

SPACECRAFT DESIGN AND FLIGHT STATUS OF HAYABUSA ASTEROID EXPLORER PROPELLED BY MICROWAVE DISCHARGE ION ENGINES

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Abstract

The microwave discharge ion engines “ $\mu 10$ s” are devoted to the main propulsion on “HAYABUSA” asteroid explorer. In the development program various kinds of tests and assessments were applied to the ion engines and the spacecraft; the endurance test, the EMI susceptibility test, the interference test between plasma and communication microwave, the beam exhaust test on the spacecraft, the assessment on plasma interference with solar array, so on. The spacecraft was input in the deep space by M-V rocket on May 9, 2003. After vacuum exposure and several runs of baking for reduction of residual gas the ion engine system established the continuous acceleration of the spacecraft toward the asteroid “ITOKAWA”. On the beginning of December 2004 the ion engines reached the total number of space operational time 20,000hour. And the spacecraft passed by the aphelion 1.7 astronomical unit on February 18, 2005 so that the microwave discharge ion engine $\mu 10$ s became the electric propulsion to arrive at the furthest space from Sun. It will rendezvous with the target asteroid on September 2005.

Introduction

At May 9th 01:29PM, 2003 the MUSES-C asteroid explorer¹⁾ was launched directly into the deep space by the M-V rocket No.5 from Kagoshima Space Center and named “HAYABUSA”, which means falcon in Japanese. It was just a moment that the microwave discharge ion engine “ $\mu 10$ ” was space-borne after the 15-year research and development. The vacuum exposure during two weeks and various kinds of test runs in a month enabled $\mu 10$ s to accelerate HAYABUSA asteroid explorer continuously at several meter per second in a day from July. Though the scientific spacecraft were thought impossible to install the electric propulsions due to penalties on weight and electric power, not only the improvement of the electric propulsion but also the technological advancement of the rocket system and the needs for the deep space exploration realize the HAYABUSA space mission, which will execute the round trip between Earth and the asteroid “ITOKAWA”. In general a round trip space mission requires too large fuel consumption of the conventional chemical thrusters, which onboard spacecraft are rated around 300sec in the specific impulse. On the other hand, the ion engines generate the thrust over 3,000sec specific impulse. The only electric propulsion makes a spacecraft approach some of the asteroids and secures a return way to Earth. HAYABUSA succeeded the Earth swing-by on May 19, 2004 after one-year acceleration by the ion engines and directed its course toward the asteroid. At the

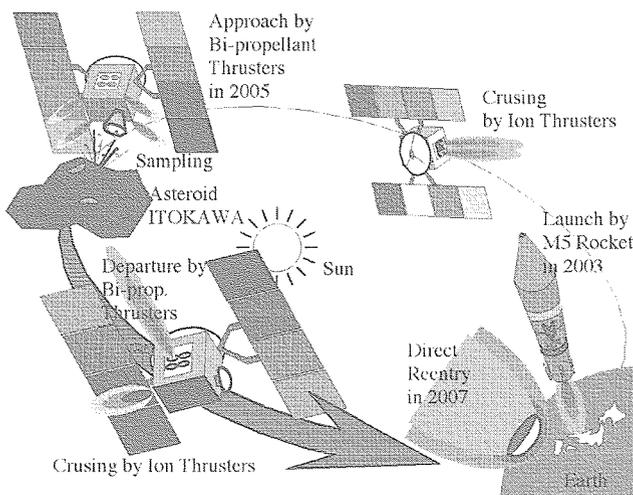


Fig.1 Sequence of events.

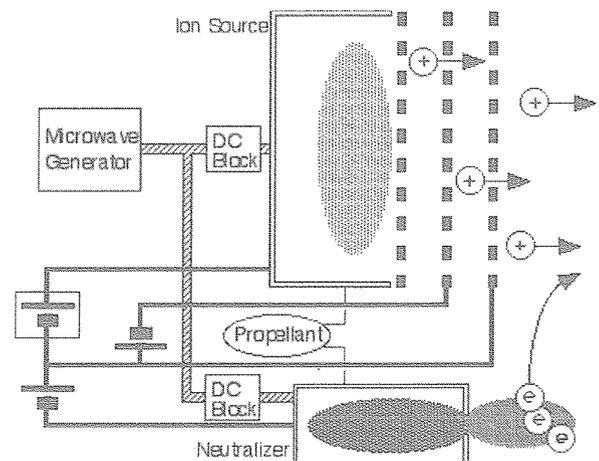


Fig.2 System concept of $\mu 10$.

end of 2004, the total numbers of the space operational time on $\mu 10$ s reached 20,000hour. And the spacecraft passed by the aphelion 1.7 astronomical unit on February 19, 2005, so that the $\mu 10$ s became the electric propulsion to arrive at the furthest space from Sun. Up to now, on April 2005 HAYABUSA locates 2.3 astronomical unit (345,000,000km) apart from Earth, and approaches the asteroid with the remaining distance 1,000,000km. It will arrive at the asteroid in September 2005 and return to Earth again in 2007 on the sequence of events illustrated in Fig.1. Table 1 summarizes HAYABUSA spacecraft. The $\mu 10$ cathode-less ECR microwave discharge ion engine system has been researched and developed by the Electric Propulsion Division in Institute of Space and Astronautical Science (ISAS) / Japan Aerospace Exploration Agency (JAXA) based on the new idea different from others. This paper will report the design and the flight status of $\mu 10$ s on HAYABUSA.

Table 1 Summary on HAYABUSA spacecraft.

Launch weight:	510kg including chemical fuel 67kg and Xe propellant 66kg
Attitude Control:	three-axis stabilization
Communication:	X band, 8kbps max.
Solar Cell Paddle:	triple-junction cells, 2.6kW at 1AU
Chemical Propulsion:	bi-propellant thrusters x 12, 290sec Isp
Electric Propulsion:	microwave discharge ion engines $\mu 10$ x 4, 3,200sec Isp
Payloads:	Telescope Cameras, Near Infra-red Spectrometer, Laser Altitude-meter, X-ray Induced Fluorescence Spectrometer, Small Landing Robot, Sampling Mechanism, Reentry Capsule

System Descriptions

Figure 2 represents the system concept of $\mu 10$. The technological features are summarized as follows. The details are described in Ref. 2.

- 1) Xenon plasmas are generated by ECR microwave discharge without solid electrodes, which are ones of life critical components in the conventional ion engines. Elimination of the solid electrode makes the ion source durable and high reliable.
- 2) Neutralizers are also driven by ECR microwave discharge. Deletion of the hollow cathodes releases the ion engine system from the performance degradation due to oxygen contaminating propellant and time limitation for air exposure during the satellite assembling.
- 3) A single microwave generator feeds simultaneously both an ion source and a neutralizer. This feature reduces the system mass and simplifies the control logic.
- 4) Carbon-carbon composite material is applied to the electro-static grid system. The clearance between the grids is kept stable regardless of temperature due to zero thermal expansion. And it prolongs the life of the acceleration grid because of the low sputtering rate against xenon ions.

The Ion Engine System (IES) was designed and assembled using four $\mu 10$ ion engines. The character “ μ ” is pronounced “mju:” and means “microwave” and the upper stage of “ μ ” rocket. The following number “10” indicates the effective diameter 10cm of the electro-static grid system. The IES system is shown in Fig.3. Each Ion Thruster (ITR) is connected to its own Microwave Power Amplifier (MPA), which emits 4.25GHz microwave to generate high density plasma by means of the electron cyclotron resonance (ECR). High voltage power generated by three IES Power Processing Units (IPPU) is distributed to four ITRs through relay switches so that simultaneous operation is limited three ITRs. The IES Pointing Mechanism (IPM) mounting four ITRs aligns a combined thrust vector through the center of gravity of the spacecraft. The propellant is supplied to ITRs through flow restrictors by means of brow-down from a sub tank, which is charged gas from the spherical titanium alloy main tank. The Ion Thruster Control Unit (ITCU) manages the IES. The dry weight of IES is 59kg. Figure 4 shows the configuration of HAYABUSA deploying a pair of the solar cell paddle (SCP). The high gain antenna (HGA) is mounted on upper surface of the body. SCP and HGA have no rotational and tilt mechanisms. IES is mounted perpendicular to the z-axis, with which HGA aperture is aligned. At the high bit rate communication the spacecraft orientates HGA to Earth without IES firing. In case of the cruising the spacecraft makes SCP face Sun in order to generate electrical power and rotate its attitude around the Sun direction to steer the thrust direction of IES. The SCPs are located out of the cone region of 40 degree along the centerlines of ITRs in order to avoid the ion beam impingement. The solar cell layout is designed to take into account as follows:

- 1) The maximum operational voltage of the solar array circuit is 120V.
- 2) The voltage gap between the neighboring cells is less than 50V.
- 3) No care of canceling on the magnetic field generated by the current loop on SCP because of the deep space operation.
- 4) The cells along the SCP edge expose voltage less than 70V.
- 5) Positive bus bars of the solar cells are coated by RTV.

The completed spacecraft before the launch is snapshot in Fig.5.

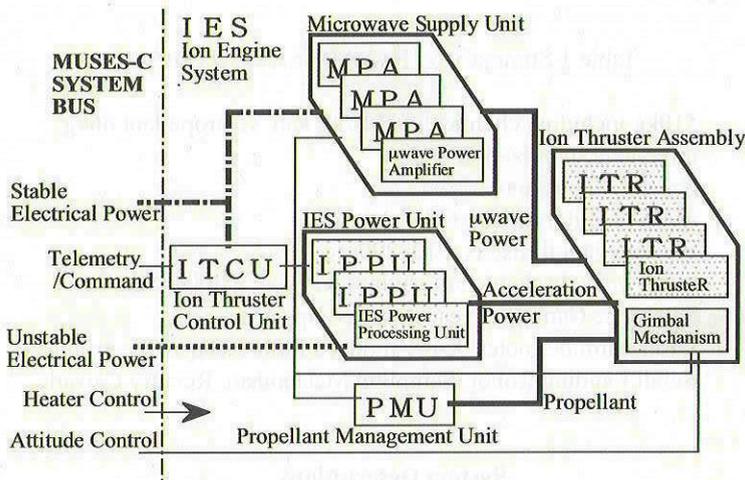


Fig.3 System diagram of IES.

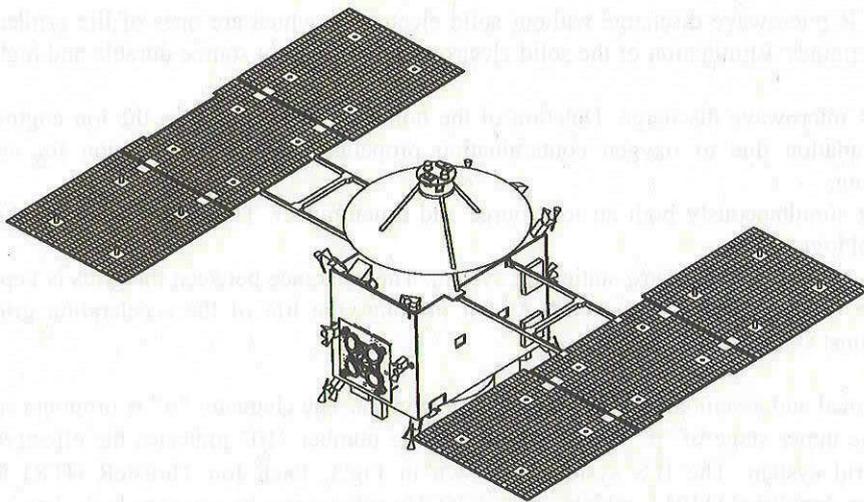


Fig.4 Configuration of HAYABUSA spacecraft.

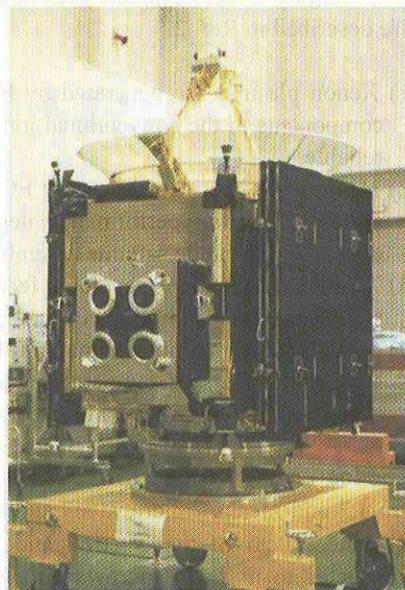


Fig.5 HAYABUSA spacecraft.

Developments

The real time endurance test is a unique way to verify durability of the electric propulsion up to now. In the development scheme on $\mu 10$ two runs of the endurance tests were performed. The first test using the engineering model (EM) was initiated on August 1997 and ended on July 1999 achieving 18,000hour accumulated operational time. The second for the prototype model (PM) starting from April 2000 was terminated on January 2003 marking 20,000hour operational time. The vacuum chamber of 2m diameter and 5m long is evacuated by four cryogenic pumps in 800mm diameter below $2E-4$ Pa gas pressure during the thruster operation. The internal surface is covered with a titanium shroud panel and a beam damper refrigerated below $-40degC$. The operation and data acquisition of the thruster and the vacuum facility are fully automated by a UNIX workstation. The chronology of the second endurance test using PM is shown in Fig.6. Figure 7 shows the thruster firings of EM and PM in the endurance test chamber.

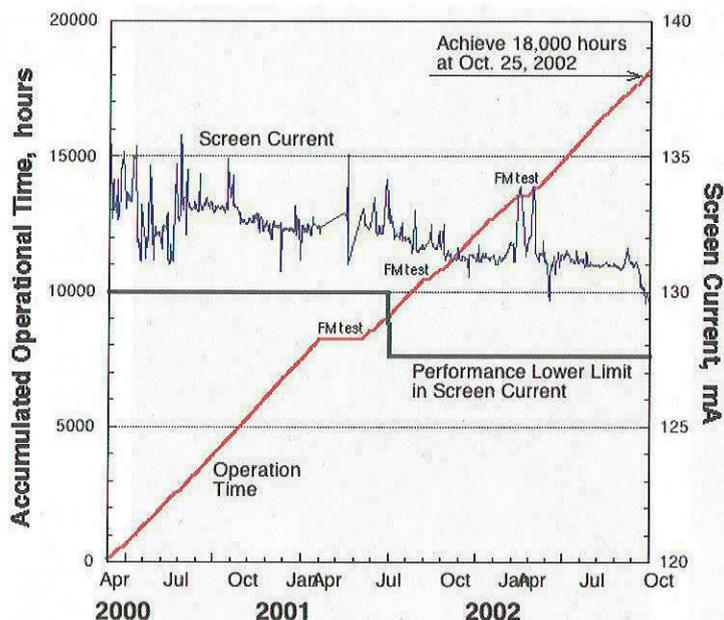
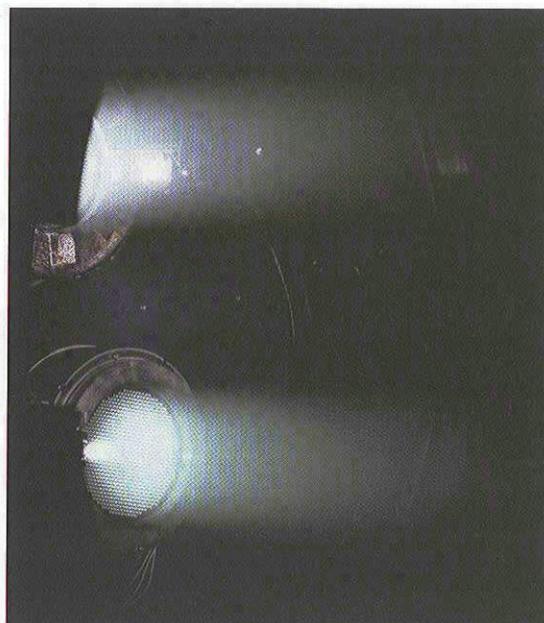


Fig.6 PM endurance test.

Fig.7 Plasma beams from $\mu 10$ s.

The interference between the plasma beam and the solar array is assessed by the ground experiment and the numerical simulation. The former is described in Ref.3 and the latter in Ref.4. The main finding is that the spacecraft may be charged down at the voltage equivalent to the cell operation voltage even if the plasma beam and the solar array interfere with each other. However that kind of charging is not so serious for the deep space maneuvering. The dumping of the communication microwave across the plasma beam was tested with the result that no effect was found⁵⁾. In order to assess the susceptibility of the spacecraft system against the IES, a $\mu 10$ thruster was operated in the glass chamber near the flight model in the clean room as seen in Fig.8.



Fig.8 Operation of the ion engine in the vicinity of the spacecraft on the susceptibility test.

The spacecraft was assembled in the clean room of ISAS and was devoted to the thermal vacuum test and the beam extraction test at the end of January 2003. The thermal design of the spacecraft system was verified in the thermal vacuum test, where three of the ion engines were ignited simultaneously without the beam extraction to dissipate heat load over 45hour. Figures 9 and 10 show the ion engines on spacecraft in the thermal vacuum chamber and the plasma ignitions. After that the test configuration was changed to the beam extraction test, which purpose is to confirm the workmanship to assemble IES. The spacecraft itself and the internal surface of the thermal vacuum chamber were taken care to protect sputtering and contamination originated from the plasma beam. The beam extraction was demonstrated one thruster by one. Four thrusters successfully exhausted the plasma beam at their full voltage 1.5kV in 2hour total. In the both tests the ITRs were supplied propellant from PMU in the same manner as the flight configuration. On March the spacecraft was transported to the launch site Kagoshima Space Center. The xenon propellant 66kg was charged into the onboard main tank by means of the liquefaction method newly developed⁶⁾. References 7 and 8 explain the R&D effort on $\mu 10$.



Fig.9 Ion engines on the spacecraft in the thermal vacuum chamber.



Fig.10 Plasma ignition during 45hour for thermal equilibrium on the thermal vacuum test.

Space Operations

The M-V rocket No.5 input MUSES-C spacecraft into the planned orbit in the deep space on May 9 2003. The telemetry data from Goldstone tracking center showed to deploy SCP. In the evening of the same day Kagoshima space center caught the signal from HAYABUSA, which is the new name of MUSES-C. The precise orbit determination revealed to need a trim maneuver about 30m/s, which is not little for the chemical propulsion. But the maneuver was cancelled because IES has an enough capability to recover it in the cruising. In the first visible opportunity the launch lock mechanism of IPM was released successfully. IES was exposed to vacuum under keeping around 0degC. At the end of May IES was turned on one by one, in which each ITR ignited plasmas and accelerated it around one hour. The first step was cleared up. In the next step two ITR parallel operation was executed, but lots of large discharges around

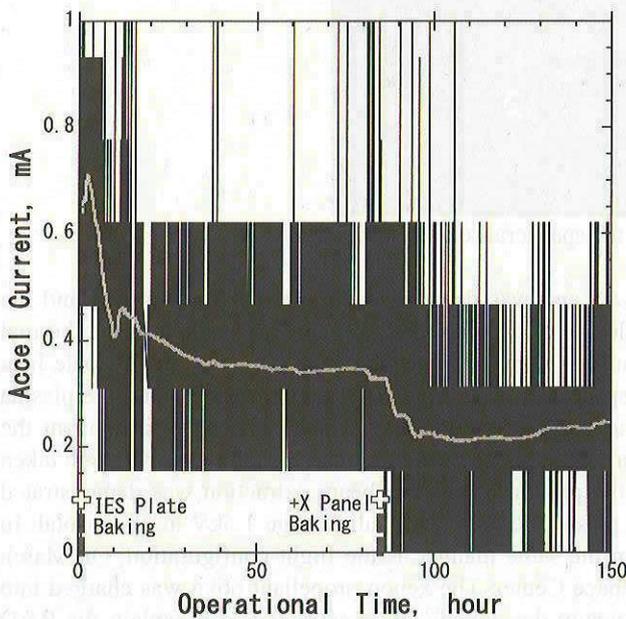


Fig.11 Time profile of leak current to the accel grid.

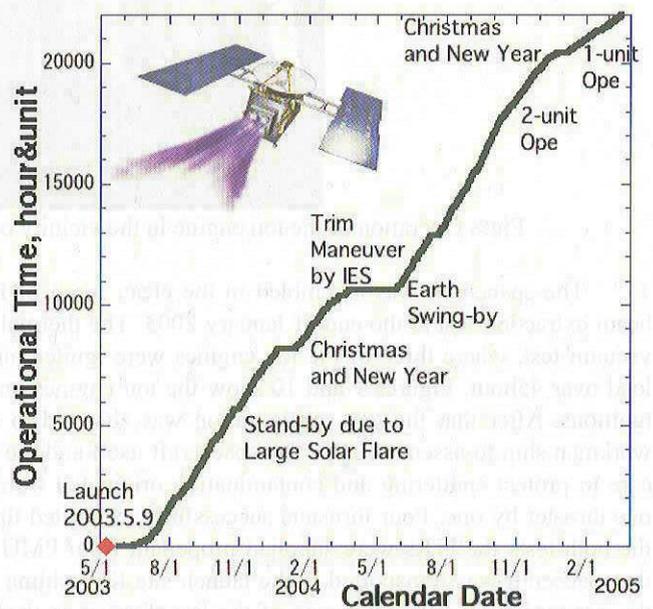


Fig.12 Chronology of the total operational time.

ITRs were caused by outgas due to temperature rise. Then IES was devoted to baking around 50degC during two days by replacement heaters and solar radiation. It enabled to operate several ITRs simultaneously. The onboard software and operational parameters were tuned for standalone firing of IES without supervision from Earth. The 24hour operations of single ITR and then of double ITRs were achieved step by step. But triple ITR operations were interrupted by large discharges several times so that the baking including IES and the +X panel was tried again. Figure 11 shows the time profile of the leak current to the acceleration grid of a specific ITR, which tends to increase under low vacuum condition. The time in the horizontal axis means the accumulated operational time of the ITR. The profile is scattered because of current resolution 0.15mA, so that numerically smoothed curve is also indicated in Fig.11. The baking was executed at the elapsed time 0.8hour and 85hour. Just after two occasions of the baking the accel current decreased dramatically. In the ground test at $1.5E-4$ Pa surrounding the accel current was 0.46mA, which is almost the same to the flight. And then the continuous acceleration by three ITRs was achieved and planned delta-V maneuver by IES was executed from July. The $\mu 10$ in space is evaluated the thrust 8mN, the thrust factor 93%, the specific impulse 3,200sec, the thrust power ratio 23mN/kW, the ion production cost 240eV and the propellant utilization efficiency 87% based on the acceleration measurement using Doppler shift of the communication microwave.

Figure 12 represents the chronology of the total accumulated operational time, which is defined time duration under a nominal operational point multiplied by number of ITRs. The continuous acceleration was started from July 2003 after the test operation on June. The largest solar flare on the record broke out at the end of October 2003, when HAYABUSA kept a safe mode without the IES firing. In three weeks at the season of the year-end and the new-year the IES was suspended depending on the plan of the orbit maneuver. Almost all the delta-V by IES in the face of the Earth swing-by has been completed by the end of February 2004. And the IES firing was executed in order to adjust the orbit on March 2004 so that the operational time increased at a slow rate. At the end of March 2004 the total number of the operational time exceeded 10,000hour. The spacecraft was accelerated about 700m/s consuming about 12kg propellant. The orbit of HAYABUSA in the rotational coordinate system is seen in Fig.13, where Earth and Sun are fixed on the horizontal axis and shows the relative location of HAYABUSA. Though just after the launch the extrapolated orbit never reached Earth, it changed gradually in accompany with the IES maneuver. HAYABUSA arrived at Earth on May 19, 2004 and bended the relative velocity vector toward the asteroid ITOKAWA by means of the Earth swing-by. On the transfer orbit to the asteroid IES has continued to accelerated HAYABUSA. At the end of March 2005 the total number of operational time reached 22,000hour with 20kg xenon propellant consumption generating 1,300m/s delta-V. The $\mu 10$ s and IES on HAYABUSA achieved the space flight heritage of the top in Japan and of the world standard. The north and south station keeping of the geo-stationary equatorial satellites, which is the most popular application of electric propulsions, needs about 50m/s delta-V per year. HAYABUSA passed through the aphelion 1.7 astronomical unit from Sun on February 18, 2005 so that $\mu 10$ s arrived at the furthest space as the electric propulsion. It locates 345,000,000km (=2.3 astronomical units) apart from Earth and 1,000,000km from ITOKAWA, which is 2.5 times as far as the distance between Earth and Moon on April 2005.

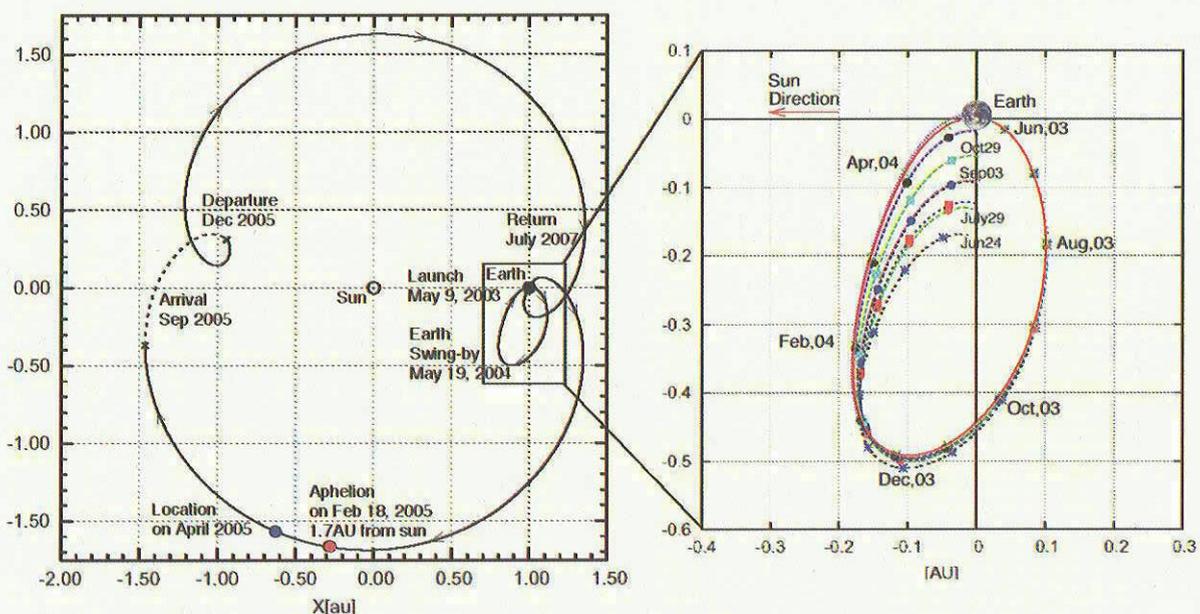


Fig.13 Orbit of HAYABUSA on rotational coordinate system.

Summary

The asteroid explorer HAYABUSA was launched into the deep space by M-V rocket No.5 on May 9 2003 from Kagoshima Space Center. It will execute a round trip space mission between Earth and the asteroid ITOKAWA propelled by four microwave discharge ion engines $\mu 10$ s, of which the space flight was realized based on the R&D during 15 years. The initial operation brought us a lot of space experience and flight data, which are never got on the ground. The $\mu 10$ in space is evaluated the thrust 8mN, the thrust factor 93%, the specific impulse 3,200sec, the thrust power ratio 23mN/kW, the ion production cost 240eV and the propellant utilization efficiency 87%. On April 2005 the total number of operational time reached 22,000hour, which is enough for the space flight heritage. It is very interesting that the ion engine types particular at US, Europe and Asia have achievements in space. Variety of the ion engines system in the world proves the healthy engineering challenge. Independent R&D effort not to imitate well-developed systems realizes the $\mu 10$ s. They are commented that a lot of works were devoted to the MUSES-C space mission in order to adapt the new system $\mu 10$ s to the existing space technology and the flight bus system and the ground support system performs appropriately to continue the acceleration of IES on HAYABUSA. The technology of $\mu 10$ s is the "foothold in deep space" and will support concretely the future space missions. HAYABUSA spacecraft has already secured the way⁷⁾ to rendezvous with the asteroid on September 2005.

Reference

- 1) J. Kawaguchi, et. al, "MUSES-C: Technologies Development and Nereus Exploration Scenario", Space Technology, 17(1997), pp. 265-279.
- 2) H. Kuninaka and S. Satori, "Development and Demonstration of a Cathode-less Electron Cyclotron Resonance Ion Thruster", Journal of Propulsion and Power, 14(1998), pp.1022-1026.
- 3) H. Kuninaka and Pedro Molina-Morales, "Spacecraft Charging due to Lack of Neutralization on Ion Thrusters", Acta Astronautica, Vol.55, No.1, pp. 27-38, 2004.
- 4) M. Cho, K. Shiraishi, M. Hikita and H. Kuninaka, "Plasma Interaction of High Voltage Solar Arrays for MUSES-C", Journal of the Japan Society of Aeronautical and Space Science, Vol.49, No.571, 2001, written in Japanese.
- 5) N. Onodera, "Interaction between Plasma Plume of Electric Propulsion and Spacecraft Communication", IEPC 99-228, 26th International Electric Propulsion Conference, Kitakyusyu, 1999.
- 6) Y. Shimizu, H. Kuninaka, K. Nishiyama and K. Toki, "Xenon Propellant for Ion Engines of the Space-Explorer HAYABUSA", Application of Rare Gas Xenon to Science and Technology, XeSAT 2005, Waseda, March 2005.
- 7) K. Toki, H. Kuninaka, K. Nishiyama and Y. Shimizu, "Flight Readiness of the Microwave Ion Engine System for MUSES-C Mission", IEPC 03-0098, International Electric Propulsion Conference, Toulouse, March 2003.
- 8) H. Kuninaka, I. Funaki, K. Nishiyama, Y. Shimizu and K. Toki, "Result of 18,000-Hour Endurance Test on Microwave Discharge Ion Thruster Engineering Model", AIAA 2000-3276, Joint Propulsion Conference, July 2000.