Total Impulse Increase of a Micro-Solid Rocket Using a Stack of B/KNO₃ Pellets

By Jun ASAKAWA,¹⁾ Hiroyuki KOIZUMI,²⁾ Shunichi KOJIMA,¹⁾ Masakatsu NAKANO,³⁾ Nobu OKADA⁴⁾
and Kimiya KOMURASAKI¹⁾

¹⁾Department of Aeronautics and Astronautics, The University of Tokyo, Tokyo, Japan

²⁾Department of Advanced Energy, The University of Tokyo, Tokyo, Japan

³⁾Tokyo Metropolitan College of Industrial Technology, Tokyo, Japan

⁴⁾ASTROSCALE PTE. LTD., Singapore

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A micro-solid rocket as the propulsion system for 1–10 kg-class micro-spacecraft is proposed here. The micro-solid rocket uses a boron/potassium nitrate pellet as propellant and its total impulse is about 1.5 Ns. Higher total impulse is needed for a propulsion system on small spacecraft to perform advanced space missions such as sample return, formation flight, and active debris removal. To increase the total impulse, it is necessary to increase the propellant mass. However, there is a difficulty in producing new sizes of solid propellant. The author designed a 20–50 Ns-class micro-solid rocket which uses a stack of existing multiple B/KNO₃ pellets. The side of the propellant pellets was sealed with epoxy resin to prevent an abnormal combustion chamber pressure rise. As a result, all the propellant was burned without an abnormal pressure rise in all combustion tests.

Key Words: Micro Solid Rocket, Micro Thruster, Solid Propellant, Boron/Potassium Nitrate, Solid Booster

Nomenclature

n	: pressure index
p	: combustion chamber pressure
r	: burning velocity

Subscripts

0	: initial
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1. Introduction

In recent days, universities, research institutions, and start-up companies have taken more interest in small spacecraft. 394 small spacecraft with a mass of less than 200 kg were launched in the 11 years from 2003 to 2013. In addition, more and more 1–10 kg-class micro-spacecraft were launched in 2013. This is because America launched 60 micro-spacecraft in 2013¹⁾.

Micro-spacecraft require a propulsion system to perform advanced space missions such as lunar exploration or observation using formation flight. A propulsion system for micro-spacecraft should have a low weight, be small, and have low power consumption. Therefore, the propulsion systems of standard spacecraft are not suitable for micro-spacecraft. It is necessary to develop micro-thrusters to meet the mission requirement for micro-spacecraft.

The required propulsion system performance depends on the space mission. An electric propulsion system, such as an ion thruster or a Hall thruster, is suitable for performing an orbital transfer in a long time period. A chemical propulsion system is suitable for performing an orbital transfer in a short duration of time, such as atmospheric re-entry, and orbital insertion.

Monopropellant or bipropellant propulsion systems have high thrust and high impulse. However, the complexity of needing

separate feeding systems for the fuel and oxidizer, for example, valves and injectors, is a disadvantage.

A cold gas thruster simply expels gas without chemical reaction from a pressurized propellant tank. In the tank, the propellant is stored at 1 to 200 atm. Its greatest advantage is the simple structure compared with other gas thrusters. Nevertheless, the thrust decreases over time because the propellant tank pressure decreases with use. Additionally, the specific impulse is small, and usually ranges from 30 to 100 s.

Solid rocket motors have advantages such as a compact size, and not needing a propellant feeding system. However, a disadvantage is difficulty in throttling the thrust.

Many researchers have researched micro-solid rockets. In 2003, a digital micro-thruster using MEMS technology was developed at the University of Tohoku and JAXA^{2,3)}. The digital micro-solid rocket had 10,000 solid propellant pellets with diameters of 0.80 mm, and controlled the thrust using an integrated circuit.

Since 2003, the University of Tokyo has researched micro-solid rockets which have a 1.0 g solid propellant pellet ignited by a diode laser. A schematic of a micro-solid rocket is shown in Fig. 1. Propellant pellets are stored inside the arrayed combustion chambers. Solid propellant pellets are ignited by a diode laser. A diode laser is located at the center of annularly arrayed combustion chambers. All of the pellets could be ignited with a single laser and actuator by rotating the arrayed combustion chamber⁴⁾.

In previous studies, a 1 Ns-class micro-solid rocket was studied. The micro-solid rocket uses multiple B/KNO₃ pellets and single pellet weighed about 0.90 g⁵⁻⁷⁾. This thruster provides micro-spacecraft with a mass of 5.0 kg about 10 Ns of impulse. However, 500 Ns is needed for micro-spacecraft with a mass of 5.0 kg to perform re-entry.

Higher total impulse is needed for micro-spacecraft to expand their capabilities, for example, to perform a sample

return or re-entry. However, it is costly to produce new solid propellant sizes which are suitable for various required impulse values. In addition, some qualifications are required to produce solid propellant. Therefore, the author proposes to increase the propellant mass by stacking multiple B/KNO₃ pellets. The thrust can be adjusted by increasing or decreasing the number of pellets. This enhances the flexibility of the micro-solid rocket.

One of the concerns when stacking multiple pellets is uncontrolled combustion chamber pressure rise with an expansion of the combustion area. The author proposes to seal the side of propellant pellets using epoxy resin to prevent the combustion chamber pressure from rising abnormally. The objective of this research is to suppress the combustion chamber pressure in less than 1.5 MPa by sealing the pellet side using epoxy resin. The author designed thrusters and measured the combustion chamber pressure.

Fig. 1. A schematic of a micro-solid rocket.⁴⁾

2. Experimental Setup

2.1. Propellant

For the propellant, boron/potassium nitrate (B/KNO₃) pellets were used because of its ignition ability in a vacuum. In atmosphere, all of the solid propellant were easily ignited. However, several solid propellants, for example, composite and double-base propellant, were not ignited in a vacuum.

The chemical reaction of B/KNO₃ is shown in Table 1. When the temperature reaches 720 K, the first reaction occurs. This reaction is exothermic reaction and generates abundant heat. When the temperature reaches 820 K, the second reaction begins. This reaction is endoergic reaction. Unreacted KNO₃ is decomposed in the second reaction. Finally, when the temperature reaches 1150 K, the third reaction occurs. KBO₂ generated in the first reaction is decomposed in this reaction. The first reaction occurs between solid boron and melted KNO₃ because the melting points of boron and KNO₃ are 2340 K and 612 K, respectively. Therefore, the oxidation of boron occurs only on the surface of boron particles. The diameter of a boron particle has an influence on the reaction. It was confirmed that the combustion velocity changes when the diameter of the

boron particles change^{8,9)}.

In general, the burning velocity of solid propellant is represented as a function of the combustion chamber pressure. The experimental equation is called Vieille's equation and shown in Eq. (1).

$$r = ap^n \quad (1)$$

The coefficient a and the pressure index n change depending on the environment. It is evident that the burning velocity is very sensitive to the pressure index. In general, $n < 1$ is a good condition for stable combustion.

According to the data acquired by NiGK Corporation which produces B/KNO₃ pellets, the burning velocity of a B/KNO₃ pellet at a background pressure of 1 atm can be calculated using Eq. (2). The pellet is ignited at normal temperature.

$$r = r_0 \left(\frac{p}{p_0} \right)^{0.08} \quad (2)$$

$$r_0 = 24 \text{ mm/s}, p_0 = 100 \text{ kPa}$$

The burning velocity of B/KNO₃ is insensitive to pressure because the pressure index is 0.08. The lower the background pressure, the more the combustion velocity deviates from Eq. (2). Furthermore, the burning velocity is constant when the background pressure is less than 1 kPa. The burning velocity of B/KNO₃ depending on the background pressure is shown in Fig. 2¹⁰⁾.

Table 1. A burning process of B/KNO₃.⁸⁾

Temperature /K	Reaction
403	Crystal transformation of KNO ₃
612	Melting of KNO ₃
720-810	Exothermic reaction $B + KNO_3 \rightarrow KBO_2 + NO$
820-970	Decomposition of excess KNO ₃ (Endothermic reaction) $2KNO_3 \rightarrow 2KNO_2 + O_2$ $4KNO_2 \rightarrow 2K_2O + 4NO + O_2$
1150	Decomposition of KBO ₂ ($\rightarrow K_2O, B_2O_3$)

Fig. 2. The burning velocity of B/KNO₃ against to a back ground pressure.¹⁰⁾

2.2. Single propellant pin thruster (thruster #1)

Thruster #1 uses one propellant pin which consists of a stack of 20 pellets. A schematic of thruster #1 is shown in Fig. 3. Each propellant pellet is stacked by applying epoxy adhesive on the side. A propellant pin is inserted in an acrylic pipe. The gap between the side of the propellant and the inner wall of the acrylic pipe is filled with epoxy resin. The combustion chamber and nozzle are made of epoxy resin. The propellant pin is ignited by irradiating with a 1 W diode laser through an acrylic window from an oblique direction.

Fig. 3. A schematic of single propellant pin thruster.

2.3. Clustered propellant pins thruster (thruster #2, #3)

Thrusters using clustered propellant pins are also designed. This thruster consists of a combustion chamber, nozzle, propellant, ignition device, and propellant holder. Propellant pins are ignited by electrical heat. A schematic of the thruster is shown in Fig. 4.

Fig. 4. A schematic of clustered propellant pins thruster.

Two thruster types are designed. Thruster #2 uses 4 propellant pins. Each pin consists of 3 stacked pellets. Thruster #3 uses 7 propellant pins. Each pin consists of 3, 5 or 8 pellets. The stack number is changed in 3 steps. A gap between the propellant pellet side and the inner wall of the combustion chamber is filled with epoxy resin.

2.4. Diode laser

A propellant pin is ignited by irradiating with a 1 W diode laser through an acrylic window in thruster #1. The diode laser is produced by Hamamatsu Photonics, model L10451-42. The laser beam from the diode laser is concentrated to an energy density of 5 MW/m² by a small condensing lens. The condensing lens is produced by THORLABS, model C230260P-B. Both the diode laser and the condensing lens are mounted in a socket. The configuration of the laser system is shown in Fig. 5.

Fig. 5. The configuration of the laser system

2.5. Electrical ignitor

The electrical ignition device consists of 2 electrical ignitors designed for model rockets, 12 B/KNO₃ pellets with a diameter of 3.2 mm and casing. A schematic of the ignition device is shown in Fig. 6. First, heat is generated at the resistor by flowing electrical current through the electrical ignitor. Subsequently, B/KNO₃ pellets with a diameter of 3.2 mm mounted on the ignition device are ignited by transferring heat to the pellets. Finally, propellant pins are ignited by collision with particles which have high velocities and temperatures generated by the combustion of pellets. The case of the ignition device is made from acrylic resin and produced using a 3D printer.

Fig. 6. Cross-sectional photo of thruster #1 after a combustion test

2.6. Vacuum chamber

All the experiments in this research are conducted in a cylindrical vacuum chamber with a diameter of 360 mm and a depth of 1400 mm, at a background pressure of less than 100 Pa during operation. In past studies, it was found that the background pressure has negligible influence on the burning characteristics and thruster performance¹⁰⁾.

2.7. Pressure sensor

The combustion chamber pressure is measured in combustion tests of thrusters #2 and #3. The pressure transducer is made by Kulite, model XTEH-7L-190M-500A, and connected to the port on the side of the combustion chamber through a stainless tube. Configuration of thruster and sensor is shown in Fig. 9.

Fig. 9. Thruster #1 before a combustion test.

3. Results and Discussion

3.1. Combustion test of thruster #1

Combustion tests were conducted 3 times using thrusters with the same design. All the propellant was burned without structural failure of the thruster casing. Fig. 10 and Fig. 11 show photos of thruster #1 before and after a combustion test.

Localized erosion was observed at the throat. An enlarged photo of the throat after a combustion test is shown in Fig. 12. It is expected that the combustion chamber pressure decreases due to local erosion at the throat. In general, in the terms of the continuation of combustion, a high combustion chamber pressure is preferable. The method of stacking propellant pellets by applying epoxy adhesive on the side and sealing the side of the propellant using epoxy resin is useful because combustion continues even under the condition that the combustion chamber pressure decreases.

Fig. 10. Thruster #1 before a combustion test.

Fig. 11. Thruster #1 after a combustion test.

Fig. 12. A throat of thruster #1 after a combustion test.

3.2. Combustion test of thruster #2

Combustion tests were conducted 2 times using thrusters with the same design. All the propellant was burned without an abnormal pressure rise. Fig. 13 – 16 show the inside of the thruster before and after the combustion tests. Fig. 17 shows measurement results of the combustion chamber pressure for the two combustion tests. Combustion from the side of the propellant did not occur because the maximum combustion chamber pressure was about 0.30 MPa.

When the throat diameter is 6.9 mm and the thrust coefficient is 1.95, the experimental total impulse from the combustion chamber pressure measurements are 8.2 Ns (the first test) and 12 Ns (the second test)¹¹⁾. This experimental total impulse is in good agreement with the estimated value of 12 Ns.

From the pressure measurement, the internal combustion mode is different between the first and second combustion tests. In the first combustion test, the combustion chamber pressure rose slightly and maintained the condition after electrical current began to flow. Subsequently, the pressure rose suddenly and decreased to the background pressure after reaching an extreme value. On the other hand, in the second combustion test, the combustion chamber pressure rose suddenly after electrical current began to flow and decreased to the background pressure after passing some extreme values. The initial pressure rise is due to combustion gas generated from the ignition device. A maintained initial pressure rise means a time lag from the beginning of combustion to ignition of the main propellant pins.

Fig. 13. The inside of the thruster #2 before the 1st combustion test.

Fig. 14. The inside of the thruster #2 before the 2nd combustion test.

Fig. 15. The inside of the thruster #2 after the 1st combustion test.

Fig. 16. The inside of the thruster #2 after the 2nd combustion test.

Fig. 17. Pressure measurement result of thruster #2.

3.3. Combustion test of thruster #3

Combustion tests were conducted 8 times in total while changing stack number of pellet. As for 3 stack thruster, 2 combustion tests were conducted. As for 5 stack thruster, 3 combustion tests were conducted. As for 8 stack thruster, 2 combustion tests were conducted. All the propellant was burned in all tests without an abnormal pressure rise. Fig. 18 and Fig. 19 show the inside of the thruster before and after the test. Fig. 20 - 22 shows the measurement result of the combustion chamber pressure for each stack number. Combustion from the side of the propellant did not occur because the maximum combustion chamber pressure is less than 1.5 MPa.

The first and second pressure histories of 5 stack thrusters were different from the third one. In the first and second tests, pressure reached 1.0 MPa just after ignition. However, the pressure history of the third test did not have such a peak. This is because heat is transmitted from the ignitor to propellant pins in a different process. In the first and second tests, heat is transmitted to not the entire surface but a part of surface. This leads to point ignition. Burning area increases spherically from the ignition point. As a result, pressure histories have a peak just after the ignition.

In the second test of 8 stack thruster, pressure reached 1.0 MPa in the middle of operation. This abnormal pressure rise can be explained by two causes. One explanation is that a throat area decreases temporally when a part of epoxy resin filled in thruster is broken and exhausted with combustion gas. The other explanation is that burning area expands rapidly when filled epoxy resin is affected by thermal stress and cracks.

When the throat diameter is 6.9 mm and the thrust coefficient is 1.95, experimental total impulse from the combustion chamber pressure measurement and estimated total impulse are shown in Fig. 23. Experimental total impulse in each stack number is in good agreement with the estimated one.

Fig. 18. The inside of the thruster #3 before a combustion test.

Fig. 19. The inside of the thruster #3 after a combustion test.

Fig. 20. Pressure measurement result of 3 stack thrusters.

Fig. 21. Pressure measurement result of 5 stack thruster.

Fig. 22. Pressure measurement result of 8 stack thruster.

Fig. 23. Experimental and estimated total impulse against to stack number.

4. Conclusion

The propellant mass of a micro-solid rocket was increased by using a stack of $\varnothing 10$ mm propellant pellets. 3 types of thrusters with an increased propellant mass were designed and combustion tests were conducted. The total impulse was calculated from measurement results of the combustion chamber pressure and was in good agreement with the design values. The results of the combustion tests showed that by sealing the side of the propellant using epoxy adhesive and epoxy resin, all the propellant was burned without an abnormal pressure rise.

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