

A Study on Methods for Optimizing Thrust and Thermal Efficiency in Rocket Engines

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Abstract. Nowadays, the global commercial space industry is developing rapidly; deep-space exploration missions keep increasing year by year, and countries around the world are speeding up the construction of both manned and unmanned space stations; rocket engines are the necessary core components of all space launch systems, and they play a very key role in the whole space launch process. Their two basic performance indicators, thrust and thermal efficiency, directly determine the payload capacity of spacecraft, affect the overall cost of each launch mission, and define the farthest reach that deep-space exploration can achieve. This paper sorts out and reviews relevant studies both at home and abroad, and summarizes the latest research progress on optimizing rocket engine thrust and thermal efficiency; it introduces three typical propulsion systems, which are variable specific impulse magnetic plasma engines, air-turbine rocket engines, and liquid oxygen/methane rocket engines, explaining their basic technical principles, main performance strengths and possible future application scenarios; it also analyzes three core efficiency modules, including combustion efficiency, thermodynamic cycle efficiency and nozzle efficiency, sums up the inner influence rules of key structural design parameters on overall thermal efficiency, and makes a quantitative comparison of the efficiency gaps between different thermodynamic cycle structures.

Keywords: Rocket engine, thrust, thermal efficiency

1. Introduction

In recent years, with the advancement of space technology, the global commercial space industry has rapidly emerged, and countries around the world have been continuously expanding deep-space exploration missions and space station construction. As the core of space launch systems, rocket engines directly determine a spacecraft's payload capacity, reliability, and launch costs; they also determine a nation's space launch capabilities, the boundaries of its deep-space exploration, and its ability to ensure space security. Thrust and thermal efficiency are two key indicators for evaluating the performance of rocket engine technology. The thrust of a rocket engine directly determines its payload capacity, while thermal efficiency reflects the proportion of propellant's chemical energy converted into gas kinetic energy, making it strongly correlated with cost. Currently, the global space propulsion sector is undergoing a critical period of technological evolution. The rapid development of reusable rockets has placed higher demands on engine thrust regulation capabilities, variable-

thrust propulsion systems, and thermal efficiency, while space station construction urgently requires propulsion systems with greater thrust [1-3].

In response to the global demand for optimizing the thrust and thermal efficiency of propulsion systems, researchers both domestically and internationally have proposed numerous solutions. Shi et al. [4] pointed out that variable-specific-impulse plasma engines possess significant advantages in terms of high thrust and high power, and hold promising prospects for application in deep-space exploration. Li et al. [5] conducted systematic ground-based thermal test experiments to investigate the effects of local momentum ratio, radial hydraulic diameter of the orifices, and the number of orifices on the combustion efficiency of liquid oxygen/gaseous methane pintle injectors. They found that combustion efficiency first increases and then decreases with an increase in the local momentum ratio, and that an increase in the number of orifices leads to a decrease in combustion efficiency. Direct impinging injector, with their excellent atomization and mixing performance as well as high combustion efficiency, play a significant role in high-thrust engines. Zhang et al. [6] conducted high-pressure combustion tests on direct-current counter-flow nozzles for self-igniting propellants, revealing the influence of the mixing ratio, orifice ratio, momentum ratio, and impact angle on combustion efficiency, thereby providing a reference for the design of injectors in high-chamber-pressure engines.

This paper addresses the issue of optimizing the thrust and thermal efficiency of rocket engines. It focuses on existing methods and concepts for optimizing these parameters, analyzing domestic and international theoretical research, software studies, and experimental analyses across multiple aspects—from thrust optimization of the rocket engine's propulsion system to combustion efficiency, thermal cycle efficiency, and nozzle efficiency—and presents the author's own ideas and recommendations.

2. Thrust-optimized rocket engines

Thrust optimization for rocket engines primarily involves two aspects: high thrust and variable thrust. High thrust effectively enhances a rocket engine's payload capacity, while both high thrust and variable thrust can increase the rocket's range, offering promising applications in crewed lunar missions, Mars exploration, and deep-space exploration.

2.1. Variable specific impulse plasma engine

The Variable Specific Impulse Magnetoplasma Rocket Engine (VASIMR) is a high-power radio-frequency electric propulsion system; its operating principle is illustrated in Figure 1. The Variable Specific Impulse Magnetoplasma Rocket Engine offers numerous advantages over traditional liquid rocket engines. Its Helicon plasma source (HPS) generates high-density plasma, and radio-frequency energy is coupled to the ions via an Ion Cyclotron Resonance Heating (ICRH) unit. Finally, under the action of a magnetic nozzle, the cyclotron energy of the ions is converted into axial kinetic energy, and the combined effects of electromagnetic and aerodynamic forces form a high-speed plasma jet that generates thrust. Theoretical calculations indicate that the ions can reach speeds of up to 300 km/s, which is 60 times that of a conventional chemical rocket engine. Its high specific impulse can reach 3,000–10,000 s, whereas that of a conventional chemical rocket engine is less than 500 s [2]. The thrust of a variable-specific-impulse magnetic plasma engine is flexibly adjustable (1–20 N) [4], which can effectively expand the scope of deep-space exploration and increase the payload capacity of spacecraft.

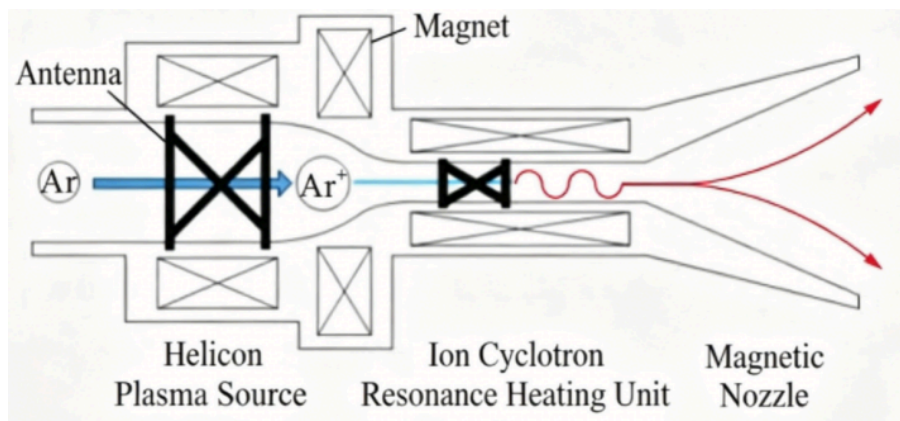


Figure 1. Schematic of VASIMR [4]

2.2. Air turbo rocket engine

The air-turbo rocket engine (ATR) is a type of turbine-based combined cycle (TBCC) engine. In an ATR, the propellant undergoes rich-burn combustion in the rich-burn gas generator to produce high-temperature, high-pressure rich-burn gases, the rich gases drive the turbine to perform work and drive the compressor to compress air; the pressurized air flows through the outer chamber, where it undergoes secondary combustion with the rich gases discharged from the turbine passage (inner chamber) in the downstream combustion chamber; the resulting high-temperature, high-pressure lean gases expand and accelerate through the nozzle to generate thrust [7]. The ATR combines the advantages of both traditional aircraft engines and rocket engines. Compared to traditional chemical rocket engines, it offers higher specific impulse and serves as an ideal solution for the first stage of a two-stage-to-orbit system. Additionally, the air-turbine rocket engine can achieve greater payload capacity under the same thrust through a two-stage-to-orbit configuration: using a horizontal takeoff method, it first accelerates horizontally using specific wings; once sufficient speed is reached, the second stage of the two-stage-to-orbit system propels the vehicle into space.

2.3. Liquid oxygen/methane rocket engines

Liquid oxygen-methane propellant offers advantages such as abundant availability, excellent cooling performance, low combustion temperature, and resistance to coking and carbon buildup. It holds broad application prospects in rocket engines, making high-performance, high-thrust liquid oxygen-methane rocket engines a key research focus for major space missions such as future deep-space exploration [5, 8, 9]. Compared to liquid oxygen/liquid hydrogen rocket engines, liquid oxygen/methane rocket engines consume less propellant and have a density specific impulse that is approximately 10% higher. Take the Raptor engine as an example: as the world's first full-flow staged combustion cycle liquid oxygen/methane rocket engine, it delivers a thrust of up to 269 metric tons at sea level and features a thrust regulation range of 40% to 100%.

3. Optimization of rocket engine thermal efficiency

The thermal efficiency of a rocket engine refers to the ratio of the energy converted into gas kinetic energy from the chemical energy released by the combustion of chemical propellants to the total energy. The thermal efficiency of a rocket engine is primarily determined by three factors: combustion efficiency, thermodynamic cycle efficiency, and nozzle efficiency. Optimizing the

thermal efficiency of rocket engines is an indispensable issue in current aerospace rocket engine research. With the development of commercial spaceflight and breakthroughs in reusable rocket technology, the thermal efficiency of rocket engines—which is directly related to launch costs—has a direct impact on the industry's prospects.

3.1. Combustion efficiency of rocket engines

Combustion efficiency is a key indicator of how thoroughly the chemical energy of a rocket engine's propellant is converted into thermal energy in the exhaust gases. In a direct impinging injector, two or more high-speed propellant jets collide, breaking the propellant into micron-sized droplets to achieve atomization and mixing; the quality of this atomization directly determines combustion efficiency. For unsymmetrical dimethylhydrazine (UDMH)/dinitrogen tetroxide hypergolic propellant (which undergoes violent combustion upon contact of the two components without an external ignition source), research has found that five key parameters—the mixing ratio (the ratio of the oxidizer mass flow rate to the fuel mass flow rate), the orifice ratio (the ratio of the oxidizer orifice diameter to the fuel orifice diameter), the momentum ratio (the ratio of the oxidizer jet momentum to the fuel jet momentum, which characterizes the atomization effect of jet impact), impact angle (the angle between the axes of the two jets), and combustion chamber pressure—have a significant impact on combustion efficiency. In a test system with a maximum operating pressure of 10 MPa, when the mixing ratio is between 1.71 and 2.31, the orifice ratio is between 1.2 and 1.4, the momentum ratio is between 0.68 and 1.28, the impact angle is between 70 and 80 degrees, and the combustion chamber pressure exceeds 3 MPa, combustion efficiency can reach a high level of 97% to 98%. Furthermore, for every 1 MPa increase in combustion chamber pressure, the combustion time is reduced by approximately 0.2 ms, and the loss due to incomplete combustion is reduced by about 0.5% [6].

A pintle injector is a type of injector with a central structure; the oxidizer is sprayed out along the axis from the central pin, and the fuel is ejected in the radial direction through small holes between the pin and the outer wall, then the two streams of fluid hit each other and mix at the tip of the pin. This kind of injector can adjust thrust within a range up to 10 times, and it is widely applied in many top rocket engines around the world, such as the Raptor engine developed by SpaceX. Research results show that three factors have obvious influences on combustion efficiency: the local momentum ratio, the equivalent flow diameter of the radial small holes, and the number of these holes; the local momentum ratio means the ratio between the momentum of radial fuel jet and axial oxidizer jet, and it is a core parameter that affects spray distribution and mixing effect, the equivalent flow diameter of radial holes is a kind of calculated diameter for non-round pipes, and people use it to calculate the flow features of fluid. Among all these factors, the local momentum ratio plays the most important role; combustion efficiency rises first and then falls as this ratio increases, and it reaches the peak when the ratio is 1.2, at which point the combustion efficiency is about 94.7%. Besides, when people reduce the equivalent flow diameter of radial holes from 0.8 mm to 0.4 mm, the average droplet size will decrease by about 40%, and combustion efficiency will increase by around 1.5%; in comparison, the number of holes has a relatively small impact, when the number of holes changes between 12 and 24, the change range of combustion efficiency is less than 0.3% [5].

3.2. Thermal cycle efficiency of rocket engines

Thermal cycle efficiency is a key factor that decides the total thermal efficiency of a rocket engine, and it has a direct effect on how people use propellant energy and the specific impulse performance of the engine; different thermal cycles take different ways to use energy to drive the turbopump and arrange the combustion process, which causes big differences in efficiency, and this is also the main reason why different engines have different thermal efficiency levels.

The gas generator cycle is the most widely used traditional cycle; people burn a small amount of propellant in a separate gas generator to produce high-temperature gas to drive the turbopump, and the exhaust gas from the turbine is directly discharged into the atmosphere. This cycle has a simple structure and mature technology, but about 2%–5% of the propellant energy is directly discharged with the exhaust, so the cycle efficiency is relatively low, and it is generally between 35% and 45%. For the staged combustion cycle, all the exhaust gas from the drive turbine is sent into the main combustion chamber for secondary combustion; this avoids propellant waste and greatly improves the cycle efficiency to 45%–55%. According to the composition of combustion products in the pre-combustion chamber, the staged combustion cycle can be further divided into oxygen-rich staged combustion cycle and fuel-rich staged combustion cycle; in the former, the oxidizer flow is more than the proportion required for complete combustion, while in the latter, the fuel flow is more than that proportion. The full-flow staged combustion cycle is the most advanced thermodynamic cycle at present; it uses two pre-combustion chambers to produce oxygen-rich gas and fuel-rich gas respectively, all propellant passes through the turbopump before entering the main combustion chamber for combustion, and 100% propellant utilization is achieved. Its theoretical cycle efficiency is 55%–60%, which is about 10%–15% higher than that of the gas generator cycle, and about 5%–8% higher than that of the conventional staged combustion cycle [8, 9].

Research shows that the differences in thermal cycle efficiency mainly come from three factors: propellant utilization, combustion chamber pressure, and combustion organization and temperature; the gas generator cycle has the lowest propellant utilization, because the turbine exhaust is directly discharged. Although the staged combustion cycle realizes secondary combustion of turbine exhaust, its single pre-combustion chamber design limits the further increase of combustion chamber pressure; higher pressure will make the gas density higher and combustion more complete, and it also raises the upper limit of Carnot efficiency. For example, the combustion chamber pressure of the Raptor engine reaches 350 bar, which makes its theoretical cycle efficiency 15% higher than that of traditional 100-bar class engines [10]; the full-flow staged combustion cycle not only achieves the highest propellant utilization, but also supports higher combustion chamber pressure, at the same time, gas-gas combustion is more complete than traditional liquid-gas combustion, so it has the highest thermal efficiency.

3.3. Nozzle efficiency of rocket engines

As the final component for energy conversion in a rocket engine, the efficiency of the nozzle directly determines the final utilization of propellant energy and the working performance of the engine; the overall efficiency of a nozzle is divided into geometric efficiency, boundary layer efficiency and chemical dynamic efficiency. Among them, geometric efficiency reflects the radial kinetic energy loss during gas expansion; boundary layer efficiency reflects the energy loss caused by friction between gas and wall; chemical dynamic efficiency reflects the energy loss caused by incomplete dissociation and recombination of high-temperature gas. Increasing chamber pressure or throat radius, or reducing the area ratio, can improve chemical efficiency; reducing the area ratio and

chamber pressure can increase geometric efficiency, but the effect of nozzle radius on geometric efficiency is not a simple one-way change. Increasing chamber pressure and throat radius while reducing the area ratio can improve boundary layer efficiency; as shown in the figure, the overall nozzle efficiency increases with the increase of chamber pressure and throat radius, but decreases with the increase of area ratio. The increase of nozzle exit angle will lead to the decrease of nozzle efficiency; this is because a larger exit angle will make the radial kinetic energy component of gas larger and the axial kinetic energy component smaller, thus reducing the geometric efficiency, and the relationship between nozzle efficiency and chamber pressure, throat radius and exit area ratio is shown in Figure 2 [11].

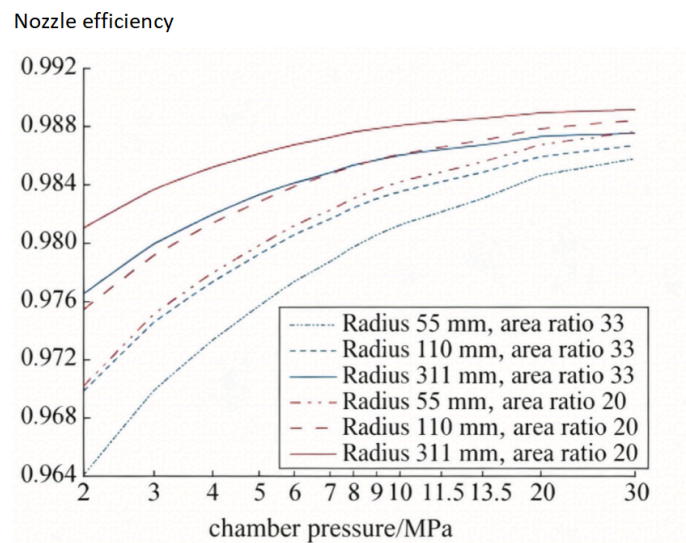


Figure 2. Nozzle efficiency as function of chamber pressure, throat radius, and exit area ratio [11]

4. Conclusion

This paper mainly studies the core problem of improving thrust and thermal efficiency of rocket engines, and this problem is a key challenge in the whole field of aerospace propulsion. It gathers and sorts out relevant research results from both domestic and foreign related studies, and makes a full analysis from two major aspects: thrust improvement and thermal efficiency improvement; the content of these analysis can provide some theoretical reference for the design work and technical iteration of high-performance rocket engines.

Relevant research results show that the work of thrust improvement has developed into a situation where many different technical paths develop in parallel at the same time; each technical path is specially made to fit the diverse demands of different space missions; magnetic plasma engines with adjustable specific impulse have ultra-high specific impulse and flexible thrust adjustment ability, and they have unique advantages for deep-space exploration tasks; air-turbine rocket engines combine the features of aviation propulsion and rocket propulsion together, and they are an ideal solution for the first stage of two-stage-to-orbit launch vehicles; liquid oxygen and methane engines using full-flow staged combustion cycle have many good characteristics like high thrust, wide thrust regulation range and low cost, and they have become the core propulsion choice for major missions like crewed lunar landings and Mars exploration. People can raise thermal efficiency by coordinating and optimizing three key components of the engine: combustion process, thermodynamic cycle and nozzle; both direct-impact injectors and pintle injectors have clear optimal

parameter ranges, and they can make combustion efficiency exceed 94%; the full-flow staged combustion cycle can achieve 100% propellant utilization, its theoretical efficiency ranges from 55% to 60%, and this is 10% to 15% higher than the efficiency of traditional gas generator cycle; the efficiency of nozzle increases with the growth of chamber pressure and throat radius, but it decreases with the growth of area ratio and exit angle.

At present, these current technologies still face some key bottlenecks that are hard to break through; electric propulsion is constrained by the demand for high-power power sources that work in space; combined-cycle engines show insufficient performance when they work across a wide speed range; the full-flow staged combustion cycle also brings great challenges to practical engineering work. Future research work should mainly focus on these core technologies.

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