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THE USE OF GYROSCOPIC EXECUTIVE SYSTEM FOR HOMING OF THE MISSILE ON THE AERIAL TARGET

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Abstract: *The paper considers the possibility of use of the executive system for missile flight control in the form of fast rotating rotor, versatile suspended in the rocket body. The constrained deviations of the rotor axis (gyroscope) relative to the body axis generate the moments of force which change the missile flight direction and cause its homing on the moving target. Some of the numerical simulation results were presented in the graphic form.*

Keywords: Homing, Missile, Control, Rotor, Gyroscope, Target.

1. Introduction

One of the most important problems in the analysis of the anti-aircraft missile homing systems is the choice of the method for homing the missile to the meeting point with the target. It is equivalent to the choice of the missile flight path determined by so-called homing algorithm, namely the equation describing constraints folded on the rocket motion. In theory there is an infinite number of possibilities for formulating such algorithms. However, only the algorithms, which completed lots of additional and necessary conditions should be chosen (like for example minimal overload, minimal power demand, ease of homing algorithm realization, etc.). In general the homing algorithm formulating is very complex task, which in the most cases can only be solved with use of digital methods. It is caused by the complex rocket dynamics equations, control circuit dynamics, homing executive element and other. In the paper the attempt was made to analyze one of the possibilities for anti-aircraft missile flight control by means of the method similar to proposed in the patent Epperson (1984) and in the paper Osiecki (2001).

2. The description of the executive anti-aircraft missile control element

Inside the missile the bulky rotor (Fig. 1) suspended on the Cardan's joint is placed. Before the shot the rotor is put into rotating motion in relation to the missile body by means of externally charged electric motor. The control of missile flight happens by acting on the rotor suspension with four pneumatic actuators, activated from gas generators by autopilot. The pneumatic actuators are pairwise placed in two planes perpendicular to each other. The actuators deviate the rotor axis in relation to missile body, what causes that the rotor takes features of gyroscope. In the same time when the gyroscope axis direction changes, the moments of gyroscopic forces appear, which act on the missile body and proper change of its attack angles and sideslip angles, and thereby on change of missile flight direction. The missile realizes homing on the target, by use of the optical homing head (Gapiński et al., 2014) with autopilot (Koruba, 2015). The difference in relations to well-known solutions applied in anti-aircraft rockets is the executive system, which does not realizes homing by means of aerodynamic forces, but of inertia force. Before the shot of the missile the rotor is brought up to specific speed with electric motor, and then the motor is disconnected, but the rotor is still rotating with free motion (it is enough for several seconds missile flight).

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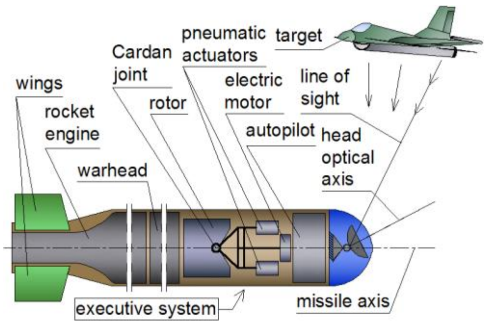

Fig. 1: General diagram of elements of the proposed guided missile.

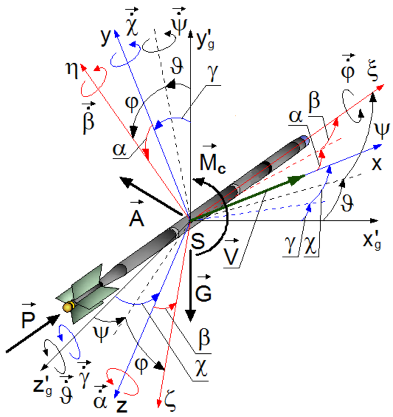

Fig. 2: Coordinate system with angles of rotation.

3. The missile flight equations

Fig. 2 summarizes the coordinate systems in which the missile flight equations were introduced and the forces that act on the moving missile. The following symbols are used (Koruba, 2015): α , β – attack angle and sideslip angle [rad]; ψ , ϑ , φ – pitch angle, yaw angle and roll angle of the missile [rad]; γ , χ – flight-path angle in vertical plane and horizontal plane – pitch angle and yaw angle of missile velocity vector [rad], $S\xi\eta\zeta$ – coordinate system for the missile; $Sxyz$ – velocity coordinate system; $Sx_g^{'}y_g^{'}z_g^{'}$ – coordinate system with the missile as an origin, parallel with the starting system $\vec{V}$ – missile velocity vector; $\vec{P}$ - thrust; $\vec{A}$ – resultant of aerodynamic forces; $\vec{G}$ – gravitational force; $\vec{M}_c$ – control moment; ε, σ – pitch and yaw angles of the line-of-sight (LOS) [rad].

For the purposes of these calculations, the missile is assumed to be a rigid body which does not rotate around its longitudinal axis. With these assumptions applied, the missile dynamic equations are as follows (Krzysztofik, 2014 and Grzyb, 2012):

$$
\begin{gathered}
\dot{V} = \frac{P}{m}\cos\alpha\cos\beta - g\sin\gamma\cos\chi - \lambda_x V^2, \qquad V\dot{\gamma}\cos\chi = \frac{P}{m}\sin\alpha\cos\beta - g\cos\gamma + \lambda_y V^2 \alpha \\
-V\dot{\chi} = -\frac{P}{m}\sin\beta - g\sin\gamma\sin\chi + \lambda_z V^2 \beta \\
\dot{\omega}_\zeta + \left(\frac{J_{ok}}{J_k} - 1\right)\omega_\eta \omega_\xi = -D_1 \frac{V^2}{L}\alpha - D_2 V\dot{\alpha} - D_3 V\dot{\vartheta} + \frac{M_{c\zeta}}{J_k} \\
\dot{\omega}_\eta - \left(\frac{J_{ok}}{J_k} - 1\right)\omega_\xi \omega_\xi = -D_1 \frac{V^2}{L}\beta - D_2 V\dot{\beta} - D_3 V\dot{\psi} + \frac{M_{c\eta}}{J_k} \\
\omega_\xi = -\dot{\vartheta}\sin\psi, \qquad \omega_\eta = \dot{\psi}, \qquad \omega_\zeta = \dot{\vartheta}\cos\psi, \qquad \lambda_y = \lambda_z
\end{gathered} \tag{1}
$$

where: L – length of the missile body [m]; m – mass of the missile [kg]; J_{ok} , J_k – moments of inertia of the missile in relation to its transverse axis and longitudinal axis [kg.m^2]; $M_{c\eta}$, $M_{c\zeta}$ – missile flight control moments [N.m]; g – acceleration of gravity [m/s^2]; $\lambda_x, \lambda_y, D_{1,2,3}$ – relative aerodynamic coefficients of aerodynamic forces and moments [1/m].

Kinematic relationships between the missile and target have the form (Koruba et al., 2010):

$$
\begin{gathered}
\dot{r} = V_t\left[\cos\chi_t \cos\sigma\cos(\varepsilon - \gamma_t) + \sin\chi_t \sin\sigma\right] - V\left[\cos\chi\cos\sigma\cos(\varepsilon - \gamma) + \sin\chi\sin\sigma\right] \\
r\dot{\varepsilon}\cos\sigma = -V_t \cos\chi_t \sin(\varepsilon - \gamma_t) + V\cos\chi\sin(\varepsilon - \gamma) \\
-r\dot{\sigma} = V_t\left[\cos\chi_t \sin\sigma\cos(\varepsilon - \gamma_t) - \sin\chi_t \cos\sigma\right] - V\left[\cos\chi\sin\sigma\cos(\varepsilon - \gamma) - \sin\chi\cos\sigma\right]
\end{gathered} \tag{2}
$$

where: V_t – target velocity [m/s]; γ_t, χ_t – angles of the target flight-path [rad]; r – distance between the missile and the target [m].

The equations of the rotor axis according to the technical gyroscope theory (Gapiński, 2014) are as follows:

$$J_B(\ddot{\vartheta}_g + \ddot{\vartheta}) + c_B\dot{\vartheta}_g - J_0 n(\dot{\psi}_g - \dot{\psi}) = M_{c\eta}, \quad J_B(\ddot{\psi}_g + \ddot{\psi}_g) + c_C\dot{\psi}_g + J_0 n(\dot{\vartheta}_g - \dot{\vartheta}) = M_{c\zeta} \qquad (3)$$

where: ϑ_g, ψ_g – pitch and yaw angles of rotor axis [rad]; c_B, c_C – coefficient of viscous friction in the gimbal bearings [N.m.s]; n – rotational speed of the rotor [1/s]; J_0 – moment of inertia of the rotor in relation to its rotation axis [kg.m^2]; J_B – moment of inertia of the rotor in relation of the lateral axis running through the centre mass [kg.m^2].

The control moments were formulated according to the algorithm of proportional navigation (Yanushevsky, 2011 and Takosoglu, 2016b).

4. Digital simulation results

Numerical simulations were conducted for a hypothetical missile attacking an aerial target from the front hemisphere. The following numerical values were used: starting missile position: x_{m0} = 0 [m], y_{m0} = 0 [m], z_{m0} = 0 [m]; starting target position: x_{t0} = 4000 [m], y_{t0} = 3000 [m], z_{t0} = 0 [m]; angle of a missile launch: γ_{m0} = 25.41 [deg]; χ_{m0} = 0 [rad]; starting angle of pitch and yaw of a target velocity vector: γ_{t0} = 180 [deg], χ_{t0} = 1.15 [deg]; starting missile velocity: V_0 = 20 [m/s]; target velocity: V_t = const = 300 [m/s]; L = 1.6 [m]; m = 10.8 [kg]; J_{ok} = 0.018 [kg.m^2], J_k = 2.4 [kg.m^2]; λ_x = 0.000171, λ_y = 0.0051 [1/m]; D_1 = 0.081, D_2 = 0.0821, D_3 = 0.00041 [1/m]; J_B = 0.0132 [kg.m^2]; J_0 = 0.00754 [kg.m^2]; $c_B = c_C$ = 0.05 [N.m.s]; P = 3150 [N] for $t <= 1$ [s] and P = 700 [N] for $t > 1$ [s]; t – time. The flight path of the target was described as follows: $\gamma_t(t) = \gamma_{t0} - 0.0001 \cdot t$; $\chi_t(t) = -\chi_{t0} \cdot 0.6 \cdot t^2$.

Graphical presentation of the results is shown in Figs. 3 - 8.

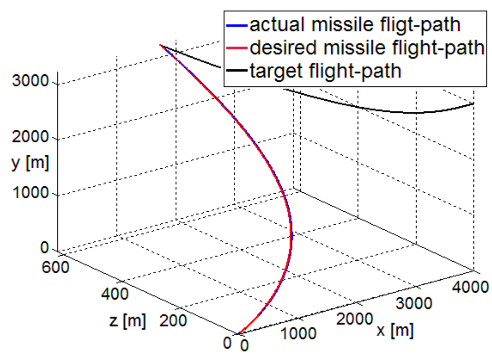

Fig. 3: The missile and target flight-paths.

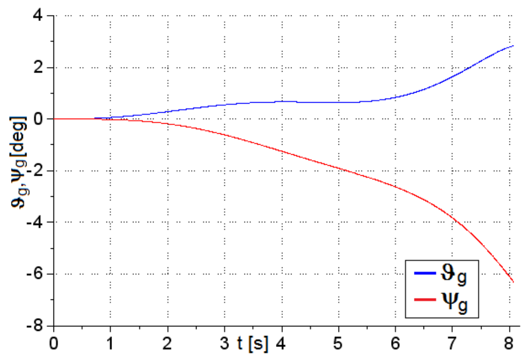

Fig. 4: Pitch and yaw angles of rotor axis.

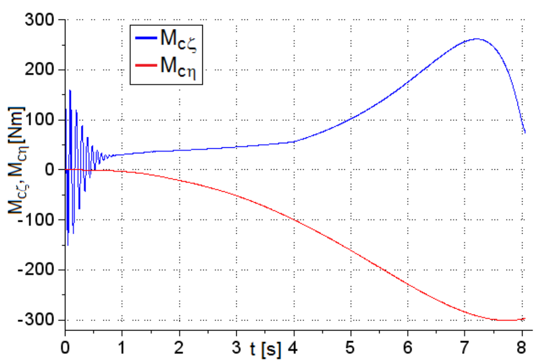

Fig. 5: Values of control moments required for homing the missile on the target.

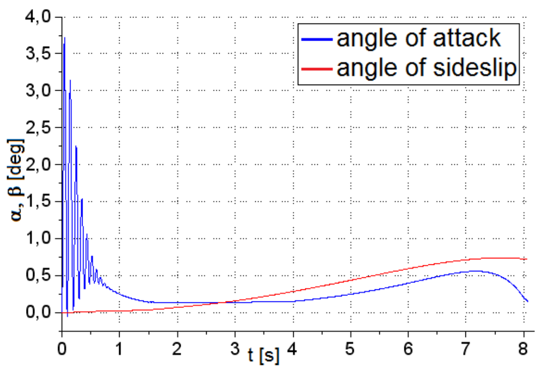

Fig. 6: The angles of attack and angles of sideslip.

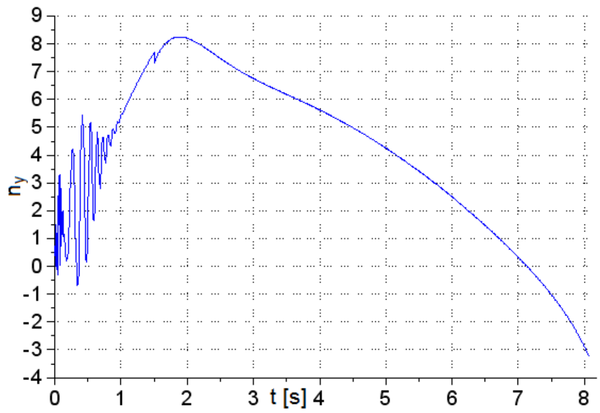

Fig. 7: Lateral overload acting on the missile during the flight.

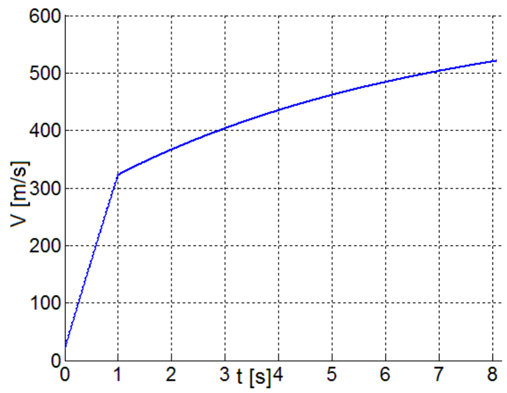

Fig. 8: Flight velocity of the missile.

Numerical simulation results presented above indicate that use of proposed executive system for homing of missile is practicable. Pitch and yaw angles of the rotor axis amount ca. a few degrees and are attainable. Similarly, the values of control moments are moderate.

5. Conclusions

The proposed executive system for control of missile flight is original, but its efficacy is needed to be tested thoroughly. This solution lacks the control forces. Homing is carried out only by the moment of forces. However it also has advantages. The rotor can be placed in any place of the missile body (the action of the couple of forces does not depend on the position). The disadvantage of this solution is the dimension of the rotor, which are limited in relation to its mass. This rotor should have relatively big mass to enable the generating bigger control moments by little pitch angles and yaw angles of its axis. Therefore I suggest the rotor to be made of high density material, e.g. tungsten or depleted uranium.

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