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Powered Flight of Electron Cyclotron Resonance Ion Engines on Hayabusa Explorer

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The electron cyclotron resonance ion engine has long life and high reliability because of electrodeless plasma generation in both the ion generator and the neutralizer. Four μ10s, each generating a thrust of 8 mN, specific impulse of 3200 s, and consuming 350 W of electric power, propelled the Hayabusa asteroid explorer launched on May 2003. After vacuum exposure and several baking runs to reduce residual gas, the ion engine system established continuous acceleration. Electric propelled delta-V Earth gravity assist, a new orbit change scheme that uses electric propulsion with a high specific impulse was applied to change from a terrestrial orbit to an asteroid-based orbit. In 2005, Hayabusa, using solar electric propulsion, managed to successfully cover the solar distance between 0.86 and 1.7 AU. It rendezvoused with, landed on, and lifted off from the asteroid Itokawa. During the 2-year flight, the ion engine system generated a delta-V of 1400 m/s while consuming 22 kg of xenon propellant and operating for 25,800 h.

I. Introduction

DEEP space exploration has expanded from exploring major planets to exploring minor celestial objects, such as comets and asteroids. Several spacecrafts, such as Giotto [1], Galileo [2], and so on, have flown by comets and asteroids. Deep Impact [3] even collided with a comet. The space missions, including rendezvous, landing, and return advanced from the flyby and hard collision, have attracted attention. The spacecraft NEAR Shoemaker succeeded in rendezvousing with and touching down on an asteroid [4]. Genesis [5] and Stardust [6] executed round trips between Earth and a heliocentric orbit. These missions in deep space require high maneuverability. High-order space missions designed using conventional chemical thrusters cause the fuel to take up an exorbitant mass and, thus, require extremely large launch vehicles. These spacecraft use flyby inertia after an initial acceleration caused by the launch vehicles. Thus, these objects are called "artificial planets" in deep space and "artificial satellites" around Earth. However, technological advancements have dramatically changed space missions. Several different types of electric propulsion systems have been used for spacecraft sent into deep space, including the NSTAR ion engine on Deep Space 1 [7], the PPS1350 Hall thruster on SMART-1 [8], and the μ10 ion engines on Hayabusa [9]. Electric propulsion efficiently generates thrust force while consuming less propellant; in other words, they have a high specific impulse. Planetary space missions will be enhanced, given that electric propulsion can provide an orbit change capability comparable to each stage of the launch vehicles. Spacecraft equipped with electric propulsion are able to cruise by themselves in space without using large launch vehicles for the initial acceleration and may thus be classified as "spaceships" with powered flight in deep space.

A spaceship needs electric propulsion that has a long operational life and high reliability. In most conventional ion engines, the hollow cathodes and the discharge electrodes in the ion generator and the neutralizer are the life determining components [10]. However, in electron cyclotron resonance (ECR) ion engines, such as the μ10 ion engines, all the electrodes used for plasma generation are eliminated, making ECR ion engines suitable for deep space missions. Four μ10s propelled the Hayabusa asteroid explorer [11], launched in May 2003, aiming at rendezvous, soft landing, liftoff, and Earth return. Hayabusa reached a distance of 0.86 AU (astronomical units) from the sun in February 2004 and 1.7 AU from the sun in February 2005. These distances are the farthest that an electric propulsion system has yet attained in the solar system. Furthermore, Hayabusa succeeded in rendezvousing with the asteroid Itokawa in September 2005 after a 2-year flight, producing a delta-V of 1400 m/s, while consuming 22 kg of xenon propellant and operating for 25,800 h. After a series of scientific observations Hayabusa landed on and lifted off the asteroid in November 2005 [12]. Finally, it is hoped that the spaceship will return to Earth. This paper reports the system and the flight experiences of the cathodeless ECR ion engines μ10 used on the Hayabusa explorer during the outward journey.

II. System Descriptions

A. Hayabusa Spacecraft

The Hayabusa space mission is focused on demonstrating the technology needed for a sample return from an asteroid, using electric propulsion, optical navigation, material sampling in a zero gravity field, and direct reentry from a heliocentric orbit. Table 1 summarizes the characteristics of the Hayabusa spacecraft, and Fig. 1 shows its configuration, including the pair of stowed solar cell paddles (SCP). The high gain antenna (HGA) is mounted on the upper surface of the body. The SCP and the HGA have no rotational or tilt mechanisms. The ion engine system (IES) is mounted on the side panel perpendicular to the z axis, with which the HGA aperture is aligned. At high bit rate communication, the spacecraft orientates the HGA toward the Earth without IES firing. In cruise mode, the

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Table 1 Summary on the Hayabusa spacecraft.

Launch mass:	510 kg including 67 kg of fuel and 66 kg of propellant
Attitude control:	3 axis stabilization, 3 reaction wheels
Communication:	X band, 8 kbps max.
Solar cell panel:	Triple-junction cells, 2.6 kW at 1 AU
Chemical thruster:	12 bipropellant RCS thrusters, 20 N thrust, 290 s I_{sp}
Electric propulsion:	Four ECR ion engines μ10, each producing a thrust of 8 mN, an I_{sp} of 3200 s, a thrust power ratio of 23 mN/kW, a dry mass of 59 kg
Thermal control:	CPU-controlled replacement heater, variable liquid-crystal radiator, high-conductivity super graphite, loop heat pipe
Payloads:	Telescope cameras, near infrared spectrometer, x-ray-induced fluorescence spectrometer, laser altitude meter, small landing robot, sampling mechanism, reentry capsule

spacecraft orients the SCP face toward the sun to generate electrical power and rotates its attitude around the solar direction to steer the thrust direction of the IES.

B. Cathodeless Electron Cyclotron Resonance Ion Engines

Figure 2 and Table 2 represent schematically the system configuration of the μ10 [13]. The technological features are summarized as follows:

1) Xenon ions are generated using an ECR microwave discharge without solid electrodes, which in conventional ion engines can cause flaking leading to electrical grid shorts. Thus, the elimination of the solid electrodes makes the ion engine more durable and highly reliable.

2) Neutralizers are also driven using an ECR microwave discharge [14]. The removal of the hollow cathodes precludes IES from heater failures and hollow cathode emitter performance degradation due to oxygen contaminating the propellant, as well as air exposure during satellite assembling.

3) A single microwave generator simultaneously feeds both the ion generator and the neutralizer. This feature reduces the system mass and simplifies the control logic.

4) dc power supplies for ion acceleration have been reduced to 3. This feature has the advantage of making the system lighter and requiring simpler operational logic.

5) The electrostatic grid system is fabricated from a carbon–carbon composite [15]. The clearance between the grids is stable regardless of the temperature because there is no thermal expansion. The carbon–carbon composite prolongs the life of the acceleration grid due to a low sputtering rate against the xenon ions. Also the low wettability of carbon minimizes electrical shorts between the grids.

The discharge chamber and the neutralizer of the μ10 ion engine employ samarium-cobalt permanent magnets. Input microwave power is fed through a waveguide or a coaxial cable as shown in Fig. 3. Figure 4 shows a system diagram of the IES designed and assembled using four μ10 ion engines with an effective diameter of 10 cm in the 3-grid electrostatic system. The upper left, upper right, lower left, and lower right ion thrusters (ITRs) shown in Fig. 1 are identified as ITR-A, B, C, and D, respectively. Each ITR is connected to its own microwave power amplifier (MPA), which emits 4.2 GHz microwaves that generate high-density plasma using ECR. Microwave dc blocks are used for electrical isolation as seen in Fig. 2. High voltage power generation is provided by three IES power processing units (IPPUs), which are distributed to the four ITRs through relay switches, so that simultaneous operation is limited to any combination of three ITRs. The IES pointing mechanism (IPM) mounting with the four ITRs aligns the spacecraft so that the resultant thrust vector is through the spacecraft's center of gravity. The propellant is supplied through gas isolators and flow restrictors using blowdown from a plenum, which is charged with gas from a spherical

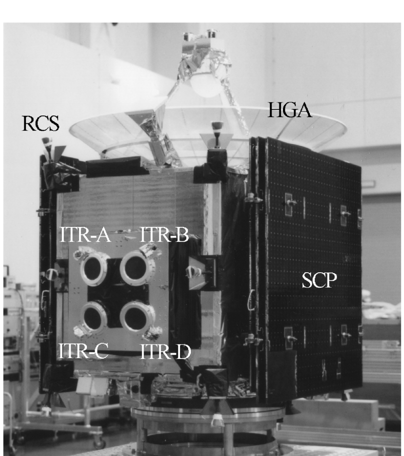

Fig. 1 Photograph of the Hayabusa spacecraft before launch.

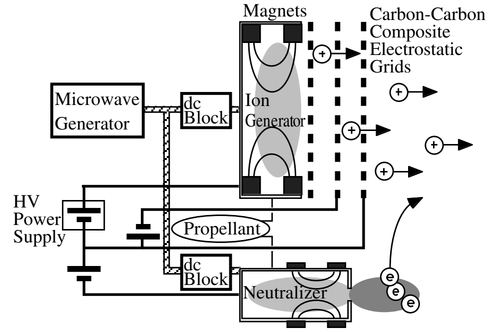

Fig. 2 System configuration of μ10 thruster.

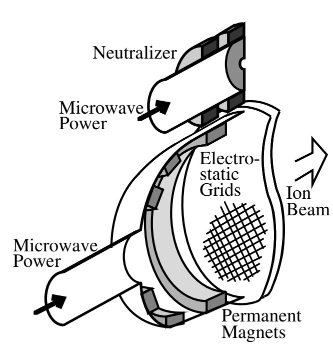

Fig. 3 Configuration of ITR.

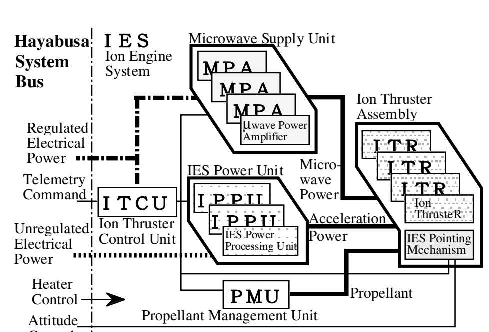

Fig. 4 System diagram of IES.

Table 2 Specifications of IES on Hayabusa.

Ion thruster (ITR)	Four μ10, cathodeless ECR plasma generation carbon-carbon composite 3-grid electrostatic acceleration 10 cm effective diameter, 8 mN nominal thrust
Microwave power amplifiers (MPA)	Traveling wave tubes, 4.2 GHz, four units A single MPA driving both an ion generator and a neutralizer simultaneously 32 W for an ion generator, 8 W for a neutralizer, 110 W total power consumption
IES power processing units (IPPU)	Three units distributed to four ITR via relay switches 1.5 kV to screen grid, −330 V to acceleration grid 240 W total power consumption
Propellant management system (PMU)	Titanium alloy pressure tank, 51 liters in volume, 73 kg maximum xenon loading Two propellant flow controllers for redundancy Blowdown via flow restrictors
IES pointing mechanism (IPM)	Two axis gimbal, ±5 deg

titanium alloy main tank in the propellant management unit (PMU). The ion thruster control unit (ITCU) operates the IES. The dry mass of the IES is 59 kg. A single ITR generates a maximum thrust of 8 mN while consuming 350 W of power and a minimum thrust of 5 mN while consuming 270 W of power. Before the launch, two endurance tests of 20,000 h each demonstrated the long life and high reliability of the μ10 ion engines [15]. To prevent plasma interaction [16], electrical grounding of the IES, surface conductivity of the spacecraft, and plasma protection of the SCP were taken into consideration when designing the Hayabusa system. The details are described in [17].

III. Space Operation

On 9 May 2003, the M-V rocket launched Hayabusa spacecraft into deep space. The Goldstone Tracking Station confirmed deployment of the SCP. In the evening of the same day, the Uchinoura Space Center received signals from Hayabusa. A precise orbit determination revealed a need to trim the velocity by about 30 m/s, which was large for the spacecraft's chemical propulsion system. An immediate orbital correction maneuver was canceled because, during cruising, the IES had enough capability to make up the deficiency of velocity. At the end of May, the four ITRs were turned on one by one. After test operations and parameter tuning, the cruise phase of the mission was started in July. In the first year, the spacecraft stayed on a 1-year Earth-synchronous orbit while changing its orbital eccentricity by using the IES. The purpose of this space operation was to accumulate a relative velocity compared to Earth, which could then be converted into orbital energy at the moment of the Earth swing-by. The 1-year Earth-synchronous orbit supplies the spacecraft with enough solar power, a moderate temperature environment, and enough time for acceleration so that the IES can attain its maximum capabilities. The 1-year Earth-synchronous electric propelled delta-V Earth gravity assist (EPΔVEGA) should theoretically result in a velocity change equal to twice the delta-V generated using onboard thrusters [18]. Figure 5 shows the relative position of the Hayabusa spacecraft in the rotational coordinate system, where the sun is located at the origin and the Earth on the horizontal axis. Just after launch, the spacecraft's orbit, which was determined based on the June 24th data, implied that the spacecraft would never return to Earth. A maneuver by the IES gradually changed the orbit and finally achieved an orbit crossing the Earth at the end of 2003. The spacecraft passed through perihelion at a distance of 0.86 AU from the sun receiving its most severe solar radiation on 23 February 2004. In the first year, the IPPU for ITR-D experienced relatively high temperatures, which were mitigated by operational cycling, where ITR-D was turned on for 2 h followed by a 1-h off period. Guidance toward the swing-by point was also accomplished using IES maneuvers. Figure 6 shows the transition of the crossing point in the ballistic plane, which includes Earth. Orbital determination indicated that the IES maneuvers made this crossing point gradually approach Earth. By the end of March 2004, the IES had reached 10,000 h of operational time and had generated a delta-V of 600 m/s while consuming 10 kg of propellant. The bipropellant thrusters executed the final guidance on 20 April and 12 May 2004; each thruster firing contributed a delta-V of only 0.1 m/s. On 19 May 2004, Hayabusa passed by the Earth at a point 4000 km above the Pacific Ocean, where its speed increased by 4 km/s in the inertial coordinate system, and was inserted into the transfer orbit toward the target asteroid. The image of Earth in Fig. 6 was taken by Hayabusa at the moment of the Earth swing-by.

On the transfer orbit to the asteroid, the IES continued to accelerate Hayabusa. Three of the four ITRs were supplied enough electric power to accelerate the spacecraft with their full capability until August 2004. From September 2004, the IES was throttled down to adapt to a reduction in solar power due to an increase in the solar

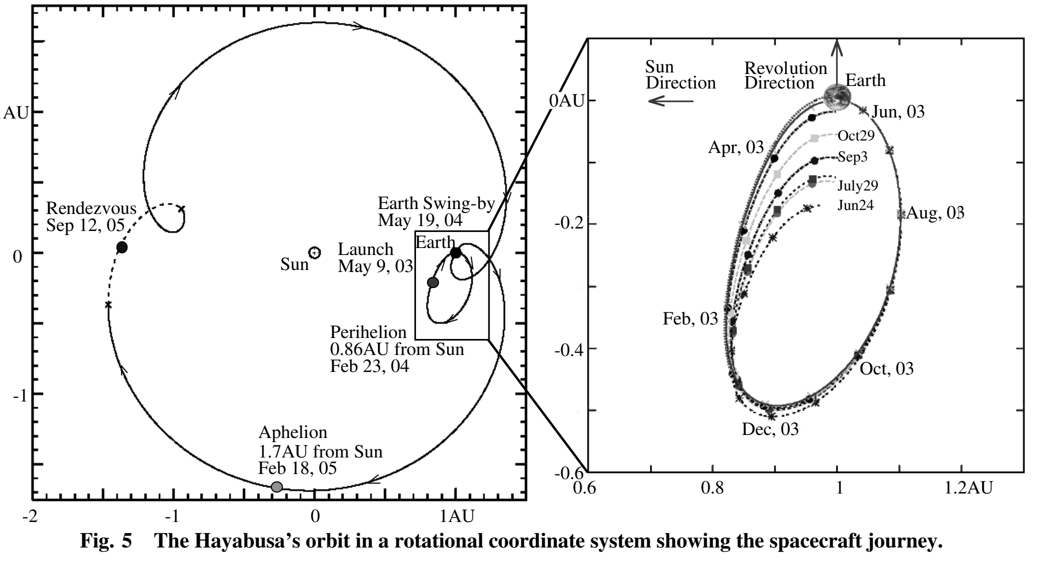

Fig. 5 The Hayabusa's orbit in a rotational coordinate system showing the spacecraft journey.

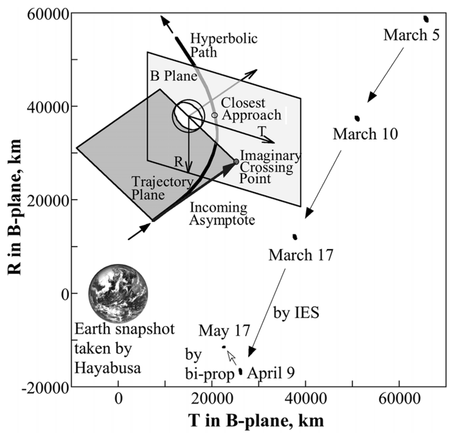

Fig. 6 Guidance toward the Earth swing-by point using IES's thrust.

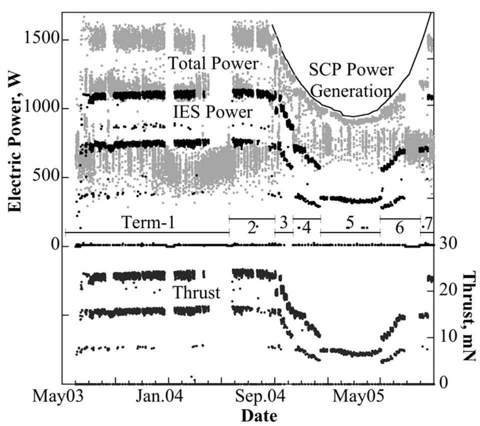

Fig. 7 Profile of electrical power and thrust force in outward journey.

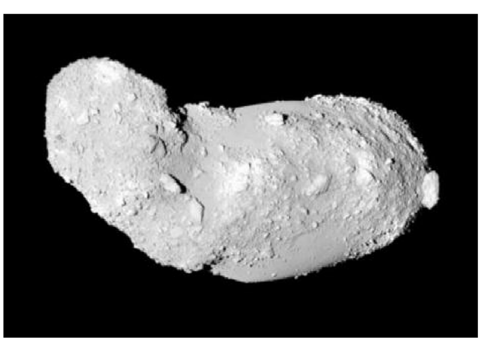

Fig. 8 Photograph of asteroid Itokawa taken by Hayabusa.

distance. The largest solar flare in history that occurred in November 2003 damaged the SCP and changed the plan for orbit maneuvers during the throttling phase of operation. Two ITRs were operated in October, and only one ITR was turned on from the end of December 2004. Figure 7 shows profiles of electrical power and thrust during the outward flight. The IES generated a maximum

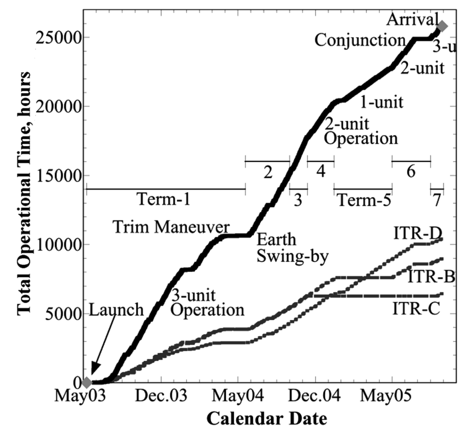

Fig. 9 Profile of the accumulated operational time for each of the ITRs.

thrust of 25 mN combining three ITRs while consuming 1.1 kW of electrical power and a minimum thrust of 5 mN using a single ITR while consuming 270 W of electrical power. On 18 February 2005, Hayabusa reached aphelion at a distance of 1.7 AU from the sun. On May 2005, two ITRs were again turned on because of an increase in solar power. In July, Hayabusa was in solar conjunction, and delta-V maneuvers by the IES were paused due to insufficient orbital determination. From August, three ITRs accelerated Hayabusa to reduce the distance to and relative velocity with respect to Itokawa. On 28 August 2005, the IES completed the outward journey and handed over the space maneuvers of Hayabusa spacecraft to the bipropellant thrusters at a distance 4800 km from the target with an approach speed of 9 m/s. Hayabusa sent Earth a complete picture of Itokawa, which is shown in Fig. 8. The total operational time of the IES, whose profile is shown in Fig. 9, was 25,800 h, while consuming 22 kg of xenon propellant and generating a delta-V of 1400 m/s. ITR-D operated for a total of 10,400 h, which was the longest space operation among the four ITRs, and completed 1720 operational cycles, which is the most frequent. ITR-A was not used on the outward journey after its initial checkout as it had been reserved for later use. In the following sections, the behavior of the IES in space for each of the time periods referred to as terms 1–7, is discussed. Table 3 presents a summary of the operational terms.

IV. Operation of the Ion Engine μ10

A. Initial Operation

After launch, the IES exposed to space vacuum with a temperature around 0°C. At the end of May 2003, each ITR was separately turned on and ionized the propellant and accelerated the spacecraft for about 1 h. In the next step, two ITRs parallel operation was executed. Large discharges around the ITRs caused by outgassing due to a temperature rise were observed. Then, the IES was exposed to baking-out at about 50°C for 2 days from two different sources: replacement heaters and solar radiation. This allowed the simultaneous operation of the three ITRs, which provided maximum capability for IES. Figure 10 shows the time profile of the leakage current to the accel grid, which tended to increase under low vacuum conditions, in ITR-D. The time in the horizontal axis represents the total operational time of ITR-D. The profile is scattered because the current resolution was only 0.15 mA; a numerically smoothed curve

Table 3 Operational terms

Term	Duration	Used ITR	Comments
Term 1	From launch (May 2003) to Earth swing-by (May 2004)	ITR-B, C, and D	Maximum thrust, on–off cycle operation of ITR-D
Term 2	From Earth swing-by (May 2004) to Aug. 2004	ITR-B, C, and D	Maximum thrust
Term 3	September 2004	ITR-B, C, and D	Throttling down
Term 4	From October to November 2004	ITR-B and D	Throttling down
Term 5	From December 2004 to April 2005	ITR-D	Throttling
Term 6	From May 2005 to conjunction (July 2005)	ITR-B and D	Throttling up
Term 7	From conjunction (July 2005) to rendezvous (Aug. 2005)	ITR-B, C, and D	Maximum thrust

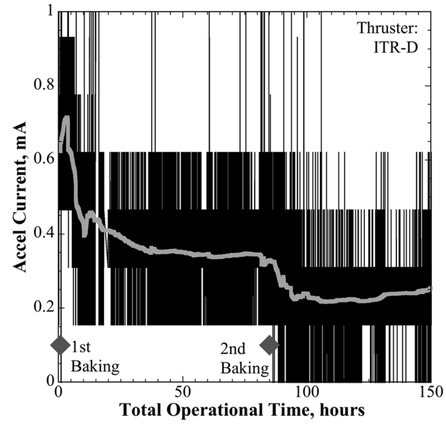

Fig. 10 Time profile of the leakage current to the accel grid during initial operation.

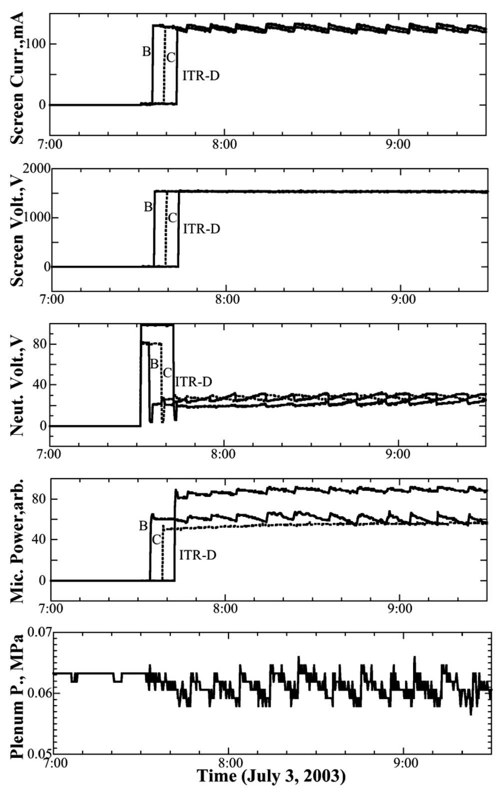

Fig. 11 Ignition sequence of IES.

is shown in Fig. 10. After two space baking-out runs executed at an elapsed time of 0.8 and 85 h, the accel current decreased dramatically. In ground tests with a pressure of 1.5×10^{-4} Pa, the accel current was 0.46 mA.

Continuous acceleration by three ITRs was achieved and the planned IES delta-V maneuver was executed in July. Figure 11 depicts the sequence to turn on the IES. First, the neutralizers were biased to around 100 V, and then a simultaneous injection of the propellant gas and microwave power ionized the propellant, of which a low impedance dropped the neutralizer voltage of each ITR. Applying 1.5 kV to the screen grids exhausted the plasma beams. The time profiles of the screen current equivalent to the plasma beam showed sawtooth fluctuations synchronized with the pressure of the plenum, which indicated the propellant flow rate. Once the IES was turned on, the maneuver monitor on the ground station caught the spacecraft velocity change along the line of sight (LOS) by means of a Doppler shift in the measurement of the communication frequency as seen in Fig. 12. The external force on the spacecraft can be identified using the delta-V measurement, the total mass of the spacecraft, and the angle between the thrust and LOS. Based on the

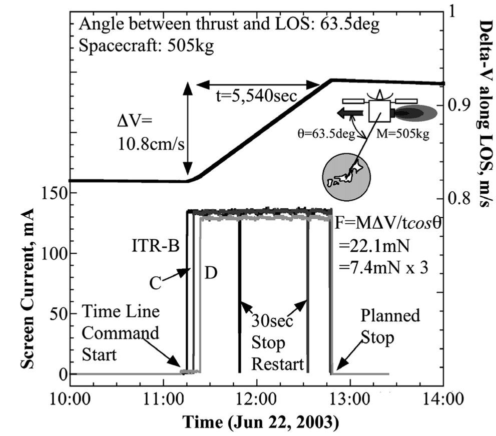

Fig. 12 Doppler measurement to reveal the IES's thrust force.

Doppler shift measurement, a single $\mu 10$ ion engine in space was able to deliver a maximum thrust of 8 mN with a thrust factor of 92%, a specific impulse of 3200 s, a thrust power ratio of 23 mN/kW, an ion production cost of 240 eV, and a propellant utilization efficiency of 87%. The thrust factor was attributed to a thrust lost due to leakage current to the accel grid, beam divergence, and multiply charged ions. The thrust power ratio was calculated based on the input powers to the IPPU and the MPA.

B. Throttling

In term 2, after the Earth swing-by, Hayabusa spacecraft entered into the transfer orbit as the IES fired. From June to August 2004, sufficient electric power and moderate temperature conditions allowed the IES to propel the spacecraft with full capability. Every Tuesday, using the parabolic HGA, a weekly command set was uploaded to the spacecraft with Earth-based attitude. The spacecraft then changed its attitude to an acceleration mode and turned on the planned ITRs. The ITRs continued to fire during a week at the acceleration attitude using a horn-shaped medium gain antenna with low speed communication. If the beam exhaust was interrupted by a spark discharge between grids, it was restarted automatically after 30 s.

On September 2004, the distance to the sun was 1.3 AU, so the maximum power from SCP was reduced to 1.5 kW. However, the spacecraft requires more electrical power to operate three ITRs. Thus, the spacecraft entered term 3 by performing to throttle the IES to save power. The IES was gradually throttled down week by week so as not to exceed the supply electric power as can be seen in Fig. 7, where the four curves represent the thrust, the IES's electric power, the total power consumed in the spacecraft, and the capability of SCP. In October, ITR-C was turned off, and ITR-B and ITR-D propelled the spacecraft into term 4. Figure 13 shows the daily change in power consumption during these 2 weeks. The spacecraft

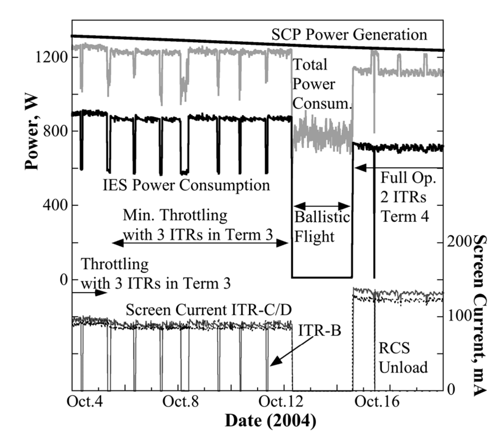

Fig. 13 Daily fluctuation in power and screen current during the throttling operation.

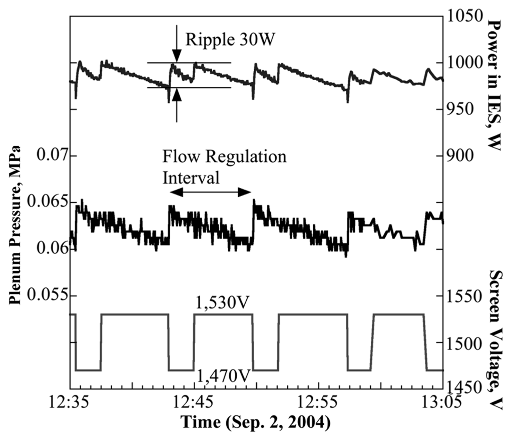

Fig. 14 Compression of the IES power ripple by adjustment of the screen voltage during the propellant flow rate fluctuation.

supplied the IES with as much electric power as possible given that the onboard transponders had been turned off. Whenever communications with Earth were performed, one of the ITRs was turned off so that the transponders could be turned on and transmit the day's stored data. After transmitting the data, the spacecraft turned on one of the ITRs after turning off the transponders. Propellant flow was proportional to the pressure in the plenum, which was controlled to within 10% of the target as can be seen in Fig. 14. The ripple in the propellant flow rate caused oscillations in the screen current; this resulted in fluctuations in the consumption power of the IES, which was compressed by about 30 W due to adjustments in the screen voltage synchronizing with the perturbations in the propellant flow rate, as can be seen in Fig. 14. The IPPU had the capability to output screen voltages of 1000, 1470, and 1530 V. In term 5, which started at the end of December 2004, only ITR-D accelerated the spacecraft, which passed through aphelion 1.7 AU from the sun on February 2005 consuming 320 W.

C. Thrust Factor

During throttling in term 3, the orbital determination technique gave average characteristic thrust factor for the three ITRs as shown in Fig. 15. The thrust factor is defined as the ratio between the actual thrust force and the calculated value based on the screen voltage and current of the ITR. It represents the thrust degradation caused by leakage current to the accel grid, beam divergence, and creation of multiply charged ions. During throttling operation of the IES, the thrust factor had a tendency to be reduced by μ10 plasma generation with a small propellant flow rate and microwave power identical to that of the maximum operation.

D. Chronology of Ion Thruster Performance

The screen current, accel current, specific impulse, and thrust are presented in Figs. 16–19 as functions of total operational time.

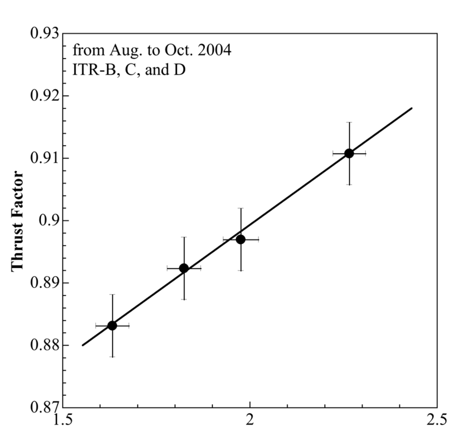

Fig. 15 Thrust factor dependence on propellant flow rate.

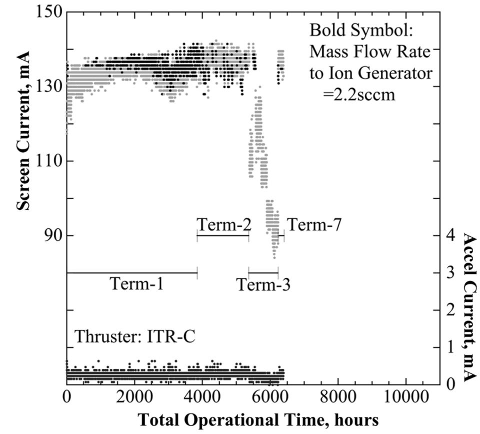

Fig. 16 Profiles of screen and accel currents on ITR-C.

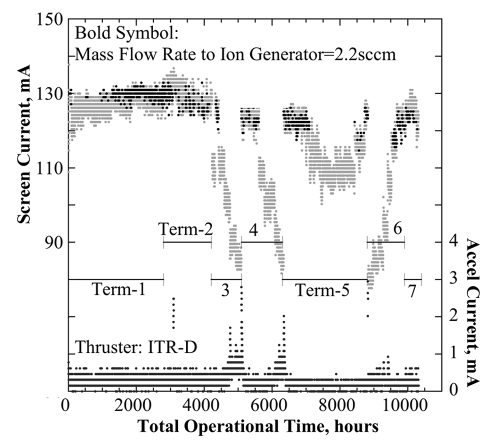

Fig. 17 Profiles of screen and accel currents on ITR-D.

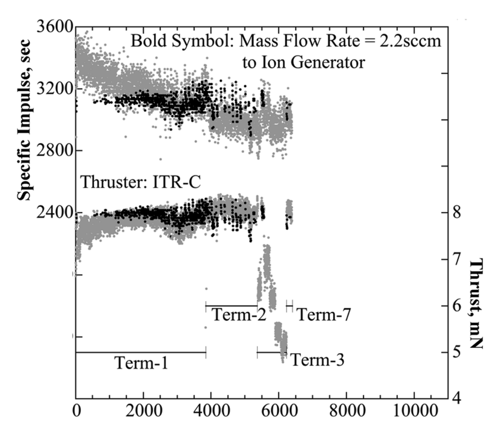

Fig. 18 Profiles of specific impulse and thrust on ITR-C.

Specific impulse and thrust were calculated based on the screen current and voltage of the telemetry data and the thrust factor in Fig. 15. The IES was operated during a wide range of throttling conditions, and the results show that the parameters diverged. This was most noticeable with ITR-D, which was throttled down in both terms 3 and 4, and throttled up once in term 6. In Figs. 16–19, the data at a mass flow rate of 2.2 sccm (standard cubic centimeter per minute) to the ion generator are marked using bold symbols to highlight the parameter trends. Though all of the parameters marked by the bold symbols for ITR-C were stable, the screen current, specific impulse, and thrust on ITR-D dropped during term 2. At this moment, ITR-D experienced a high accel current, which also happened during deep throttling in terms 3, 4, and 6. It is speculated that ion impingement

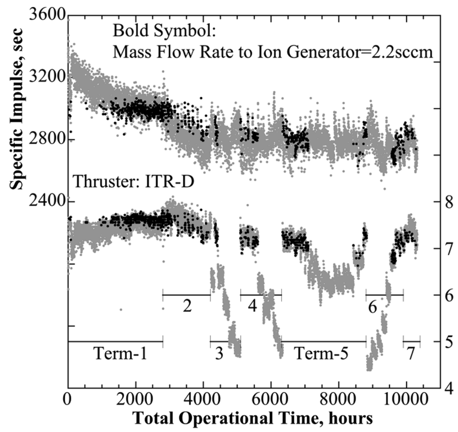

Fig. 19 Profiles of specific impulse and thrust on ITR-D.

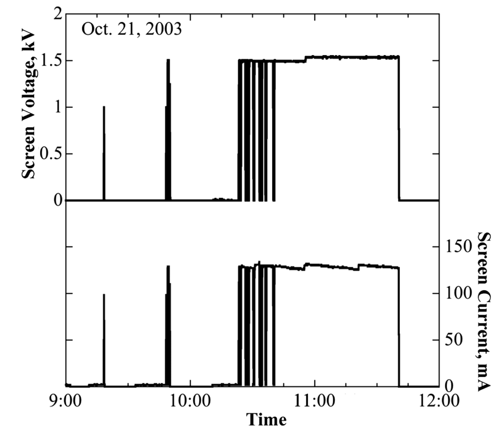

Fig. 22 Operation sequence to remove contamination between grids.

enlarged the holes of the accel grid so as to degrade the confinement of the neutral gas in ITR-D. However, performance was still good: a specific impulse of 2800 s, a thrust of 7 mN, and a thrust power ratio of 22 mN/kW at the operational point of 2.2 sccm propellant flow to the ion generator after 10,400 h of operation and 1720 cycles of switching. For ITR-C, the performance characteristics include a specific impulse of 3100 s, a thrust of 8 mN, and a thrust power ratio of 22 mN/kW after 6000 h of operation at the same operational point.

E. Thrust Vector Control

The IPM installed with the four ITRs could tilt on two axes within 5 deg as shown in Fig. 20. The active control of the thrust direction was applied so as to cancel the torque due to the misalignment between the thrust axis and the spacecraft's center of gravity, as well as to maintain the reaction wheels (RW) at a proper rotational rate. The latter function was called "IES unloading" for the RW-Y and Z. The momentum stored in RW-X was dumped during firing of the bipropellant reaction control system (RCS) thrusters. This sequence was termed "RCS unloading," which was triggered automatically about once per week and interrupted the IES operation for about 10 min, as can be seen in Fig. 13. Using IES unloading reduced a fuel consumption 1.2 kg of the bipropellant RCS thruster over 2 years. Figure 21 shows the initial IES thrust deviation, estimated from the flight data based on the tilt angles of the IPM, the momentum stored in the RWs, and the thrust of the IES neglecting any natural disturbance torques, which acted for a short period of time, that is, produced no more than a 0.12 deg amplitude and a damping time of no more than 30 min.

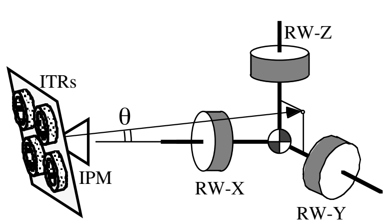

Fig. 20 Method to unload RW by the thrust of IES.

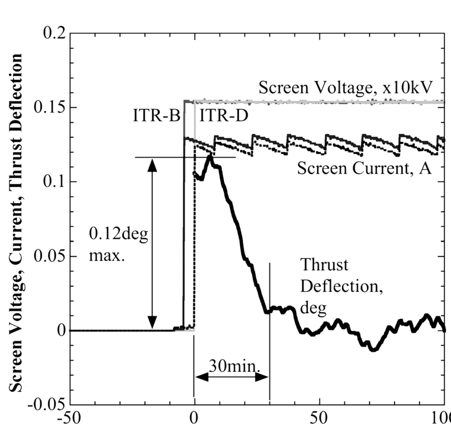

Fig. 21 Deviation of the thrust vector at cold start.

F. Removal of Contamination Among Grids

In the middle of October 2003, ITR-D's electrical isolation of the electrostatic grid system degraded. Thus, ITR-D was automatically turned off by the ITCU. On 21 October, a rescue operation was executed from the ground station using real time commands. Figure 22 presents the sequence used to recover grid isolation. After two short plasma emissions, a continuous acceleration gradually removed the contamination among the grids, which had caused frequent spark discharges. Eventually, stable operation was achieved. A similar grid short occurred again in October 2004 in ITR-D, which was solved using a similar procedure. Because this process easily removed the contaminants, the carbon materials used were considered more suitable for the construction of the electrostatic grid than metals, because carbon grids were never welded together.

V. Summary

In May 2003, the asteroid explorer Hayabusa was launched into deep space to complete a round-trip space mission between Earth and the asteroid Itokawa. Hayabusa was propelled using four μ10 ECR ion engines, featuring electrodeless plasma generation, providing a long life and high reliability. A single μ10 thruster in space was determined to produce a maximum thrust of 8 mN with a thrust factor of 92%, a specific impulse of 3200 s, a thrust power ratio of 23 mN/kW, an ion production cost of 240 eV, and a propellant utilization efficiency of 87% just after launch. In February 2004, the IES passed through a severe thermal condition at a perihelion, located 0.86 AU from the sun. Hayabusa spacecraft flew by Earth in May 2004. At this moment, the spacecraft accelerated to a velocity of about 4 km/s in the inertial coordinate system using an Earth swing-by. In February 2005, it arrived at an aphelion, which was 1.7 AU from the sun. Thus, Hayabusa reached the farthest points in the deep space ever achieved for a spacecraft using a solar electric propulsion system. At the end of August 2005, Hayabusa had reached a position 4800 km away from Itokawa. The outward journey was accomplished by a combination of the ΔVEGA orbital transfer scheme and electric propulsion, that is EPΔVEGA. The total combined operational time of the ITRs reached 25,800 h and generated 1400 m/s during delta-V, while consuming 22 kg of propellant. The IES fulfilled its function over an input power range from 270 W to 1.1 kW. ITR-D attained 10,400 h of operation, the longest accumulated operational time, and 1720 switching cycles, the most frequent among four ITRs. The ECR ion engines μ10 demonstrated the possibility of powered flight in deep space and succeeded in performing an Earth flyby and an asteroid rendezvous. Hayabusa is being prepared to return to Earth in 2010.

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