27th ABCM International Congress of Mechanical Engineering
December 04-08, 2023, Florianópolis-SC, Brazil

COB-2023-2341
PERFORMANCE ANALYSIS OF AN ADDITIVE-MANUFACTURED PROPYLENE-FED ELECTROTHERMAL THRUSTER FOR SMALL SATELLITES

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Abstract. *The aerospace industry has recently expressed a growing interest in electric propulsion systems, especially for use in small satellite missions due to their small size and higher specific impulse when compared to chemical propulsion systems. The presence of a propulsion system on these spacecrafts is important to attitude control, orbital corrections and de-orbiting, rising significantly the lifespan of the machine. In order to meet these requirements, the Combustion and Propulsion Laboratory of the National Institute for Space Research (LCP-INPE) is developing an additively-manufactured resistojet thruster. This paper presents an analytical model to predict and optimize its performance. The advantages and limitations of the model are also discussed. Initial simulations suggest that the resistojet can deliver a specific impulse of over 65s with a thrust of nearly 60mN operating at low power (~200W). Higher values can be easily obtained by rising the heater temperature, lowering heat transfer losses, increasing the propellant mass flow rate or changing the propellant. The model is going to be validated against a series of performance tests at INPE's atmospheric conditions thrust stand.*

Keywords: *electrothermal thruster, micro-propulsion, resistojet, green propellants, metallic 3D printing*

1. INTRODUCTION

The realm of space exploration has experienced momentous growth and development across diverse nations in recent years, owing to the confluence of ambitious government initiatives and the pioneering efforts of private enterprises, culminating in the advent of what is commonly referred to as the "New Space Race." This emergent phase has witnessed the active involvement of key public players, including esteemed space agencies such as NASA in the United States, ROSCOSMOS (Russian Federal Space Agency) in Russia, ESA (European Space Agency) in Europe, JAXA (Japan Aerospace Exploration Agency) in Japan, CNSA (China National Space Administration) in China, and ISRO (Indian Space Research Organization) in India. Concurrently, the private sector has observed a proliferation of collaborations with public entities, propelling the transformation and cost optimization of space technologies. Remarkable entities actively engaged in this dynamic domain comprise SpaceX, Rocket Lab, Relativity Space, and the highly regarded Brazilian company, Pion Labs.

Within this context, satellites have assumed a pivotal and ever-increasing role, facilitating Earth monitoring (e.g., deforestation, wildfires, glacier melt, and natural disasters), providing positioning systems and offering low-latency communications and internet services (e.g., Starlink).

Among the various systems that constitute a space project, the propulsion system plays a critical role in a mission's success. In a launch vehicle, it is responsible for the ascent of the spacecraft or satellite beyond the Earth's atmosphere and, if needed, for orbital insertion. In a spacecraft, it is crucial for executing planned maneuvers, while in a satellite, it handles orbit corrections, orbit changes, and attitude control. Without a properly functioning propulsion system, a satellite's operational life can be significantly shortened, or it may be unable to fulfill its mission.

There are numerous propulsion systems that can be classified based on different criteria, such as the energy source. These systems encompass chemical propulsion, nuclear propulsion, electric propulsion, solar propulsion, laser propulsion, and more. Within chemical propulsion, further classifications can be made based on the physical state of the propellants, including solid, liquid, hybrid, cold gas, gel, emulsion, and more. Electric propulsion systems can be categorized as electrothermal, electrostatic, and electrodynamic. Chemical propulsion can also be classified based on the number of propellants used, leading to monopropellant, bipropellant, and multipropellant systems. Propulsion systems can also be categorized based on their application as primary propulsion or secondary propulsion, or in terms of space mission as interorbital, interplanetary, or Earth-to-orbit propulsion.

Electrothermal thrusters have the capability to offer specific impulses superior to those typically achieved by monopropellant chemical propulsion systems commonly used in satellites, featuring relatively simple construction and operation, low cost, and reliability (Sutton and Biblarz, 2017). Moreover, they have reduced mass and volume compared to other electrical systems, and thrust and specific impulse can be controlled through the regulation of propellant flow and energy supply.

The operation of electrothermal thrusters relies on the addition of thermal energy to the propellant through electrical means. This enthalpy in the form of energy is converted into kinetic energy of exhaust when the propellant passes through the thruster nozzle. Heating can be achieved through various methods, such as electrical resistance (resistojet), electrical discharges (arcjet), and high-frequency radiation or electrodeless electrical discharges.

Among the electric propulsion systems, the resistojet presents a simple, reliable, light and inexpensive option. Figure 1 provides visual examples of resistojets that have been successfully manufactured using additive manufacturing techniques, highlighting the feasibility and potential of this approach for producing such propulsion systems. Through the application of additive manufacturing, the thruster's intricate design can be faithfully replicated, ensuring optimal performance and reliability.

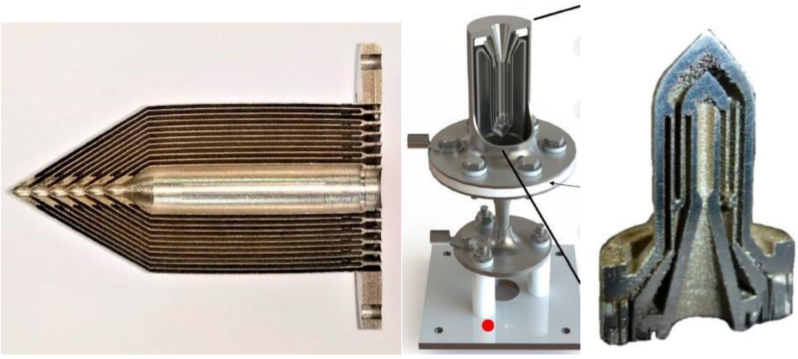

Figure 1. Examples of resistojet thrusters manufactured using metal additive manufacturing. a) Inconel heat exchanger (Coral et al., 2021); b) and c) Stainless steel heat exchanger (Romei and Grubišić, 2020).

This work aims to present the initial design of an electrothermal thruster prototype utilizing propylene as the propellant for orbit correction and attitude control in small satellites. Propylene is a green, self-pressurized propellant known for its low toxicity and minimal environmental impact. A simplified theoretical model is used to estimate the temperature of the heated gas and the specific impulse (the ratio of thrust to the weight of the propellant used) with perfect expansion in a vacuum, considering different heating powers and propellant mass flow rates.

2. METHODS

2.1 Propellant

The chosen propellant for the propulsion system is propylene, also known as propene, primarily due to its exceptional self-pressurizing properties, since its saturation pressure at 298 K is around 12 bar, sufficiently high to eliminate the need for an external pressurization system or a pumping system in most cases, making the propulsion system less complex, more cost-effective, lightweight, and reliable. The selection of propylene is further justified by its low cost, wide availability, and favorable propulsive performance.

2.2 Simplified Flow Model in the Thruster

A simplified theoretical analytical model of the flow in the electrothermal thruster has been developed based on the mass balance, momentum balance, and energy balance equations to determine the temperature of the heated gas and the specific impulse of the thruster. Initially, a one-dimensional steady flow of inert gas with heat exchange and no friction along the heat exchanger, as well as an isentropic flow along the nozzle, were considered. A schematic of the flow in the thruster is shown in Figure 2.

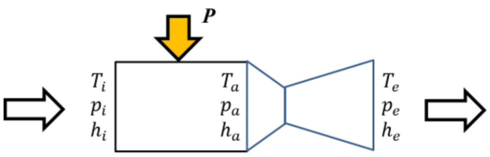

Figure 2. Schematic of the flow in the thruster

The first law of thermodynamics (Borgnakke and Sonntag, 2013) applied to the heat exchanger provides:

$$h_i + \frac{P}{\dot{m}} = h_a, \tag{1}$$

where h_i, P, $\dot{m}$ and h_a are the enthalpy on the inlet of the heat exchanger, the supplied power, the mass flow rate and the enthalpy on the outlet of the heat exchanger, respectively.

Considering a perfect gas and calorically perfect gas, with $h = c_P T$, where T is the temperature and cp is the specific heat of the gas, we can express it as:

$$T_a = T_i + \frac{P}{c_p \dot{m}}, \tag{2}$$

where T_a, T_i and c_p are the temperature on the outlet of the heat exchanger, the enthalpy on the inlet of the heat exchanger and the specific heat in constant pressure, respectively.

Once the heating temperature is determined as a function of the supplied power and propellant mass flow rate, the specific impulse (Isp) of the thruster can be estimated using the following equation (Sutton and Biblarz, 2017):

$$Isp = \frac{1}{g_0} \sqrt{\frac{2\gamma}{\gamma - 1} R T_a \left[1 - \left(\frac{p_e}{p_a} \right)^{\frac{\gamma}{\gamma - 1}} \right]}, \tag{3}$$

where $Isp, g_0, \gamma, R, p_e \; and \; p_a$ are the specific impulse, the acceleration of gravity on the surface of the earth, the specific heat ratio, the pressure on the nozzle exit and the pressure on the outlet of the heat exchanger, respectively.

The optimal specific impulse is calculated assuming gas expansion in a vacuum, that is, pe = 0, resulting in:

$$Isp_{optimal} = \frac{1}{g_0} \sqrt{\frac{2\gamma}{\gamma - 1} R T_a}, \tag{4}$$

Figure 2, 3, and 4 depict, respectively, the temperature of the heated gas, the optimal specific impulse, and the theoretical thrust for different heating powers and propellant mass flow rates.

Finally, the theoretical impulse can be obtained by the relation (Hill and Peterson, 2015):

$$F = \dot{m} g_0 Isp, \tag{5}$$

2.3 Thruster design

The thruster consists of a heat exchanger and a conical nozzle with an expansion ratio of 5 for laboratory testing or an expansion ratio of 100 for vacuum operation. The mass and length of the chamber and nozzle should be minimized.

A maximum chamber pressure of 10 bar, a maximum temperature of the heated gas of 2000 K, and a maximum heating power of 500 Watts are considered.

The heat exchanger will be composed of a cartridge heater surrounded by concentric channels. The contact area and temperature difference between the gas and the heater should be maximized. The designs of the heat exchanger and the thruster are being performed using the CAD software *Autodesk Fusion 360*. Figure 3 shows a preliminary computational cutaway view of the thruster.

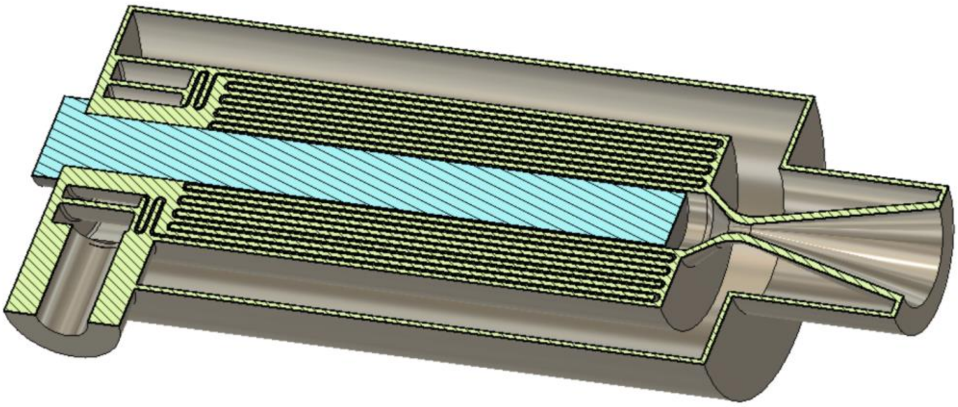

Figure 3. Preliminary computational cutaway view of the thruster.

The decision to manufacture the thruster using additive manufacturing is driven by the intricate and complex geometry involved in its design. Additive manufacturing enables the creation of intricate structures with high precision, making it a suitable and efficient method for producing the thruster components.

3. RESULTS AND DISCUSSION

Based on the theoretical relationships presented above, graphs could be generated that depict the relationships between propellant mass flow rate and supplied power with outlet temperature, specific impulse, and theoretical thrust, as depicted in Figure 4. Furthermore, in the implementation of the code, the NASA-CEA was used to obtain the gas properties needed.

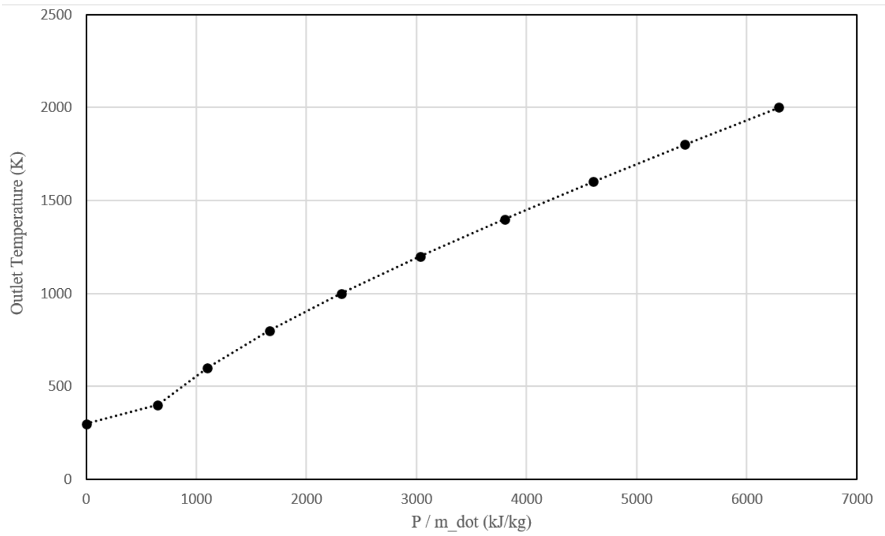

Figure 4. Outlet Temperature vs $\frac{P}{\dot{m}}$.

The exit temperature can be represented as a function of the supplied power for each propellant mass flow rate. Thus, it can be observed that the exit temperature for each mass flow rate is directly proportional to the supplied electrical power, as depicted in Figure 5.

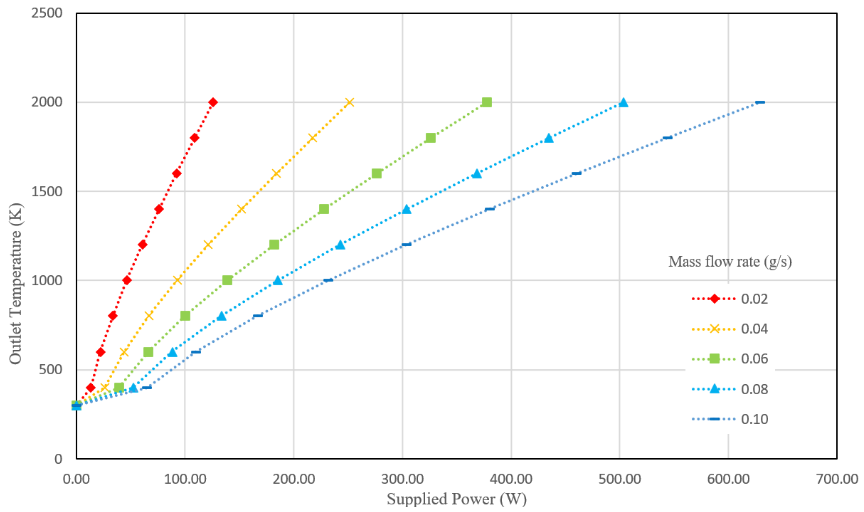

Figure 5. Outlet Temperature vs Supplied Power.

Using the equation for the optimal specific impulse, it is possible to obtain the specific impulse as a function of the outlet temperature, Figure 6. It can be observed that the specific impulse is proportional to the outlet temperature, as expected. From a thermodynamic perspective, this can be explained by the fact that a higher temperature provides more thermal energy to be converted into kinetic energy by the convergent-divergent nozzle. Since the specific impulse can be regarded as an indirect measure of the exhaust velocity of the heated gases, it will increase with the outlet temperature.

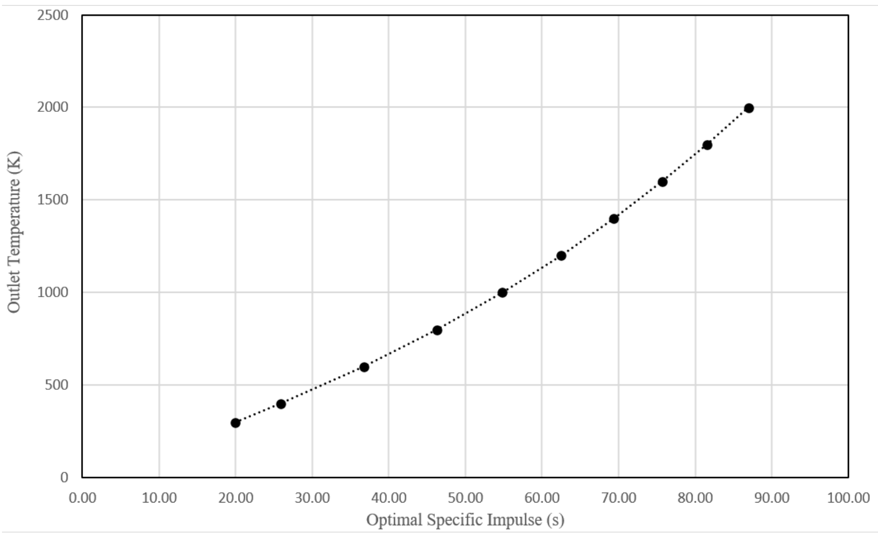

Figure 6. Outlet Temperature vs Optimal Specific Impulse.

The same can be done for the specific impulse to obtain a graph of specific impulse against the supplied electrical power for each propellant mass flow rate. It can be observed that for each mass flow rate, the specific impulse is directly proportional to the supplied power, as can be seen in Figure 7.

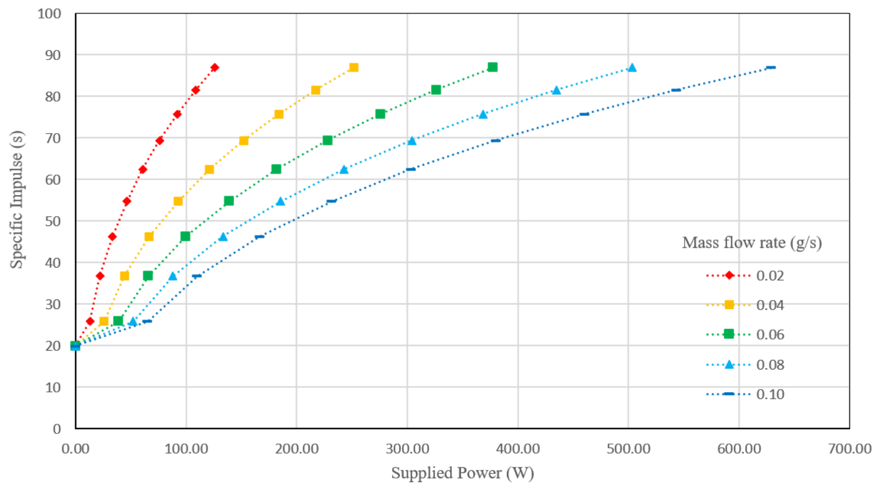

Figure 7. Specific Impulse vs Supplied Power.

In terms of the theoretical thrust, in Figure 8, a slightly different behavior can be observed as it increases with both outlet temperature and mass flow rate. This behavior is expected, as thrust is proportional to the propellant mass flow rate and the exhaust gas velocity, which, in turn, is proportional to the outlet temperature.

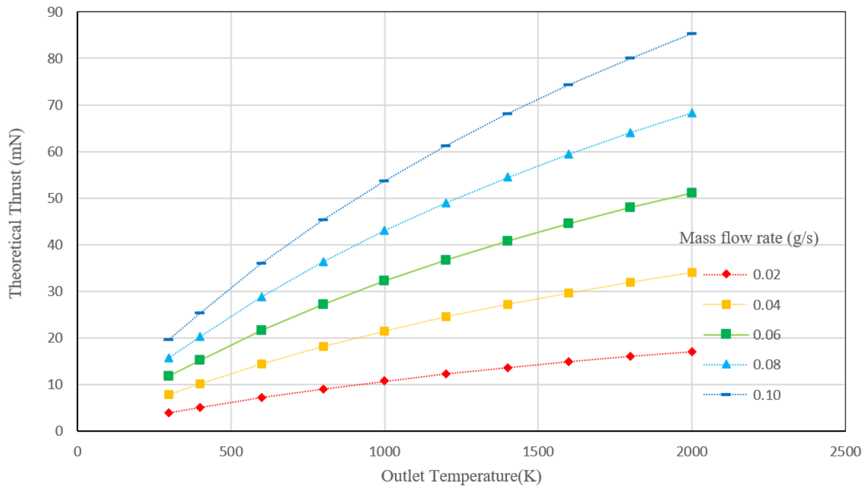

Figure 8. Theoretical thrust vs Outlet Temperature.

The findings derived from the conducted experiments reveal a clear positive correlation between the heating temperature and the optimal specific impulse, as both parameters exhibit an increase with the rise in supplied power and a decrease in propellant mass flow rate. However, it is crucial to consider the material's operating temperature limit and the available power resources within the satellite or spacecraft, as these factors impose practical constraints on the maximum achievable heating temperature.

Furthermore, the permitted mass flow rates for attaining heating temperatures around 500 K are found to be in the order of tenths of grams per second, resulting in a substantial limitation on the provided thrust capabilities. These results shed light on the intricate interplay between heating temperature, specific impulse, power supply, and mass flow rate, offering essential insights for the optimization and design of propulsion systems for future space missions.

Regarding the limitations of the model, it is possible to understand that since it is based only on the first law of thermodynamics, it does not consider the heat transfer phenomena occurring inside the heat exchanger and with the thruster vicinity. In this sense, in order to achieve more precise values, a more complete analysis needs to be carried out.

Following the manufacturing process, the thruster will undergo rigorous testing on test benches, where critical parameters such as temperature and thrust will be meticulously measured and analyzed. Additionally, the specific impulse, a fundamental performance metric, will be calculated to validate the accuracy and efficacy of the thruster's theoretical model. Through comprehensive testing, the thruster's operational capabilities and adherence to the theoretical predictions will be thoroughly evaluated.

4. CONCLUSIONS

In conclusion, the findings derived from this study reveal a positive correlation between the heating temperature and the optimal specific impulse, a relationship further influenced by the supplied power increment and the reduction in propellant mass flow rate. It can be seen that for achieving temperatures of 1200 K, a power supply in the range of 50 to 300 W is needed, depending on the mass flow rate chosen. In this case, a specific impulse of 65 s can be obtained. Regarding the theoretical thrust, the values are contained in the range of 10 to 60 mN.

Nevertheless, it is crucial to exercise caution and account for the material's operating temperature limit and the available power resources within the satellite or spacecraft. These considerations impose practical constraints on the maximum attainable heating temperature. Moreover, the permissible mass flow rates for achieving heating temperatures around 1200 K are constrained to the order of tenths of grams per second, substantially restricting the provided thrust capabilities.

As space missions demand a delicate balance between performance and resource limitations, the insights garnered from this research contribute valuable guidance towards informed decision-making in the design and implementation of propulsion systems for future space endeavors.

5. ACKNOWLEDGEMENTS

This work was funded by CNPq.

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7. RESPONSIBILITY NOTICE

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