = \frac{T_{\vartheta}}{K_{\gamma}T^{2}_{SS\,\vartheta}}. \tag{28}$$

Parameters of the transfer functions that describe the aircraft properties are related to the dynamic coefficients by the following calculated dependencies [17]:

$$K_{\vartheta} = \frac{a_{12}a_{43} - a_{13}a_{42}}{a_{12} + a_{11}a_{42}};\; T_{1\,\vartheta} = \frac{-a_{13}}{a_{12}a_{43} - a_{13}a_{42}};$$
$$T_{\vartheta} = \frac{1}{\sqrt{a_{12} + a_{11}a_{42}}};\; \xi_{\vartheta} = \frac{a_{11} + a_{42}}{2\sqrt{a_{12} + a_{11}a_{42}}}$$

— for pitch channel;

$$K_{\vartheta} = -\frac{c_{13}}{c_{11}};\; T_{\vartheta} = \frac{1}{c_{11}}$$

— for roll channel.

Calculated dependences for the a_{ij}, c_{ij} dynamic coefficients obtained after linearizing the motion equations based on the aircraft state parameters are provided in [17, 18]. Thus, the resulting system implements automatic search for the required value of the controlled value under the object changing external operating conditions according to the described algorithm. The main difficulty lies in the fact that such a system is applicable only in the case, when the object properties and external disturbances are well known. If parameters of the object itself are not known reliably enough and if they could change randomly within certain limits during operation, the controller parameters could only be selected approximately [14, 17]. In addition, the use of such a principle in determining the SS parameters throughout the entire controlled flight requires mandatory presence on board of special equipment that determines the aircraft speed, the angles of attack and slip, the current values of the speed of sound and of the atmosphere density. As of today, it is technically unrealizable for the aircraft of such a class.

In this regard, another approach is proposed based on indirect assessment of the aircraft state as a control object. To implement this approach to determining the state parameters, it is proposed to use an accelerometer sensor oriented along the aircraft longitudinal axis, which determines the axial accelerations. The signal from the axial accelerometer output is transmitted to the onboard computer, where according to the dependences of the following form:

$$K_{1\vartheta,\,2\vartheta,\,1\gamma,\,2\gamma} = K_{1\vartheta,\,2\vartheta,\,1\gamma,\,2\gamma}(j_X(t)) \tag{29}$$

coefficients are formed that are part of the control equations.

To determine dependencies (29), it is proposed to approximate the corresponding expressions for the control coefficients depending on the axial acceleration.

Direct implementation of this technique is reduced to transferring the measured axial acceleration from the accelerometer sensor to the onboard computer, which, in turn, controls the regulator devices of the SS elements amplification factor level. These devices are inertial. However, their inertia is not taken into account, since $j_X(t)$ is a slowly varying function, when simulating the HS dynamics (see Fig. 2).

The curves obtained as a result of calculating the coefficients according to the presented technique and the curves approximating the obtained dependences are shown in Fig. 5.

Dependences of control coefficients on the axial acceleration have the following form (see Fig. 5):

$$K_{1\vartheta} = \begin{cases} 1185.0872\, j_X^{-1.1741}, & \dot{j}_X > 0; \\ -0.00057668\, j_X + 3.1125589, & \dot{j}_X < 0; \end{cases}$$

$$K_{2\vartheta} = \begin{cases} 14.4398\, j_X^{-1.5}, & \dot{j}_X > 0; \\ -0.000203\, j_X + 0.110762, & \dot{j}_X < 0; \end{cases} \qquad (30)$$

$$K_{1\gamma} = \begin{cases} 0.0000166\, j_X - 17.2986719, & \dot{j}_X > 0; \\ 0.0000019\, j_X - 17.2928544, & \dot{j}_X < 0; \end{cases}$$

$$K_{2\gamma} = -31.4522.$$

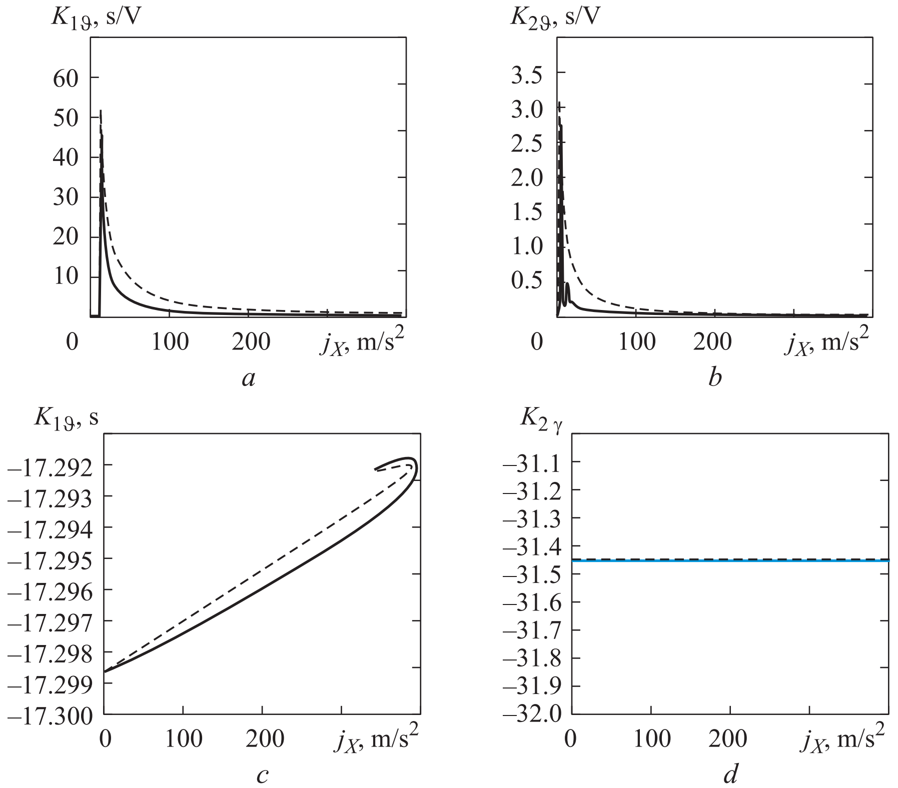

Fig. 5. True (solid curves) and approximated (black dasheds (*a–c*), blue (*d*)) dependences of control coefficients on the axial acceleration:
a — $K_{1\vartheta}(j_X)$; *b* — $K_{2\vartheta}(j_X)$; *c* — $K_{1\gamma}(j_X)$; *d* — $K_{2\gamma}(j_X)$

The polynomials order for approximating the corresponding functions is chosen iteratively based on the trajectory analysis simulated for one or another parameter. The order should be minimum possible, but able to generalize the pattern with the required accuracy level.

To evaluate effectiveness of the created adaptive control system, it is necessary to calculate and compare the aircraft reachability areas obtained for cases, where the control coefficients are determined according to formulas (25)–(28), and where they are functions of only the axial acceleration, i.e., they are calculated during the flight using the system (30).

Target engagement zones are presented in Fig. 6 for the cases of calculating the control coefficients according to the developed method using formulas (25)–(28), as well as for expressions (30) obtained by approximating the control coefficients depending on the axial acceleration. Since a specific warhead is not considered in the simulation, the value of the maximum permissible miss at the target, taken in all calculations equal to 5 m, was chosen as a conditional characteristic of hitting the target. In accordance with the zones in Fig. 5, the engagement areas obtained using the adaptive technique in determining the control coefficients have similar sizes in range and direction, which indicates effectiveness of introducing the considered approach to determining the aircraft guidance and stabilization coefficients.

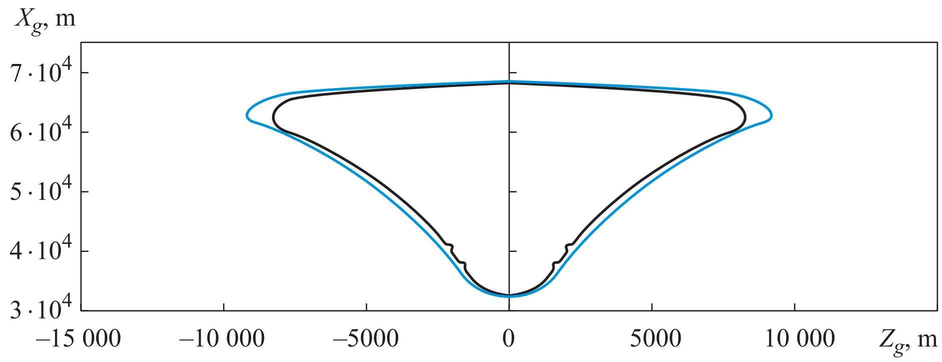

Fig. 6. Target engagement zones according to analytical formulas (black curve) and approximated expression dependences (blue curve) for calculating the control coefficients

An important role in creating the described technique of the coefficients self-tuning is played by the dependences approximation accuracy shown in Fig. 5. This directly affects coincidence of the engagement zones (Fig. 7).

Due to the fact that some of these dependencies (see Fig. 5) are not unambiguous, it is advisable to apply the classical methods of curve approximation using information about the derivative of the function, or to apply the least squares method [19].

The first proposed option is used in the considered example. When the dj_X / dt derivative sign is changed, the coefficient dependences have different

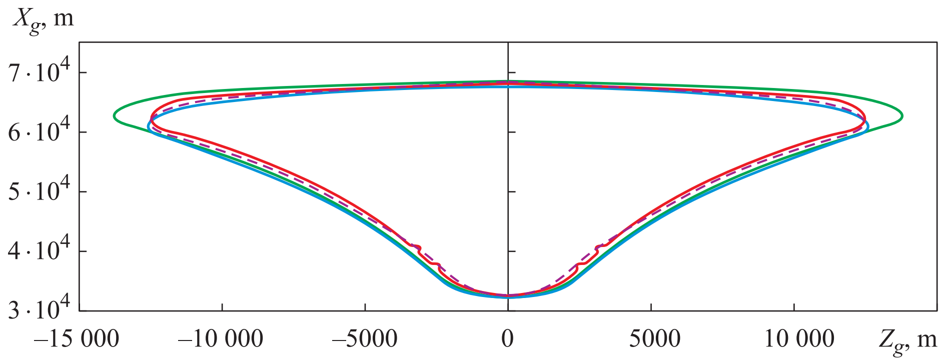

Fig. 7. Target engagement zones according to analytical formulas (red curve), approximation by power function and first-order, second-order Newton's polynomials (green, blue curves) and approximation by the least square method (dashed) for determining the parameters of the adaptive guidance system and of the control coefficients

character making it possible to divide them into two sections. The $dj_X / dt \geq 0$ derivative is on the first, and the $dj_X / dt < 0$ derivative is on the second.

Using this approximation method requires continuous derivative calculation of the axial acceleration during the flight and switching from one control law to another depending on the specified value. Introduction of this principle makes it possible to improve regulation quality in the HS (see Fig. 7).

Conclusion. A technique for controlling amplification factor of the control system elements was prepared for the homing phase compensating nonlinearity and nonstationarity of the system under consideration. Based on the described technique, its introduction on board the aircraft using the accelerometer sensor that determines axial acceleration is proposed. Such an implementation is quite simple in execution, and does not impose additionally high requirements to the onboard computer.

The conducted study demonstrates consistency of the considered guidance approach and determines feasibility of further research in determining the guidance system structure based on information obtained from various instrument sensors that are part of the onboard measurement system, since this technique is one of the possible examples of implementation of the developed adaptation algorithm.

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