# Initial Checkout after Launch of Hayabusa2 Ion Engine System

**IEPC-2015-334 /ISTS-2015-b-334**

*Presented at Joint Conference of 30th International Symposium on Space Technology and Science 34th International Electric Propulsion Conference and 6th Nano-satellite Symposium, Hyogo-Kobe, Japan July 4 – 10, 2015*

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**Abstract: The ion engine system (IES) for Hayabusa2 is based on that developed for Hayabusa with modifications necessary to improve durability, to slightly increase thrust, and to reflect on lessons learned from Hayabusa mission. Hayabusa2 will rendezvous with a near-earth asteroid 1999 JU3 and will take samples from its surfaces. More scientific instruments than Hayabusa including an impactor to make a crater and landers are on board thanks to the thrust enhancement of the IES. After 2.5 years of short development time, the spacecraft was launched from Tanegashima Space Center in Kagoshima Prefecture on-board an H-IIA F26 rocket on December 3, 2014. Initial checkout of the IES in orbit was quickly and efficiently completed by the end of February 2015 with full use of hayabusa1 heritages. All thrusters are generated thrust in space. Multi thruster and automatic operation functions were confirmed. HAYABUSA2 is moving to the cruising phase.**

## I. Introduction

Japan's first asteroid explorer "Hayabusa" came back to the earth on June 13, 2010. Hayabusa completed more than 7-year space mission, and finally Hayabusa could return the capsule to the earth, although Hayabusa had a lot of troubles and difficulties. [1] After careful inspections of about 1,500 grains found in the sample container, it turned out that we actually obtained rocky particles from the surface of Asteroid Itokawa. In 2011, Japan Aerospace Exploration Agency (JAXA) started the second project "Hayabusa2" for the asteroid sample return mission to another Asteroid 1999 JU3 which is one of different type (C-type) of asteroids from Itokawa (S-type). The Hayabusa2 flight system was designed based on the Hayabusa in order to make the development time as short as possible for the best launch window at the end of 2014. The critical design review (CDR) of the Hayabusa2 system was conducted in the beginning of FY2012, then flight model manufacturing and testing started.[1] depicts instruments on the spacecraft and component layout of the IES is completely the same as Hayabusa. Hayabusa2 was launched on December 3, 2014. It will arrive at the asteroid mid-2018 and return to Earth at the end of 2020[2]. In this paper, we reports the results of initial checkout phase of Hayabusa2 ion engine system.

## I. Subsystem and Component Descriptions

Figure 1 shows the system architecture of IES. Most of the components of Hayabusa2 are more massive than those of Hayabusa mainly due to replacement of discontinued foreign parts and partly due to the increase in maximum thrust level.

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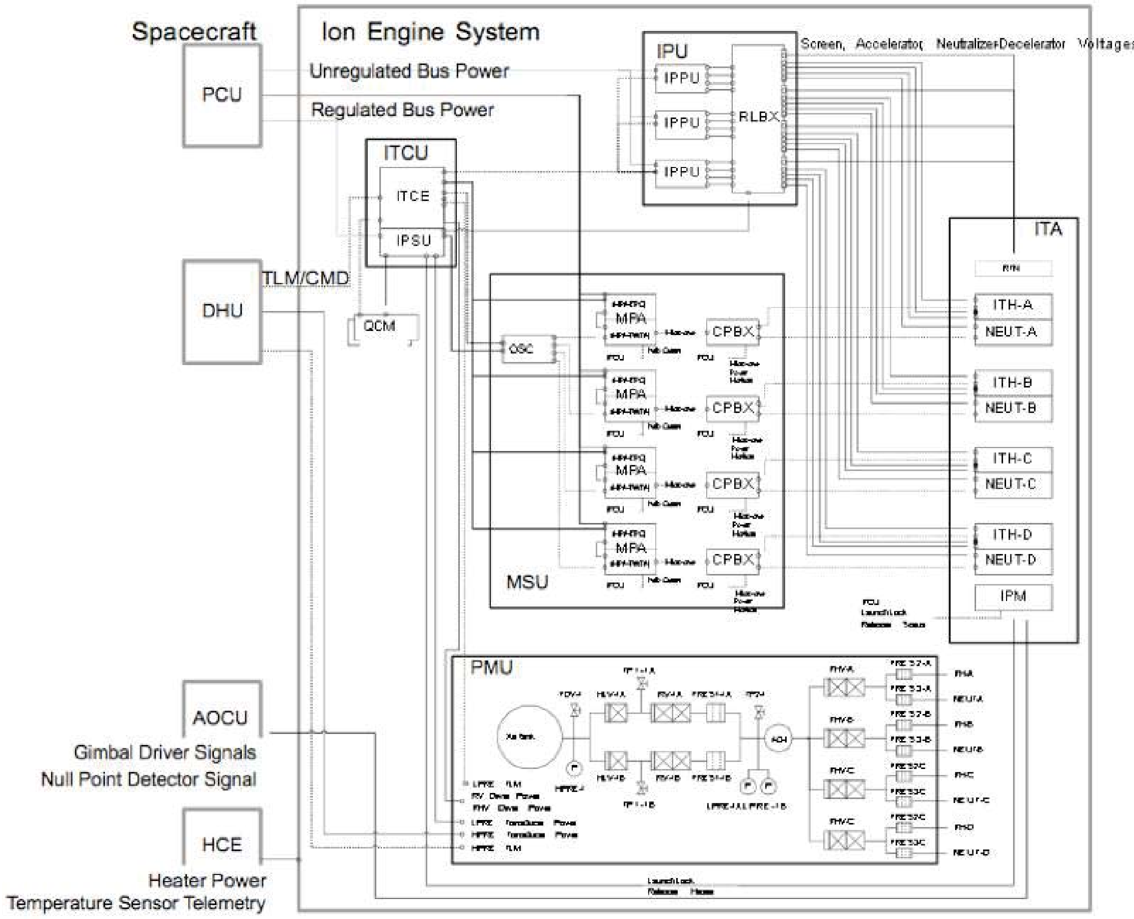


**Figure 1. Architecture of the Ion engine subsystem.**

## A. Ion Thruster Head (ITH)

The original well-tuned μ10 ion thruster can generate 8 mN at a specific impulse (including neutralizer flow) of 3000 s with consuming 32 W of microwaves and 0.23 mg/s of xenon flow. Our work indicated that design change of gas injector layout had the large impact on thrust enhancement[2]. The highest thrust of 10 mN is generated when the xenon flow is divided at the ratio of 1:2 between the original injector deep in the waveguide and newly added injectors between magnets, respectively, when the total flow rate is 0.34 mg/s. Mass utilization efficiencies are decreased from the original but can be improved by using a small hole accelerator grid whose aperture diameter was decreased from 1.8 to 1.5 mm. Decrease of screen grid thickness from the original value of 0.95 mm to 0.8 mm also helped the mass utilization recovery. The combination of above mentioned modifications (gas distributors, small-hole accelerator grid and thinner screen grid) achieved the maximum thrust enhancement shown in Ref. 3 with acceptable decrease in specific impulse. This thrust enhancement was indispensable for Hayabusa2 mission because of the increase of spacecraft mass. The former laboratory experiments were conducted using two mass flow controllers to investigate the flow rate distribution ratio effects on thruster performance. To realize the optimum propellant feed balance in flight system as simply as possible, a gas flow divider was newly inserted between the gas isolator and the original gas port near the waveguide-end without changing design of the PMU. The divider consists of a small orifice connected to the waveguide and a branched gas pipe connected to the inter-magnet gas ports. Several orifice plates with different aperture diameters were manufactured and the ion source performance was tested whose result will be described in the later section. Because addition of the gas divider leaded to the upstream pressure rise which affected the gas isolator performance, the gas isolator design was also modified so that the number of stacked internal metal meshes and ceramics was increased. It was verified that the new gas isolator had a sufficiently high withstanding voltage at xenon flow rates lower than 0.5 mg/s. Material of the support ring of ion optics was changed from a discontinued Al-Si alloy to a metal matrix composites of SiC and Al. By this change, thermal expansion coefficient of the ring material decreased from $16\times10^{-6}$ $K^{-1}$ to $7\times10^{-6}$ $K^{-1}$ that contributes to decrease thermal grid-gap change and to relax thermal stress of the carbon-carbon composite grids. A microwave feedthrough made in USA (an antenna with a TNC coaxial connector) was replaced with a newly developed domestic one for better availability.

### B. Neutralizer (NEUT)

Lifetimes of Hayabusa neutralizers were ranges between 9579 and 14830 hours which are both much shorter than the ground test's achievement of 20000 hours by a prototype model (voluntarily stopped). Number of on/off cycles does not seem to be dominant because the longest life was achieved by the one most frequently switched on and off. Most remarkable difference between in-space and on-ground would be the temperature range. Hayabusa ion engines experienced extremely low temperatures several times by accidents and it might have somehow shorten the life time of neutralizers. We have analyzed the stored prototype neutralizer by disassembling it and found many magnetized metal flakes stuck on 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 microwave launcher by ion sputtering and sputtered metal deposition, and may increase plasma loss. All the flight and ground test data have a common feature that there is a synchronized increase of neutralizer coupling voltage and neutralizer backward power. This may be caused by above mentioned metal coating of dielectric surfaces of a microwave launcher. A start point of the synchronized increases would correspond to an occurrence of delamination and sticking of a piece of flake. Improvement of the neutralizer reliability was the most important work of the Hayabusa2 IES, and design of the magnetic field was improved by increasing the number of SmCo magnets and changing the magnetic yoke shape. The new neutralizers have 20% stronger magnetic field for better plasma confinement. About 500 hours of diode mode operations of a laboratory model of neutralizer with different magnetic field strengths indicated us that the stronger magnetic field decreased the erosion or deposition rates of metallic parts used in the discharge chamber.

Another improvement is about neutralizer bracket's electrical insulation. Leakage currents were observed through an electric short circuit at the neutralizer's insulator surfaces contaminated by deposition of sputtered metallic materials generated during crossed operation of the neutralizer A and the ion source B in the final approach to Earth by Hayabusa in which the spacecraft was highly negatively charged relative to the ion engine plume plasma. We improved shadow shielding of the dielectric insulator between the neutralizer bracket and the IES plate. A microwave feedthrough made in USA (an antenna with an SMA coaxial connector) was replaced with a newly developed domestic one for better availability.

During the Hayabusa2 IES development, the neutralizer was the only component that required production of engineering models to conduct qualification tests including long term endurance test in diode mode operation.

### C. IES Power Processing Unit (IPPU)

The IPPU converts the 60 – 120 V input power from spacecraft solar arrays (unregulated power bus) into the currents and voltages needed by thruster operation. The IPPU is almost the same as the Hayabusa IPPU with the following exceptions:

1. One of three Hayabusa IPPUs showed unstable oscillatory current limiting behavior at high temperatures in the first year in space. This has already been fixed in the qualification model development for the "generalized" IES.
2. Higher beam current operation up to 180 mA is possible and qualified.
3. Options for constant current operation target of neutralizer current were changed. Original IPPUs had three options: ×1 of the corresponding screen current (nominal operation), ×1.5 (two neutralizers for three ion sources on one neutralizer failure) and ×2 (one neutralizer for two ion sources on one neutralizer failure). We will have different three options for Hayabusa2: NOMINAL (screen current + 3 mA), HIGH (screen current + 10 mA) and OFF (the neutralizer will be fixed to the spacecraft potential without negative bias voltage).
4. The high voltage output connector was replaced with another type due to availability. The mass increase of the IPPU was partly caused by this.

### D. Propellant Management Unit (PMU)

The PMU feeds xenon gas to ion sources and neutralizers. Flow rates for higher thrust operation are a little larger than in the nominal thrust operation of Hayabusa IES. However, no design change is required about this. Only the design change will be replacement of discontinued solenoid valves for regulation valves (RV) and thruster valves (ITHV) to different space-qualified models which were also used in DS1 and Dawn. The low pressure transducer (LPRE) of Hayabusa showed noisy output when the IPPUs were switched on, which degraded flow rate (or thrust level) control accuracy. These transducers were replaced with different models which were equipped with internal current limiters according to JAXA's standard, and the electromagnetic interference as mentioned before was not observed any longer.

In the Hayabusa2 project sinusoidal vibration testing was required which were not applied in the format Hayabusa project because of the change of launch vehicle from the Japan’s M-V solid rocket to the H-IIA liquid propellant rocket. To cope with this change four xenon tank brackets were slightly modified but mass change due to this was negligible. Mass gain of the PMU is mainly due to the change of solenoid valves.

### E. Microwave Supply Unit (MSU)

The MSU is for ECR plasma generation and includes traveling wave tube amplifiers (TWTA) and passive microwave components such as flexible cables, semi-rigid cables, coupler boxes (CPBX) for microwave distribution between an ion source and a neutralizer, and DC blocks. Several parts such as microwave receptacles and detectors used in the CPBX were replaced with alternatives due to availability, but there was essentially no design change. TWTAs were replaced by successors of Hayabusa models of the same manufacturer. The preheat time for the TWT cathode was changed from 180 s to 200 s that was acceptable because this time interval was a variable in the control logic of the IES thruster control unit (ITCU). The vender of flexible coaxial cables were switched and unified for entire spacecraft including IES and communication subsystem in accordance with JAXA's space electronic parts selection standard. Figure 1 shows pictures of new low-loss DC blocks jointly developed by JAXA and NEC. The new one has an insertion loss of 0.2 dB which is 0.3 dB smaller than the Hayabusa's made in USA. This slightly improve electrical efficiency of the thruster and reduce heat dissipation. The oscillator was replaced by the one newly developed by NEC due to non-availability of the original parts. The new oscillator has a built-in power divider for four outputs.

### F. IES Thruster Control Unit (ITCU)

The ITCU is the controller of the IES. Details of its function were described in a reference[1]. In the generalization and commercialization program, a qualification model of enhanced ITCU was developed. However, most of the enhancement will be removed and lightweight for Hayabusa2. Only the new capability will be the generation of user packet telemetry that contains accumulated impulse (time integration of the thrust calculated from voltages and currents). This compact telemetry data will be quickly downloaded and used for radiometric orbit determination without downloading much larger spacecraft housekeeping data which also contains the same telemetry data. This improvement is based on lessons learned from seven-year Hayabusa mission.

Another improvement is about the xenon flow rate control. The original ITCU for Hayabusa controlled the accumulator tank pressure by open and close the regulation valve (RV) referring the low pressure transducers (LPRE) output, which we called “delta-P control”. Pressure (or flow rate) ripple was relatively large because of the lack of pressure telemetry’s resolution and slow response due to the ITCU’s internal analog filter whose time constant was about 1 s. As a result, minimum valve opening duration was limited to 2 s and pressure ripple was twice as large as we expected. Now the new ITCU has another control method “delta-T control” in which valve opening duration can be set as a multiplier of the minimum value 62.5 ms. At the initial main tank pressure of 7 MPa after launch, the valve opening duration of 500 ms is the nominal setting for cruise operation.

### G. IES Pointing Mechanism (IPM)

The IPM consists of two-axes gimbals and launch lock mechanisms. Electrical harnesses for wax motors and hall sensors were improved for better stress relief and electromagnetic compatibility. In order to comply with the sinusoidal vibration specifications, some parts were slightly improved in its mechanical strength by increasing thickness of the supporting legs and number of fasteners and by adding shear pins. Mass increase by these improvement was only 32 grams (+1% of the original).

### H. Ion Thruster Assembly (ITA)

Hayabusa2’s ion thruster cant angles were slightly changed from those of Hayabusa because relative position to the spacecraft center of mass was different from the Hayabusa. The thickness of the IES plate was also changed so that the center of mass of the plate and four thrusters is close to the gimbal’s pivot point because the thruster’s mass distribution was different due to design changes for the thrust enhancement or the material change as already mentioned. Two quarts crystal microbalances (QCM) made in Japan were installed on the outer surface of the ion engine plate as depicted in Fig. 3a in order to monitor surface erosion or deposition which might be caused by ion thruster operation. Those locations are similar to but not exactly the same as those of solar cell contamination monitors of Hayabusa IES[4]. The QCMs were the same lot as the one that has already been space qualified in the JAXA's Small Demonstration Satellite-4 (SDS-4) satellite mission and has been operating for three years in a low earth orbit[4].

## II. Initial checkouts of the Ion engine system

Table 1 shows the sequence of events of HAYABUSA2 checkout phase. HAYABUSA2 was launched by H-IIA F26 rocket at 4:22:43 (UTC) from Tanegashima Space Center. One hour, 47 minutes and 21 seconds after liftoff, the separation of the Hayabusa2 to earth-escape trajectory was confirmed. After the separation, hayabusa2 moved to the critical operation phase. In the critical phase, the launch lock mechanism of IPM was released successfully.

**Table 1. Summary of HAYABUSA2 SOE (Yellow hatchings are corresponding to the IES events)**

| Date | | Confirmed point list |
|---|---|---|
| 2014 | Dec. 3 | Launch by H-IIA F26 at 4:22:43 (UTC) |
| | | IPM gimbal launch lock release |
| | Dec. 7,8 | X-band mid-gain antenna beam pattern measurement, Data acquisition at a radio station, X-band communication equipment function verification. |
| | Dec. 9 | Power system (batteries) functional confirmation |
| | Dec. 10 | Near Infrared Spectrometer (NIRS3) inspection |
| | Dec. 11 | Thermal Infrared Imager (TIR) / Deployable Camera (DCAM3)/ Optical Navigation Camera (ONC) inspection |
| | Dec. 12-15 | Attitude and orbit control system (each unit) function confirmation |
| | Dec. 16 | Small rover (MINERVA-II) / Small lander (MASOT) inspection |
| | Dec. 17 | Re-entry capsule / Small Carry-on Impactor (SCI) inspection |
| | Dec. 18 | X-band high-gain antenna (XHGA) five-spot pointing test, IES pre-operation arrangement |
| | Dec. 19-20, 22 | IES baking (24-hour thruster warm-up to 54-55 degrees Celsius by heaters and solar irradiations without plasma ignition, 2-hour plasma ignition for two combinations of thrusters, AB and CD, respectively) |
| | Dec. 23-26 | Ion engine test operation (ignition, aging with low screen voltage, high screen voltage and low/high screen voltage switching operation for power management)<br>*Unit by unit operation (Ion thruster A (ITR-A) on the 23rd, ITR-B on the 24th, ITR-C on the 25th, ITR-D on the 26th) |
| | Dec. 27-Jan. 4 | Precise orbit determination, DDOR (Delta Differential One-way Range) operation |
| 2015 | Jan. 5-7 | Ka-band communication equipment; radio station communication data acquisition; antenna pattern measurement |
| | Jan. 9-10 | DOR by each Ka-band DSN station, ranging test |
| | Jan. 12-15 | Two ion thrusters combination test <AC on the 12th> <CD on the 13th> <AD on the 14th> <AC on the 15th> |
| | Jan. 16 | Three ion thrusters combination test for 3.5 hours <ACD> |
| | Jan. 19-20 | Two ion engines combination test for 24-hour continuous autonomous operation <AD> |
| | Jan. 20- Mar.2 | Functional confirmation to move to the cruising phase (regular operation) including combined operation of multiple instruments. Solar light pressure impact evaluation; solar tracking behavior data acquisition; combined operational function confirmation of solar light pressure, attitude and orbit control system (such as a reaction control wheel) and ion engines. |

### A. Ion Engine System baking

The importance of Ion thruster system baking for an stable ion beam acceleration was confirmed by that system of HAYABUSA explorer[5]. In the situation of the HAYABUSA explorer is as follows; IES was exposed to vacuum in space 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 smoothly. In the next step two ITR parallel operation was executed, but lots of large discharges around 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 program and driving parameters were tuned for standalone firing of IES without monitoring 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.

From these heritages, we planned to the two stage IES baking. At first, +X panel, the microwave components were equipped on the backside of that panel, and IPPU were devoted to baking during 26hour with 50degC and 0degC, respectively. Figure 2 shows the temperature histories of IES components. +X panel baking was carried out by the spacecraft heaters and sun light with 45 degree pitching attitude control. Thruster heads, +X panel and IPPU were kept above the target temperatures. Next, we carried out the baking of grid and inside wall of discharge chamber of thruster heads by plasma ignition without ion beam acceleration. Grid baking was carried out during 2hour with two thrusters. The durability of plasma sustain without ion beam acceleration was confirmed at system thermal vacuum test. These baking sequences get good results in quick checkout of IES.

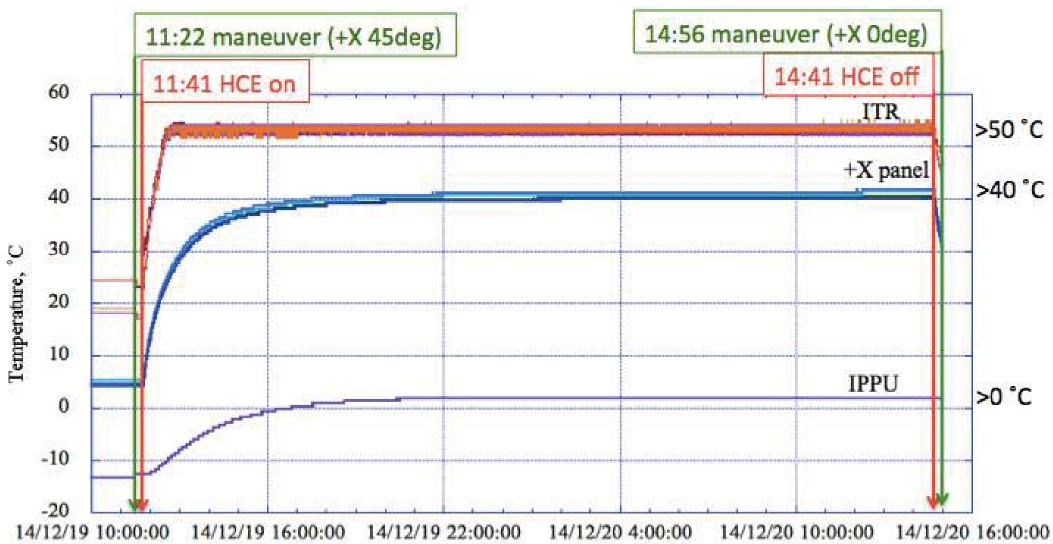


**Figure 2. Historical trends of IES components temperatures. Time is described at UTC.**

### B. Propellant Management Unit (PMU) initialize

During the baking operation extra xenon gas was dumped from the accumulator tank and the plenum pressure was lowered to about 0.08 MPa so that the PMU was able to deliver suitable xenon mass flow to the thrusters for the subsequent checkout steps. The ITCU was slightly improved in xenon flow rate control logic. The original ITCU for Hayabusa controlled the plenum tank pressure by open and close the regulation valve (RV) referring the low pressure transducers (LPRE) output, which we called "delta-P control". Pressure (or flow rate) ripple was relatively large because of the lack of pressure telemetry's resolution and slow response due to the ITCU's internal analog filter whose time constant was about 1 s. As a result, minimum valve opening duration was limited to 2 s and pressure ripple was twice as large as we expected. Now the new ITCU has another control method "delta-T control" in which valve opening duration can be set as a multiplier of the minimum value 62.5 ms. Figure 3 shows the result of valve opening time adjustment demonstration conducted on December 19, 2014 with three ion thruster valves A-C open. At the initial main xenon tank pressure of 7 MPa after launch, the valve opening duration of 500 ms is selected as the nominal setting for cruise operation. Figure 7 indicates the Hayabusa2's successful compression of plenum pressure (i.e. xenon flow rate) ripple compared with that of Hayabusa.

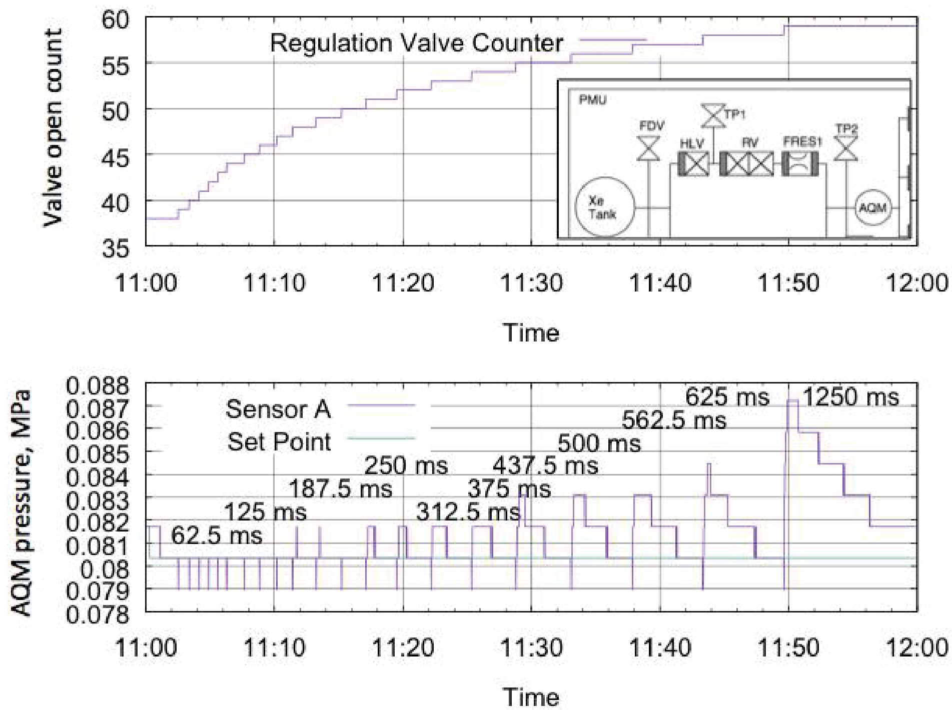


**Figure 3. Adjustment of opening duration (shown in milliseconds) of the regulation valve to obtain an acceptable IES input power ripple and reasonable plenum charge period to meet the valve specification on open/close cycles.**

### C. Ion thruster initial checkout

Unit by unit operation including the thruster aging (Ion thruster A (ITR-A) on the 23rd, ITR-B on the 24th, ITR-C on the 25th, ITR-D on the 26th) at larger xenon flow rates than the optimum flow rate for the maximum beam current was carried out carefully. This is because we experienced rather smooth cold-start of the thruster beam acceleration with less frequent high voltage breakdowns (HVBDs) in these flow rate range in ground tests. All the thrusters exhibited very frequent high voltage breakdowns during the first 30 minutes. If the ITCU detects HVBD, it turns off high voltage power supplies in the IPPU for 30 seconds and restarts them. An acceptable frequency of HVBDs can be set to the ITCU by commands. Figure 4 shows the result of ITR-A initial operation. Horizontal axis is corresponding to the universal time. This graph including two pass operation. One is the domestic visible pass from 10:02 to 16:49 (UTC) at Dec.23rd in 2014 using USUDA Deep Space Center, another is the NASA DSN visible pass from 19:50 to 25:55 at Dec.23rd in 2014 using MADRID Deep Space Network. Vertical axes are corresponding to the xenon flow rate, screen current, screen voltage, accel current and neutralizer contact voltage. X After the ion beam acceleration with lower screen voltage (~1460V) due to thruster aging, although a frequent HVBD and higher accel current were observed, thruster operation was getting stably at the first pass. Then, we continue to the thruster aging with higher screen voltage (~1520V). Neutralizer contact voltage is around 20V at the end of the aging operation, which is the enough lower for performance degradation due to sputtering.

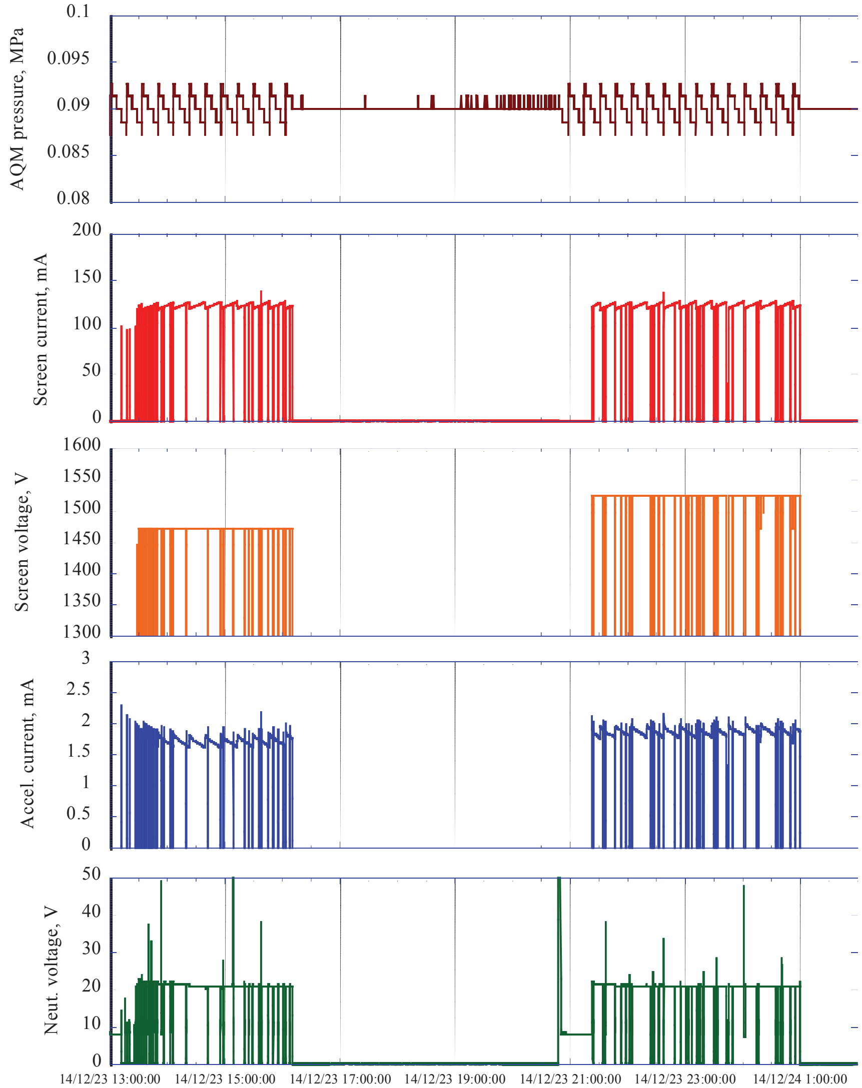


**Figure 4. 1st beam acceleration of ITR-A.**

### D. Multi-thruster operation and automatic operation checkout

Figure 5 shows the telemetries of two thruster operations. Vertical axes are corresponding to the Screen currents, IPM angles and the periods of Reaction Wheel (RW). IPM installing four thrusters is able to tilt on two axes. The active control of thrust direction are applied to cancel torque force due to misalliance between the thrust axis and the center of gravity of spacecraft, as well as to maintain the reaction wheels (RW) at a proper rotation rate. This function is called "IES unloading", which is seen in Fig. 6. After the spacecraft reoriented its attitude for IES acceleration, IPM was set at the initial position which was estimated from before operation. 10 minutes after the IES generate thrust force at 10:30, IPM inclined intentionally the thrust vector in order to reduce the rotation rate of RWs. About one hour later RWs were controlled within a proper rotation rate and IPMs got back to an even keel. Momentum stored in RW-X due to the roll torque of ITR and other reason is dumped firing the bi-propellant thrusters as the reaction control system (RCS) because it is not controlled by IES.

Figure 6 shows the result of three thruster operation. ITR-A, C and D were operate at three and half hour. Estimated averaged thrust from the telemetry of the screen voltage and ion beam current was 30.8mN. On the other hand, calculated thrust from 2-way Doppler monitor is 28.0mN, so that effective thrust efficiency is about 92%.

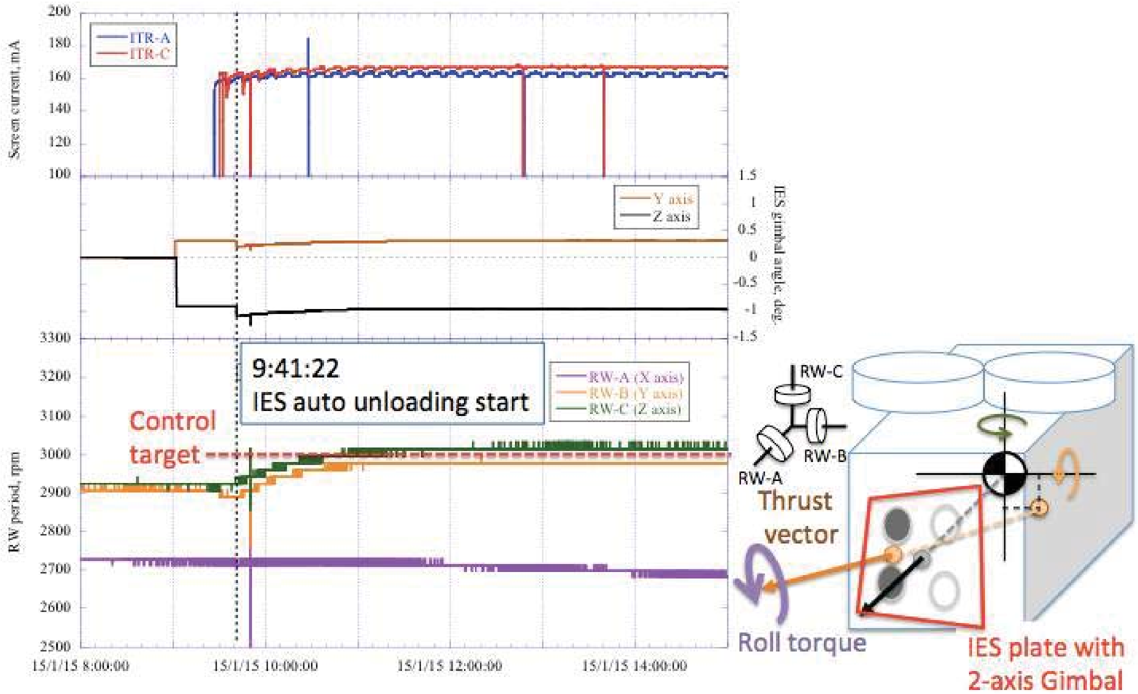


**Figure 5. IES auto unloading maneuver with two ITR operation.**

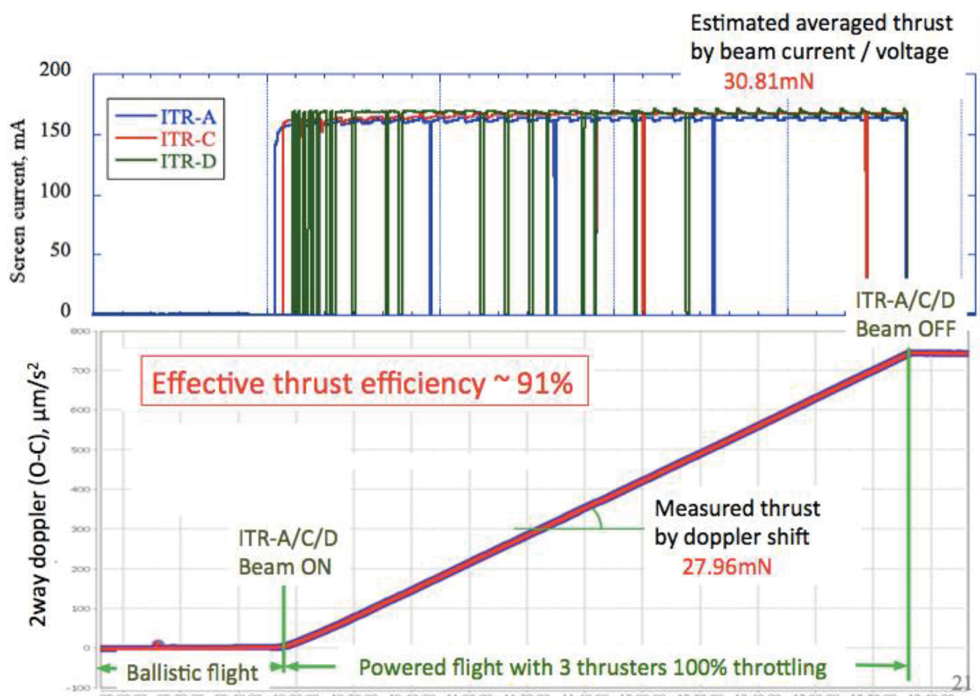


**Figure 6. Screen current of the three thrusters operation and 2-way Doppler monitor. A gradient of the Doppler monitor is corresponding to the spacecraft acceleration along the earth – spacecraft direction.**

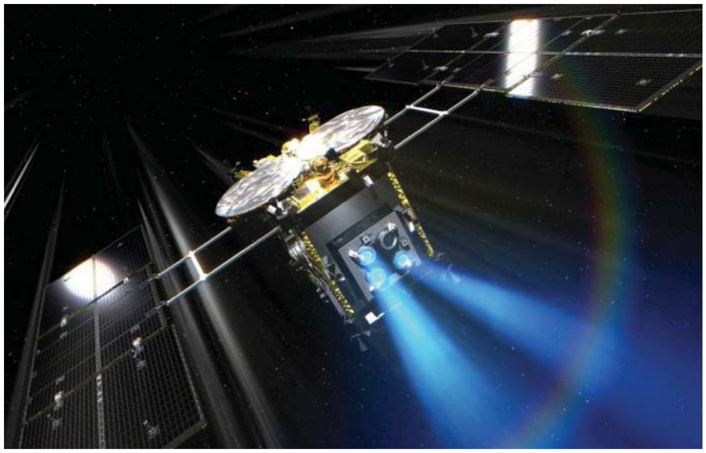

**Figure 7. Image illustration of the deep space ion thruster operation with 3 thrusters.**

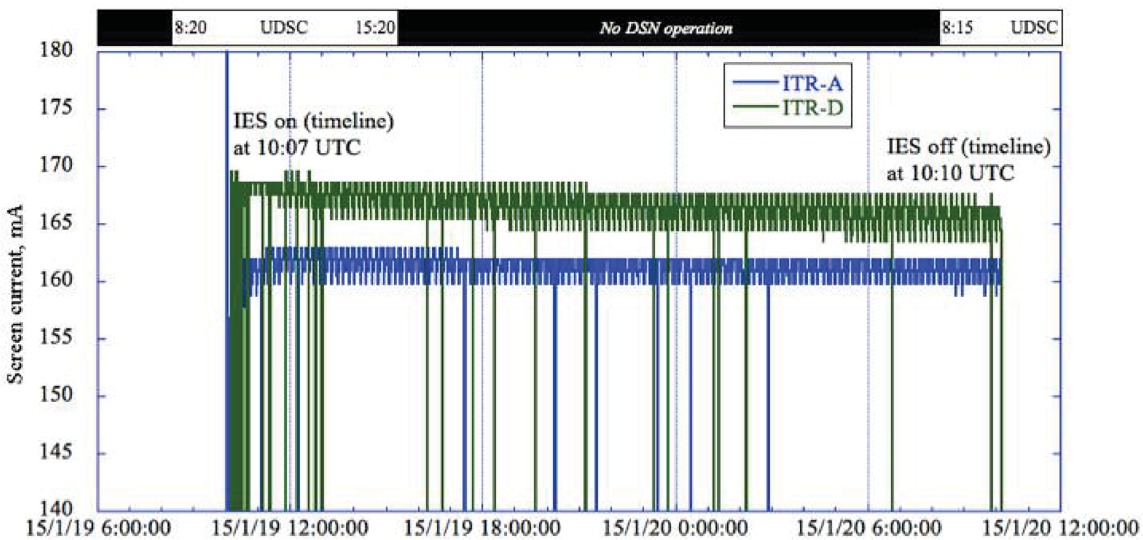


**Figure 8. Result of the 24hour automatic operation of two Ion thrusters.**

Figure 8 shows the result of automatic IES operation. ITR-A and D Thrusters were ignited and operated automatically by timeline command at 10:07 in Jan.19th in 2015. All thrusters were controlled automatically by ITCU during visible and non-visible operation. Finally, thrusters were stopped by timeline command without no-trouble.

## III. Conclusion

The flight model of Hayabusa2 ion engine system was developed within 2.5 years, which was very challenging for authors. The spacecraft was launched from Tanegashima Space Center in Kagoshima Prefecture on-board an H-IIA launch vehicle on 3 December, 2014. Initial check out of the IES in orbit was successfully completed by the end of February 2015 and powered flight for Electric Delta-V Earth Gravity Assist (EDVEGA) started in March 2015[6].

Initial checkout of the IES in orbit was quickly and efficiently completed by the end of February 2015 with full use of hayabusa1 heritages. All thrusters are generated thrust in space. Multi thruster and automatic operation functions were confirmed. HAYABUSA2 is moving to the cruising phase.

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