A study on developing a computational model for determining operating parameters of a dual-flow solid rocket engine

Mai Duy Phuong, Bui Dinh Tan*, Duong Quoc Viet

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

*Corresponding author: buidinhtantb@gmail.com

Received 26 Oct. 2024; Revised 24 Dec. 2024; Accepted 5 Feb. 2025; Published 25 Feb. 2025.

DOI: https://doi.org/10.54939/1859-1043.j.mst.101.2025.140-147

ABSTRACT

The aerodynamic launch principle operates with a complex mechanism, combining both artillery-like principles and solid-fuel engine concepts. Calculating and determining the working parameters of the launch motor is a crucial step in designing launch systems, especially when researching, developing, and manufacturing new systems. This is a mandatory requirement to ensure the functionality and reliability of the equipment. This paper focuses on developing a physical model to calculate the operating parameters of a dual-flow solid rocket engine over time, based on a set of hypothetical system parameters. The computational model presented in this paper, along with the resulting calculations, supports the investigation and design of systems while serving as a theoretical foundation for further, more comprehensive research. It also provides a basis for applications in research projects, improvements, upgrades, and the design and manufacturing of similar launch devices.

Keywords: Engine; Launch principle; Launch system; Aerodynamic.

1. INTRODUCTION

The principle of aerodynamic launching is applied to various launch systems depending on the characteristics of the object or device being launched as well as the intended user. Although it is referred to as an aerodynamic launch type, each system has different structural designs. As a result, the computational models for determining operating parameters also vary. Several studies on high-pressure and low-pressure aerodynamic launching techniques [1, 2] have shown that it is impossible to develop a universal model applicable to all types of aerodynamic launch systems. Therefore, it is necessary to establish specific computational models for the operating parameters of each particular system. This paper presents a dual-flow aerodynamic launch model with a structure illustrated in figure 1. The main components of this model include the launch tube (1), the rocket (2), the launch motor (LM) (3), and the aerodynamic chamber (4).

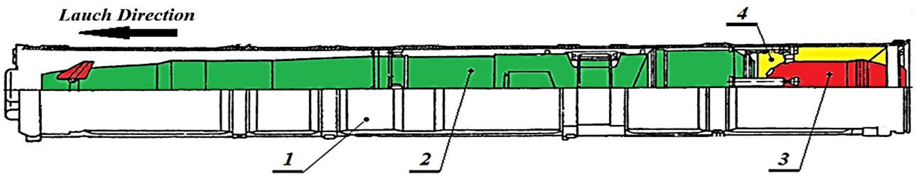


Figure 1. *Structure of the aerodynamic launch system:*
1. Launch tube; 2. Rocket; 3. Launch motor; 4. Aerodynamic chamber.

Before launch: The rocket is positioned inside the launch tube, with the LM rigidly connected to the launch tube. The launch tube and the rocket’s base form an open chamber referred to as the aerodynamic chamber.

During the launch phase, multiple physical processes occur simultaneously, including The combustion process that generates gas (taking place inside the LM). The expansion of combustion products (occurring within the aerodynamic chamber or inside the launch tube). These processes simultaneously produce two components of work: Thrust work and propelling the rocket forward. Exhaust jet work: The combustion products create a rearward jet to balance the rocket's kinetic energy, reducing the recoil force acting on the launch tube.

The two flow streams of combustion products are generated by:

- Gas expansion in the aerodynamic chamber: Producing thrust that propels the rocket forward, based on the working principle of artillery systems.

- Gas jet exhaust to the rear: Generating a counter-thrust that pushes the entire system forward, partially neutralizing the recoil force caused by the rocket's motion. This follows the working principle of a rocket engine. This phenomenon is referred to as the dual-nature characteristic of the dual-flow aerodynamic launch principle.

2. THEORETICAL MODEL AND PROBLEM FORMULATION

2.1. Modeling of physical processes during the launch phase

The launch phase is modeled to include the following processes (figure 2):

2.1.1. Launch motor (3)

- Ignition: The LM starts operating. The propellant burns, generating gas; the combustion products create high pressure inside the LM, leading to changes in the gas state.

- Gas exhaust through nozzles: Due to the LM's structure, which includes 6 front nozzles (5) and 6 rear nozzles (7), the combustion products are expelled in two streams: A forward jet through the front nozzles into the aerodynamic chamber (6). A rearward jet through the rear nozzles into the atmosphere (8). Energy exchange and mass loss: the system undergoes energy transformations and experiences mass losses due to gas expulsion.

2.1.2. Aerodynamic chamber (6)

- Receives high-pressure gas from the LM through the 6 front nozzles, creating a low-pressure jet (9) that exhausts into the atmosphere.

- State changes: The gas undergoes thermodynamic state changes due to expansion and pressure differences. Energy transfer: converts internal energy into mechanical energy, propelling the rocket forward.

2.1.3. Rocket (2)

When the pressure in the aerodynamic chamber reaches a critical threshold, the rocket begins to move. Chamber expansion: As the rocket accelerates, the aerodynamic chamber expands, increasing its volume.

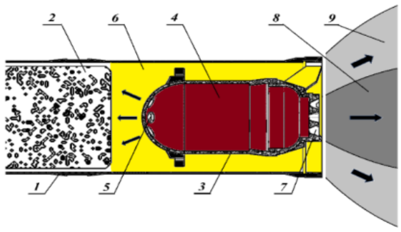


Figure 2. *Launch process modeling:*
1. Launch tube; 2. Rocket; 3. Launch motor; 4. High-pressure chamber; 5. Front nozzle; 6. Aerodynamic chamber; 7. Rear nozzle; 8. High-pressure jet stream; 9. Low-pressure jet stream.

2.2. Assumptions

- The propellant burns completely following geometric burning laws, and the burning rate adheres to an exponential burning rate equation;

- The adiabatic exponent, adiabatic index, and gas constant (R) are considered constants;

- The problem is analyzed from the moment after the ignition phase, with an initial ignition pressure of $p_m = 50.0x10^5$ (N/m^2); The entire propellant surface is ignited simultaneously, and the analysis continues until the rocket base reaches the launch tube's exit, excluding the post-burn gas effects. The launch process is conducted under standard conditions, neglecting heat losses.

2.3. Formulation of the internal ballistics

Basis for problem formulation: The internal ballistics for the dual-flow aerodynamic launch system are established based on the following principles: Gas generation rate of the propellant (1 equation), Heat loss in the LM (1 equation), Mass conservation, Energy exchange, Gas state equations within the LM and the aerodynamic chamber (6 equations). Additionally, the formulation includes Equations describing the volume variation of the aerodynamic chamber and equations for the displacement and velocity of the rocket during the launch process.

1. Relative burn rate of propellant in the LM (gas generation rate) [3]

$$\frac{d\psi}{dt} = \frac{S(e).u.\rho_T}{\omega} \tag{1}$$

2. Gas state in the LM: neglecting the accumulation of combustion products

$$p_1.w_1 = m_1.\chi_n.R.T_{b1} \tag{2}$$

$$\frac{dp_1}{dt} = \frac{\chi_n.m_1.R.T_{b1}}{w_1}.\left(\frac{1}{m_1}.\frac{dm_1}{dt} + \frac{1}{T_{b1}}.\frac{dT_{b1}}{dt} + \frac{1}{\chi_n}.\frac{d\chi_n}{dt} - \frac{1}{w_1}.S.u\right) \tag{3}$$

3. Mass conservation equation in the LM

$$\dot{m}_1 = \dot{m}_+ - \dot{m}_- - \dot{m}_{td1} \tag{4}$$

Where: $\dot{m}_1$ - Variation of gas flow retained in the LM; $\dot{m}_+$ - Gas flow generated by the propellant in the LM; $\dot{m}_-$ - Gas flow through the nozzle; $\dot{m}_{td1}$ - Gas flow exchanged from the LM to the aerodynamic chamber.

Parameters in equation (4):

$$\dot{m}_+ = \omega.\frac{d\psi}{dt} = S(e).u.\rho_T \tag{5}$$

Considering $p^* \approx p_1$ and $T^* \approx T_{b1}$, we have:

$$\dot{m}_- = \frac{\phi_2.K_0(K).F_{th}.p^*}{\sqrt{\chi_n.R.T^*}} \approx \frac{\phi_2.K_0(K).F_{th}.p_1}{\sqrt{\chi_n.R.T_{b1}}} \tag{6}$$

Substituting into equation (4), we get:

$$\frac{dm_1}{dt} = S.u.\rho_T - \frac{\phi_2.K_0(K).F_{th}.p_1}{\sqrt{\chi_n.R.T_{b1}}} - \dot{m}_{td1} \tag{7}$$

4. Energy exchange equation in the LM

The thermal energy of the propellant gas is converted into the kinetic energy of the rocket and the propellant gas. This process follows the first law of thermodynamics, and the following equation is applied to the aerodynamic chamber:

$$\frac{dQ}{dt} = \frac{dU}{dt} + \sum_{i=1}^{n}\frac{dL_i}{dt} \tag{8}$$

Where: Q - Heat generated by the propellant combustion; U - Internal energy of the propellant gas; L - Total work done by the propellant gas during launch.

Using the relational equations $R = (k-1).C_V$ and $C_P = k.C_V$, by transforming equation (8), we get:

$$\frac{dT_{b1}}{dt} = \frac{T_{b1}}{m_1}.\left(\omega.\frac{T_1}{T_{b1}}.\frac{d\psi}{dt} - \frac{dm_1}{dt} - k.\dot{m}_{td1} - \frac{k.\phi_2.K_0(K).F_{th}.p_1}{\sqrt{\chi_n.R.T_{b1}}}\right) \tag{9}$$

5. Heat loss rule in the aerodynamic chamber

$$\frac{d\chi_n}{dt}=\frac{d}{dt}\left(1-\frac{a}{1+b.\psi}\right)=-\frac{-a\dfrac{d}{dt}\left(1+b.\psi\right)}{\left(1+b.\psi\right)^2}=\frac{a.b}{\left(1+b.\psi\right)^2}.\frac{d\psi}{dt} \quad (10)$$

6. Gas state equation in the aerodynamic chamber

$$p_2.w_2=m_2.\chi_n.R.T_{b2} \quad (11)$$

$$\frac{dp_2}{dt}=\frac{d}{dt}\left(\frac{m_2.\chi_n.R.T_{b2}}{w_2}\right)=R.\frac{d}{dt}\left(\frac{m_2.\chi_n.T_{b2}}{w_2}\right) \quad (12)$$

Applying the transformation and differentiation similar to the equation p_1, we get:

$$\frac{dp_2}{dt}=\frac{\chi_n.m_2.R.T_{b2}}{w_2}.\left(\frac{1}{m_2}.\frac{dm_2}{dt}+\frac{1}{T_{b2}}.\frac{dT_{b2}}{dt}+\frac{1}{\chi_n.}.\frac{d\chi_n}{dt}-\frac{1}{w_2}.\frac{dw_2}{dt}\right) \quad (13)$$

in which, χ_{n2} is the heat loss function in the aerodynamic chamber.

7. Mass conservation equation in the aerodynamic chamber

$$\dot{m}_{2+}=\dot{m}_{td1}=\dot{m}_2+\dot{m}_{kq} \quad (14)$$

$$dm_2/dt=\dot{m}_{td1}-\dot{m}_{kq} \quad (15)$$

In which: $\dot{m}_{2+}$ - Gas flow rate exchanged (received) from the LM; $\dot{m}_2$ - Variation of the gas flow rate retained in the aerodynamic chamber; $\dot{m}_{kq}$ - Gas flow rate from the aerodynamic chamber to the atmosphere.

8. Energy exchange equation in the aerodynamic chamber

Similar to the process of deriving equation (9) from the general equation (8):

$$\frac{dT_{b2}}{dt}=\frac{T_{b2}}{m_2}.\left(k.\frac{T_{b1}}{T_{b2}}.\dot{m}_{td1}-k.\dot{m}_{kq}-(k-1)\frac{V_d.S_d.p_2}{R.T_{b2}}-\frac{dm_2}{dt}\right) \quad (16)$$

9. The change in volume in the aerodynamic chamber due to the rocket's movement

$$\frac{dw_2}{dt}=S_d\frac{dl}{dt}=S_d.V_d \quad (17)$$

10. The equation of motion of the rocket: considering the positive direction of velocity as the same as the launch direction (figure 1), according to Newton's second law

$$dl/dt=V_d \quad (18)$$

11. Determination of the function $\dot{m}_{td1}$ [4, 5]

- When $p_1>p_2$, if the gas exchange rate exceeds the speed of sound:

$$\dot{m}_{td1}=\phi_2.F_{td}.\frac{p_1}{\sqrt{RT_{b1}}}\left(\frac{2}{k+1}\right)^{\frac{1}{k-1}}\sqrt{\frac{2k}{k+1}} \quad (19)$$

- When $p_1>p_2$, if the gas exchange rate is subsonic:

$$\dot{m}_{td1}=\phi_2.F_{td}.\frac{p_1}{\sqrt{RT_{b1}}}\sqrt{\frac{2k}{k-1}\left[\left(\frac{p_2}{p_1}\right)^{\frac{2}{k}}-\left(\frac{p_2}{p_1}\right)^{\frac{k+1}{k}}\right]} \quad (20)$$

- When $p_2 > p_1$, if the gas exchange rate exceeds the speed of sound:

$$\dot{m}_{td1} = -\phi_2.F_{td}.\frac{p_2}{\sqrt{RT_{b2}}}\left(\frac{2}{k+1}\right)^{\frac{1}{k-1}}\sqrt{\frac{2k}{k+1}} \tag{21}$$

- When $p_2 > p_1$, if the gas exchange rate is subsonic:

$$\dot{m}_{td1} = -\phi_2.F_{td}.\frac{p_2}{\sqrt{RT_{b2}}}\sqrt{\frac{2k}{k-1}\left[\left(\frac{p_1}{p_2}\right)^{\frac{2}{k}} - \left(\frac{p_1}{p_2}\right)^{\frac{k+1}{k}}\right]} \tag{22}$$

12. Determination of the function $\dot{m}_{kq}$

- If the gas velocity from the aerodynamic chamber to the atmosphere exceeds the speed of sound:

$$\dot{m}_{kq} = \phi_2.F_{th2}.\frac{p_2}{\sqrt{RT_{b2}}}\left(\frac{2}{k+1}\right)^{\frac{1}{k-1}}\sqrt{\frac{2k}{k+1}} \tag{23}$$

- If the gas velocity from the aerodynamic chamber to the atmosphere is subsonic:

$$\dot{m}_{kq} = \phi_2.F_{th2}.\frac{p_2}{\sqrt{RT_{b2}}}\sqrt{\frac{2k}{k-1}\left[\left(\frac{p_{kq}}{p_2}\right)^{\frac{2}{k}} - \left(\frac{p_{kq}}{p_2}\right)^{\frac{k+1}{k}}\right]} \tag{24}$$

By combining equations (1), (3), (7), (9), (10), (12), (13), (15), (16), (17), and (18), we obtain the system of differential equations describing the launch phase of the aerodynamic launch system:

$$\begin{cases}
1.\dfrac{d\psi}{dt} = \dfrac{S.u.\rho_T}{\omega}; \qquad 2.\dfrac{d\chi_n}{dt} = \dfrac{a.b}{\left(1+b.\psi\right)^2}.\dfrac{d\psi}{dt} \\
3.\dfrac{dp_1}{dt} = \dfrac{\chi_n.m_1.R.T_{b1}}{w_1}.\left(\dfrac{1}{m_1}.\dfrac{dm_1}{dt} + \dfrac{1}{T_{b1}}.\dfrac{dT_{b1}}{dt} + \dfrac{1}{\chi_n}.\dfrac{d\chi_n}{dt} - \dfrac{1}{w_1}.S.u\right) \\
4.\dfrac{dm_1}{dt} = S.u.\rho_T - \dfrac{\phi_2.K_0(K).F_{th}.p_1}{\sqrt{\chi_n.R.T_{b1}}} - \dot{m}_{td1} \\
5.\dfrac{dT_{b1}}{dt} = \dfrac{T_{b1}}{m_1}.\left(\omega.\dfrac{T_1}{T_{b1}}.\dfrac{d\psi}{dt} - \dfrac{dm_1}{dt} - k.\dot{m}_{td1} - \dfrac{k.\phi_2.K_0(K).F_{th}.p_1}{\sqrt{\chi_n.R.T_{b1}}}\right) \\
6.\dfrac{dp_2}{dt} = \dfrac{\chi_n.m_2.R.T_{b2}}{w_2}.\left(\dfrac{1}{m_2}.\dfrac{dm_2}{dt} + \dfrac{1}{T_{b2}}.\dfrac{dT_{b2}}{dt} + \dfrac{1}{\chi_n.}.\dfrac{d\chi_n}{dt} - \dfrac{1}{w_2}.\dfrac{dw_2}{dt}\right) \\
7.\dfrac{dm_2}{dt} = \dot{m}_{td1} - \dot{m}_{kq} \\
8.\dfrac{dT_{b2}}{dt} = \dfrac{T_{b2}}{m_2}.\left(k.\dfrac{T_{b1}}{T_{b2}}.\dot{m}_{td1} - k.\dot{m}_{kq} - (k-1)\dfrac{V_d.S_d.p_2}{R.T_{b2}} - \dfrac{dm_2}{dt}\right) \\
9.\dfrac{dV_d}{dt} = \dfrac{S_d.p_2}{\phi.m_d}; \qquad 10.\dfrac{dw_2}{dt} = S_d.V_d \qquad 11.\dfrac{dl}{dt} = V_d
\end{cases} \tag{25}$$

And the condition equations (19) - (24) to determine $\dot{m}_{td1}$ và $\dot{m}_{kq}$.

The meaning of the parameters in system of equations (25) is as follows: ψ - Relative amount of propellant burned; S - Initial total burning surface area of the propellant (m^2); $u = f(p).f_1(T_{bd}).\phi(w)$ - Burning rate of the propellant (m/s); ρ_T - Density of the propellant (kg/m^3); ω - Initial mass of the propellant (kg); $\chi_n = \chi_n(t)$ - Heat loss in the combustion chamber; a,b - Shape factor of the propellant; p_1, p_2 - Pressure in the LM and the aerodynamic chamber (N/m^2); m_1, m_2 - Mass of gas retained in the LE and the aerodynamic chamber (kg); $\dot{m}_+$ - Gas flow rate generated by the propellant in the LM (kg/s); $\dot{m}_{td1}$, $\dot{m}_-$ - Gas flow rate through the front nozzle and rear nozzle (kg/s); $\dot{m}_{2+}$ - Gas flow rate exchanged (received) between the LE and the aerodynamic chamber (kg/s); $\dot{m}_{kq}$ - Gas flow rate from the aerodynamic chamber to the atmosphere (kg/s); T_{b1}, T_{b2} - Gas temperature in the LM and the aerodynamic chamber (K); k - Adiabatic exponent; $K_0(K)$ - Adiabatic index function; R - Gas constant; ϕ_m - Rocket mass increase coefficient; V_d - Velocity of the rocket relative to the launch tube (m/s); S_d - Base area of the rocket (m^2); w_2 - Aerodynamic chamber volume (m^3).

3. RESULTS OF THE PROBLEM SOLUTION AND OBSERVATIONS

3.1. Initial conditions of the system of equations

The system of equations (25) is a first-order differential system, so it can be solved using numerical methods with the Runge-Kutta algorithm. To solve it, the initial conditions of the system must be determined. The initial conditions are derived from the parameters of the CTX rocket, which is modeled after the Russian 9K111 rocket. The parameters are categorized into the following groups:

- Parameters of the LM: Diameter: $D_k = 0.077$ *(m)*; Length: $L_k = 0.15$ *(m)*; Number of nozzles: $n_l = 6$; Diameter of the rear nozzle: $d_{th1} = 0.0105$ *(m)*; Diameter of the front nozzle: $d_{th2} = 0.012$ *(m)*; Nozzle exit area of the aerodynamic chamber: $S_{th2} = 0.0054$ *(m^2)*;

- Propellant parameters: Single-hole cylindrical shape; Density: $\rho_t = 1590$ *(kg/m^3)*; Outer diameter: $D_n = 0.0047$ *(m)*; Inner diameter: $d_t = 0.0029$ *(m)*; Length: $L_t = 0.130$ *(m)*; Number of propellant rodes: 179; Propellant mass: $\omega = 0.375$ *(kg)*; Burn rate law function: $u = u_1.p^{\nu} = 31.10^{-6}.p^{0.392}$;

- Rocket parameters: Diameter $d_đ = 0.12$ *(m)*; Mass $m_đ = 11.3$ *(kg)*; Mass increase coefficient $\varphi = 1.03$; Projectile ejection pressure (required for the rocket to move) $P_{tđ} = 0.9x10^5$ *(Pa)*;

- Launch tube parameters: Length: $L_P = 0.8$ *(m)*; Mass: $m_L = 3.2$ *(kg)*; Spring stiffness coefficient: $k_L = 2510$ *(N/m)*; Damping coefficient: $c_L = 12.5$ *(kg/s)*;

- Initial conditions of the problem for each equation:

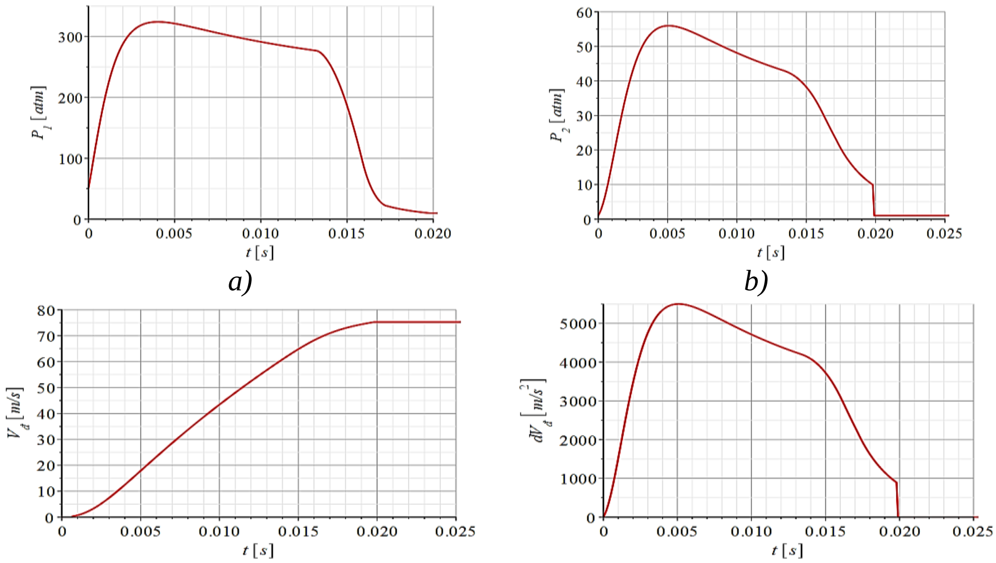
$\psi\big|_0 = 0$; $\chi_n\big|_0 = 0.715$; $p_1\big|_0 = p_m = 50.0x10^5$ *(N/m^2)*; $m_1\big|_0 = 0.01055$ *(kg)*; $T_{b1}\big|_0 = 2700$*(K)*; $p_2\big|_0 = p_{kq} = 1.0x10^5$ *(N/m2)*; $m_2\big|_0 = 0.0022$ *(kg)*; $T_{b2}\big|_0 = 288.15$*(K)*; $V_d\big|_0 = 0.0$*(m/s)*; $w_2\big|_0 = w_0 = 0.0025$ *(m^3)*; $x_P\big|_0 = 0.0$ *(m)*; $V_P\big|_0 = 0.0$ *(m/s)*.

3.2. Results of internal ballistics problem

The system of differential equations (25) was solved numerically using a computer, yielding a table of calculation results. Some operating parameters of the LM are presented in graphical form in figure 3. In addition to the 11 variables of the system of equations (25), such as in figures 3a, 3b, and 3c, other results were also obtained, including parameters like the forces acting on the rocket, the rocket's acceleration in figure 3.d, and the thrust forces generated by exhaust jets acting on the launch system.

In the experiment, the pressure profile of the LM was measured. This serves as an important basis for validating the theoretical framework established in this paper.

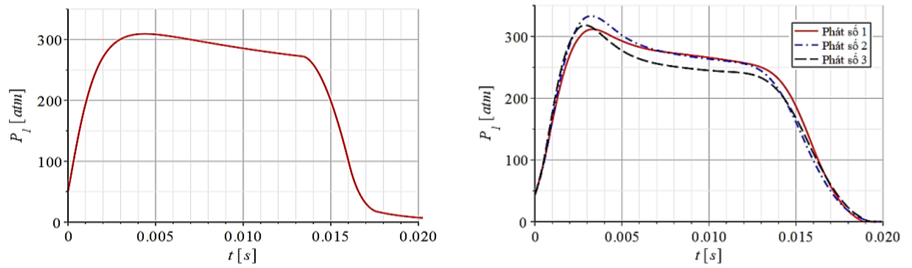
The comparison of the LM pressure results between theoretical calculations and experimental data in figure 4 shows that the profiles of the pressure characteristic curve $P_1(t)$ in both graphs are quite similar. The pressure in the LM spikes to a peak value (around 320 atm), then gradually decreases at a relatively slow rate (forming a saddle-shaped curve). When the propellant is nearly completely burned, a collapse phenomenon occurs, causing the pressure to drop rapidly until it equals atmospheric pressure. The operating time of the LM ranges from 17 to 18 milliseconds. In terms of impulse pressure, the theoretical and experimental curves are also nearly equivalent, with the experimental impulse pressure being about 5% lower than the theoretical value.



a) b) c) d)

***Figure 3.** Some calculation results:*
a) Pressure in the combustion chamber; b) Pressure in the aerodynamic chamber;
c) Rocket velocity; d) Rocket acceleration.

The calculated result of the rocket's launch exit velocity. $V_d\big|_{ts=0.022(s)} \approx 76.0(m/s)$, this value is consistent with the technical specifications of the 9K111 rocket: 80.0 (m/s) [6, 7].



a) b)

***Figure 4.** Theoretical and experimental results:*
a) Calculated pressure in the LM; b) Digitized experimental pressure in the LM.

Remarks: The theoretical calculation results closely match the experimental values. This reflects two key aspects: (1) Qualitatively - The description of the launch phase through sequential processes aligns well with the actual operational principles observed in practice; (2) Quantitatively - The theoretical model accurately determines the fundamental parameters of the launch process (as shown in the pressure graphs), including propellant burn time, pressure impulse, peak pressure, and the rocket's launch exit velocity, with an error margin of less than 5%.

4. CONCLUSIONS

Based on the actual launch system, this paper has modeled the launch phenomenon into distinct phases, applying physical laws to develop the internal ballistics problem of the launch system according to aerodynamic principles. The problem was solved, and several results were obtained using an assumed set of parameters. This serves as a foundation for investigating models in subsequent studies, such as launcher dynamics, launch tube oscillations, recoil impulse of the launch system, launcher stability on elastic ground, and the influence of structural parameters on launch characteristics.

The scientific methodology employed in this paper is rigorously argued, utilizing physical models that align with ballistic phenomena, as referenced in various internal ballistics publications [3, 4]. The problem-solving process was conducted using Maple software, achieving the necessary accuracy as well as efficiency in computational time and volume. In addition to the parameters already investigated, this method can be applied to study various other parameters. However, due to the limited scope of the paper, not all aspects could be covered. This problem will become more accurate when standardized and calibrated with experimentally measured parameters. Moreover, this problem, along with the theoretical approach developed in the paper, serves as a foundation for calculations, design calibration, and experimental result prediction.

REFERENCES

[1]. N. H. Hoan et al., *"Mô hình thuật phóng trong của hệ súng phóng lựu có tính đến thuốc phóng cháy không hoàn toàn trong buồng cao áp"*, J. of Military Science and Technology, No. 72, (2021) (in Vietnamese).

[2]. T. D. Dien, *"Thuật phóng trong của vũ khí đặc biệt (tập 1, 2)"*, University of Military Technology, (1975) (in Vietnamese).

[3]. P. T. Phiet, *"Lý thuyết động cơ tên lửa"*, Military Technial Academy, (1995) (in Vietnamese).

[4]. L. S. Tung et al., *"Tính toán thiết kế động cơ tên lửa nhiên liệu rắn"*, People's Army Publishing House, (2013) (in Vietnamese).

[5]. Орлов Б. В., *"Термодинамические и баллистические основы проектирования РДТТ"*, Издательство Машиностроение, Москва, (1968).

[6]. *"Переносный противотанковый комплекс 9К111 - Техническое описание"*, Военное издательство, Министерства обороны СССР, Москва, (1981).

[7]. *"Техническое описание и инструкция по эксплуатации ПТУРС 9М111М"*, Военное издательство, Министерства обороны СССР, Москва, (1975).

ABSTRACT

Nghiên cứu xây dựng mô hình tính toán các tham số làm việc cho động cơ phóng theo nguyên lý khí động hai luồng

Nguyên lý phóng khí động có cơ chế làm việc phức tạp, kết hợp cả nguyên lý kiểu súng pháo và động cơ sử dụng nhiên liệu rắn. Việc tính toán và xác định các tham số làm việc cho động cơ phóng là bước quan trọng trong công tác thiết kế hệ thống phóng, đặc biệt khi nghiên cứu, phát triển và chế tạo các hệ thống mới. Đây là yêu cầu bắt buộc để đảm bảo tính năng và độ tin cậy của thiết bị. Bài báo này tập trung vào việc xây dựng mô hình vật lý nhằm tính toán các thông số làm việc của động cơ phóng theo tham số thời gian, dựa trên một bộ thông số giả định của hệ thống. Mô hình tính toán của bài báo này, cũng như kết quả tính toán nhận được phục vụ việc khảo sát, thiết kế hệ thống, đồng thời làm cơ sở lý thuyết để phát triển những công trình nghiên cứu sâu hơn và hoàn thiện hơn, ứng dụng cho các đề tài nghiên cứu, cải tiến, nâng cấp, cũng như thiết kế chế tạo loại thiết bị phóng có kiểu phóng tương tự.

Từ khóa: Động cơ; Thuật phóng; Hệ thống phóng; Khí động.