Research and Development of PROITERES Micro/Nano-Satellite Series at Osaka Institute of Technology

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Abstract: The Project of Osaka Institute of Technology Electric-Rocket-Engine onboard Small Space Ship (PROITERES) was started at Osaka Institute of Technology in 2007. In PROITERES, a nano satellite named the 1st PROITERES satellite with electrothermal pulsed plasma thrusters (PPTs) was launched by Indian PSLV C-21 rocket on September 9, 2012. The 2nd PROITERES satellite has been developed since 2010. The main mission is to change long-distance orbital altitude with high-power electrothermal PPTs for micro/nano-satellites. The 2nd PROITERES satellite was determined to be launched by H-IIA rocket of JAXA on July in 2018. And, we are planning the 3rd and 4th PROITERES satellites. In this paper, we introduce the R&D feature and final checking, launch process and final operation of the 1st PROITERES satellite FM. Furthermore, the research and development of the 2nd, 3rd, and 4th PROITERES satellites with high-power electric thrusters, i.e., high-power electrothermal PPT and cylindrical Hall thruster etc., are also introduced.

I.Introduction

The Project of Osaka Institute of Technology Electric-Rocket-Engine onboard Small Space Ship (PROITERES) was started at Osaka Institute of Technology in 2007. In PROITERES, a nano satellite named "1st PROITERES" with electrothermal pulsed plasma thrusters (PPTs) was successfully launched on a Sun-synchronous orbit of 660 kilometers in earth altitude by PSLV C-21 launcher at Satish Dhawan Space Center (SDSC), Indian Space Research Organization (ISRO) on September 9th, 2012. Figure 1 shows photographs of the launch feature of the PSLV C-21 rocket[1,2)].

We developed Bread Board Model (BBM) and Engineering Model (EM) of the satellite, including electrothermal PPT system, high-resolution camera system, onboard system and etc, in 2007-2009. Finally, the development of the satellite Flight-Model (FM) was completely finished in 2010. Now, the 1st PROITERES was blackout and we are sending FM telemetry on thruster firing and continuing observation.

Now, we are developing the 2nd PROITERES satellite, as shown in Fig. 2. The R&D of the 2nd PROITERES was started in 2010. The main mission is to achieve longer distance orbital transfer, i.e. changing 50-100 km in altitude on near-earth orbits by longtime operation and high power PPT system. The 2nd PROITERES satellite was determined to be launched to 613 km in earth altitude by H-IIA rocket at Tanegashima Space Center, Japan Aerospace Exploration Agency (JAXA) on July in 2008. The 2nd PROITERES satellite will be launched by piggyback payloads of GOSAT-2/Khalifasat.

In this paper, we introduce the R&D of launch process, operation history of the 1st PROITERES. Furthermore, the R&D of the attitude control devices of 2nd PROITERES, 3rd PROITERES and 4th PROITERES with high-power electric thrusters, i.e., high-power electrothermal PPT, magnetic torquer and cylindrical Hall thruster, respectively, are also introduced.

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Figure 1. Launch of PSLV C-21 rocket (Photo Courtesy of ISRO).

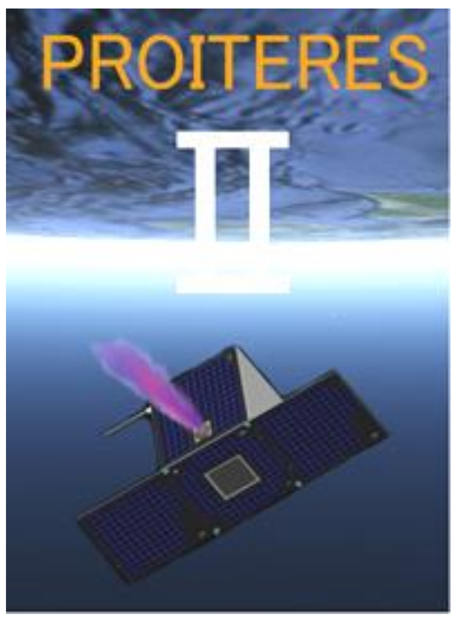


Figure 2. Image of the 2nd PROITERES satellite.

II.Overview of the 1st PROITERES satellite

Figure 3 shows the FM of the 1st PROITERES satellite. The specification of the satellite is shown in Table 1. The weight is 14.5 kg, the configuration is a 0.29 m cube, and the minimum electric power is 10 W. The altitude is 660 km in Sun-synchronous orbit. The lifetime is above one year. The main mission of the satellite are to achieve powered flight of nano-satellite by an electric thruster and to observe Kansai district in Japan with a high-resolution camera[3].

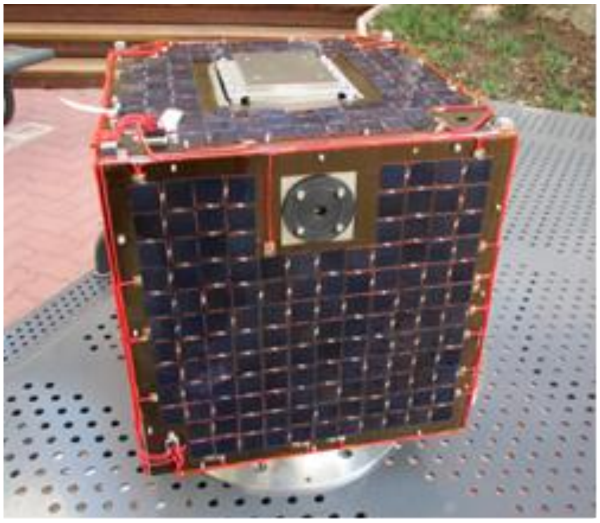


Figure 3. Flight-model of the 1st PROITERES satellite.

Table 1. Summary of the 1st PROITERES satellite.

Mass, kg	14.5
Outside dimension	290 mm x 290 mm x 290 mm (Without extension boom)
Orbit	Orbital inclination: 99.98deg, Eccentricity : 0
Altitude, km	660
Commencing time	April 2007
Life time, years	1-2
Attitude control	Magnetic attitude control Gravity-gradient stabilization

A. Electrothermal pulsed plasma thruster

Pulsed plasma thrusters, as shown in Fig. 4, are expected to be used as a thruster for micro/nano satellites. The PPT has some features superior to other kinds of electric propulsion. It has no sealing part, simple structure and high reliability, which are benefits of using a solid propellant, mainly Teflon® (poly-tetrafluoroethylene: PTFE). However, performances of PPTs are generally low compared with other electric thrusters.

At Osaka Institute of Technology, the PPT has been studied since 2003 in order to understand physical phenomena and improve thrust performances with both experiments and numerical simulations[4-6].

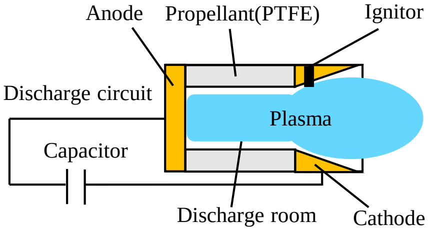


Figure 4. Electrothermal pulsed plasma thruster.

We mainly studied electrothermal-acceleration-type PPTs, which generally had higher thrust-to-power ratios (impulse bit per unit initial energy stored in capacitors) and higher thrust efficiencies than electromagneticacceleration-type PPTs. Although the electrothermal PPT has lower specific impulse than the electromagnetic PPT, the low specific impulse is not a significant problem as long as the PPT uses solid propellant, because there is no tank nor valve for liquid or gas propellant which would be a large weight proportion of a thruster system.

In our study, the length and diameter of a Teflon discharge room of electrothermal PPTs were changed to find the optimum configuration of PPT heads in very low energy operations for PROITERES satellite. Initial impulse bit measurements were conducted, and long operations and endurance tests were also carried out with the optimum PPT configuration. Figure 5 shows schematic of a thrust stand in a vacuum chamber for precise measurement of an impulse bit. The PPT and capacitors are mounted on the pendulum, which rotates around fulcrums of two knife edges without friction. The displacement of the pendulum is detected by an eddy-current-type gap sensor (non-contacting micro-displacement meter) near the PPT, which resolution is about ±0.5 m.

The vacuum chamber is 1.25 m in length and 0.6 m in inner diameter, which is evacuated using a turbo-molecular pump with a pumping speed of 3,000 l/s. The pressure is kept below 1.0 x 10^{-2} Pa during PPT operation. We carried out endurance tests with the optimum cavity shape 9.0 mm in length and 1.0 mm in diameter at a discharge energy per one shot of 2.43 J/s. Table 2 shows the operational condition of endurance test. The repetitive frequency is 1.0 Hz.

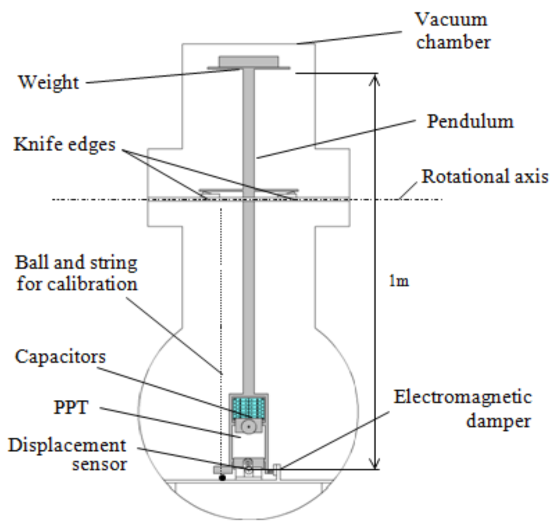
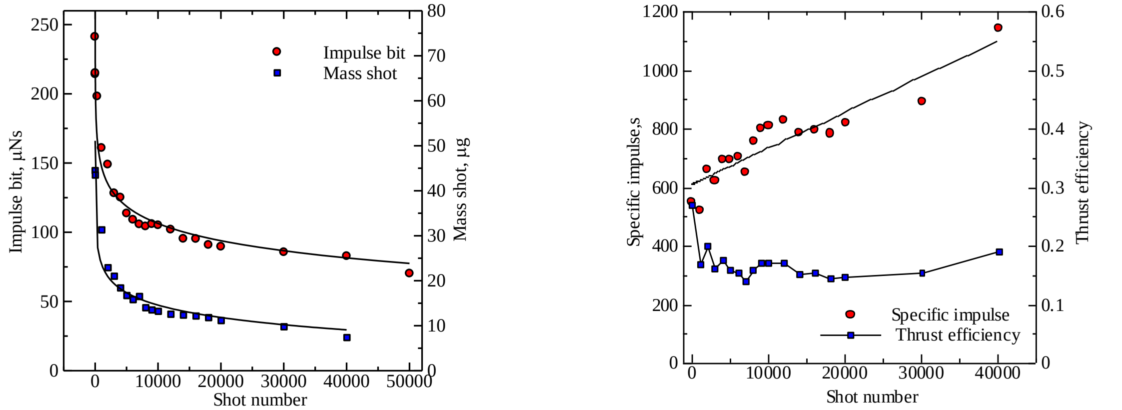


Figure 5. Thrust stand installed in the vacuum chamber.

Table 2. Experimental conditions of endurance test.

Capacitor, μF		1.5
Charging voltage, kV		1.8
Stored energy, J		2.43
Cavity	Length, mm	9.0
	Diameter, mm	1.0
Nozzle	Length, mm	23
	Half angle, degree	20

Figure 6 shows the shot-number history of impulse bit, mass loss, specific impulse and thrust efficiency. Both the impulse bit and the mass loss, as shown in Fig. 6 (a), rapidly decrease with increasing shot number. Specially, the impulse bit decreases from 250 μNs at initial condition to 75 μNs after about 50,000 shots. Although a few miss fires occurred around 53,000 shot, a total impulse of about 5 Ns was achieved. As shown in Fig. 6 (b), the specific impulse increases with increasing shot number, and the thrust efficiency is around 0.2 during the repetitive operation.



(a) Impulse bit and mass shot vs. shot number (b) Specific impulse and thrust efficiency vs. shot number

Figure 6. Result of endurance test.

We designed the FM of PPT head and its system. Figures 7 and 8 show the structure, illustrations, and photos. The PPT head has a simple structure, and two PPT heads are settled on the outer plate of PROITERES satellite. As shown in Fig. 8 (b), the power processing unit (PPU) and the 1.5-µF capacitor are mounted in the satellite. The final endurance test of the PPT system was successfully finished.

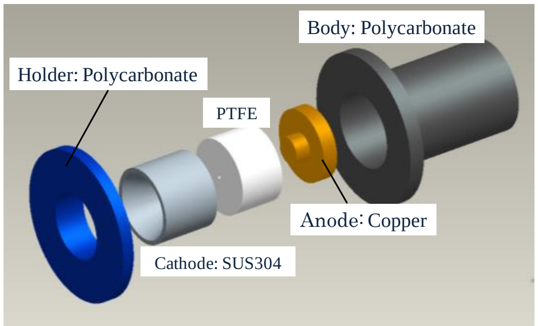
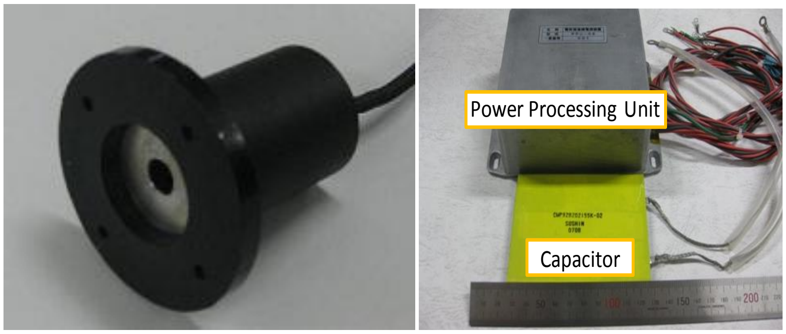


Figure 7. Inner structure of PPT.



(a) Photos of PPT head (b) PPU and capacitor

Figure 8. FM PPT system parts.

B. Final Checking at SDSC, ISRO

The 1st PROITERES satellite and its mass-dummy were carried to SDSC, ISRO by air transportation to Chennai International Airport, India from Kansai International Airport, Osaka, Japan on July 18, 2012, and final checking tests of the satellite were conducted in July-August 2012. After those, the PROPTERES satellite was installed to the PSLV C-21 rocket.

The satellite was attached to the satellite deck of the PSLV C-21 launcher with five degrees in inclination shown in Fig. 9. An interface ring was used to give an inclination of five degrees. Finally, we prayed to Japanese God by Japanese style for success of launch on the satellite deck just before launching.

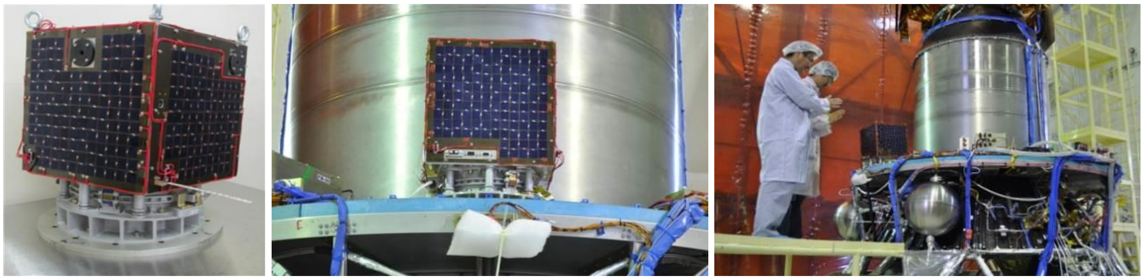


Figure 9. 1st PROITERES with separation device and interface ring, on PSLV C-21 rocket.

C. Operation after Launch

Table 3 shows the operation history of 1st PROITERES in 2012 and 2013. The launch on September 9th succeeded, and after that we could receive telemetry data of Morse signals from the satellite nine hours later at the ground station, Omiya campus, Osaka Institute of Technology, Osaka, Japan. The receiving operational data of current and voltage values of the power supply system are shown in Figs. 10 and 11. From these data, the discharging current is stable between 0.2 and 0.31 A. The voltage was lower than 11.5 V at the time of charging and higher than 9.8 V at the time of discharging. These data show that the satellite works normally[7-9].

Table 3. Operation history of the 1st PROITERES.

Date	Time(JST)	
Sep. 9, 2012	13:23	We succeeded to launch satellite.
	16:36	We received reports that German ham operator received Morse of the PROITERES.
	22:48	We succeeded to receive morse of the PROITERES at OIT.
Sep. 10-13		We succeeded to receive data of power supply.
Sep. 14	10:36	We could not gradually receive morse of the PROITERES.
	10:36	The communication was blackout.
Sep. 21	10:25	The other station received Morse of the PROITERES. (Then the night path could not receive.)
Sep. 23	21:41	We succeeded to receive again morse of the PROITERES at OIT. Then the morse of PROITERES was abnormal.
Dec. 4	00:16	We received warning from JSOC that approaching two debris.
Dec. 5	00:01	A debris was closest to the PROITERES. (First debris) Then we didn't receive clashing of the PROITERES and the debris.
Dec. 6	03:27	A debris was closest to the PROITERES. (Second debris) Then we didn't receive clashing of the PROITERES and the debris.
Apr. 11, 2013	09:00	We succeeded to receive again morse of the PROITERES only morning path at OIT. But the morse of PROITERES was abnormal.
Apr. 12	09:00	We succeeded to receive again morse of the PROITERES only morning path at OIT. But the morse of PROITERES was abnormal.
Apr. 13		The communication was blackout again. Now that still continues. We are sending FM telemetry on thruster firing and continuing observation.

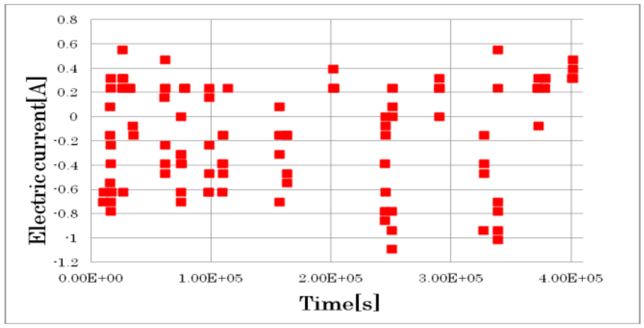


Figure 10. Current history data of power supply.
(Plus values present discharging, and minus values present charging)

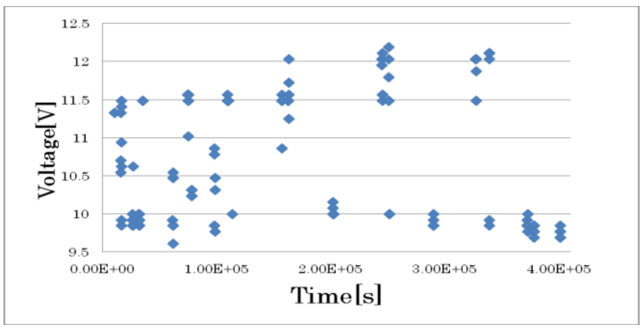


Figure 11. Voltage history data of power supply.

Then we could keep receive of Morse signals between the satellite and the ground station during about a week. However, the communication was blackout. After that, the satellite is repeating receive of Morse signals of revival and communication blackout. As shown in Fig. 12, we conducted a ground experiment to find a cause of this malfunction. As a result, the communication system of transmitter and receiver on the satellite was inferred to be not so good, i.e. electrically unstable.

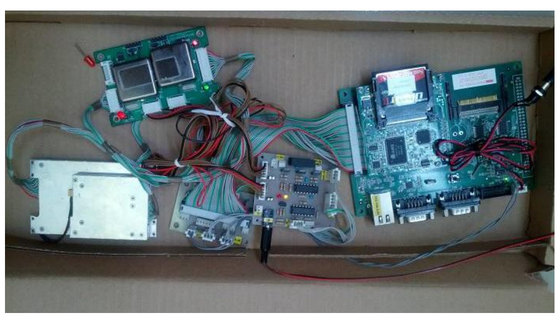


Figure 12. Ground experiment on malfunction of satellite.

Furthermore, we received the warning information that two debris approached the satellite with about 100 m in distance and 15 km/s in relative velocity in the end of December 2012 from the Joint Space Operations Center (JSOC), USA. However, because we succeeded in the communication with the satellite later than that accident, the satellite works now. Just now we are sending mission signals of FM telemetry at fixed intervals on thruster firing and continuing observation.

III.R&D of the 2nd PROITERES

We started the research and development of the 2nd PROITERES satellite in October 2010. 2nd PROITERES basic design has ended. Currently it is the design phase of Structure Thermal Model (STM). The 2nd PROITERES is divided into five systems of engine system, structure system, attitude control system, communication system and C & DH system and developing. Application of PPTs technology mounted on the 1st PROITERES, and installing PPT with increased power in the 2nd PROITERES. We are researching and developing each mounted equipment. Table 4 shows the specifications of the 2nd PROITERES satellite. It has been decided that the 2nd PROITERES is launched to 613 km in earth altitude by H-IIA rocket at Tanegashima Space Center, Japan Aerospace Exploration Agency (JAXA) on July in 2008. The 2nd PROITERES satellite will be launched by piggyback payloads of GOSAT-2/Khalifasat.

Table 4. The specifications of the 2nd PROITERES.

Mass, kg	45
Dimensions, mm^3	470 x 470 x 450
Electrical power , W	60
Altitude, km	613
Life time, years	1-2

A. Missions

Micro satellites and nano satellites often become a piggyback payloads. Therefore, the trajectory and orbit altitude of the satellite is limited by the main satellite. It is considered that if we can change the orbital altitude by obtaining velocity increment by firing of PPT, it will lead to expansion of mission of satellite. It is also expected to be useful for preventing debris by firing PPT in the direction of satellite to lower the speed and the altitude. The main mission is to achieve longer distance orbital transfer, i.e. changing 50-100 km in altitude on near-earth orbits by longtime operation and high power PPT system.

The method of mission, it is changed by stabilizing the satellite in the direction of the earth's center of the earth, fired plasma in the direction of travel by the operation of PPT, lower the speed of satellite, lower the altitude. Figure 13 shows a schematic of the mission.

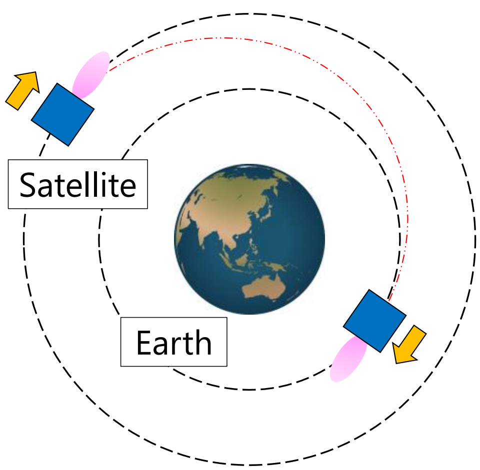


Figure13. Schematic of the mission.

B. Equipment list

Table 5 shows the equipment mounted on the 2nd PROITERES satellite. A reaction wheel and magnetic torquers are used as attitude control actuators. Sensors required for posture control are five solar sensors, three gyro sensors, one earth sensor and one magnetic sensor. Magnetic torquers and PPT system are made at OIT.

Table 5. Equipment list.

Onboard equipments	Number of units	Self-made
PPT system	1	Y
Reaction wheel	4	N
Magnetic torquer	3	Y
Solar sensor	5	Y
Angular rate sensor	3	N
Earth sensor	1	N
Magnetic Sensor	1	N
Transmitter and receiver	1	N
Antenna	2	Y
On board computer	2	N
Battery and power unit	1	Y

(Y : Yes, N : No)

C. Communication system

Because the 2nd PROITERES circles the low orbit of the Earth, it needs to be communicated in a limited time. It is assumed that the communication system is used for operation and maintenance of the satellite, data transmission / reception at the mission, and confirmation of the orbit of the satellite. Transceiver was developed for nano satellite. Also, development is under way so that down-link and up-link frequencies can be used in different bands so that transmission and reception can be performed at the same time. Transceiver as shown in Fig. 14, is mounted on the 2nd PROITERES satellite.

Figure 14. Transceiver.

D. Onboard computer

The C & DH system is an abbreviation for Command & Data Handling system. In the C & DH system, the design and interface of attitude control system, communication system, power system, optical system, engine system bus devices are developed. The C & DH system of the 2nd PROITERES has two onboard computers (OBC) as the main computer, and the subsystem is formed mainly around these. The attitude control system and the communication system, which are important in the mission, are connected to both OBCs. Two OBCs are called OBC - 0 and OBC - 1 for convenience. The two OBCs build a local network, and the security of the system is enhanced by monitoring each other. The OBC used in the 2nd PROITERES is Raspberry Pi as shown in Fig. 15.

Figure 15. The appearance of OBC.

E. Magnetic torquer

Magnetic torquer is continued to be installed from the 1st PROITERES as the attitude control device in the 2nd PROITERES. Air-core type magnetic torquer was mounted on the 1st PROITERES. However, as the satellite becomes larger, the required magnetic moment becomes larger, so the cored magnetic torquer is mounted on the 2nd PROITERES. Magnetic torquer is an actuator suitable for nano satellite because it structure is simple and lightweight. In the 2nd PROITERES, not only posture stabilization and posture control are performed, but also unloading of accumulated angular momentum of the reaction wheel is performed by magnetic torquer.

In order to obtain the magnetic moment required for magnetic torquer, the input orbit altitude was set to 600 km which is the same as the 2nd PROITERES. In that case, the geomagnetic torque, the gravity gradient torque, the solar radiation pressure torque, and the aerodynamic torque which are the expected environmental disturbance to the satellite were calculated. The calculated values of each environmental disturbance and the total disturbance, which is the total value, as shown in Fig. 16. From Fig. 16, it can be seen that the residual magnetic torque and the gravitational inclination torque exceed the other disturbance torques in the vicinity of the planned input orbit altitude of 600 km. The total disturbance torque "T" is 3.09×10^{-5} Nm at this time. In order to perform attitude control with the cored magnetic torquer, it is the design condition that torque exceeding this total disturbance is generated. If the magnetic flux density of the Earth is "B", the minimum required magnetic moment "M" of magnetic torquer is expressed by the following equation.

$$M = T / B \tag{1}$$

From the formula (1), the minimum required magnetic moment is 0.65 Am^2. However, this value is the minimum necessary magnetic moment value. The generated torque per unit time of the magnetic torquer is small. For this reason, the magnetic torquer was fabricated under the design conditions of 10 times 6.0 Am^2 or more and power consumption 0.5 W or less in order to give a margin to the control power. Figure 17 shows the appearance of the magnetic torquer produced, and Table 6 shows the specifications.

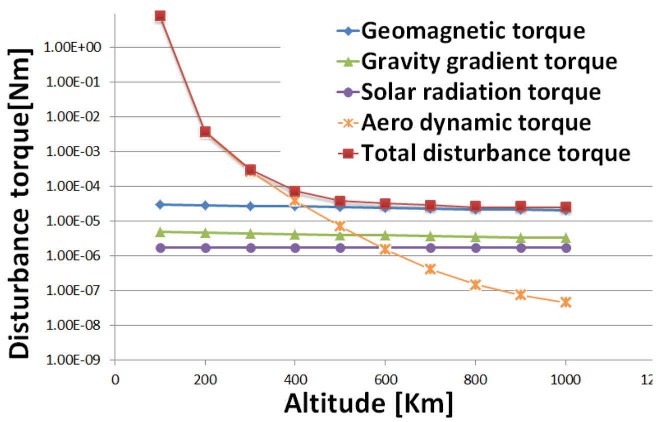


Figure 16. Environment disturbance and total disturbance.

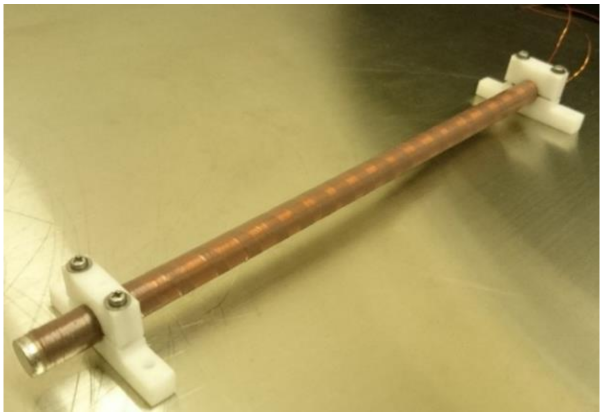


Figure 17. The appearance of the magnetic torquer produced.

Table 6. The specifications of the magnetic torquer produced.

Material of core	Permalloy PB
Max output of magnetic moment [Am^2]	6.0
Upper limit of an electrical consumption [W]	0.5
Diameter of core [mm]	8
Length of core [mm]	300
Material of lead	Copper (ϕ 0.5 mm)
Number of turns	2,174

A current was applied to the manufactured magnetic torquer using a stabilized power supply, and a magnetic flux density of magnetic torquer was measured by setting a gauss meter at a fixed distance. Magnetic moment was converted from the measured magnetic flux density to evaluate the performance of magnetic torquer[10]. The measurement results are shown in Fig. 18. From Fig. 18, it was confirmed that both the theoretical value and the measured value attained the required performance of 6.0 Am^2 at 0.125 A. In addition, the power consumption at this time was 0.096 W, which is lower than the target value of 0.5 W, and the design condition could be satisfied.

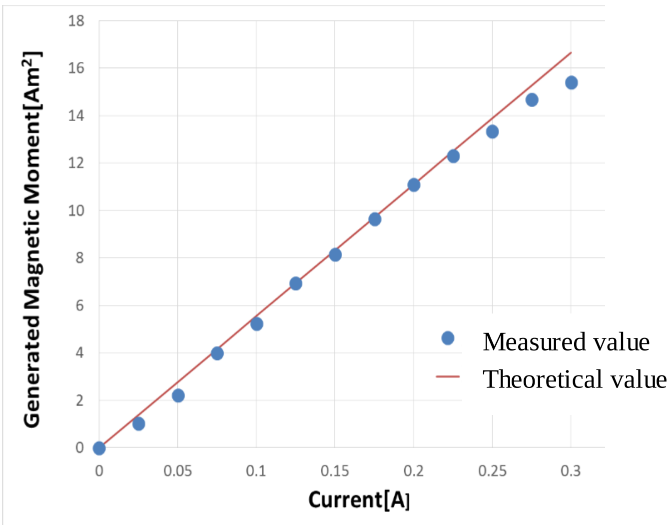


Figure 18. The measurement results.

In the Pulse Width Modulation (PWM) control, the output is controlled by changing the duty ratio of the pulse wave. It is used for the purpose of suppressing power saving and heat generation by switching on / off of the power supply at high speed.

Since magnetic torquer is a simple coil, when frequency control such as PWM control is performed, it is necessary to define the inductance in order to know the high frequency characteristics, and the time constant is calculated from the value. Inductance "L" and time constant "τ" are expressed by the following equation.

$$L = \mu_0 \cdot \mu_{eff} \cdot n^2 \cdot S \cdot l^{-1} \tag{2}$$

$$\tau = \frac{L}{R} \tag{3}$$

The time constant is a concept to estimate the circuit transient response to the current change. It represents the time when the current rises to 63.2% of the maximum value from the power supply or decays to 36.8% of the maximum value after discontinuing the current. In order to perform PWM control of a coil such as magnetic torquer, it is necessary to calculate the time constant and determine the frequency. The values of inductance and time constant calculated from equations (2) and (3) are shown in Table 8.

Table. 7. Coil inductance and time constant.

Inductance L [H]	0.506
Time constant τ [s]	0.081

Measurement of the time constant of the manufactured magnetic torquer was performed. It is assumed that the magnetic flux density also rises at the same time as the current value rises, and the change of the magnetic flux density from the start of the current flow is measured. A linear Hall IC and an oscilloscope were used to measure magnetic flux density. The linear Hall IC outputs 2.5 V which is the reference voltage when the sensitivity is 0.1 mV / mT and the magnetic flux density is 0. The measurement result is shown in Fig. 19. The voltage of the magnetic torquer instantaneously rises after turning on the power. On the other hand, the voltage of the linear Hall IC rises gently. For this result, assuming that the magnetic torquer's power-on time is 0 s, the time to reach around 63% of the maximum value indicated by the time constant was 0.08 s. It was possible to measure almost the same value as the theoretical value found. Also, it was confirmed that it takes about 0.4 s to output 100%.If this is set as one period of the PWM control, the PWM frequency has to be set to 2.5 Hz or less. Therefore, from this condition, the PWM cycle is set to 2 Hz and the duty ratio is set to 20% or more to perform the PWM control.

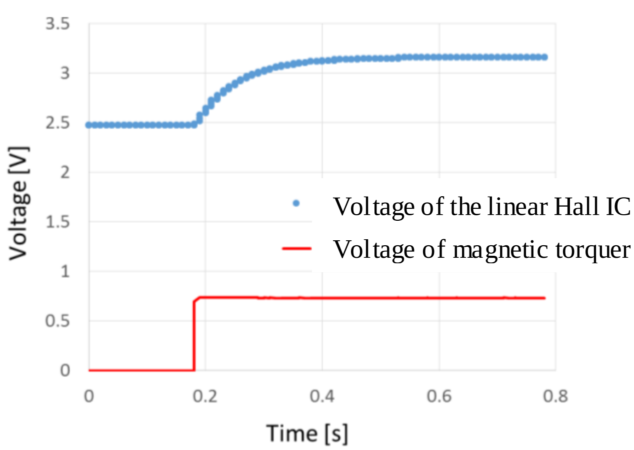


Figure 19. Voltage change of linear Hall IC and magnetic torquer.

F. PPT

Conventional single discharge room PPT (S-PPT) has limited propellants that can be used. Therefore, long-time operation cannot be expected. In order to achieve long distance power flight for the purpose of changing the orbital altitude, it was decided to install a Multi-Discharge-Room PPT (MDR-PPT). The MDR-PPT has seven S-PPTs. An ignitor, a nozzle and an anode are established every each discharge room. Figure 20 and Table 8 show a MDR-PPT head installed in the 2nd PROITERES and specification of MDR-PPT. Further, a single discharge room PPT constituting MDR-PPT is shown in Fig. 21[11-13].

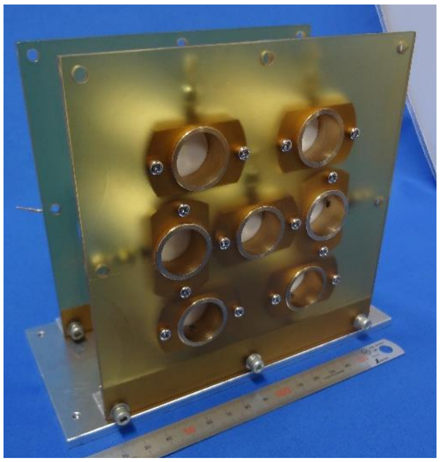


Figure 20. The MDR-PPT head.

Figure 21. The single discharge room PPT.

Table. 8. Specification of MDR-PPT.

Size, mm	90 x 226 x 155
Mass, kg	1.33
Number of discharge room	7
Discharge room diameter, mm	4
Discharge room length, mm	50
Cathode diameter, mm	20
Cathode length, mm	14

The S-PPT (Fig. 21) was used to do a repetitive operation experiment. The experimental conditions are shown in Table 9.

The S-PPT was able to accomplish round about 114,000 shots with a total impulse of 100 Ns. The impulse bit results during the experiment are presented in Fig. 22. The generated total impulse would be sufficient to propel a 45 kg nano-satellite for approximately 30 km.

Table. 9. Experimental conditions with S-PPT.

Discharge room diameter, mm	4
Discharge room length, mm	50
Cathode diameter, mm	20
Cathode length, mm	14
Input energy, J	31.59
Operational frequency, Hz	0.67

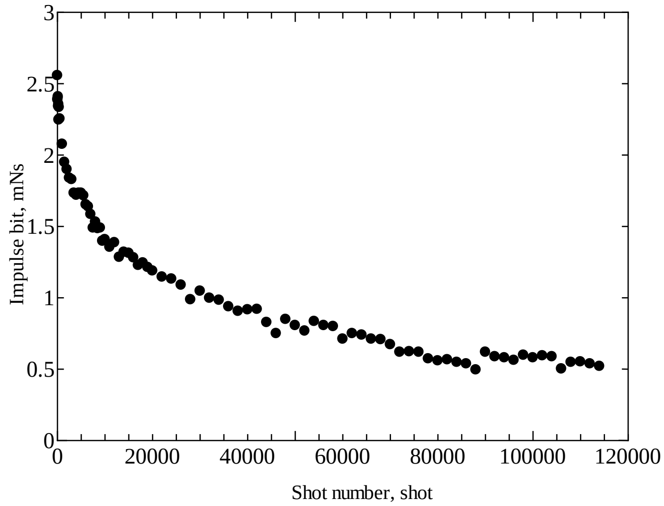


Figure 22. Impulse bit vs. shot number.

Finally, a mass of 2.47 kg for the capacitors, 0.5 kg for the circuit board 1.33kg for the MDR-PPT could be achieved. Thus, the mass of the entire MDR-PPT system is approximately 5.3 kg. The assembled MDR-PPT system is shown in Fig. 27.

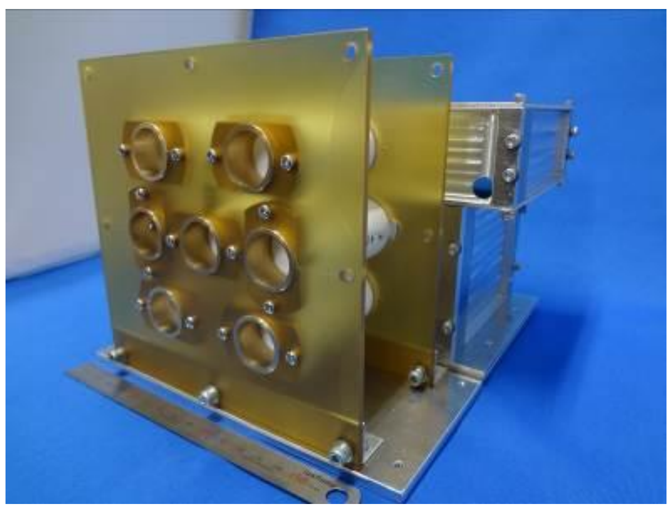


Figure 23. MDR-PPT system.

IV.3rd PROITERES

The 3rd PROITERES satellite, as shown in Fig. 24, is a 50 kg class moon-exploration satellite with cylindrical-type Hall thruster system for powered flight from the low earth orbit to the moon orbit. The Hall thruster system will produce specific impulses of 1500-2000 sec at xenon mass flow rates of 0.1-0.3 mg/s with an input power of 30 W. The trip time to the moon is within 3 years. Figure 25 shows a cylindrical-type Hall thruster of present laboratory. Now, we are developing the cylindrical-type Hall thruster and the satellite[14-17].

Figure 24. Illustration of 3rd PROITERES satellite.

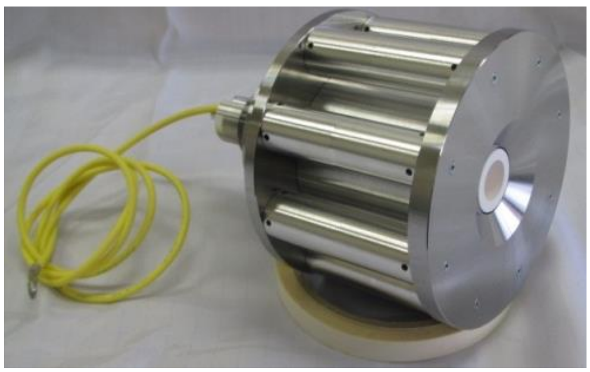


Figure 25. Photo of the cylindrical-type Hall thruster.

V.4th PROITERES

In recent years, development of micro/nano-satellites has been putting into practical use by cutting launching cost and shortening development period. Accordingly, the number of space debris has been increasing around the earth. Therefore, we must decrease the amount of space debris for future space development. The 4th PROITERES satellite, as shown in Fig.26, is planned as a micro/nano-satellite in order to achieve main mission that space debris makes deorbit by electric propulsion. The satellite is supposed 50 kg class and 50 cm class satellite. The principle of deorbiting space debris, as shown in Fig. 27, is exposure of thruster plume to space debris by an electric propulsion; that is, reaction impulse is given to debris, and after that debris decreases velocity and deorbits. Accordingly, the 4th PROITERES satellite can deorbit space debris with safety without contacting to space debris. OIT is developing four kinds of electric propulsion. These electric propulsions will be investigated, and for the 4th PROITERES satellite for deorbiting space debris a suitable electric propulsion will be selected[18,19].

Figure 26. The conceptual diagram of the 4th PROITERES.

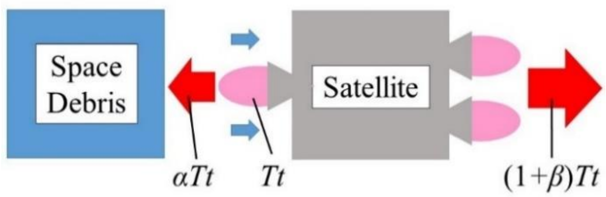


Figure 27. The relationship between the PROITERES 4th and debris.

VI. Conclusion

The Project of Osaka Institute of Technology Electric-Rocket-Engine onboard Small Space Ship (PROITERES) was started at Osaka Institute of Technology in 2007.

1) In PROITERES, a nano satellite named the 1st PROITERES satellite with electrothermal pulsed plasma thrusters (PPTs) was launched by Indian PSLV C-21 rocket on September 9, 2012. And, we succeeded the communication with the satellite in ground station of our university nine hours later. However, the communication with the satellite was cut off a few days later. After that, the satellite repeats revival and communication blackout and reaches it at the present. Therefore we experiment it to find a cause of this malfunction.

2) As next projects, we started the research and development of the 2nd PROITERES satellites in Oct. 2010. It has been decided that the 2nd PROITERES is launched to 613 km in earth altitude by H-IIA rocket at Tanegashima Space Center, Japan Aerospace Exploration Agency (JAXA) on July in 2008. The 2nd PROITERES satellite will be launched by piggyback payloads of GOSAT-2/Khalifasat. Now we development of magnetic torque and MDR-PPT system installed in the 2nd PROITERES satellite and the body of 2nd PROITERES.

3) In 3rd and 4th PROITERES satellite, we are developing the electric propulsion and conception of satellite.

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