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Research Article

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Research on the Combustion Mode and Thrust Performance of Rotating Detonation Scrarmjet Engines

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Abstract
Decades of research in hypersonic science and engineering have shown that scramjet engines are the preferred choice for powering hypersonic vehicles. In these engines, oblique shock combustion and rotating detonation combustion are two common combustion modes. This article primarily focuses on the numerical simulation and theoretical analysis of these two combustion modes. The research indicates that as the combustible mixture injected into the combustion chamber increases, the combustion mode transitions from deflagration combustion to self-sustaining detonation combustion and then to forced detonation combustion to maintain stable combustion in the chamber. Under self-sustaining detonation combustion mode, as the amount of combustible mixture injected into the chamber increases, the number of detonation waves gradually increases from one to multiple and the angle between the detonation waves and the inflow changes from acute to right angles. In the forced detonation combustion mode, the minimum shock wave intensity required to initiate detonation combustion can be obtained by drawing a tangent from the Rayleigh line. The structure generating oblique shock waves in the combustion chamber needs to match the shock wave intensity. In the detonation combustion mode, both the exit total temperature and specific impulse of the combustion chamber increase with the increase in Mach number at the inlet, while the pressure ratio decreases. This article provides a reference for the design of the combustion chamber in supersonic combustion ramjet engines.

Keywords: Scramjet; Deflagration; Detonation; Thrust Performance

1. Introduction
Although subsonic combustion ramjet engine technology has become mature and has been applied to some extent in military fields such as unmanned reconnaissance aircraft and missiles, as the flight speed of aircraft continues to increase, decelerating the hypersonic incoming flow to subsonic speeds will result in a significant increase in static air temperature. The high static temperature would cause thermal cracking of the fuel and decrease the heat release efficiency, limiting the continuous improvement of aircraft flight speeds powered by subsonic combustion ramjet engines. The upper limit of flight speed is around Mach 4-5 [1]. Decades of research in the field of hypersonic science and engineering have shown that supersonic combustion ramjet engines are the preferred choice for powering hypersonic vehicles. The flight tests of the X-43A with hydrogen fuel and the X-51A with hydrocarbon fuel have proven their feasibility and potential [2-4].

The main components of a supersonic combustion ramjet engine include the inlet, isolator, combustion chamber and nozzle [5], as shown in Figure 1. The inlet is responsible for capturing air and decelerating and pressurizing it. The isolator, arranged between the inlet and the combustion chamber, ensures good pressure matching between the inlet and the combustion chamber. The combustion chamber is where supersonic combustion occurs. The high-temperature and high-pressure gas flow in the combustion chamber expands through the nozzle, generating reactive thrust in the opposite direction.

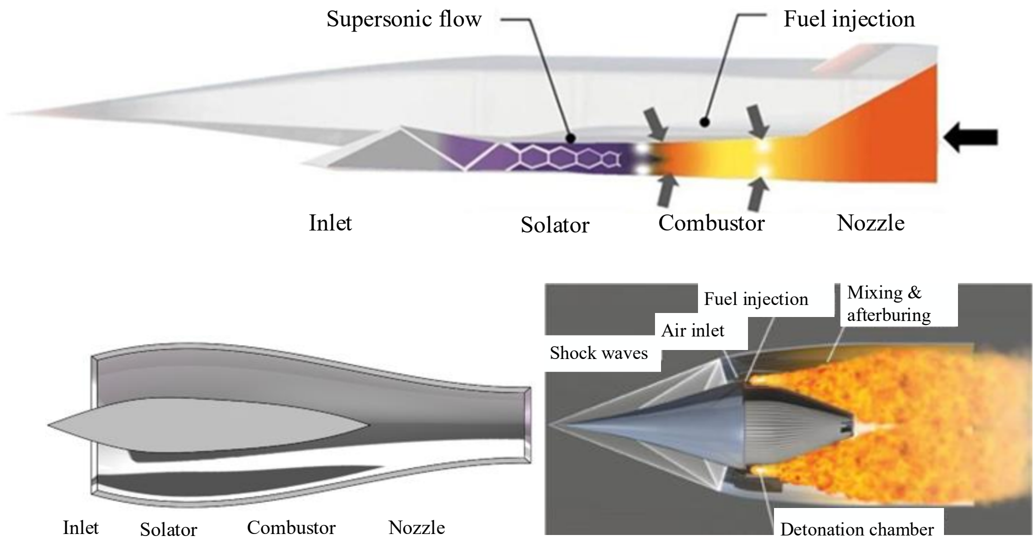


Figure 1: Scramjet Engine Schematic Diagram

The combustion chamber is the core component of a scramjet engine. It must address a series of issues such as rapid fuel mixing, reliable ignition, and stable combustion within the combustion chamber of a supersonic combustion ramjet engine. Depending on the different detonation combustion modes used in the combustion chamber, the propulsion devices as energy conversion methods are mainly divided into Pulsed Detonation Engine (PDE), Oblique Detonation Engine (ODE), and Rotating Detonation Engine (RDE). For instance, the ODE does not require additional ignition [6]. The fuel is injected into the inlet and mixed with supersonic airflow, utilizing the oblique shock waves formed by high-speed airflow to induce compression and heating, resulting in stationary detonation waves. The combustible mixture fully combusts in the combustion chamber in the form of stationary oblique detonations, generating thrust. The RDE is an engine that utilizes one or multiple detonation waves propagating continuously in a annular combustion chamber to generate thrust, as depicted in Figure 2 [7].

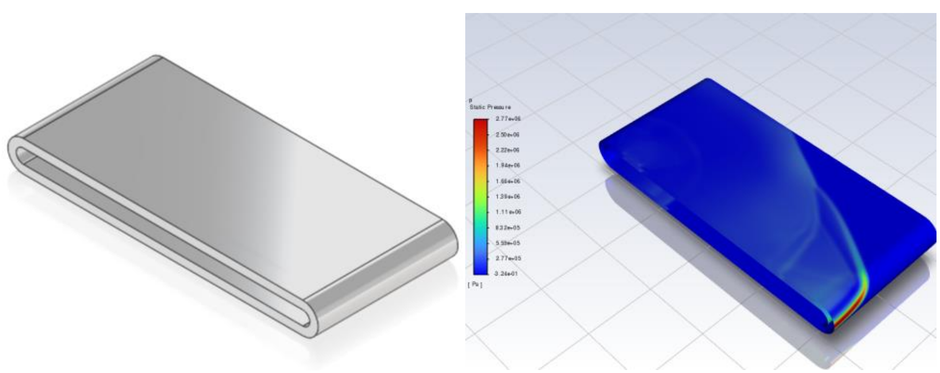


Figure 2: One Type of Structure of a Rotating Detonation Engine

In detonation combustion mode, the propagation speed of detonation can reach the level of kilometers per second. The combustion wave and shock wave are closely coupled together, and the pressure and temperature increase sharply during the combustion process, while the volume decreases slightly. It is usually approximated as constant volume combustion. Detonation combustion releases heat quickly and generates relatively small entropy increase, resulting in high thermal efficiency. The advantage of continuous rotating detonation combustion lies in the sustained rotation and propagation of detonation waves. Due to the selfsustaining and self-compressing nature of detonation waves, the combustible mixture can be pressurized to a certain pressure by the detonation wave, allowing for greater effective work to be produced at lower compression ratios. It can achieve stable operation at supersonic inlet flow velocities and the average flow rate of incoming fuel can be greatly adjusted [8].

The combustion chamber of a continuous rotating detonation engine is usually of annular cavity structure, where fuel and oxidizer are injected through slots or round holes on the intake wall [9-12]. One or more detonation waves propagate and rotate along the circumferential direction at the head of the combustion chamber [13]. The detonation wave completely burns the fuel and the resulting high-temperature and high-pressure products rapidly expand predominantly in the axial direction, generating thrust. In addition, oblique shock waves and contact discontinuity occur behind the detonation wave. During the propagation of the detonation wave, the combustible mixture continuously enters the combustion chamber from the head, forming a triangle of unburned combustible mixture in front of the detonation wave for combustion [14-15].

Assuming a uniform combustible mixture at the inlet of the combustion chamber, the combustion chamber exhibits different combustion modes under different fuel supply conditions. As the fuel filling rate increases from low to high, the combustion chamber goes through stages of deflagration combustion, detonation combustion and complete detonation combustion [16-19].

2. Transition of Combustion Modes

Thrust performance indicators of the engine under different combustion modes [20].

$$I_2 = \int dI_2 = \int Ma_2\sqrt{\gamma RT_2}dm_2$$

$$= \int Ma_2\sqrt{\frac{\gamma RT_{02}}{1+0.5(\gamma-1)Ma_2^2}}dm_2$$

$$= \int \sqrt{\frac{2\gamma}{\gamma-1}}\sqrt{RT_{02}}\sqrt{1-\frac{1}{(P_{02}/P_1)^{(\gamma-1)/\gamma}}}dm_2 \qquad (1)$$

Where: P_1 is the ambient static pressure; A_2 is the area at the exit section; P_2 is the exit static pressure; P_{02} is the total exit pressure; T_2 is the exit static temperature; T_{02} is the total exit temperature; Ma_2 is the Mach number of the axial exit; dm_2 is the exit mass flow; γ Is the specific heat ratio of gas; R is the gas constant, the gas physical parameters are generally a function of temperature.

a) The inlet velocity of the combustion chamber is the minimum flame propagation speed for deflagration combustion. It is the flow velocity of the combustible mixture entering the combustion chamber. If the combustible mixture entering the combustion chamber is insufficient, even in the deflagration combustion mode, it is not enough to sustain the flame. The flame moves upstream and leaves the combustion chamber, becoming unstable. In this situation,P_{02}=P_{01}.

$$I_2 = \int \sqrt{\frac{2\gamma}{\gamma-1}}\sqrt{RT_{01}}\sqrt{\frac{0.5(\gamma-1)Ma_1^2}{1+0.5(\gamma-1)Ma_1^2}}dm_2 \qquad (2)$$

Ma_1 is the inlet Mach number for the combustion chamber.

b)When the inlet velocity of the combustion chamber is the maximum flame propagation speed for deflagration combustion. The combustible mixture entering the combustion chamber can maintain stable deflagration combustion, producing a stable deflagration flame that remains in the combustion chamber. The specific impulse calculation for the engine is the same as in case a), where the inlet Mach number for the combustion chamber is taken as the Mach number corresponding to the speed of deflagration combustion.

$$I_2 = \int \sqrt{\frac{2\gamma}{\gamma-1}}\sqrt{RT_{02}}\sqrt{\frac{0.5(\gamma-1)Ma_{Df}^2}{1+0.5(\gamma-1)Ma_{Df}^2}}dm_2 \qquad (3)$$

Where W_{Df} is the average flame propagation speed of deflagration combustion.

c)The inlet velocity of the combustion chamber. In the equation, is the minimum combustible mixture height required for detonation combustion, below which detonation combustion will not occur. is the perimeter of the combustion chamber. The combustion is in the transitional zone between deflagration combustion and detonation combustion, within which detonation occurs, but it is unstable. Deflagration combustion is dominant with a small amount of detonation combustion. The proportion of deflagration combustion is w_{Df}/w_{in} and the proportion of detonation combustion is $(1-w_{Df}/w_{in})$.

$$I_2 = I_{2Df}+I_{2Dw}$$

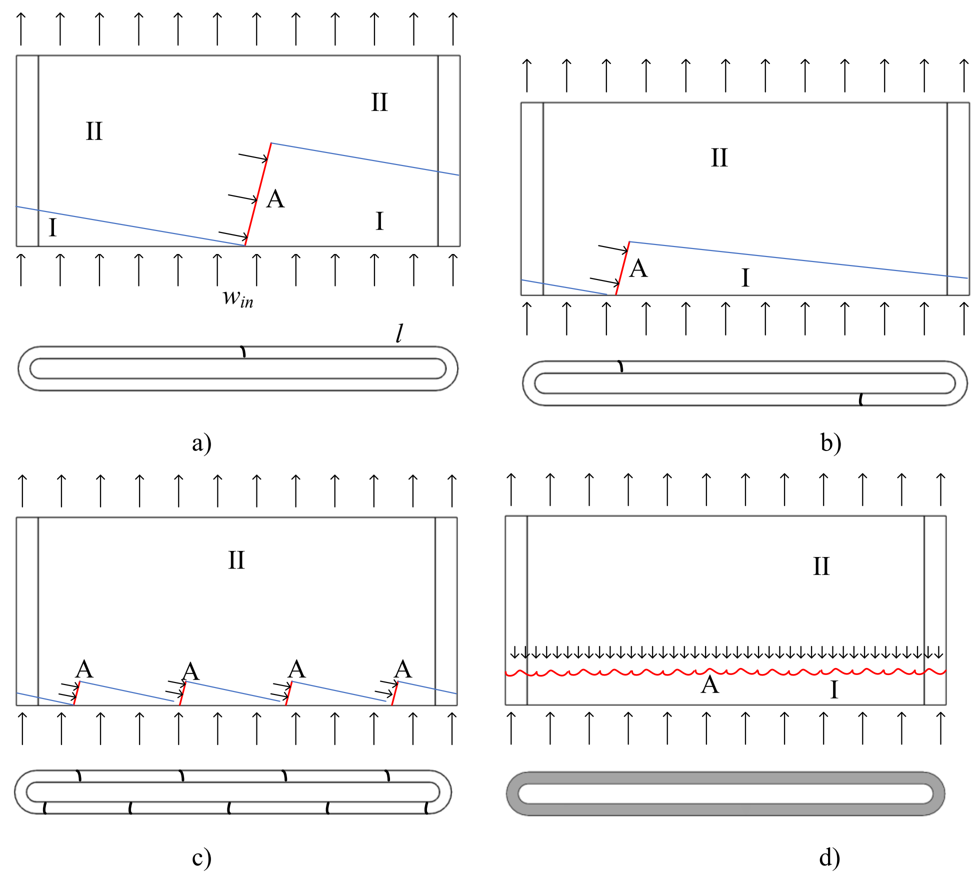
$$= \int \sqrt{\frac{2\gamma}{\gamma-1}}\sqrt{RT_{02}}\left[\sqrt{\frac{0.5(\gamma-1)Ma_{Df}^2}{1+0.5(\gamma-1)Ma_{Df}^2}}\frac{w_{Df}}{w_{in}}+\sqrt{1-\frac{1}{(P_{02}/P_\infty)^{(\gamma-1)/\gamma}}}\left(1-\frac{w_{Df}}{w_{in}}\right)\right]dm_2 \qquad (4)$$

Where I_{2Df}, I_{2Dw} represent the impulse contribution corresponding to deflagration combustion and detonation combustion respectively.

After the detonation wave, the static temperature and static pressure are higher than the total temperature and total pressure at the exit of the combustion chamber. This is because the working process of the detonation wave first pre-compresses the combustible mixture through a shock wave, and then releases heat in the compressed combustible mixture. This causes the energy contained in the airflow after the detonation wave to include not only the energy carried by the incoming flow and the energy released by combustion, but also the work done by the pre-compression. This pre-compression work, which exists to maintain the operation of the detonation wave, is not used to propel the engine, but is reflected in the static temperature and static pressure after the detonation wave.

d) When the combustion enters the detonation phase at the inlet of the combustion chamber, also known as continuous rotating detonation, the part of the combustible mixture in contact with the high-temperature combustion products will continue to burn in a deflagration combustion mode. As more combustible mixture is injected into the combustion chamber, the number of detonation waves around a circumference of the combustion chamber will increase. Initially, with a small amount of combustible mixture inject into the combustion chamber, there is only one detonation wave in a full circumference, as shown in Figure3 a), which is sufficient to completely burn the combustible mixture. As the amount of combustible mixture increases, one detonation wave is not enough to completely burn all the combustible mixture, so an additional detonation wave is added. Here, only the scenario of detonation waves propagating in the same phase is considered, and the scenario of detonation waves colliding in reverse directions is not taken into account. In this way, two stable detonation waves are formed in one circumference, as shown in Figure3 b), and as more combustible mixture is added, the number of detonation waves increases, as shown in Figure3 c), and eventually increases to eight detonation waves in one circumference. Finally, with the increasing amount of combustible mixture inject into the combustion chamber, multiple detonation waves occur, and at this point, the detonation waves become completely erpendicular to the inflow velocity (as shown in Figure3 d), instead of forming a certain angle.

The proportion of deflagration combustion is w_{Df}/w_{in}, and the proportion of detonation combustion is$(1 - w_{Df}/w_{in})$. The calculation of engine specific impulse is the same as c).



I Premixed gas II Combusted gas A Detonation wave

Figure 3: As the Combustible Mixture Increases, the Number of Detonation Waves Increases

e) When the inlet velocity of the combustion chamber $W_{in} > D_w$,Combustion can no longer be sustained in the combustion chamber solely by the self-sustaining detonation waves; external energy must be utilized to maintain combustion in the combustion chamber. Typically, shock wave compression is employed to compress the inflow of combustible mixture. During this stage, the entire combustible mixture relies on shock wave ignition, leading to combustion in the form of detonation waves.

$$I_2 = \int \sqrt{\frac{2\gamma}{\gamma-1}}\sqrt{RT_{02}}\sqrt{1-\frac{1}{(P_{02}/P_1)^{(\gamma-1)/\gamma}}}dm_2 \quad (5)$$

Raising the temperature of the incoming combustible mixture helps to increase the propagation speed of the detonation wave, which is beneficial for keeping the detonation wave from being blown out of the combustion chamber.

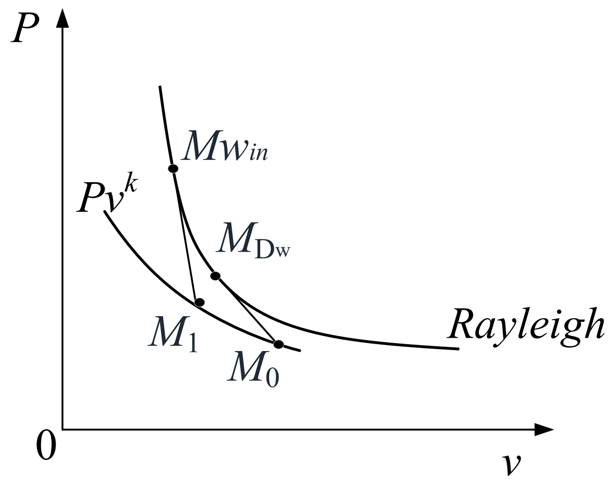


Figure 4: Rayleigh Line that Generates Detonation Waves

During this stage, it is necessary to use shock waves to boost and heat the incoming combustible mixture in order to maintain high combustion efficiency. Generally, a lower temperature increase through shock waves is used to prevent high-pressure waves from ignition. Instead, the temperature is increased to achieve a detonation wave velocity (D_w) that matches the flow velocity of the combustible mixture, ensuring that the detonation wave remains within the combustion chamber and not blown out by incoming mixture flow, remaining confined within the chamber. As shown in Figure 4, the thermodynamic parameters of the combustible mixture transition from m0 to m1 through shock wave pressurization. The region between m0 and m1 experiences weaker shock wave effects, and the shock wave intensity in this range is insufficient to sustain detonation within the chamber. The curve on the left side of m1 represents the range of shock wave intensity where shock wave-induced detonation can be sustained in the combustion chamber.

It is not advisable to use higher shock wave temperature increase, as the post-detonation wave temperature is much higher than the maximum temperature achievable during deflagration combustion. Elevated static temperature implies that the detonation wave is more prone to dissociation, chemical reactions, and other real gas effects, which can result in total temperature losses. Higher shock wave heating would enhance such effects and losses, hence lower shock wave temperature increase is preferred.

3. Performance of Detonation Combustion Mode

This section specifically analyzes the specific impulse generated by detonation combustion.

$$I_2 = \int \sqrt{\frac{2\gamma}{\gamma-1}}\sqrt{RT_{02}}\sqrt{1-\frac{1}{(P_{02}/P_\infty)^{(\gamma-1)/\gamma}}}dm_2 \quad (6)$$

The total temperature at the exit:

$$T_{02} = T_{01} + \Delta T$$

$$T_{01} = T(1 + \frac{\gamma-1}{2}M_{a1}^2)$$

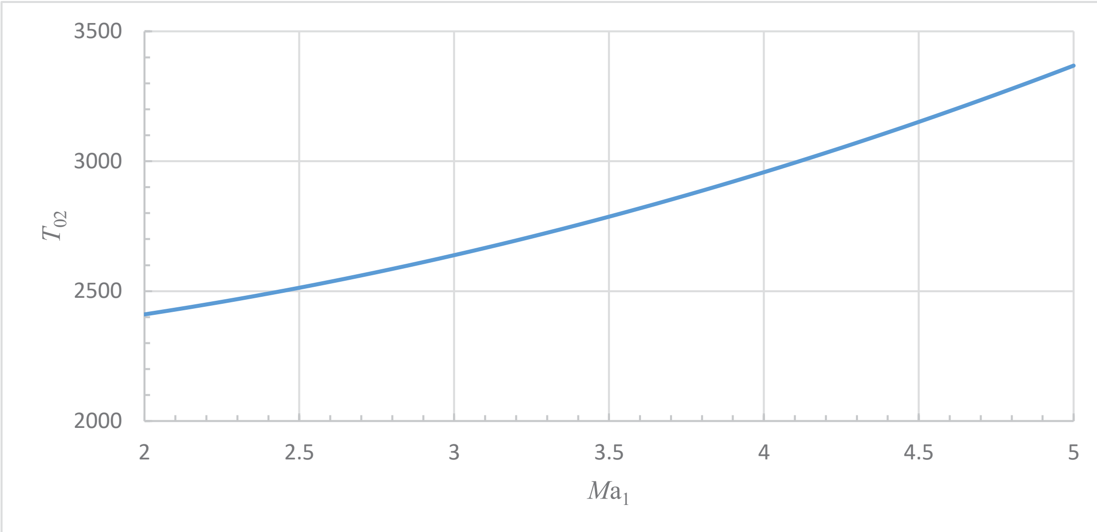


Figure 5: Variation of the Total Temperature at the Exit of the Combustion Chamber with the Mach Number at the Inlet

It can be observed that the total temperature (T_{02}) at the exit of the combustion chamber increases as the Mach number at the inlet of the combustion chamber increases.

Assuming a constant gas specific heat ratio, and considering that the pre-combustion flow static temperature is set to, as the aircraft is located in the atmosphere isothermal layer. Due to the nature of detonation combustion, a Mach number greater than 2 is chosen, ranging from 2 to 5 here.

The temperature rise produced by the heat release during combustion is set to 2000K. The analysis assumes the same fuel proportion for the same combustible mixture, with the same heat release during combustion. Through a derived formula, the ratio between the total pressure at the exit of the combustion chamber and the total pressure at the inlet can be obtained [20].

$$\frac{P_{02}}{P_{01}} = \frac{P_1}{P_{01}} \frac{1+\gamma Ma_D^2}{\gamma+1}\left[1+\frac{(\gamma-1)}{2}\right]^{\frac{\gamma}{\gamma-1}} \left\{\frac{T_1\left[1+\frac{(\gamma-1)}{2}Ma_1^2\right]+\Delta T}{T_1\left[1+\frac{(\gamma-1)}{2}Ma_D^2\right]+\Delta T}\right\}^{\frac{\gamma}{\gamma-1}}$$

$$= (1+\gamma Ma_D^2)(\gamma+1)^{\frac{1}{\gamma-1}} \left\{\frac{1+\frac{\Delta T}{T_{01}}}{[2+(\gamma-1)Ma_D^2]+\frac{\Delta T}{T_{01}}[2+(\gamma-1)Ma_1^2]}\right\}^{\frac{\gamma}{\gamma-1}} \quad (7)$$

$$Ma_D^2 \approx 1+(1+\gamma)\frac{\Delta T}{T_{01}}\left[1+\frac{(\gamma-1)}{2}Ma_1^2\right] + \sqrt{\left\{1+(1+\gamma)\frac{\Delta T}{T_{01}}\left[1+\frac{(\gamma-1)}{2}Ma_1^2\right]\right\}^2}$$

$$=2\left\{1+(1+\gamma)\frac{\Delta T}{T_{01}}\left[1+\frac{(\gamma-1)}{2}Ma_1^2\right]\right\}$$

$$P_{01} = P_1\left(1+\frac{\gamma-1}{2}M_{a1}^2\right)^{\frac{\gamma}{\gamma-1}}$$

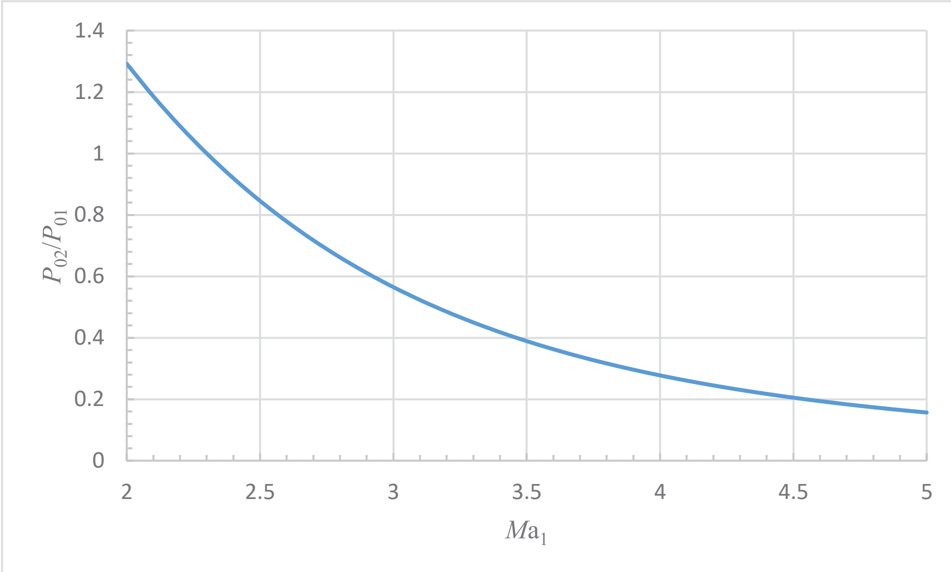


Figure 6: Variation of Combustion Pressure Ratio with the Mach Number at the Inlet of the Combustion Chamber

Figure 6 shows the variation of the pressure ratio of the detonation engine with the Mach number at the inlet of the combustion chamber. From the Figure6, it can be observed that at low Mach numbers, specifically below 2.3, the detonation engine exhibits a self-pressurizing characteristic, where the total pressure at the exit of the combustion chamber exceeds the total pressure of the incoming flow.

The design objective of an engine is to produce the highest possible thrust with limited fuel. Therefore, the total temperature and total pressure of the exhaust flow can be considered as performance indicators for thrust performance. Since this study focuses on the influence of the combustion mode on engine performance, in order to simplify the analysis, it is assumed that there are no total temperature and total pressure losses in the exhaust nozzle. To analyze the thrust performance of the engine, it is sufficient to obtain the total temperature and total pressure of the flow at the exit of the combustion chamber.

Without considering the total temperature losses caused by real gas effects, the total temperature of the exhaust flow is equal to the sum of the total temperature of the incoming flow and the temperature rise (Δ□) caused by the heat release during combustion. By controlling the fuel flow rate, different levels of heat release temperature rise can be achieved. The total temperature of the incoming flow is related to the compression effect of detonation combustion, Mach number at the inlet of the combustion chamber (□□1), and the ratio of heat release temperature rise to the total temperature at the inlet. The exit flow total temperature can be calculated accordingly. Once the total temperature and total pressure of the engine exhaust flow are obtained, the thrust performance of the engine can be calculated using formula (1).

For detonation engines, it can be considered that the gas flow is uniform at the exit. The impulse at the exit can be set as constant along the circumference, and the total impulse(on a unit area basis) can be written as:

$$I_2 = \rho_1 \gamma R M a_1 \sqrt{\frac{2T_{02}T_1}{\gamma-1}} \sqrt{1-\frac{1}{(P_{02}/P_1)^{(\gamma-1)/\gamma}}} \qquad (8)$$

Where, ρ_1 is the density.

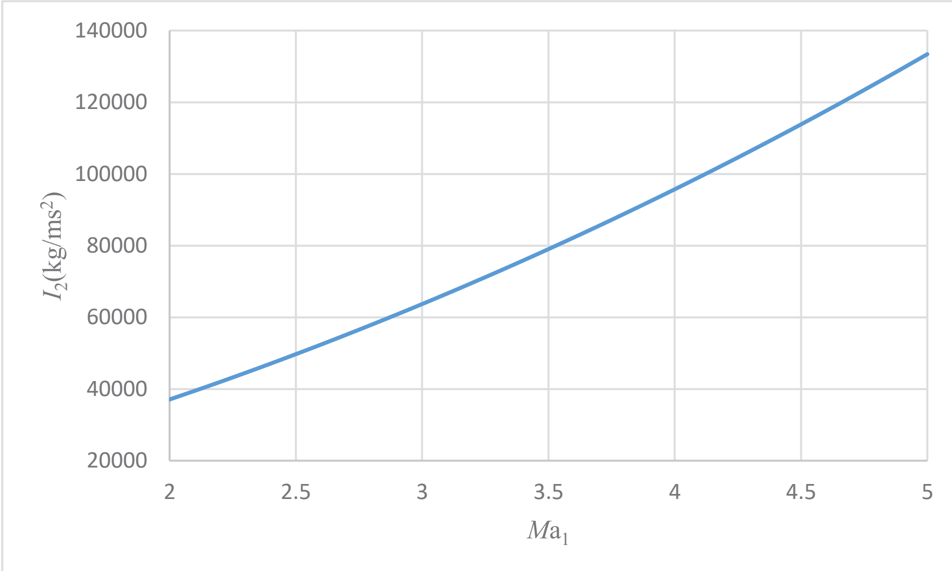


Figure 7: Variation of Specific Impulse with The Mach Number at The Inlet of The Combustion Chamber

Figure 7 shows variation of the detonation engine pressure ratio with the Mach number at the inlet of the combustion chamber. It can be observed that as the inlet Mach number increases, the engine's thrust increases. This increase is due to two factors: an increase in specific impulse and an increase in inlet mass.

4. Conclusion

As the combustible mixture in the combustion chamber increases, the combustion mode changes from deflagration combustion to self-sustaining detonation combustion, and then to forced detonation combustion in order to maintain stable combustion in the combustion chamber. When there is only a small amount of combustible mixture entering the combustion chamber, the combustion mode is deflagration. As the amount of combustible mixture entering the chamber increases to a certain extent, a transition from deflagration combustion to detonation combustion occurs. At this point, due to the inability to form a stable detonation wave, unstable detonation waves occur intermittently. When the amount of combustible mixture entering the chamber reaches a certain level to sustain a stable detonation wave, a stable detonation wave is formed. During the self-sustaining detonation combustion mode, as the amount of combustible mixture in the combustion chamber increases, the number of detonation waves gradually increases from one to multiple, and the angle between the detonation wave and the inflow changes from acute to right. The forced detonation combustion mode is necessary when further increasing the amount of combustible mixture entering the chamber, so that the detonation wave is not blown out of the combustion chamber by the inflow. External energy, such as shock waves, can be introduced for forced detonation. The minimum intensity of the shock wave can be obtained from the tangent on the Rayleigh line. Under the detonation combustion mode, the total temperature at the exit of the combustion chamber increases with the increase in the Mach number at the inlet of the combustion chamber, the pressure ratio decreases with the increase in the Mach number at the inlet of the combustion chamber, and the specific impulse increases with the increase in the Mach number at the inlet of the combustion chamber.

References

1. Andrews, E. (2001). Scramjet development and testing in the United States. In *10th AIAA/NAL-NASDA-ISAS International Space Planes and Hypersonic Systems and Technologies Conference* (p. 1927).
2. Peebles, C. (2008). *Road to Mach 10: Lessons learned from the X-43A flight research program.* American Institute of Aeronautics and Astronautics, Inc.
3. Hank, J., Murphy, J., & Mutzman, R. (2008, April). The X-51A scramjet engine flight demonstration program. In *15th AIAA international space planes and hypersonic systems and technologies conference* (p. 2540).
4. Choi, J., Noh, J., Byun, J. R., Lim, J. S., Togai, K., & Yang, V. (2011, April). Numerical investigation of combustion/shock-train interactions in a dual-mode scramjet engine. In *17th AIAA International Space Planes and Hypersonic Systems and Technologies Conference* (p. 2395).
5. Ashford, S. A. (1994). *Oblique detonation waves, with application to oblique detonation wave engines, and comparison of hypersonic propulsion engines.* The University of Oklahoma.
6. Karagozian, A. R. (2010). Transverse jets and their control. *Progress in energy and combustion science, 36*(5), 531-553.
7. Tomioka, S., Kohchi, T., Masumoto, R., Izumikawa, M., & Matsuo, A. (2011). Supersonic combustion with supersonic injection through diamond-shaped orifices. *Journal of Propulsion and Power, 27*(6), 1196-1203.
8. Smith, L. A., & Farokhi, S. (2015). The interaction of pulsed injection and a supersonic shear layer in a scramjet combustor. In *20th AIAA International Space Planes and Hypersonic Systems and Technologies Conference* (p. 3624).

9. Williams, K. E., Harloff, G. J., & Gessner, F. B. (1995). Investigation of supersonic flow about strut/endwall intersections in an annular duct. *AIAA journal, 33*(4), 586-594.
10. Zhong Fuyu, Ran Wei, Tian Ye, Yu Xin, Peng Jiangbo, & Le Jialing. (2022). Experimental study on the wave system evolution rules of scramjet engine under self-ignition conditions. *Propulsion Technology , 43* (12), 186-194.
11. Li Xuyan, Zheng Xing, Xue Rui. Development Status and Related Suggestions of Scramjet Technology [J]. Science and Technology China, 2019 (02): 5-8
12. Jiang Zonglin. On Supersonic Combustion and Hyperdynamics [J]. Advances in Mechanics, 2021,51 (01): 130-140
13. Lu Hongbo, Chen Xing, Zeng Xianzheng, Chen Yongfu, Sun Riming, Wen Shuai, ... & Jin Yi. (2022). FD-21 wind tunnel Ma= 10 hypersonic propulsion test technology exploration. *Gas Physics , 7* (2), 1-12.
14. Yang Jianing, Shen Chibing, Du Zhaobo. Mixing enhancement law of shockwave interference support plate jet [J]. Journal of Rocket Propulsion, 2023,49 (03): 34-47
15. Deng Shiyu, Jin Zhiguang, Ke Yuxiang. Resistance Characteristics of Low Speed Ramming Channel for Wide Range Combined Engines [J]. Journal of Rocket Propulsion, 2022,48 (06): 44-51
16. Qingwu, W., & Xuedong, Z. (2024). Research on the Combustion Mode and Thrust Performance of Rotating Detonation Scrarmjet Engines.
17. Jincheng, Z., Zhenguo, W., Mingbo, S., Hongbo, W., Yanan, W., & Chaoyang, L. (2022). Auto-ignition tabulated method for supersonic combustion at high Mach number. *Chinese Journal of Theoretical and Applied Mechanics, 54*(6), 1548-1556.
18. Can, H., Jianwen, X., Hao, O., Weixin, D., & Baoguo, X. (2022). FLOW FIELD AND COMBUSTION CHARACTERISTICS ANALYSIS OF SRAMJET UNDER Ma12 FLIGHT CONDITION. *Chinese Journal of Theoretical and Applied Mechanics, 54*(3), 622-632.
19. Lianjie, Y., Xu, Z., Qifan, Z., Keting, C., Jinping, L., Hao, C., ... & Hong, C. (2022). Research progress on high-Mach-number scramjet engine technologies. 力学学报*, 54*(2), 263-288.
20. Qingwu, W., & Xuedong, Z. (2024). Research on the Combustion Mode and Thrust Performance of Rotating Detonation Scrarmjet Engines.