Discussion on Performance History and Operations of Hayabusa Ion Engines

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The \(\mu 10\) cathode-less electron cyclotron resonance ion engines, have propelled the Hayabusa asteroid explorer for seven years since its launch in May 2003. The spacecraft was 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 re-entry from a heliocentric orbit. The final stage of the return cruise and the subsequent trajectory correction maneuvers have been accomplished by using a special combined operation of neutralizer A and ion source B after the exhaustion of the other neutralizers’ lives by the autumn of 2009. The total duration of the powered spaceflight was 25,590 h, which provided a delta-V of approximately 2.2 km/s and a total impulse of 1 MN·s. The degradation trends of the thruster performances have been investigated. It seems that the main cause of the degradation was the decrease in effective microwave power input to the discharge plasma induced by the increase in the transmission loss of the microwave feed system, and not due to the increase in the gas leakage through the accelerator grid apertures enlarged by erosion. Unintentional engine stop events have been summarized and analyzed. Most of them occurred due to the limit check errors of the backward microwave powers. Such errors can be decreased by carefully monitoring the trend change in microwave backward power as a function of xenon flow rate in future missions.

Key Words: Hayabusa, Asteroid Explorer, Microwave Discharge, Ion Thruster

1. Introduction

Japan’s Hayabusa asteroid explorer, launched on May 9, 2003 rendezvoused with the asteroid “Itokawa” on September 12, 2005. It executed the orbit maneuver using ECR discharge ion engines “\(\mu 10\)” (Fig. 1 and 2), which have electro-static grids with an effective diameter of 10.5 cm and established 31,400 h as the total numbers of space operational time required to generate an accumulated total impulse of 0.79 MN·s at the end of the first half of the delta-V for the return journey in 2007\(^1\), 2\(^2\). The spacecraft had lost its ability for attitude control using reaction wheels or bi-propellant thrusters due to the failure of two of the three reaction wheels before and after the asteroid encounter and the complete leakage of the hydrazine fuel just after the second touchdown. The ion engine system (IES) has not been used only for orbit maneuvers but also for attitude maneuvers of Hayabusa. A combination of ion propulsion and the last remaining reaction wheel was able to provide attitude control by changing the angles of the ion engine’s gimbals mechanism or by blowing cold xenon gas from the standby thrusters. The latter half of the long-term delta-V for the return journey started in February 2009 and ended at the end of March 2010. In the autumn of 2009, we experienced a difficult situation where neutralizer degradations limited the number of available thrusters to only one, and its thrust was also limited by the neutralizer. It was fortunate that the new combination of an ion source and a fresh neutralizer worked well during the last mission period before homing in, though the specific impulse reduced to half.

In this paper, we describe the ion engine’s operation in the latter half of the return cruise and trajectory correction maneuvers before reentry into the Earth’s atmosphere. The degradation trends of the thruster performances and its possible causes will also be discussed. Unintentional engine stop events will be summarized and analyzed. Methods to improve the operating ratio of ion engines for future missions will also be discussed and proposed.

Fig. 1. Photograph of the Hayabusa ion engine system. Ion thruster heads are referred to as ITH-A (upper left), ITH-B (upper right), ITH-C (lower left) and ITH-D (lower right).

Fig. 2. Schematic diagram of a microwave discharge ion engine \(\mu 10\).
2. Overview of the IES Operational History

We obtained 1,844,000 shots of housekeeping telemetry data during 14,000 h of spacecraft tracking in the 7-year (2592-day) round-trip mission. Cumulative operational hours and number of on/off cycles can easily be analyzed by using the database as shown in Figures 3–5. The most used thruster is thruster D, whose elapsed time was 14,830 h (forced to stop due to neutralizer failure) and number of cycles was 1805 (rapidly increased in the first year due to instability and intermittent operation of IES Power Processing Unit 3 (IPPU-3) to keep its temperature low), which is comparable to geostationary satellite missions. It was recorded as the longest powered spaceflight lasting 25,590 h, which is well over Deep Space 1’s record of 16,265 h (voluntarily stopped), although Deep Space 1 will still continue to be at the top, which will be the longest duration of a single thruster until Dawn exceeds its achievement. Fig. 5 indicates the history of the remaining xenon mass. 66.2 kg of xenon was loaded before launch and 47 kg was consumed for ion thruster operations and cold gas jet operations.

3. Engine Troubles and Workarounds

In this section, engine troubles experienced in the fall of 2009 will be summarized not in chronological order but by thruster ID for clarity. Workarounds that lead to the success of the mission will also be described in the last subsection.

It was observed that the screen current of the ion source A was unstable and dropped to 80% of its nominal value during the initial test in July 2003 when the elapsed time was only 8 h. Since then, thruster A had been reserved as the final backup source until it was found that plasma ignition of the ion source was impossible in October 2009. Most of the incident microwave power was reflected back to the isolator in the coupler box A (CPBX-A). The temperature of the flexible cable between CPBX-A and the ion source A was cold even with microwave input. These facts suggest that something was wrong with the CPBX-A or the cable. Nevertheless, neutralizer A’s plasma seemed to be ignited and it was used along with ion source B later, as will be described in detail hereafter.

The neutralizer voltage was controlled so that the electron emission current equaled the screen current of the paired ion source. If something degrades the neutralizer performance or efficiency, the neutralizer voltage would increase under the condition of constant xenon flow rate and microwave power. Neutralizer B showed a stepwise increase in the voltage on June 22, 2005 before arriving at the asteroid Itokawa, as shown in Fig. 6. After the commencement of the return journey in April 2007, the voltage gradually increased to 50 V and its operation was terminated at an elapsed time of 9579 h. A test conducted on October 22, 2009 showed a rapid voltage increase to 90 V within a few hours after engine start and the stand alone operation of thruster B turned out to be impossible.

Fig. 7 shows the history of neutralizer C’s performance. It had been stable for a long time until a stepwise increase in voltage was observed on October 12, 2009. After this, the current vs. voltage characteristics were measured by changing the flow rates and it was confirmed that thruster C operation was possible at a thrust of about 5 mN by maintaining the coupling voltage lower than 35 V. No further voltage increase was observed at the 5 mN operation for more than 400 h, and thruster C was chosen as a backup for the final mission phase.
along with the main thruster of neutralizer A and ion source B. Due to the successful operation of the A+B thruster, thruster C was not used in 2010.

Thruster D performed most of the required delta-V in 2007 after the failure of neutralizer B. The second term of the homeward delta-V in 2009 was also initiated by thruster D, however, its neutralizer showed a sudden increase in the coupling voltage at the end of March and the voltage reached 35 V at the beginning of April. Hereafter, thruster C was used as the main thruster. After a total of 150 h of short duration operations during May, September and October, thruster D was automatically stopped by the ITCU’s (IES Thruster Control Unit) protective function on November 4, 2009 because the neutralizer voltage exceeded the upper limit of 80 V. Thruster D could not be started by any means after the completion of 14830 h, which is the maximum operational hour limit for all the flight thrusters (Fig. 8). Therefore, the only remaining thruster was C, which was limited to 5 mN operation. This was a very serious situation in which return to the Earth became impossible.

It was very fortunate that we succeeded in achieving the combined operation of neutralizer A and ion source B, both of which were part of failed ion thrusters as on November 10, 2009. Fig. 9 indicates the operation concept. Neutralizer A was ignited before the extraction of ion source B’s beam. Xenon gas was supplied to all the plasma sources, which doubled the total flow rate for generation of thrust equivalent to that of one normal thruster, i.e., the specific impulse dropped to half of the nominal value. Xenon availability was not a concern. Ion source A did not produce discharge plasma due to the microwave feeder’s failure, and the increase in the coupler box A’s (CPBX-A) operating temperature due to microwave reflection was a concern. Heaters around the microwave supply unit (MSU) were almost disabled to lower the high temperature of CPBX-A. Power processing unit No. 1 (IPPU-1) was connected to thruster A, but it was not switched on because of the absence of ions in ion source A. Nevertheless, neutralizer A could emit electrons from the spacecraft potential through the bypass diode in IPPU-1. We have no telemetry for the bypass current monitor. It was very likely that the Hayabusa spacecraft was negatively charged so that electron emission from neutralizer A was promoted. There were two circumstantial evidences. The first is the observed floating potential of the ion source A, which was between 70 and 80 V. This suggests that the spacecraft was charged at least up to that value. The second is the discrepancy between the thrust observed by the radio signal’s Doppler shift and the thrust calculated from the telemetry. The observed thrust was about 92.5% of the calculated thrust using an empirical thrust factor of 0.92. This suggests that the effective ion beam energy was the square of 0.925 and the spacecraft was charged by –220 V (= (0.925^2 – 1) × 1500 V). Of course, it is probable that the charging was smaller and the beam divergence was much larger. Another combination of neutralizer A and the ion source D failed probably due to the longer diagonal distance between the two sources than in the case of neutralizers A and B. The combination of A and B was the only practical workaround for the neutralizer failures.
Let us compare neutralizer performance histories in a ground endurance test and in spaceflight. The prototype model (PM) phase endurance test of a thruster was carried out from March 31, 2000 to January 22, 2003. Fig. 10 shows the time history of the neutralizer voltage at a constant flow rate. Although it was stable below 25 V before an elapsed time of 18,000 h, it gradually increased to 40 V by the end of a 20,000-h operation. As for flight models on board the Hayabusa, the lifetimes of neutralizers range between 9,579 and 14,830 h, which is much lower than the ground test’s achievement. The number of on/off cycles does not seem to be dominant with respect to neutralizer lifetime. The most remarkable difference between in-space and on-ground operations would be the temperature range and throttling operations. Hayabusa ion engines experienced extremely low temperatures several times and this might have somehow shortened the lifetime of neutralizers. Throttling slightly increases the coupling voltage and the number density of doubly charged ions in the discharge chamber may increase and this may make the ion sputtering on the internal surfaces severer.

We have analyzed the stored PM neutralizer by disassembling it and found many magnetized metal flakes stuck on the magnetic yokes where local magnetic surface flux densities are large. These metal flakes may be immersed into the discharge plasma, may become a new source of surface contamination of the dielectric part of the microwave launcher by ion sputtering and sputtered metal deposition, or may increase the plasma loss. All the flight and ground test data shown in Figures 6–8Fig. 10 have a common feature that there is a synchronized increase in neutralizer coupling voltage and neutralizer backward power. This may be caused by the above mentioned metal coating of the dielectric surfaces of a microwave launcher. The starting point of the synchronized increases would correspond to the occurrence of delamination and sticking of a piece of flake. Improvement of the neutralizer reliability is the most important issue in our current work with microwave discharge ion engines, and we are considering covering the magnetic yokes with thin molybdenum protectors to reduce the metal flake delamination and sticking5).
Fig. 11. Concept of trajectory correction maneuvers.

Fig. 12. Control direction in $\delta R_p$-$\delta T_p$ plane at TCM-0.

Fig. 13. TCM plan and results (Orbits determined by JPL and JAXA, independently and in good agreement with each other) in $\delta R_p$-$\delta T_p$ plane.

Table 1. Plan of trajectory correction maneuvers (TCMs).

<table>
<thead>
<tr>
<th>Dates in 2010</th>
<th>Delta-V</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Duration (h)</td>
</tr>
<tr>
<td>TCM-0</td>
<td>4/4–4/6</td>
</tr>
<tr>
<td>TCM-1</td>
<td>5/1–5/4</td>
</tr>
<tr>
<td>TCM-2</td>
<td>5/22–5/26</td>
</tr>
<tr>
<td>TCM-3</td>
<td>6/3–6/5</td>
</tr>
<tr>
<td>TCM-4</td>
<td>6/9</td>
</tr>
</tbody>
</table>

Fig. 14 shows the history of the combined operation of the neutralizer A and ion source B until Hayabusa’s reentry to the Earth’s atmosphere on June 13, 2010. The failed neutralizer, neutralizer B seldom showed electron current emissions (referred to as “Neutralizer 2 (NEUT-2) (mA)” in Fig. 14) for the first 3100 h of the combined operation. However, continuous currents from neutralizer B restarted in the middle of TCM-1, although the assistance from neutralizer A was still indispensable for the start of thruster B and the spacecraft charging was at the same level as for the combination of A and B. Perhaps this was not the actual revival of neutralizer B and its electron emission, but the leakage current through an electric short circuit at the neutralizer’s insulator surfaces contaminated by the deposition of sputtered metallic materials generated during the combined operation of A and B in which the spacecraft was highly negatively charged relative to the ion engine plume plasma.

Fig. 14. Time history of the combined operation of the neutralizer A and ion source B. IPPU-1, and IPPU-2 were connected with thruster A and B, respectively. For example, NEUT-1(V) refers to neutralizer A’s voltage and SCREEN-1(mA) refers to ion source B’s current. Temperatures of coupler boxes (CPBX) and flexible cables (FLX) are also plotted. Cruise phase was over by the end of March 2010. The following five short-duration burns are called trajectory correction maneuvers (TCM)-0–4 before the reentry on June 13, 2010.

5. Ion Engine Performance Degradation History

Figures 15–17 show the evolution of screen current as a function of mass flow rate of the ion source for every 1000 h. Screen currents were averaged for discrete flow rates. We can
see an almost monotonous curve shift to the lower right side as the elapsed time increases for all the thrusters which indicates the decrease in propellant utilization efficiency due to accelerator aperture erosion and some other performance degradation of the microwave discharge. We have analyzed the accelerator grid aperture enlargement after completion of the 20,000-h PM ground endurance test (Fig. 10). This analysis suggested that the largest aperture’s diameter was only 6% larger than the original (at 0 h) which corresponds to 11% enlargement of the open area that determines neutral gas leakage from the discharge chamber of the ion source. Most of the apertures at different locations had smaller enlargements. Judging from these observations, the performance change of flight thrusters was not entirely due to the grid erosion. Some other gas leakage from feed line interfaces, decrease in microwave transmittance through passive components or some kind of contamination inside the discharge chambers would be the suspected causes of performance degradations. It is strange that some degradation was seen in the long term engine hibernations in 2006 and 2008. These happened at 8950 h (thruster B), 8360 h (thruster C) and 13,390 h (thruster D) and corresponding small gaps between performance curves can be observed in Figures 15–17. The reason for this is not clear.

Other characteristic changes were observed in the neutralizer’s microwave forward power history, though the graphs of these trends are not shown in this article. Although the forward power should be almost constant regardless of flow rates, it gradually decreased or increased as the time approached each neutralizer’s end of life. According to the results of our ground experiments, change in microwave powers generally affects neutralizer coupling voltages. Any shift from the designed value is not preferable. Some characteristic change in the microwave feed lines might have degraded both the ion source and neutralizer performances.

6. Summary of Interruptions of Ion Engine Operation

Table 2 summarizes the ion engine activities over the seven-year Hayabusa mission; these are already shown in Figures 3 and 4. In addition, the accidental (i.e., not planned) shutdowns of each thruster were investigated and are listed here. The ion drive of Hayabusa was interrupted 68 times. The overall success probability of powered spaceflight using 1–3 thrusters is limited to 0.84. The causes of interruptions are categorized into five types and summarized in Table 3. Category 1 is called mis-operation and it happened only in the early stage of the mission. Accidental stops of ion engine operation were most frequently caused by the upper limit errors of backward microwave power (in Category 2), which turned out to be not actually critical to the engine hardware. Figures 18–20 show the history of microwave backward power as a function of the mass flow rate of the ion source. Backward microwaves are absorbed and converted into heat by a dummy load inside a coupler box (CPBX) which distributes the microwaves to the ion source and the neutralizer. Although the CPBX can tolerate the perfect reflection of microwave power of 60 W for more than one hour, we were very cautiously and closely maintaining the
upper limit of the thruster control logic to the most recently observed values. On the other hand, the backward power as a function of mass flow rate was far from constant and its trend curve gradually changed as the accumulated operational hours increased as shown in Figures 18–20. The ion engines’ throttling levels (i.e., flow rates) were also gradually changed according to the orbit maneuvering plan. These were the reasons why the ion engine operation was unstable and flawed. As for thruster C, its initial operation started from the full throttle level, where the microwave backward power was a maximum of 15 W as shown in Fig. 20, however, the limit had always been set to a value higher than 15 W. This value had been appropriate every time and had never been updated. This is why thruster D stopped only once due to this error. The only exception was due to a slightly different error mode on an increase in the microwave reflection in the plasma-discharge-only mode not in the beam-acceleration mode. Because the microwave power reflection trend of thruster B was unstable and unforeseen, it was not suited to be limited by a constant strict upper limit.

Although accidental shutdowns due to Category 4 and 5 errors in Table 3 can not be avoided by changing the control logic settings, the Category 1–3 errors can be decreased or eliminated by proper planning and reasonable relaxation of limits. If these errors were eliminated, the successful operational ratio of powered spaceflight would be increased. As for thruster A, although the error occurred in the upper limit of the thruster control logic on an increase in the microwave reflection in the plasma-discharge-only mode not in the beam-acceleration mode, it was not suited to be limited by a constant strict upper limit. As for thruster C, its initial operation started from the full throttle level, where the microwave backward power was a maximum of 15 W as shown in Fig. 20, however, the limit had always been set to a value higher than 15 W. This value had been appropriate every time and had never been updated. This is why thruster D stopped only once due to this error. The only exception was due to a slightly different error mode on an increase in the microwave reflection in the plasma-discharge-only mode not in the beam-acceleration mode. Because the microwave power reflection trend of thruster B was unstable and unforeseen, it was not suited to be limited by a constant strict upper limit.

Although accidental shutdowns due to Category 4 and 5 errors in Table 3 can not be avoided by changing the control logic settings, the Category 1–3 errors can be decreased or eliminated by proper planning and reasonable relaxation of limits. If these errors were eliminated, the successful operational ratio of powered spaceflight would be increased from 0.84 to 0.94 as shown in Table 2. Actually, shutdown due to the upper limit of microwave backward power occurred only twice on the return journey from asteroid Itokawa due to appropriate settings and careful trend monitoring.

Table 2. Summary of in-space operations of Hayabusa IES.

<table>
<thead>
<tr>
<th>Thruster A</th>
<th>Thruster B</th>
<th>Thruster C</th>
<th>Powered Spaceflight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hours</td>
<td>ON/OFF Cycles</td>
<td>Accidental Shutdowns</td>
<td>Successful Operational Ratio</td>
</tr>
<tr>
<td>Neutralizer A</td>
<td>2324</td>
<td>25</td>
<td>0.95</td>
</tr>
<tr>
<td>Thruster A</td>
<td>12904</td>
<td>43</td>
<td>28</td>
</tr>
<tr>
<td>Thruster B</td>
<td>11953</td>
<td>23</td>
<td>21</td>
</tr>
<tr>
<td>Thruster C</td>
<td>7663</td>
<td>18</td>
<td>20</td>
</tr>
<tr>
<td>Powered Spaceflight</td>
<td>25599</td>
<td>20</td>
<td>26</td>
</tr>
<tr>
<td>Total (A+B+C+D)</td>
<td>56337</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Table 3. Summary of accidental shutdowns of ion engines.

<table>
<thead>
<tr>
<th>Cause of Accidental Shutdown</th>
<th>Error Category</th>
<th>Number of Shutdowns Per Thruster</th>
</tr>
</thead>
<tbody>
<tr>
<td>False Start Before Enable by Power Control Unit</td>
<td>B C D</td>
<td>0 0 0 0</td>
</tr>
<tr>
<td>Upper Limit Error of Power Control Unit</td>
<td>1 1 1 1</td>
<td></td>
</tr>
<tr>
<td>Neutralizer Failure (Voltage Upper Limit)</td>
<td>5 1 1 1</td>
<td></td>
</tr>
<tr>
<td>Neutralizer Failure (Voltage Lower Limit)</td>
<td>5 1 1 1</td>
<td></td>
</tr>
<tr>
<td>Total (All Categories)</td>
<td>12 21 26</td>
<td></td>
</tr>
<tr>
<td>Total (Categories 4–5)</td>
<td>5 9 16</td>
<td></td>
</tr>
</tbody>
</table>

Error Category
1. Mistake of autonomy programming or ground operations
2. Too strict limit check conditions
3. Low power margin due to challenging operation
4. Very low possibility but inevitable events
5. Hardware errors or failures

Fig. 18. Averaged microwave backward power of thruster B vs. mass flow rate of the ion source.

Fig. 19. Averaged microwave backward power of thruster C vs. mass flow rate of the ion source.

Fig. 20. Averaged microwave backward power of thruster D vs. mass flow rate of the ion source.

7. Conclusion

The µ10 cathode-less electron cyclotron resonance ion engines, have propelled the Hayabusa asteroid explorer for seven years since its launch in May 2003. The spacecraft was 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 re-entry from a heliocentric orbit. The final stage of the return cruise and the subsequent trajectory correction maneuvers have been accomplished by using a special combined operation of neutralizer A and ion source B after exhaustion of the other neutralizers’ lives by the autumn of 2009. The total duration of the powered spaceflight was 25,590 h which provided a delta-V of approximately 2.2 km/s and a total impulse of 1 MN·s. The degradation trends of the thruster performances have been investigated. It seems that the main cause of the degradation was the decrease in effective microwave power input to the discharge plasma induced by the increase in the transmission loss of the microwave feed system, and it was not due to the increase in gas leakage through the accelerator grid apertures enlarged by erosion. Unintentional engine stop events have been summarized and analyzed. Most of them occurred due to the limit check errors of the backward microwave powers. Such errors can be decreased by carefully monitoring the trend change in microwave backward power as a function of xenon flow rate in future missions.

Acknowledgments
The authors sincerely thank all the people whose contribution made the longest spaceflight ever powered by small ion engines successful. The Hayabusa mission guidance and navigation was partly supported by NASA’s Jet Propulsion Laboratory (JPL) using its Deep Space Network. The authors appreciate the full support of NEC Aerospace Systems, Ltd. That helped make Hayabusa’s difficult cruise operation and final approach to the Earth successful.

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