Method of commanding with amplitude pulses for a single-channel missile controller

Pham Cong Tu, Nguyen Quang Vinh*, Phan The Son

Institute of Missile, Academy of Military Science and Technology, 17 Hoang Sam, Cau Giay, Hanoi, Vietnam.

*Corresponding author: vinhquang2808@gmail.com

Received 16 Apr. 2025; Revised 29 May 2025; Accepted 10 Jun. 2025; Published 25 Jun. 2025.

DOI: https://doi.org/10.54939/1859-1043.j.mst.104.2025.15-24

ABSTRACT

Single-channel missile controlling is a method of using a rudder to control the spinning missile to change both pitch and yaw. In essence, the single-channel missile (SCM) control system still follows the dynamic equations of the general aircraft control system, except that the rudder angle signal will simultaneously create control force in both pitch and yaw. Therefore, the method of commanding the single-channel missile rudder has its own characteristics and plays a role in distinguishing the single-channel missile control system from other aircraft control systems. The commanding methods that have been proposed are ON-OFF commanding and continuous sinusoidal signal commanding. This paper presents a method for establishing a single-channel missile control command using the amplitude pulse method. By approaching the desired total control force according to the error, the rudder angle changes according to the amplitude pulse law. This commanding method can be applied to similar lines of rudders, making it more convenient in modern control.

Keywords: Single-channel missile control; Command control; Spinning flight vehicle; Single ON-OFF actuator.

1. INTRODUCTION

Missiles are increasingly diverse and rich in both quantity and type. However, tactical missiles in modern warfare cannot be absent on the battlefield, such as anti-tank missiles (e.g., Maluytka and Refleks) and shoulder-launched missiles (e.g., Igla and Stinger). These missiles often require high mobility; the body length is much larger than the diameter, the stabilizer is designed to be small and always spin around the body while flying, and it cannot be fixed to control independently according to each pitch and yaw channel. The single-channel missile control system will be responsible for controlling the aircraft to change simultaneously for both pitch and yaw. Taking on this special task in the single-channel missile control system is the control command system, the core of which is the command method. The command system of the single-channel missile is implemented as a MISO system, in which the input signal is the error according to the pitch channel and yaw channel, and the output signal is the steering control signal. In the past, there have been published research works about the method of establishing a command for a single-channel missile controller or related to the spinning flight vehicle command system. In [1, 4] propose a command method for the ON-OFF actuator, and in [9] propose a command method for the linear steering gear. Specifically, in [1], [2], and [8], a command method for the ON-OFF actuator is established; it generates the output signal changes from a positive maximum level or a negative maximum level, equivalent to creating a maximum opening angle of the rudder in both directions ($+\delta_{max}$ and $-\delta_{max}$). The control torque acting on the aircraft is calculated by the average in a rotation cycle of the aircraft. The solution for the average torque in a rotation cycle to have a magnitude that varies according to the input error of the range channel and direction channel is implemented by a linear signal with a frequency twice the rotation frequency around the longitudinal axis of the aircraft. This commanding method is only applicable to the ON-OFF actuator, and when the control error is zero, the instantaneous control torque still exists, so the aircraft will fly in a spiral. In the work

[3], the commanding method is implemented by generating a square pulse signal to the ON-OFF actuator in each rotation cycle of the SCM. The width of the square pulse varies according to the mean square of the error signal in the direction channel and the range channel; the phase is taken as the ratio of the error signal between the two channels. This method allows the control of the SCM with the ON-OFF actuator and still ensures that when the zero deviation, the control signal is zero; however, the maximum control moment is only implemented in half of the rotation cycle of the SCM. The single-channel missile commanding method, according to the continuous function [9], generates a sine wave signal with a frequency equal to the rotation frequency around the longitudinal axis of the aircraft, and the phase and amplitude of the sine wave signal are determined by the pitch and yaw error signal. This method has the advantage of bringing the instantaneous control torque to zero when the control error is zero, but the maximum combined torque in a rotation cycle of the aircraft is smaller than the ON-OFF control signal generation actuator, which is less effective. With the desire to propose a command establishment method that can solve both the advantages of the above two methods. We have researched and proposed a method of establishing control commands according to pulse amplitude that can be applied to a linear actuator. With this control method, the signal given by the controller has an amplitude and pulse width that vary according to the square of the error of the pitch channel and the yaw channel, and the initial phase is established according to the ratio between the error of the pitch channel and the yaw channel. The method also provides a specific calculation formula for command formation and the maximum oscillation amplitude caused by the trajectory deviation, thereby serving as a basis for developing and applying modern control methods for spinning flight vehicles.

2. METHOD OF CONTROL COMMAND

Consider the controlled movement of a flying device in space (figure 1).

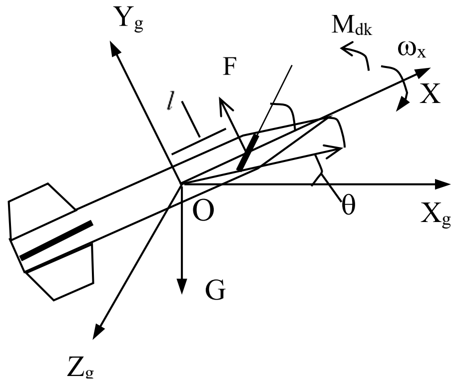


Figure 1. *Control forces and moments acting on an aircraft.*

O - center of gravity; $OX_gY_gZ_g$ - fixed coordinate system; ω_x - angular velocity of the aircraft around the longitudinal axis; F - aerodynamic force acting on the rudder; M_{dk} - control torque; α - angle of attack; θ - pitch angle; δ - rudder angle.

Suppose that the control of a spinning flight vehicle is proportional to some quantity of pitch error Δy and yaw error Δz. The values of Δy and Δz error can be information provided by the control station or by the homing head. To control the SCM, it is necessary to create forces that cause a torque to change the direction of the SCM with the following values:

$$F_{\Delta y} = k_1 \cdot \Delta y \tag{1}$$

$$F_{\Delta z} = k_2 \cdot \Delta z \tag{2}$$

The force applied at a point (generated by a rudder) and the total effect on the aircraft will be:

$$\vec{F}_{\Delta} = \vec{F}_{\Delta y} + \vec{F}_{\Delta z} \tag{3}$$

Since $\overrightarrow{F_{\Delta y}}$ is perpendicular to $\vec{F}_{\Delta z}$, the total force has the value

$$F_{\Delta} = \sqrt{F_{\Delta y}^2 + F_{\Delta z}^2} = \sqrt{(k_1 \Delta y)^2 + (k_2 \Delta z)^2} \tag{4}$$

If we call the angle between the horizontal force component vector and the resultant vector as φ_{Δ}, then we have

$$F_{\Delta y} = F_{\Delta} \sin \varphi_{\Delta} \tag{5}$$

$$F_{\Delta z} = F_{\Delta} \cos \varphi_{\Delta} \tag{6}$$

$$\varphi_{\Delta} = \arctan \frac{F_{\Delta y}}{F_{\Delta z}} = \arctan \frac{k_1 \Delta y}{k_2 \Delta z} \tag{7}$$

Thus, it is necessary to synthesize the control command δ so that the resultant force is equal to the value F_{Δ}, and the angle between it and the horizontal is. With the initial rudder plane lying on the XOZ plane, this can be done by creating the rudder opening angle at time τ with the normal vector making an φ angle with the horizontal with the value:

$$\varphi = \varphi_{\Delta} - \frac{\pi}{2} = -\operatorname{arccot} \frac{F_{\Delta y}}{F_{\Delta z}} = -\arctan \frac{F_{\Delta z}}{F_{\Delta y}} = -\arctan \frac{k_2 \Delta z}{k_1 \Delta y} \tag{8}$$

While moving, the aircraft spins on the longitudinal axis with ω_x angular velocity, and the rudder plane also rotates. The instantaneous aerodynamic force $\vec{F}$ acting on the rudder creates a total moment controlling $\vec{M}_{dk}$ the aircraft with an instantaneous value:

$$F = C^{\delta} \frac{\rho V^2 S}{2} \delta(t) \tag{9}$$

Where: S - rudder area, ρ - air density, V - aircraft speed, C^{δ} - derivative of the aerodynamic force coefficient of the rudder.

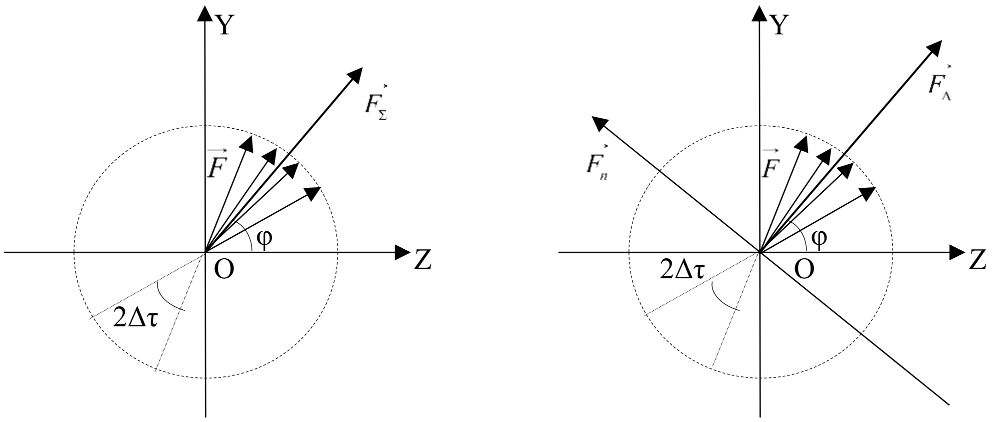


Figure 2. *Control force synthesis.*

The instantaneous aerodynamic force $\vec{F}$ can be divided into two components $\vec{F}_t$ and $\vec{F}_n$ that are perpendicular to each other. The component $\vec{F}_t$ is the projection of $\vec{F}$ onto $\vec{F}_{\Delta}$ (figure 2). Then we have:

$$F_t = C^{\delta}\frac{\rho V^2 S}{2}\delta(t)\cos(\omega_x t-\varphi)$$
$$F_n = C^{\delta}\frac{\rho V^2 S}{2}\delta(t)\sin(\omega_x t-\varphi) \qquad (10)$$

The resultant force in one rotation cycle of the SCM is:

$$F_{\Sigma t} = C^{\delta}\frac{\rho V^2 S}{2}\int\limits_{\omega_x iT}^{\omega_x (i+1)T}\delta(t)\cos(\omega_x t-\varphi)d\omega_x t = C^{\delta}\frac{\rho V^2 S}{2}\omega_x\int\limits_{iT}^{(i+1)T}\delta(t)\cos(\omega_x t-\varphi)dt \qquad (11)$$

$$F_{\Sigma n} = C^{\delta}\frac{\rho V^2 S}{2}\int\limits_{\omega_x iT}^{\omega_x (i+1)T}\delta(t)\sin(\omega_x t-\varphi)d\omega_x t = C^{\delta}\frac{\rho V^2 S}{2}\omega_x\int\limits_{iT}^{(i+1)T}\delta(t)\sin(\omega_x t-\varphi)dt \qquad (12)$$

The component $F_{\Sigma t}$ is the component that creates the normal force to control the SCM, and the component $F_{\Sigma n}$ is the component that causes disturbances to the aircraft. Thus, the problem of generating commands for the aircraft must ensure that $F_{\Sigma t} = F_{yz}$ và $F_{\Sigma n} = 0$.

It can be done by creating a solution to open the rudder angle $\delta(t)$ according to the following rule:

$$\delta(t) = \begin{cases} +\delta;\ \ iT+(\tau-\Delta\tau)\le t\le iT+(\tau+\Delta\tau) \\ 0; iT+(\tau+\Delta\tau)< t< iT+\dfrac{T}{2}+(\tau-\Delta\tau); iT+\dfrac{T}{2}+(\tau+\Delta\tau)\le t\le iT+1+(\tau-\Delta\tau); \\ -\delta;\ \ iT+\dfrac{T}{2}+(\tau-\Delta\tau)\le t\le iT+\dfrac{T}{2}+(\tau+\Delta\tau); \end{cases} \qquad (13)$$

Where T is the rotation period of the SCM; $\tau = \dfrac{\varphi}{\omega_x}$; $0\le 2\Delta\tau\le\dfrac{T}{2}$; $-\delta_{gh}\le\delta\le+\delta_{gh}$; δ_{gh} - is the maximum opening angle of the rudder.

Thus, we have the rule for changing the opening angle of the rudder blade shown in figure 3.

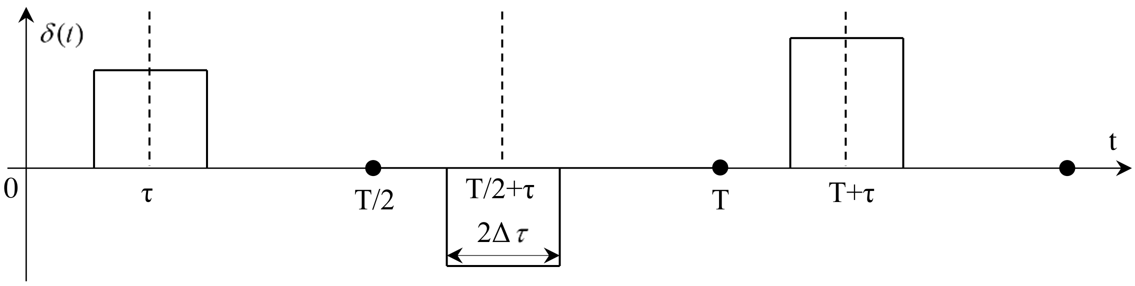


Figure 3. *Rule of changing the rudder angle.*

During the time 2Δτ, the steering angle is considered unchanged.

$$F_{\Sigma t} = C^{\delta}\frac{\rho V^2 S}{2}\omega_x\left(\int\limits_{iT+(\tau-\Delta\tau)}^{iT+(\tau+\Delta\tau)}\delta\cos(\omega_x t-\varphi)dt + \int\limits_{iT+\frac{T}{2}+(\tau-\Delta\tau)}^{iT+\frac{T}{2}+(\tau+\Delta\tau)}-\delta\cos(\omega_x t-\varphi)dt\right)$$
$$= C^{\delta}\frac{\rho V^2 S}{2}\omega_x\frac{\delta}{\omega_x}\left(\sin(\omega_x t-\varphi)\Big|_{iT+(\tau-\Delta\tau)}^{iT+(\tau+\Delta\tau)} - \sin(\omega_x t-\varphi)\Big|_{iT+\frac{T}{2}+(\tau-\Delta\tau)}^{iT+\frac{T}{2}+(\tau+\Delta\tau)}\right)$$

$$
\begin{aligned}
&= C^{\delta}\frac{\rho V^2 S}{2}\delta(\sin(\omega_x(iT+(\tau+\Delta\tau)-\varphi)-\sin(\omega_x(iT+(\tau-\Delta\tau)-\varphi)-\\
&-\sin(\omega_x(iT+\frac{T}{2}+(\tau+\Delta\tau)-\varphi)+\sin(\omega_x(iT+\frac{T}{2}+(\tau-\Delta\tau)-\varphi)).\\
&= C^{\delta}\frac{\rho V^2 S}{2}\delta(\sin(i\omega_x T+\omega_x\tau+\omega_x\Delta\tau-\varphi)-\sin(i\omega_x T+\omega_x\tau-\omega_x\Delta\tau)-\varphi)-\\
&-\sin(i\omega_x T+\omega_x\frac{T}{2}+\omega_x\tau+\omega_x\Delta\tau-\varphi)+\sin(i\omega_x T+\omega_x\frac{T}{2}+\omega_x\tau-\omega_x\Delta\tau)-\varphi)).
\end{aligned}
\tag{14}
$$

We have: $\omega_x T = 2\pi;\quad \omega_x\tau = \varphi$ (15)

$$
\begin{aligned}
F_{\Sigma t} &= C^{\delta}\frac{\rho V^2 S}{2}\delta(\sin(i2\pi+\varphi+\omega_x\Delta\tau-\varphi)-\sin(i2\pi+\varphi-\omega_x\Delta\tau-\varphi)-\\
&-\sin(i2\pi+\pi+\varphi+\omega_x\Delta\tau-\varphi)+\sin(i2\pi+\pi+\varphi-\omega_x\Delta\tau-\varphi))).\\
&= C^{\delta}\frac{\rho V^2 S}{2}\delta(\sin(\omega_x\Delta\tau)+\sin(\omega_x\Delta\tau)-\sin(\pi+\omega_x\Delta\tau)+\sin(\pi-\omega_x\Delta\tau)\\
&= C^{\delta}\frac{\rho V^2 S}{2}\delta.4\sin(\omega_x\Delta\tau) = 2C^{\delta}\frac{\rho V^2 S}{\omega_x}\delta\sin\omega_x\Delta\tau
\end{aligned}
\tag{16}
$$

$$
\begin{aligned}
F_{\Sigma n} &= C^{\delta}\frac{\rho V^2 S}{2}\omega_x\left(\int_{iT+(\tau-\Delta\tau)}^{iT+(\tau+\Delta\tau)}\delta\sin(\omega_x t-\varphi)dt+\int_{iT+\frac{T}{2}+(\tau-\Delta\tau)}^{iT+\frac{T}{2}+(\tau+\Delta\tau)}-\delta\sin(\omega_x t-\varphi)dt\right)\\
&= C^{\delta}\frac{\rho V^2 S}{2}\omega_x\frac{\delta}{\omega_x}\left(-\cos(\omega_x t-\varphi)\Big|_{iT+(\tau-\Delta\tau)}^{iT+(\tau+\Delta\tau)}+\cos(\omega_x t-\varphi)\Big|_{iT+\frac{T}{2}+(\tau-\Delta\tau)}^{iT+\frac{T}{2}+(\tau+\Delta\tau)}\right)\\
&= C^{\delta}\frac{\rho V^2 S}{2}\delta(-\cos(\omega_x(iT+(\tau+\Delta\tau)-\varphi)+\cos(\omega_x(iT+(\tau-\Delta\tau)-\varphi)+\\
&+\cos(\omega_x(iT+\frac{T}{2}+(\tau+\Delta\tau)-\varphi)-\cos(\omega_x(iT+\frac{T}{2}+(\tau-\Delta\tau)-\varphi)).\\
&= C^{\delta}\frac{\rho V^2 S}{2}\delta(-\cos(i\omega_x T+\omega_x\tau+\omega_x\Delta\tau-\varphi)+\cos(i\omega_x T+\omega_x\tau-\omega_x\Delta\tau)-\varphi)+\\
&+\cos(i\omega_x T+\omega_x\frac{T}{2}+\omega_x\tau+\omega_x\Delta\tau-\varphi)-\cos(i\omega_x T+\omega_x\frac{T}{2}+\omega_x\tau-\omega_x\Delta\tau)-\varphi)).\\
&= C^{\delta}\frac{\rho V^2 S}{2}\delta(-\cos(i2\pi+\varphi+\omega_x\Delta\tau-\varphi)+\cos(i2\pi+\varphi-\omega_x\Delta\tau-\varphi)+\\
&+\cos(i2\pi+\pi+\varphi+\omega_x\Delta\tau-\varphi)-\cos(i2\pi+\pi+\varphi-\omega_x\Delta\tau-\varphi))).\\
&= C^{\delta}\frac{\rho V^2 S}{2}\delta(-\cos(\omega_x\Delta\tau)+\cos(\omega_x\Delta\tau)+\cos(\pi+\omega_x\Delta\tau)-\cos(\pi-\omega_x\Delta\tau))
\end{aligned}
$$

$$
F_{\Sigma n} = C^{\delta}\frac{\rho V^2 S}{2}\delta(-\cos(\omega_x\Delta\tau)+\cos(\omega_x\Delta\tau)-\cos(\omega_x\Delta\tau)+\cos(\omega_x\Delta\tau)) = 0 \tag{17}
$$

so we have: $F_{\Sigma} = F_{\Sigma t} = 2C^{\delta}\rho V^2 S\delta\sin\omega_x\Delta\tau$ (18)

From (18) it can be seen that F_{Σ} is proportional to the magnitude of the rudder opening angle δ and the rudder opening time Δτ. With $0\le\Delta\tau\le\frac{T}{4}$ equivalent to $0\le\omega_x\Delta\tau\le\frac{\pi}{2}$, the function

$\sin\omega_x\Delta\tau$ is increasing in the above interval. Thus, according to (18), the magnitude of F_Σ varies with the magnitude of the rudder opening angle and the rudder opening time Δτ. F_Σ reaches its minimum value $F_{\Sigma\min}=0$ at $\delta=0$ or $\Delta\tau=0$ and its maximum value $F_{\Sigma\max}=2C^\delta\rho V^2 S\delta_{gh}$ at $\delta=\delta_{gh}$ and $\Delta\tau=\dfrac{T}{4}$.

According to (10), force $\vec{F}$ consists of two components, $\vec{F_t}$ and $\vec{F_n}$. Although the total force in each rotation cycle of the aircraft, component $\vec{F_n}$ is eliminated, at the instantaneous time it causes oscillation for the SCM, so in the process of establishing control commands, it must be taken into account, especially the largest instantaneous value. We have the largest value as:

$$\begin{aligned}\max(F_n)&=\max\left(C^\delta\frac{\rho V^2 S}{2}\delta(t)\sin(\omega_x t-\varphi)\right)\\&=\max\left(C^\delta\frac{\rho V^2 S}{2}\delta(t)\sin\omega_x(t-\tau)\right)\end{aligned}\tag{19}$$

During time $\tau-\Delta\tau\le t\le\tau+\Delta\tau$, the opening angle of the rudder blade is constant $\delta(t)=+\delta$, $-\sin\Delta\tau\le\sin\omega(t-\tau)\le\sin\Delta\tau$, we have:

$$-C^\delta\frac{\rho V^2 S}{2}\delta\sin\omega_x\Delta\tau\le F_n\le C^\delta\frac{\rho V^2 S}{2}\delta\sin\omega_x\Delta\tau\ ;\tag{20}$$

$$\max(F_n)=C^\delta\frac{\rho V^2 S}{2}\delta\sin\omega_x\Delta\tau\tag{21}$$

From (18) we have: $\delta=\dfrac{F_\Delta}{2C^\delta\rho V^2 S\sin\omega_x\Delta\tau}$

Substitute into (21) we have:

$$\max(F_n)=C^\delta\frac{\rho V^2 S}{2}\frac{F_\Delta}{2C^\delta\rho V^2 S\sin\omega_x\Delta\tau}\sin\omega_x\Delta\tau=\frac{F_\Delta}{4}\tag{22}$$

Similarly, during time $\tau-\Delta\tau+\dfrac{T}{2}\le t\le\tau+\Delta\tau+\dfrac{T}{2}$, the opening angle of the rudder blade is constant $\delta(t)=-\delta$, $-\sin\Delta\tau\le\sin\omega(t-\tau)\le\sin\Delta\tau$, we have: $\max(F_n)=\dfrac{F_\Delta}{4}$ (23)

Thus, the maximum magnitude of the oscillation from the control vane is proportional to the total control force acting on the spinning flight vehicle. This is the basis for calculating and controlling the lead error as well as the slip during the control of the spinning flight vehicle.

3. RESULTS AND DISCUSSION

On the basis of generating the control signal for the single-channel missile given above, we can build a flowchart to generate the control signal for the single-channel missile rudder, as shown in figure 4. Based on the errors Δy and Δz obtained from the guidance system, determine the equivalent normal force value F_Δ and φ according to formulas (4) and (33). From the spin angular velocity of the aircraft determined from the gyroscope at each time, calculate T and τ according to formula (15) and calculate δ and Δτ according to formulas (18) and (19). The synthesis of the instantaneous values of T, τ, δ, and Δτ forms a rudder control signal in the form of (13) sent to the steering machine controlling the SCM. This process takes place continuously after each update cycle until there is a The End signal (command to end the control process when the SCM encounters a target or exceeds the specified flight time and must self-destruct).

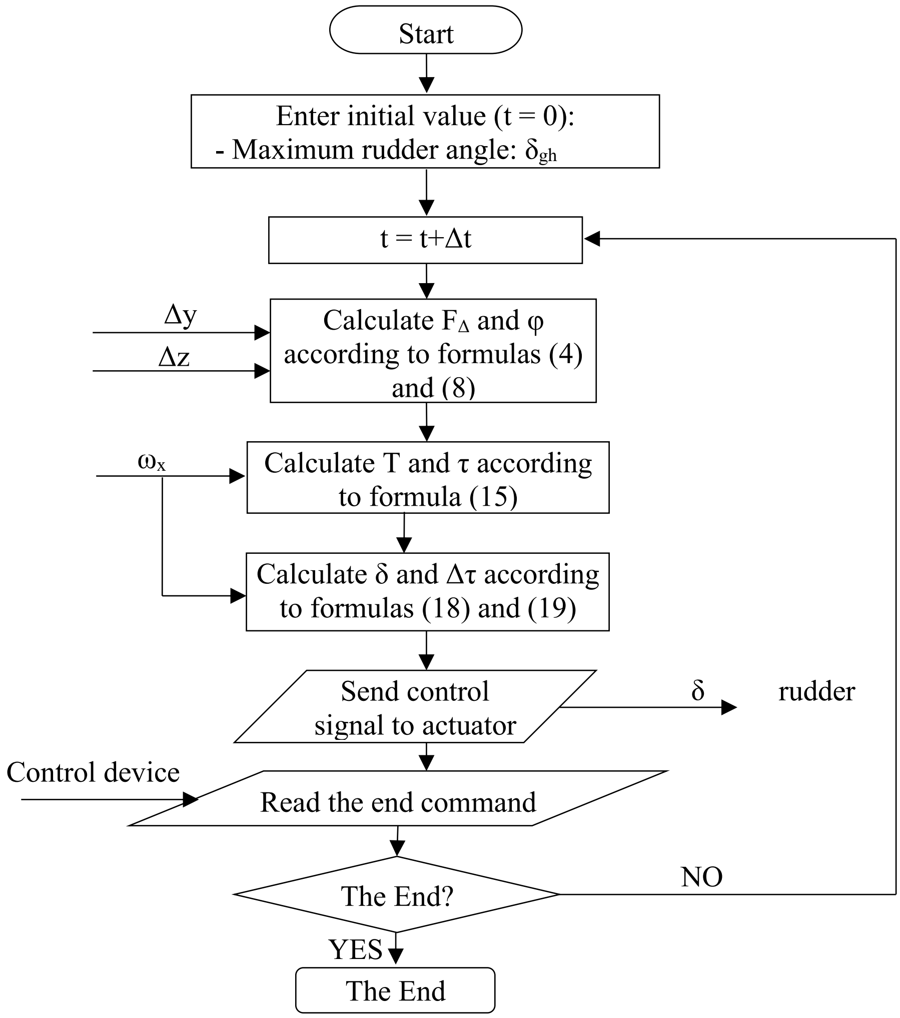


Figure 4. *Flowchart of generating control signal for single-channel aircraft rudder.*

To evaluate this single-channel control command creation method, we apply it to a spinning flight vehicle model with initial mass m_o = 10 kg. The SCM flew at a stable speed of V = 560 m/s and a spin frequency of $f_x = 20$ Hz. The aerodynamic coefficients of the SCM are preliminarily calculated according to the approximate methods of aerodynamics with the mean aerodynamic chord b_a = 0.1 m, the wing area S = 0.01 m^2 and the diameter D = 0.07 m. The moments of inertia will be Jx = 0.007 kg.m^2 and Jy = Jz = 2 kg.m^2. The simulation diagram of single-channel missile control is built on Matlab Simulink as shown in figure 5, the diagram of creating the opening angle of the rudder blade is shown in figure 6. The simulation process is performed with PID control. The simulation results shown in figures 7 to 9 reflect the method of the single-channel missile command. In figures 7a, 8a, and 9a are the results of input and output; figures 7d, 8d, and 9d are the results of rudder opening angle. The results show that the commanding method by this method has controlled a single-channel missile, specifically:

Figure 7 is the simulation result with a fixed target with coordinates Y = 50 m, Z = -20 m. The simulation result shows that the rudder opening angle is initially very large to bring the error in coordinates Y and Z to zero, then the control maintains a small error level. The rudder opening angle still maintains control when the error approaches zero because there is always a missile weight compensation signal.

Similarly, the simulation results in figure 8 were performed with a target with initial coordinates Y = 50 m, Z = 20 m, and a target moving at a constant vertical speed of 2 m/s and a horizontal speed of 4 m/s. The simulation results showed that the missile tracked well with a target moving at a constant vertical and horizontal speed.

Finally, figure 9 is the simulation result with a snake-shaped target moving vertically and horizontally. The target has initial coordinates Y = 50 m and Z = 20 m and moves vertically with a sinusoid $Y = 2\sin(2\pi t)$ and horizontally with $Z = \sin(\pi t)$. The simulation results show that the missile follows the target after 1.25 s. The rudder angle in this case, after the error approaches zero, fluctuates according to the target being followed.

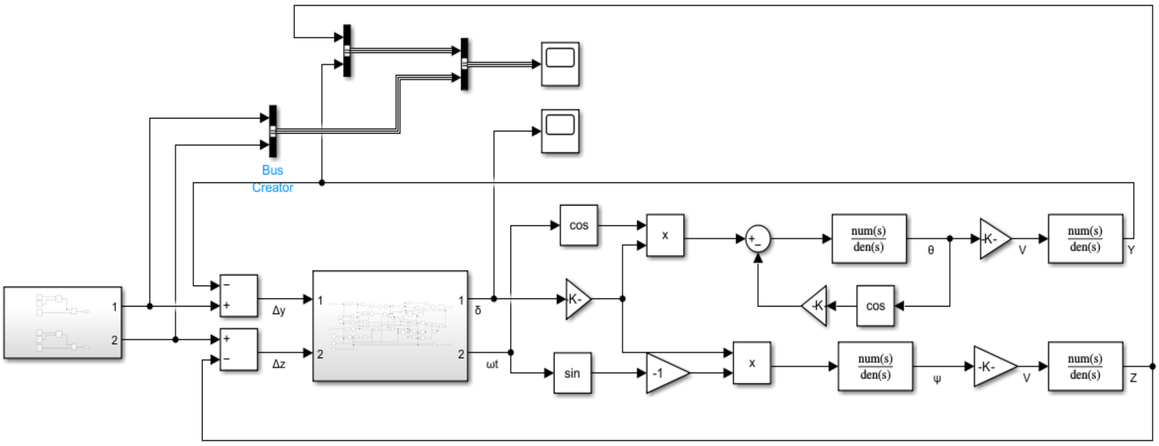


Figure 5. *Simulation diagram of single-channel aircarft control.*

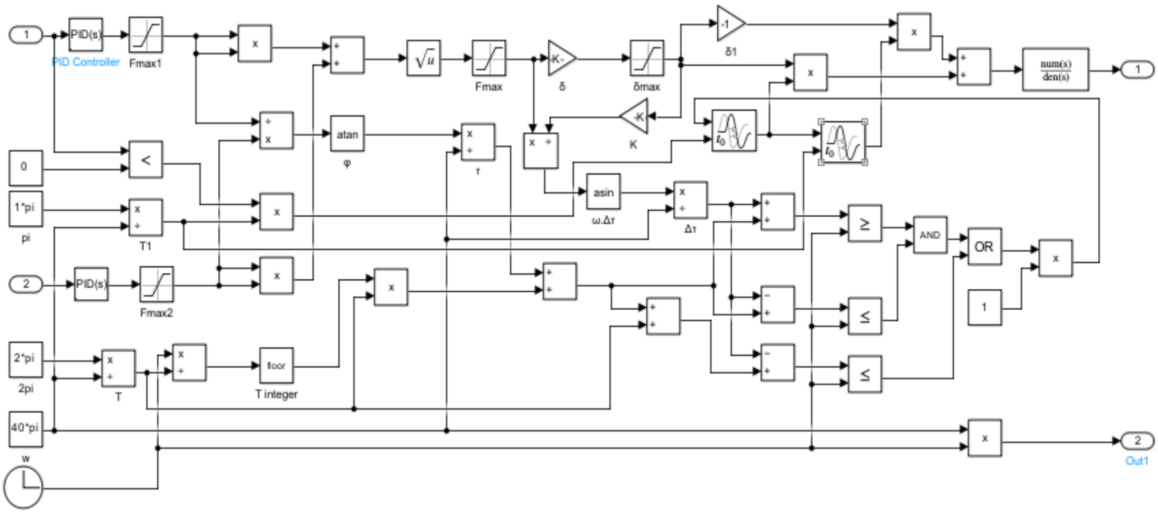
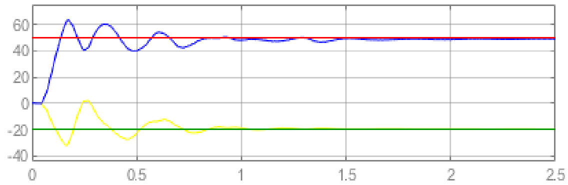


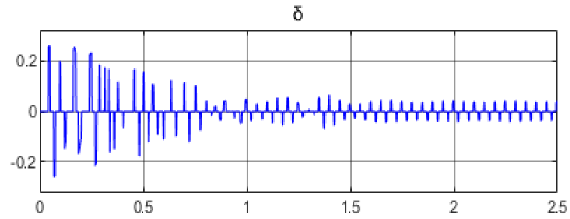
Figure 6. *Simulation diagram of single-channel missile control command.*

Color symbols on graphs a and b:

Target Y Coordinate	Spinning flight vehicle Y coordinate.
Target Z Coordinate	Spinning flight vehicle Z coordinate



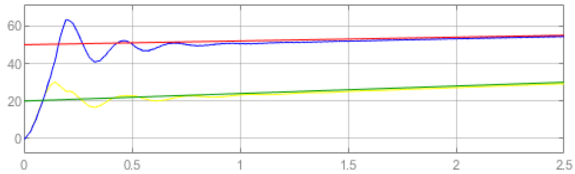
a)



b)

Figure 7. *Target coordinates Y = 50, Z = -20.*

a)



b)

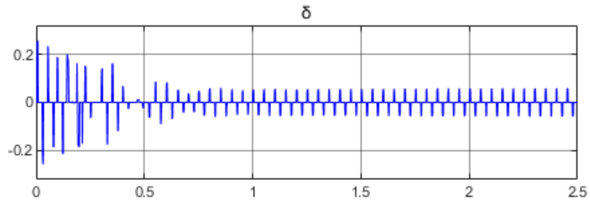
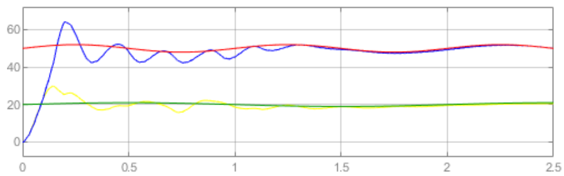


Figure 8. *Target coordinates Y = 50+2t, Z = 20+4t.*

a)



b)

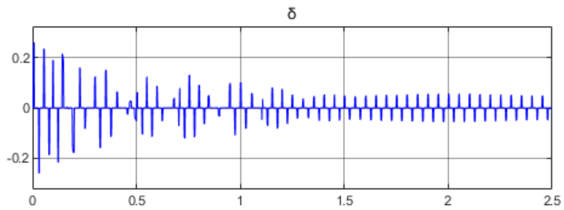


Figure 9. *Target coordinates Y = 50+2sin(2πt), Z = 20+sin(πt).*

4. CONCLUSIONS

The paper has developed a method for establishing a single-channel amplitude pulse command to control a spinning flight vehicle. Constructed a flowchart to generate a control signal for a single-channel missile rudder. Simulation evaluation of the single-channel missile control command formulation method according to pitch and yaw channel input error proportional and PID.Compared with the previous control command method, this method allows output control signals with variable amplitude and pulse width, thereby opening the largest control torque equal to the largest control torque of the ON-OFF control command method, and the smallest control torque is 0 when the input error is 0. The survey results demonstrated the correctness of the research method. Based on the research results, we will conduct research to select the optimal parameters δ and Δτ applied to the linear steering machine as well as apply to the specific guidance method for spinning flight vehicles to increase the guidance accuracy and reduce slip when controlling the SCM to follow the target.

REFERENCES

[1]. Botez, R.M.; Chelaru, V.; Parvu, P., Gheorghe, C., *"Calculus model for a rolling guided missile"*. J. Vib. Control, 7, 863–889, (2001).

[2]. Suiçmez, E.C.; Kutay, A.T., *"Single channel digital controller design for a high spinning rate rolling airframe missile", AERONAUTICAL JOURNAL,* vol. 126, no. 1305, pp. 1815-1833, (2022).

[3]. P. C. Tu, N. Q. Vinh, N. V. Son, *"Method of control of single channel aircraft by unipole relay steering gear",* Journal of Military Science and Technolog, (2024), DOI: https://doi.org/10.54939/1859-1043.j.mst.FEE.2024.12-19.

[4]. J. Karimi, *"A new closed form solution for dynamic stability analysis of rolling airframes having one pair ON-OFF actuator"*. Aviation 25(2):92-103, (2021). DOI: 10.3846/aviation.2021.13832

[5]. Chelaru, T.-V.; Constantinescu, C.E.; Pană, V.; Ene, C.; Chelaru, A, *"Stability of Single-Channel Homing Rolling Aerospace Vehicle"*. Aerospace, 11, 660, (2024).

[6]. Koohmaskan, K., Arvan, M. R., Vali, A. R., & Farid, B., "*Dynamic stability conditions for a rolling flight vehicle applying continuous actuator. Aerospace Science and Technology"*, 42, 451–458, (2015). https://doi.org/10.1016/j.ast.2015.02.007

[7]. H Nobahari; H Mohammadkarimi, *"Multiple-input describing function technique applied to design a single channel ON-OFF controller for a spinning flight vehicle"*, Proc. IMechE Vol. 226 Part G: J. Aerospace Engineering, (2012). DOI: 10.1177/0954410011414521.

[8]. Trần Đức Thuận, Nguyễn Hải Long, Đỗ Hoàng Chính, Phan Văn Từ, Nguyễn Sỹ Long, Nguyễn Văn Sơn, *"Xây dựng mô hình và phương pháp xác định tham số cho hệ thống điều khiển TBB một kênh"*, Tạp chí NCKHKT&CN của TT KHKT&CNQS, Số 13, (2005) (in Vietnamese).

[9]. Phan Văn Từ, *"Xây dựng phương pháp và thuật toán điều khiển cho thiết bị bay một kênh"*, Luận án tiến sỹ, (2010) (in Vietnamese).

[10]. Nguyễn Đức Cương, *Mô hình hoá và mô phỏng chuyển động của khí cụ bay tự động*, Sách chuyên khảo Trung tâm KHKT-CNQS, NXB Quân đội nhân dân, Hà nội, (2002) (in Vietnamese).

[11]. А. А.Лебедев, Л. С. Чернобровкин, *"Динамика полёта беспилотных летательных аппаратов"*, Машиностроение, Москва, (1973).

[12]. Пупков. К. А., Егупов Н.Д. *"Высокоточные системы самонаведения"*, Изд-во: Москва Физматлит, ISBN-13: 978-5922113113, (2011).

TÓM TẮT

Phương pháp lập lệnh điều khiển thiết bị bay một kênh xung biên độ

Điều khiển tên lửa một kênh là phương pháp sử dụng một máy lái để điều khiển thiết bị bay (TBB) thay đổi cả tầm và hướng. Về bản chất hệ thống điều khiển tên lửa một kênh vẫn tuân thủ theo các phương trình động học của hệ điều khiển TBB nói chung chỉ khác là tín hiệu góc mở cánh lái đồng thời sẽ tạo ra lực điều khiển theo cả tầm và hướng. Do đó, phương pháp lập lệnh điều khiển máy lái tên lửa một kênh có đặc điểm riêng biệt và đóng vai trò phân biệt giữa hệ thống điều khiển tên lửa một kênh so với hệ thống điều khiển TBB khác. Các phương pháp lập lệnh đã được đưa ra là lập lệnh ON-OFF và lập lệnh tín hiệu liên tục hình sin. Bài báo trình bày phương pháp lập lệnh điều khiển tên lửa một kênh theo phương pháp xung biên độ. Bằng cách tiếp cận từ lực điều khiển tổng hợp mong muốn theo sai số tạo ra góc mở cánh lái thay đổi theo quy luật xung biên độ. Phương pháp lập lênh này có thể áp dụng cả cho dòng máy lái tương tự giúp thuận lợi hơn trong điều khiển hiện đại.

Từ khóa: Điều khiển một kênh; Thiết bị bay một kênh**;** Lập lệnh điều khiển; Máy lái rơ le.