Overview of JAXA's Activities on Electric Propulsion

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Abstract: JAXA is promoting to enhance electric propulsion technology in space and on ground by ourselves and/or under international collaboration. We have just finished Hayabusa space mission and started to develop Hayabusa2 using the microwave discharge ion engines. Our efforts are also concentrated on R&D of ion engines, MPD arcjet, electrodynamic tether, plasma sail, and so on for future space realizations.

I. Introduction

JAXA (Japan Aerospace eXploration Agency) promotes variety of the electric propulsion technology in space realizations and R&D fields leading Japanese space community and collaborating with international agencies. This paper will disclose recent activities of JAXA.

II. Flight Projects

A. Hayabusa Asteroid Explorer

Hayabusa space mission aims to retrieve surface material of the asteroid to Earth. Total launch mass of the spacecraft is 510 kg. The artist image of Hayabusa asteroid explorer under the powered flight in deep space is seen in Fig.1. The solar array panels can generate 2.6 kW electrical power at 1 AU from Sun. The µ10 ion engine with 10 cm effective diameter was developed in order to dedicate to Hayabusa space mission as the ion engine system (IES). The main feature of this device is microwave plasma generation without electrodes¹, which are exhaustive. Four µ10 are installed on Hayabusa spacecraft and three of them can generate thrust simultaneously. The dry mass of IES is 59 kg including a gimbal and a propellant tank, which was filled with xenon propellant 66 kg. A single µ10 is rated at 8 mN thrust, 3,000 sec Isp, and 350 W electrical power consumption so that Hayabusa spacecraft is accelerated 4 m/s per a day by the maximum thrust 24 mN. Hayabusa spacecraft was launched in 2003. Acceleration by the microwave discharge ion engines during two years made successfully it arrive at the near-Earth Asteroid Itokawa in 2005. The spacecraft suffered from a fuel leak and was lost after its successful descent, touch-down and lift-off at the end of November in 2005. Fortunately a series of rescue operation resumed to communicate with the spacecraft again one and half months later. Xenon cold gas jets from the ion engines stabilized attitude of spacecraft on behalf of damaged chemical thrusters. The ion engines started to boost up the spacecraft for Earth return in 2010 reconsidered from the original in 2007. On the way to Earth in November 2009 an interruption of acceleration due to

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malfunction of a thruster was resolved by combined operation of the ion source B and neutralizer A, which had guided Hayabusa precisely to a landing point in Australia. The microwave discharge ion engines achieved 40,000 hours & units as accumulated operational time and 2,200 m/s as delta-V in 7-year deep space flight. The reentry capsule was separated from a mother ship Hayabusa on June 13th 10:54, 2010 UT. And Hayabusa and the capsule dived into Earth atmosphere at 12 km/s velocity. Above Woomera desert of Australia a lot of fireballs were observed by JAXA optical observation teams and NASA airborne observatory as shown in Fig.2. Hayabusa spacecraft broke apart and emitted blue and yellow lights and vanished in midair. On the other hand the capsule took a stable flight and safe landing. Hayabusa mission was deeply supported by NASA, Australia government, international organized science team and so on.

Figure 1. Hayabusa asteroid explorer executing powered flight in deep

Figure 2. Fireballs observed at Woomera desert by JAXA optical observation teams on June 13th, 2010

B. Hayabusa2

Encouraged by achievement of Hayabusa, the new space project Hayabusa2 has just started. A 570 kg spacecraft similar to Hayabusa will be lunched by H2A rocket in 2014 and propelled by the microwave discharge ion engines $\mu_{10.2}$. It aims to arrive at an asteroid 1999JU3 in 2017 and to come back Earth in 2020.
C. Microwave Discharge Neutralizer on DubaiSat-2

The microwave discharge neutralizer with 500 mA emission current enhanced from μ10 ion engine with 130 mA will be tested in low Earth orbit on DubaiSat-2, which will be launched on 2012, in the collaboration between EIAST (Emirate Institution of Advanced Science and Technology) and JAXA. The neutralizer is combined with Korean Hall effect thruster on DubaiSat-2. The flight experience of the neutralizer will feed back to JAXA’s space projects using μ20 ion engine.

D. Super Low Altitude Test Satellite

The Super Low Altitude Test Satellite (SLATS) is a small test satellite of near 300 kg shown in Fig.5. It demonstrates the flight in a super low Earth orbit under 250 km of altitude, where an ion engine system is used to compensate for air drag. The SLATS ion engine system is being developed based on the Kiku-8 ion engine system. Its schematic and main performance parameters are shown in Fig.6 and Table 1 respectively. The ion thruster has a few small dimensional changes to reflect the anomaly of the Kiku-8 ion thruster. The thrust required for the SLATS system will be changed from the Kiku-8. Hence, the thrust range test was conducted with various operating parameters. The test showed that the thrust range from 10 mN to 35 mN could be realized. A hollow cathode used in the ion thruster might be sensitive to atomic oxygen (AO) that is rich in the super low altitude orbit. So, the effect of AO for the hollow cathode was evaluated by using some cathode piece parts samples. After the samples were exposed by AO, they were analyzed physically by SEM-EDX, TDS and XPS, and the electron emission was measured. The results showed there is no damage of the samples by AO exposure. In the next step, the resistance of the hollow cathode against AO will be evaluated by using a hollow cathode similar to the flight model. The Power Processing and Control Unit (PPCU) is a component that combines the functions of power supplies and a controller. These functions were divided in two components on the Kiku-8. The SLATS system requires a compact and lightweight ion engine system. Hence, a new PPCU is being developed by using highly heat-conductive printed boards. A breadboard model of the PPCU was fabricated and a coupling test with an ion thruster showed a good performance. The power efficiency of the high-voltage power supply achieved over 90 %. The SLATS system
requires autonomous on/off control of the ion thruster in orbit. So, new control logic for orbit keeping was added to the previous control logics in the PPCU. Presently, the design of a PPCU EFM is conducted.

Figure 5. The SLATS in orbit

Figure 6. The ion engine system for the SLATS

Table 1. Main performance parameter of SLATS ion engine system

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Method</td>
<td>Kaufman type ion thruster</td>
</tr>
<tr>
<td>Thrust level</td>
<td>10mN (tentative)</td>
</tr>
<tr>
<td>Specific impulse</td>
<td>1,000sec (tentative)</td>
</tr>
<tr>
<td>Total dry mass</td>
<td>28kg</td>
</tr>
<tr>
<td>Bus voltage</td>
<td>32.5-50V</td>
</tr>
<tr>
<td>Power consumption</td>
<td>385W (tentative)</td>
</tr>
<tr>
<td>Heat dissipation</td>
<td>110W (tentative)</td>
</tr>
<tr>
<td>Signal interface</td>
<td>RS422</td>
</tr>
<tr>
<td>MEOP</td>
<td>15MPa</td>
</tr>
<tr>
<td>Propellant</td>
<td>Xenon</td>
</tr>
<tr>
<td>Initial mass of propellant</td>
<td>12kg (tentative)</td>
</tr>
<tr>
<td>Operational mode</td>
<td>IDLG (hollow cathode heating)</td>
</tr>
<tr>
<td></td>
<td>ACTV (hollow cathode activation)</td>
</tr>
<tr>
<td></td>
<td>NEUT (neutralizer hollow cathode unit operation)</td>
</tr>
<tr>
<td></td>
<td>DISC (main electrical discharge operation)</td>
</tr>
<tr>
<td></td>
<td>ORBIT (the thrust generation and the electrical discharge maintenance)</td>
</tr>
<tr>
<td>Function for failure</td>
<td>CM mode (grid short-circuit opening)</td>
</tr>
<tr>
<td></td>
<td>Neutralizer hollow cathode discharge watching function</td>
</tr>
<tr>
<td>Required life time</td>
<td>3,000hours</td>
</tr>
</tbody>
</table>
III. Flight Plans

A. DECIGO Pathfinder

Gravity wave detection requires the determination of extremely small strains, and moving to space provides an opportunity to be free of fundamental limitations from gravity gradient effects on the Earth. Following this idea, some gravity wave detection observatory missions are planned. The LISA mission by NASA and ESA is a prime example. Japanese group also proposed another gravitational wave observatory space mission, DECIGO. To test the key technologies for DECIGO, DECIGO pathfinder mission (DPF) is now studied by Kyoto University, National Astronomical Observatory Japan, JAXA, and so on.

The purposes of DPF are to test the key technologies and to make observations at 0.1-1 Hz frequency band, which is different from the LISA’s target. DPF in Fig.7 will be a small satellite with weight of about 400 kg, orbiting the earth along a sun-synchronous orbit with an altitude of 500 km. The mission part of DPF is designed to be a prototype of DECIGO, being comprised of a Fabry-Perot (FP) cavity, a stabilized laser source, and a drag-free control system. The cavity has a short baseline length of only 30 cm, so all the components can be stocked in a 1-m-cubic mission module that is located on the upside of the bus module. For the drag-free control of the satellite position, 100 micro-Newton class thrusters for DECIGO/DPF are employed. From a conceptual design of the DECIGO pathfinder, requirements for thrust precision (0.5 µN), thrust noise (0.1 µN/Hz^{1/2}) are derived and eight 10-FEEP thrusters and two 100-FEEP thrusters are to be equipped for the DPF satellite. DPF is now one of the several mission candidates for the upcoming small scientific missions by JAXA, and might be launched in 2017 in the earliest case.

![Figure 7. DECIGO Path Finder (DPF)](image)

IV. Research and Development

A. 25cm Ion Engine

A series of super low altitude practical satellites, which will be launched following SLATS in 2014, will be propelled by plural 12-cm ion thrusters compensating air drag. Larger ion thrusters are required for larger satellites and a 25-cm xenon ion thruster, whose designed maximum thrust is 100 mN or higher, is under development for their use. The power to thrust ratio, which is one of the most important parameters for air drag compensation, measured at the input of a power processing unit is expected to be lower than 27 W/mN while the specific impulse is held at 3,000 s or higher. Table 2 shows the status of laboratory model thruster performance and the target performance has not yet been obtained. The thruster is going to be modified and tested in 2012. Figure 8 shows a 25-cm thruster operating.
A preliminary duration testing to find out unexpected problems had been held in 2010 and was stopped at 4,370 hours due to grid damaging by a problematic beam supply. The beam supply is going to be replaced by another model for future testing.

Duration testing of two graphite hollow cathodes is continuing and their operating periods reached 38,700 hours and 22,200 hours, respectively on June 20, 2011. Currently, the ways of analyzing the cathodes are under consideration. The testing of the discharge cathode will be stopped intentionally some time in 2012 and the cathode will be analyzed.

### Table 2. Performance of laboratory model thruster

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust, mN</td>
<td>80.3</td>
</tr>
<tr>
<td>Beam Voltage, kV</td>
<td>1</td>
</tr>
<tr>
<td>Accel. Grid Voltage, V</td>
<td>-200</td>
</tr>
<tr>
<td>Beam Current, A</td>
<td>1.615</td>
</tr>
<tr>
<td>Discharge Voltage, V</td>
<td>30.7</td>
</tr>
<tr>
<td>Discharge Current, A</td>
<td>10.94</td>
</tr>
<tr>
<td>Propellant Utilization at Discharge Chamber</td>
<td>0.902</td>
</tr>
<tr>
<td>Ion Production Cost, W/A</td>
<td>177</td>
</tr>
<tr>
<td>Power-to-Thrust Ratio at Thruster’s Input, W/mN</td>
<td>24.9</td>
</tr>
<tr>
<td>Specific Impulse, s</td>
<td>3,080</td>
</tr>
</tbody>
</table>

**Figure 8. The 25-cm Ion Thruster**

**B. μ20 Microwave Discharge Ion Engine**

We have been developing a 20-cm diameter class microwave discharge type ion thruster "μ20". In contrast to the μ10 whose ion beam current was saturated to 150 mA at higher microwave powers than 30 W, the μ20 is expected to generate 500 ~ 530 mA ion beam current with 100 W microwave power and 1300 V acceleration voltage, yielding beam ion production cost of 200 W/A and thrust of 27 ~ 30 mN due to enlargement of the discharge chamber and moderate plasma density below cutoff. μ20 will be applied to deep space missions with larger DV and more massive spacecraft than Hayabusa or to leaving the ecliptic mission (e.g. observation of polar region of sun).

A 10,000-hour operation has been executed in order to identify any design problems. We carried out the measurements of the thrust performance variation and the statistical analysis on aperture diameters of flat C/C composite grids using the flat-head image scanner. The propellant utilization efficiency has been gradually decreased from 70% to 66% with increasing the accelerator hole diameters. It was confirmed that the grid system of μ20 ion source kept the required thrust performance with durability.
C. \( \mu 1 \) Microwave Discharge Ion Engine

An ion thruster \( \mu 1 \) is a microwave discharge ion thruster designed for 10–100 kg small spacecraft. The thruster head and operational view are shown in Fig.11. The major features of the \( \mu 1 \) thruster are low microwave power driving (~ 1 W) and full set development including a neutralizer. These days, increasing number of planned small spacecraft missions are in need of propulsive capability. Propulsion devices supply the spacecraft with attitude control, station keeping, and orbit transfer. An ion thruster is the most successful electric propulsion device and its characteristics, high specific impulse and controllable thrust, are suitable for the future small spacecraft missions. Hence ion thruster is a promising propulsion device also for small spacecraft, if the thruster is successfully miniaturized. So far, several studies on miniature ion thrusters have been conducted by a number of researchers. However, no miniature ion thruster has been utilized for small spacecraft. It has been impeded by two major problems: strict power limitation on small spacecraft and absence of the suitable neutralizer for miniature ion thrusters. The \( \mu 1 \) ion thruster has been studied to overcome these problems in ISAS/JAXA\(^8\), and currently with the University of Tokyo. ECR (electron cyclotron resonance) plasma heating by microwave realized simple structure (no electron cathode) and miniaturization enough to be installed on small spacecraft. Microwave power to generate the plasma was decreased down to 1.0 W with keeping its ion production cost 224 W/A. The ion beam current is 4.48 mA by 1.0 W microwave power, 1.5 kV beam voltage, and 14.5 \( \mu g/s \) xenon flow. At this nominal operating condition, estimated thrust is 260 \( \mu N \) using the thrust coefficient of 0.90.

The plasma source developed for the ion source was applied for a miniature neutralizer with a minor modification\(^9\). The developed miniature neutralizer supplied required electron current consuming the microwave power of 1.0 W. The contact voltage (biased negative voltage to a neutralizer) to emit electron ranged between 10–30 V. Total performance of the miniature ion thruster system was estimated based on the experimental data. We assumed energy conversion efficiencies of 0.80 and 0.50, respectively for high voltage power source and microwave.
power source. As a result, the µ1 miniature ion thruster system has the following performance: total power consumption: 13.2 W, total xenon flow rate: 26.3 µg/s (ion: 14.5 and neut.: 11.8), generating thrust: 258 µN, thrust to power ratio: 19.6 mN/kW, specific impulse: 1290 s, and thruster efficiency: 12.3%. This performance includes power and propellant consumption by the neutralizer (experimental data) and energy conversion loss at the power sources (estimated values). Additionally, the thruster performance can be adjusted to high thrust to power ratio or high specific impulse by changing the microwave power and propellant flow rate.

![Figure 11. Picture of the miniature ion thruster µ1](image)

(a) ion source and (b) its operational view (left is an ion source and right is a neutralizer).

D. MPD and RF Thruster

Self-field magnetoplasmadynamic (MPD) thrusters have been investigated in order to identify power loss mechanisms by use of magnethydrodynamics (MHD) codes whose results are compared with experimental results. In particular, discharge voltage-current characteristics near the critical current, at which transition from electrothermal to electromagnetic acceleration takes place, are investigated. Hall effect, shortage of the current carrier near the anode surface, and real gas effects, such as nonequilibrium dissociation and ionization, three temperature for hydrogen propellant, are investigated. An electrode model is under development in order to evaluate sheath voltage self consistently.

An electrodeless electromagnetic thruster was proposed by Toki et al. in 2003 in order to extend the lifetime of electric propulsion with a compact size. The research becomes a part of the HEAT (Helicon Electrodeless Advanced Thruster) project. The thruster concept is shown in Fig.12 left. First, the plasma is produced by Helicon waves and is accelerated by radio frequency (RF) power which is applied by two pairs of planer antennae. Acceleration of the plasma by this method has been observed experimentally as a small increment compared with thermal flow velocity without the RF power. In order to maximize electromagnetic acceleration, an analytical thrust model is developed to improve the design of laboratory models where the electromagnetic thrust is expected to be observed. We are fabricating the laboratory model and thrust stands for thrust measurement to confirm the scaling laws obtained from the analytical thrust model. Fig.12 right shows first Helicon plasma discharge obtained by the laboratory model of 26 mm diameter without acceleration antennae.

![Figure 12. Electrodeless Lissajous Plasma Thruster](image)

(Left) Concept of Lissajous Acceleration, (Right) Photo of Helicon Plasma (2 kW RF power at 9.5 MHz frequency was applied to Ar gas flow with 20 sccm. Axial magnetic field of 0.08 T was applied on thruster axis.)
E. Electrodynamic Tether

Electrodynamic tether (EDT) systems have been studied for ten years in the aerospace research and development directorate of JAXA.\textsuperscript{18, 19} The major purpose of the development is applying the EDT for space debris deorbit. Concerns about space debris have been realized in recent years, and the strategic discussion on the space environment remediation has become vigorous. In order to prevent the progress of “Kessler Syndrome”, it is necessary to remove a number of large-sized debris from LEO, and an efficient propulsion system is needed to realize it. The EDT is suitable for this purpose because the system can be small, simple, and light-weight. In addition, neither thrust vector control nor point-of-mass management are necessary for the EDT.

An on-obit demonstration of the EDT is being proposed in JAXA for validating the key technologies. Table 3 and Fig.13 show the tentatively specified characteristics and schematic of the demonstration, respectively. In this plan, EDT thrust generation will be demonstrated in sun-synchronous-orbit by observing the orbit change. Features of major components for the demonstration are as follows. A) The tether consists of three braided bare conductive wires, which possess unique net geometry to keep distances between the each cord. This tether geometry can lower the probability of tether cutting by small debris collisions, and improve the electron collection capability. B) The spool reel is adopted to reduce the risk of tether deployment failure. The end of the tether is connected to a rotation reel for braking. C) The release mechanism for mother-daughter separation consists of a spring and non-explosive actuators. D) The electron emitter is a field emission cathode using carbon nanotubes.\textsuperscript{20} The arrayed cathodes emit the electron current up to 40 mA. Combination of the bare tether and field emission cathode enables us to build a completely propellant-free EDT system.

\begin{table}[h]
\centering
\caption{Tentative specifications of EDT demonstration}
\begin{tabular}{|l|l|}
\hline
Tether length & 1 km \\
Max. tether current & 40 mA \\
Estimated thrust & 0.1 mN (ave.) \\
Tether & Net bare tether \\
Reel & Spool reel \\
Release mechanism & Spring and NEAs \\
Electron emitter & F. E. cathode \\
\hline
\end{tabular}
\end{table}

\begin{figure}[h]
\centering
\includegraphics[width=\textwidth]{EDT_demonstration_diagram.png}
\caption{System diagram for EDT demonstration}
\end{figure}

F. Magnetic Sail and Plasma Sail

JAXA and Japanese universities are collaborating to realize a spacecraft propulsion system utilizing the energy of the solar wind. Experimental and numerical studies showed that moderately sized magnetic sails with the ion inertial scale magnetosphere (~100 km) could produce a Newton-class thrust level.\textsuperscript{21, 22, 23} On the same scale length, magnetic cavity size was successfully increased in the laboratory experiments of magnetic sail with a plasma jet (Magnetoplasma sail, MPS), as in Fig.14.\textsuperscript{24} This type of “magnetic field inflation” is the key process to increase a thrust level of magnetic sail by increasing magnetospheric size, but so far, no significant thrust increment was observed in the series of thrust measurements. On the contrary, MPS concept was successfully demonstrated by numerical simulations: by numerical simulations on MHD scale, thrust gain ($F_{\text{mps}}/F_{\text{mag}}$) as much as 2 was obtained where $F_{\text{mag}}$ is a thrust force of pure MagSail and $F_{\text{mps}}$ is an enhanced thrust level by MPS operation.\textsuperscript{25} Also, in a transitional regime between electron-, ion- and MHD-scales, full particle-in-cell numerical simulation obtains...
$F_{\text{mps}}/F_{\text{mag}}$ as much as 5 for an electron scale MPS ($\sim$5 km magnetosphere)\textsuperscript{26} and hybrid particle-in-cell simulation indicates $F_{\text{mps}}/F_{\text{mag}} \approx 10$ is possible for an ion scale MPS ($\sim$10 km magnetosphere).\textsuperscript{27}

Figure 14. Laboratory Experiment of Magnetoplasma Sail

H$_2$ plasma flow simulating solar wind (25 km/s, $5 \times 10^{18} \text{m}^{-3}$) is introduced to miniature Magnetoplasma sail with coil and plasma source (15 km/s, $3 \times 10^{19} \text{m}^{-3}$).\textsuperscript{24} (Left) Experimental Setup, (Right) Magnetoplasma Sail with plasma source operation (upper) and Magnetic sail without plasma source operation.

References