Backflow of Charged Particles onto a Spacecraft in Operation of Ion Thrusters under Lack of the Neutralization

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Takanobu Muranaka[1]
Chukyo University, Nagoya, 466-8666, Japan

Kento Hoshi[2], Hirotsugu Kojima[3], Hiroshi Yamakawa[4]
Kyoto University, Uji, 611-0011, Japan

and

Satoshi Hosoda[5] and Kazutaka Nishiyama[6]
Japan Aerospace Exploration Agency, Sagamihara, 229-8510, Japan

Abstract: Rapid increase of spacecraft contamination had measured by HAYABUSA spacecraft when lack of neutralization on its ion thrusters had occurred. It is considered that spacecraft charging due to that enhanced the contamination by propellant ions existed in the vicinity of the spacecraft. To understand the correlation between the increase of the contamination and the spacecraft potential, a numerical code has been developed to estimate the charging status of the spacecraft in excess emission of ion beam from the thruster. By a preliminary simulation using the code, we obtained the time history of the return current of the beam ions and the spacecraft potential in the case where only the ion beam emission was considered. Further evaluation of the increase of the contamination is ongoing.

I. Introduction

CONTAMINATION on spacecraft surface is considered to cause degradation of the power generation if it occurs on solar arrays. Therefore, it can determine the spacecraft performance for long-duration mission powered by electric propulsion. In operation of ion thrusters, one source of the contamination is outgassing neutral particles from a spacecraft surface exposed in a vacuum, and other source is propellant ions from plasma plume exhausted by ion thrusters. In normal operation of ion thruster, the thermal charge-exchange (CEX) ions produced in the plasma plume could be the dominant source for spacecraft contamination, and therefore numerical analyses have been performed to estimate the contamination due to the CEX ions.[1]

Meanwhile, in Japan, HAYABUSA spacecraft powered by ion thrusters developed by JAXA had successfully returned to the earth with a small sample of the near-earth asteroid in 2010 after seven years of space flight. During the mission, the contamination on the spacecraft had been measured by the onboard sensors.[2] The sensors showed rapid increase of the contamination at the time when the lack of neutralization had occurred on HAYABUSA's ion

[1] Associate Professor, Department of Electrical and Electronic Engineering, muranaka@sist.chukyo-u.ac.jp
[2] Doctoral Student, Research Institute of Sustainable Humanosphere (RISH), kento_hoshi@rish.kyoto-u.ac.jp
[3] Associate Professor, Research Institute of Sustainable Humanosphere (RISH), kojima@rish.kyoto-u.ac.jp
[4] Professor, Research Institute of Sustainable Humanosphere (RISH), yamakawa@rish.kyoto-u.ac.jp
[5] Researcher, Space Exploration Innovation Hub, hosoda@ep.isas.jaxa.jp
[6] Associate Professor, Institute of Space and Astronautical Science (ISAS), nishiyama@ep.isas.jaxa.jp

engine system (IES) in November 2009.[3,4] The lack of the neutralization means an excess emission of the ion beam in short of the electron current from the neutralizer to compensate for the emitting beam current. We consider that spacecraft charging as expected by the ground test occurred at that time,[5] and hence the spacecraft at highly negative potential must have attracted the charged particles existed in the vicinity of the spacecraft. In this case, the return current of the emitted beam ions could be one of the dominant sources of the contamination because the magnitude of the spacecraft potential is expected to be comparable to the acceleration potential applied to the beam ions.[5] Therefore, we need to obtain the value of spacecraft potential to evaluate the contributions of the sources to the contamination: outgassing neutral particles, CEX ions, and beam ions, (ambient plasmas if considered). However, direct measurement of the spacecraft potential had not been made onboard HAYABUSA.

We have performed charging analysis by using a numerical code to estimate the charging status of a spacecraft with excess emission of an ion beam modeling HAYABUSA in its ion thruster operation under lack of the neutralization. We aim to clarify the charging status to understand the contribution of the sources of the contamination. The numerical method to evaluate the onboard measurement of spacecraft contamination is expected to contribute to the contamination monitoring scheduled in the HAYABUSA2 mission[6] that is the follow-on mission to the HAYABUSA mission by JAXA. HAYABUSA2 has the same type of the ion thruster assembly that had originally been developed for HAYABUSA[7], and the spacecraft had successfully been launched on December 3, 2014.

In this paper, we first introduce the measured data of spacecraft contamination by the sensors onboard HAYABUSA during its mission. Next, the numerical modeling to perform charging analysis for a spacecraft in ion thruster operation is described, and preliminary result of charging status for HAYABUSA under lack of the neutralization on its ion thruster is also shown.

II. Measurements of Spacecraft Contamination onboard HAYABUSA

HAYABUSA was equipped with four ion thrusters, and three of them were simultaniously used in the normal operation. An onboard sensor had been developed adopting a solar arrays to determine the spacecraft contamination in the operation of the IES during the mission[1]. The two sensors had installed onboard HAYABUSA near the exit of one of the four thrusters of its IES as shown in Fig. 1 indicated "A" and "B", respectively. The sensors show degradation of the efficiency to generate electricity if contamination on their surface occurs. We can, therefore, estimate the degree of the contamination relative to the initial surface condition of the sensors (we cannot obtain absolute value of the contamination). Figure 2 shows the power efficiency obtained by the sensors during HAYABUSA mission. The efficiency is normalized considering the incident solar flux depending on the spacecraft attitude and the distance to the sun. The lower the normalized efficiency, the more the contamination occurs in the figure. We can recognize following two characteristics from the data in Fig. 2:

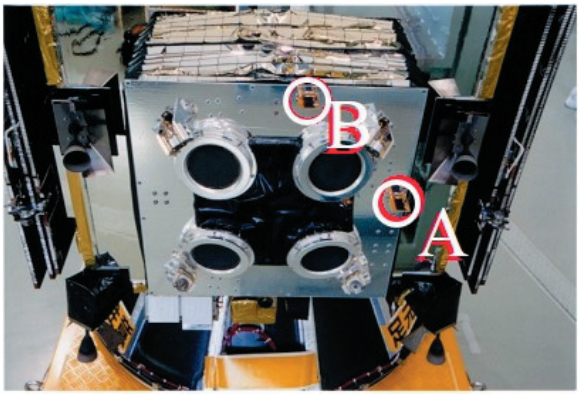


Figure 1. Ion engine system installed in HAYABUSA and the two onboard contamination sensors indicated by "A" and "B".

1) The degradation of the sensor B increases more than that of the sensor A.
2) Rapid increase of the degradation of the both sonsors is recognized at the date indicated by the block arrow.

We consider that the former is due to the more or less of the outgassing nearby each sensor depending its location, and the latter is due to the increase of the contamination by charged particles caused by spacecraft charging. During the time when the rapid increase of the contamination was recognized, the lack of neutralization on one of the four ion thrusters had occurred. Figure 3 shows the extracted telemetry data showing the instantaneous currents of the screen, the acceleration and the neutralizer of the thruster. The vertical axis shows the values of those currents and horizontal one shows the elapsed time in second from 13:01 on November 4, 2009 UT when the lack of the neutralization had occurred. We can estimate the excess of the beam current emitted from the spacecraft of 18.34 mA by subtracting the peak value of the accelerating grid current of 23.3 mA and neutralizer current of 44.01 mA from that of the screen current of 85.65 mA in Fig. 3. Because of the excess emission of the positive beam ions from the spacecraft, we consider that the body potential of HAYABUSA at that time had reached highly negative as had already been examined by the ground testing before launching HAYABUSA.[5] The spacecraft at highly negative potential would attract positive ions existed in the vicinity of it such as emitted beam ions resulting in rapid increase of the current onto it. However, the body potential of the spacecraft had not been measured directory onboard HAYABUSA to estimate the contamination as a function of the potential.

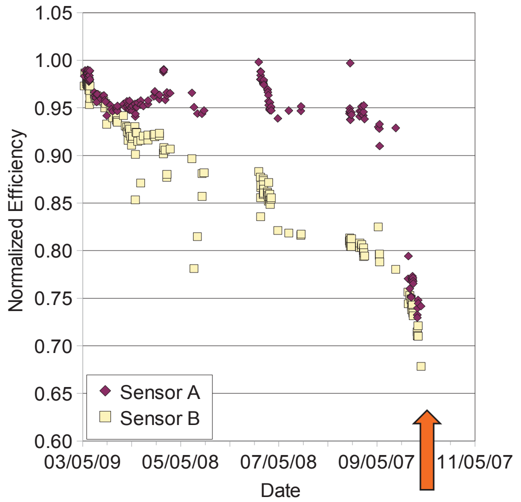


Figure 2. Time history of the contamination measured by the sensors A and B.

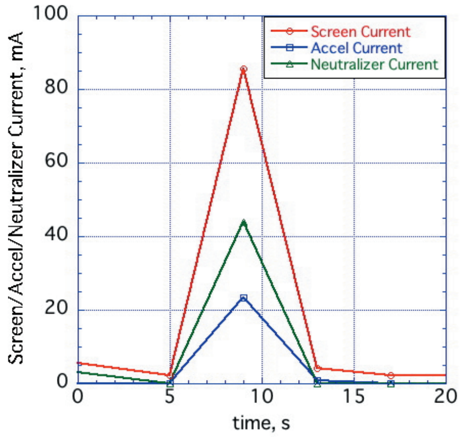


Figure 3. Extracted telemetry data showing the current components of the ion thruster under lack of the neutralization.

III. Increase of Spacecraft Contamination due to Spacecraft Charging

To estimate the increase of the contamination, we consider the current balance onto a spacecraft in the operation of an ion thruster. The net current onto a spacecraft surface I_{net} at its potential V_{s} in the operation of ion thrusters is described as Eq. (1).

$$I_{net}(V_s) = -I_e(V_s) + I_i(V_s) + I_{CEX} - \left\{ I_B - I_{NE} - I_{ph}(V_s) \right\}, \tag{1}$$

where I_{e}, I_{i}, and I_{ph} are the current of the ambient electrons, of the ambient ions and of photoelectrons, and I_{B} and I_{NE} are that of the beam ions and of the neutralizer electrons from an ion thruster, respectively. I_{CEX} is the current of charge-exchange ions onto a spacecraft. The signs of the currents in Eq. (1) are explicitly described. Magnitude of I_{B} is almost equivalent to that of I_{NE} in the normal operation of the IES. The first three currents on the right side of Eq. (1) are the incoming currents onto a spacecraft and others the outgoing ones at V_{s}=0. In the case of lack of the neutralization on ion thruster, i.e. I_{NE} is lower than I_{B}, spacecraft potential becomes negative due to the excess of the positive charge emission. If the neutralizer does not work at all, i.e. I_{NE} becomes zero, the spacecraft potential will reach up to 1000 V negative whose magnitude is almost the same order of the acceleration potential applied to the beam ions.[5]

In order to evaluate the increase of the contamination at the spacecraft potential be negative, we need to estimate the increase of the attracted current onto the spacecraft in Eq. (1), I_{i}, I_{CEX} and the return current of I_{B}. In this paper, at first, we only consider I_{e}, I_{i}, I_{B} and I_{NE} in Eq. (1), then the current collection onto a spherical conductor is extracted as Eq. (2) from Eq. (1) following the orbital motion limited (OML) theory when $|V_{\mathrm{s}}|$ is much greater than kT/e,[8]

$$I_{net}(V_s) = -A \cdot j_{0e} \exp(-e|V_s| / kT_e) + A \cdot j_{0i}(1 + e|V_s| / kT_i) - \left(I_B - I'_{NE} \right) \cdot \tag{2}$$

In Eq. (2), A is the surface of a spacecraft, k is the Boltzmann's constant, T_{e} and T_{i} are the temperature of the ambient electrons and that of ions, respectively. $j_{0\mathrm{k}}$ is the thermal current of the ambient ions and electrons which is determined as

$$j_{0_{k=i,e}} = q_k n_k (kT_k / 2\pi m_k)^{1/2}, \tag{3}$$

where q, m and n are the charge, the mass and the number density of the charged particles, respectively. I'_{NE} is the current of the neutralizer electron with lack of the neutralization as I'_{NE} is lower than I_B. Assuming a spherical conductor of A=10 m^2 at V_s= –1000 V in a thermal plasma of T_e=T_i=10 eV, n_e=n_i=10^6 m^{-3}, the magnitude of the ambient currents in Eq. (2) are calculated as I_e~0 mA and I_i~2.0x10^{-3} mA.

IV. Charging Simulation for HAYABUSA under Lack of Neutralization on Ion Thruster

A. Numerical Code

To evaluate the measured contamination by the sensors onboard HAYABUSA, we introduced a numerical simulation for charging analysis by using a charging analysis code, HiPIC.[9] HiPIC is a three-dimensional electrostatic full-Particle-In-Cell (PIC) code that had been developed by JAXA to analyze spacecraft charging. The code computes the trajectories of all kinds of charged particles (both ions and electrons) that is represented by macroparticle in PIC method, and computes the electrostatic potential due to them selfconsistently with no robustness.

The equation of motion for the macroparticles of the spieces k is shown as Eq. (4) to compute their trajectories.

$$\begin{cases} d^2\vec{x}/dt^2 = (q_k/m_k)(\vec{E}+\vec{v}_k\times\vec{B}) \\ d\vec{x}_k/dt = \vec{v}_k \end{cases}, \tag{4}$$

where m, q, x and v are the mass, the charge, the position and the velocity of the charged particle, respectively. E and B in Eq. (4) are the electric field and the magnetic field at the particle's position, respectively. The position of the particles is computed adopting the Buneman-Boris algorithm in Eq. (4) to integrate in time. The electrostatic potentialt is obtained by solving the Poisson's equation shown as Eq. (5),

$$\nabla^2\phi = -\rho/\varepsilon_0 \,. \tag{5}$$

The electrostatic potential ϕ at the numerical grids representing the surface of a spacecraft is computed by the total charge collected at the grid point and the capacitance of the point which has been obtained by the capacity matrix method.[10] The numerical grid system adopted here is a rectangular grid system with same spacing in Cartesian coordinate, X-, Y- and Z-directions because we use Fast Fourier Transform (FFT) to solve Eq. (5). HiPIC is pararellized using Message Passage Interface (MPI) in domain decomposition mannar for high-speed computing.

B. Modeling of Ion Beam Emission

We have newly developed a numerical module for the HiPIC solver to simulate the emission of charged particle beam from an ion thruster. Once the charged particles are emitted from a thruster their trajectories and the electrostatic potential due to them are computed selfconsistently by the original solver. To model the emission, we adopted a point-source model for the beam emission, in which the acceleration grid of an ion thruster is assumed as a convex on the thruster exit as illustrated in Fig. 4.[11] The curvature radius of the convex R_c is defined by the radius of the thruster exit R_T and the divergence angle of the beam α as defined in ref. 11,

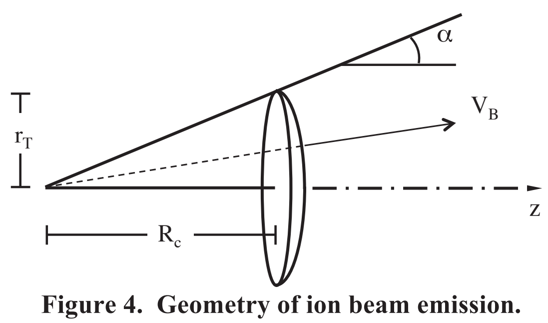


Figure 4. Geometry of ion beam emission.

$$R_c = r_T/\alpha\cos(\alpha/2)\,. \tag{6}$$

The initial distribution of the beam ions on the acceleration grid is determined as follows. We assume the point source located at the center of the sphere whose one part of area forms the convex as illustrated in Fig. 4, and assume that the ions are expanding isotropically from the point source into the conic region with its opening angle to be 2α. The magnitude of the beam velocity V_B is obtained as functions of the acceleration potential applied to the ion thruster ϕ_a as

$$V_B = \left(2q_i\phi_a / m_i\right)^{1/2}, \tag{7}$$

where m_i and q_i are the mass and the charge of the beam ion, respectively.

C. Preliminary Results of Charging Simulation

We had performed numerical simulations using HiPIC to estimate spacecraft charging in ion thruster operateion under lack of the neutralization. Because we aim to obtain the saturation value of the body potential of the spacecraft in this condition we only compute spacecraft charging due to the ion beam emission considering the excess amount of the beam current to that of the neutralizer electron obtained by the telemetry data shown in Fig. 3. The ambient plasmas are also neglected because the currents onto the spacecraft are estimated to be much lower than that of the ion beam at the expected spacecraft potential in excess emission of the ion beam from the thrusters.

Figure 5 shows the computational geometry for the charging analysis and Table 1 shows the parameters used for the simulation. We adopted the scale of the numerical domain of 25.6 m x 25.6 m x 25.6 m in Cartesian coordinate, and adopted a cubic conductor of 1.0 m on each side as HAYABUSA's body for simplicity, which was located at the center in the domain. The beam ions we compute are monovalent xenon, which are emitted from the thruster whose radius is of 0.05 m located on the center of one face of the conductor. The initial distribution of the beam ions on the thruster exit is determined by the point source model mentioned before; the spatial distribution is uniform on the acceleration grid, and the magnitude of their beam velocity on the grid V_B is determined by (7), where the acceleration potential ϕ_a is of 900 V. The beam current I_B is assumed to be of 10 mA referring the order of the excess amount of the beam current obtained by the data shown in Fig. 3. The direction of the beam emission is positive Y direction in the coordinate.

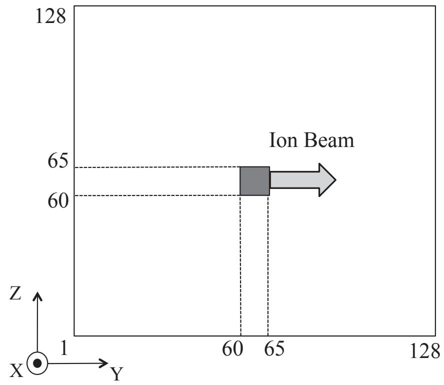
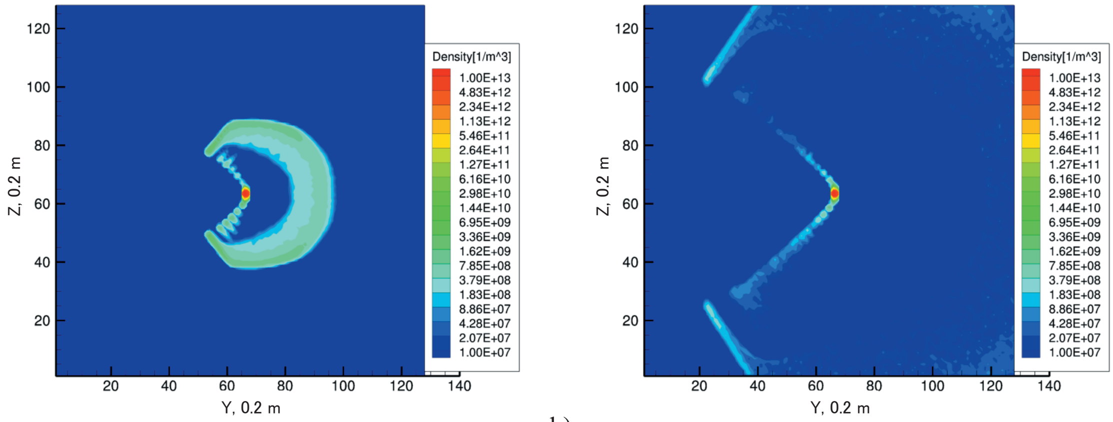


Figure 5. Computation geometry for the simulation. *Numerical domain has 128 x 128 x 128 grids with same spacing of 0.2 m, and spacecraft model has 5 x 5 x 5 grids.*

Table 1. Parameters for the simulation.

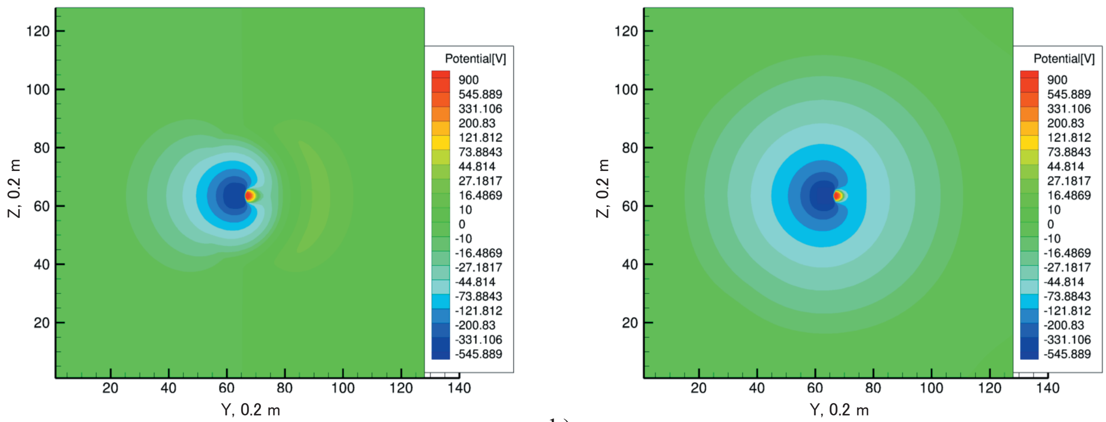
Ion Beam	
divergence angle	20 deg.
accelerating potential	900 V
beam current	10 mA
thruster radius	0.05 m
Numerical Conditions	
grid size	0.2 m
time step	10 ns
numerical domain	25.6 m*25.6 m*25.6 m
spacecraft scale	1.0 m *1.0 m*1.0 m
Magnetic field	0 T

Figure 6 shows the snapshots of the distribution of the number density of the beam ions obtained by the simulation in the *YZ*-plane at *X*=12.7 m at 105 μs and 305 μs, respectively. Figure 7 indicates the distribution of the electric potential corresponding to that of the number density shown in Fig. 6. In Fig. 6, we can recognize the expansion of the beam ions from the thruster exit almost in the whole region around the spacecraft except a part of negative *Y* direction from the spacecraft. We can also recognize a high-density region in the vicinity of the thruster exit whose value is up to 10^{13} m^{-3} in Fig. 6. The snapshots in Fig. 7 shows that the value of the electrostatic potential in this region grows up to 900 V from the early step of the simulation, meanwhile the spacecraft potential becomes negative and its absolute value grows up to 600 V. The absolute value of the spacecraft potential at the quasi-stationary state is estimated to be almost the same as the acceleration potential of 900 V applied to the ion beam, however the numerically obtained body potential is much lower than the expected value of 900 V. We consider this is due to the space charge effect of the beam ions in the vicinity of the thruster exit. If a spacecraft is highly negatively charged the potential difference between a spacecraft and the space in the vicinity of the thruster exit rapidly repels the emitting beam ions. The space charge effect of the beam ions forming a high positive potential at the exit will enhance the deceleration of the beam ions as mentioned in ref. 12.



a) b)

Figure 6. The distribution of the number density of the beam ions in the *YZ*-plane at *X*=12.7 m at 105 μs (a) and at 305 μs (b), respectively.



a) b)

Figure 7. The distribution of the electric potential in the *YZ*-plane at *X*=12.7 m at 105 μs (a) and at 305 μs (b), respectively.

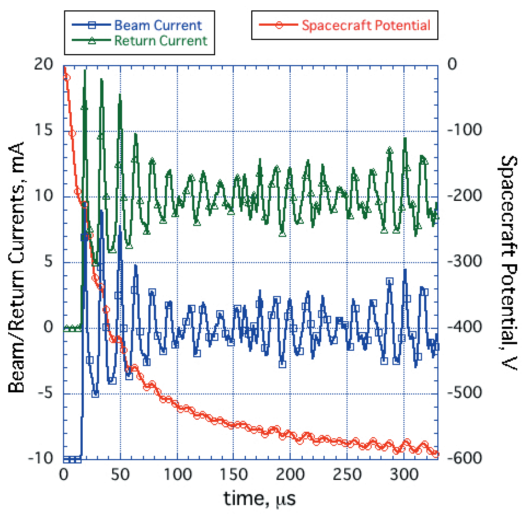


Figure 8. Time history of the net and the return current of the beam ion, and the spacecraft potential.

Figure 8 shows the time history of the net current of the beam ions onto the spacecraft and the body potential for 330 μs obtained by the simulation. The history of the return current is also plotted adding the value of the emission current of 10 mA to that of the net beam current in the figure (positive value means the incoming current of the beam ions onto the spacecraft). We can see the balance of the emission and the return current of the beam ions in the figure. We can also recognize that the amount of the return current is large until 50 μs whose maximum value is almost double of the emission current. The oscillation of the values shown in the history is considered due to the inertia of the beam ion.[13]

V. Conclusion

We have introduced a numerical simulation to estimate the charging status of HAYABUSA spacecraft under lack of the neutralization on its ion thrusters to evaluate the correlation between the increase of spacecraft contamination and the spacecraft potential. For the simulation, we have modified a charging analysis code, HiPIC, to compute spacecraft charging with ion beam emission simulating ion thruster operation. By using the code, we computed charging status of the spacecraft by only the ion beam emission modeling the thruster operation in excess emission of ion beam from the thruster. By a preliminary simulation we obtained the time history of the net and the return current of the emitted beam ions and the spacecraft potential until they reached quasi-stationary state. We also recognized high electric potential at the exit of the thruster due to the space charge effect of the emitted beam ions. The deceleration of the emitted beam ions due to the positive potential could enhance the return current of the beam ions in the downstream of the thruster exit, i.e. the increase of the contamination onto the spacecraft surface. Further evaluation of the increase of the contamination is ongoing.

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References

[1]Hyakutake, T., Nishida, M., Kuninaka, H., and Toki, K., “DSMC-PIC Analysis of a Plume from MUSES-C Ion Engines,” *Trans. Japan Soc. Aero. Space. Sci.* vol. 46, no. 151, 2003, pp.24, 30.

[2]Nishiyama, K., and Kuninaka, H., “Development of Spacecraft Surface Contamination Sonsores,” *Proc. 60th International Aeronautical Congress,* Daejeon, Republic of Korea, 2009.

[3]Nishiyama, K., Hosoda, S., Koizumi, H., Shimizu, Y., Funaki, I., Kuninaka, H., Bodendorfer, M., Kawaguchi, J., and Nakata, D., “Hayabusa's Way Back to Earth by Microwave Discharge Ion Engines”, *Proc. 46th AIAA/ASME Joint Propulsion Conference and Exhibit*, Nashville, USA, 2010.

[4]Hosoda, S., and Kuninaka, H., “The Homeward Journey of Asteroid Explorer “Hayabusa” Powered by Ion Engines,” *J. Plasma Fusion Res.* vol. 86, no. 5, 2010, pp.282, 292.

[5]Kuninaka, H., and Molina-Morales, P., “Spacecraft Charging due to Lack of Neutralization on Ion Thrusters,” *Acta Astronautica,* vol. 55, 2004, pp27, 38.

[6]Nishiyama, K., and Kuninaka, H., “Development and Flight Experiment of a Space QCM in Small Demonstration Satellite-4”, *Proc. 29th International Symposium on Space Technology and Science*, Nagoya, Japan, 2013, 2013-r-29.

[7]Nishiyama, K., Hosoda, S., Ueno, K., and Kuninaka, H., “The Ion Engine System for Hayabusa2,” *Proc. 32th International Electric Propulsion Conference,* Weisbaden, Germany, 2011, IEPC-2011-309.

[8]Hastings, D., and Garrett, H., *Spacecraft-Environment Interactions,* Cambridge Atmospheric and Space Science Series, Cambridge University Press, UK, 1996.

[9]Muranaka, T., Shinohara, I., Funaki, I., Kajimura, Y., Nakano, M., and Takaki, R., “Research and Development of Plasma Simulation Tools in JEDI/JAXA,” *Journal of Space Technology and Science,* vol. 25, no. 2, 2011, pp. 1, 18.

[10]Hockney, R. W. and Eastwood, J. W., *Computer Simulation Using Particles*, Adam Hilger, New York, 1988.

[11]Samanta Roy, R. I., Hastings, D. E., and Gatsonis, N. A., “Ion-Thruster Plume Modeling for Backflow Contamination,” *J. Spacecraft and Rockets*, vol. 33, no. 4, 1996, pp. 525, 534.

[12]Lai, S. T., “An overview of electron and ion beam effects in charging and discharging of spacecraft, ” *IEEE Transactions of Nuclear Science*, vol. 36, no. 6, 1989, p2027.

[13]Hoshi, K., Muranaka, T., Kojima, H., Yamakawa, H., Usui, H., and Shinohara, I., “Numerical Simulation of Satellite Charging Control for Propellantless Orbital Control,” *Proc. 13th Spacecraft Charging Technology Conference,* Pasadena, 2014, Paper No. 195.