TRAJECTORY DESIGN FOR JAPANESE NEW ASTEROID SAMPLE RETURN MISSION HAYABUSA-2

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Abstract: *This paper describes the trajectory design for Hayabusa2 mission. Hayabusa2 is the second Japanese asteroid sample return mission following Hayabusa mission which was completed after the successful return back to the Earth in 2010. The paper introduces the target asteroid selection and trajectory design process, the low-thrust trajectory design technique and how it is implemented to the mission scenario.*

Keywords: *Electric Propulsion, Low-Thrust Trajectory, Optimization, Solar System Exploration*

1. Introduction

The Japan Aerospace Exploration Agency is now developing the second asteroid sample return mission “Hayabusa2”[1,2]. Following the successful return back of Hayabusa from the asteroid Itokawa, Hayabusa2 aims at a round trip mission to the asteroid 1999JU3.1999JU3 is a C-type asteroid, which is supposed to contain organic matters and hydrated minerals. Thus it is expected that, after the successful sample collection, more knowledge on the origin and evolution of the planets, especially the origin of water and organic matters could be acquired.
The paper introduces the target asteroid selection and trajectory design process, the low-thrust trajectory design technique and how it is implemented to the mission scenario.

2. Selection of Target Asteroid

The target asteroid 1999JU3 has been selected by searching for known NEOs (Near-Earth Objects) which are reachable with the ion engine applicable for Hayabusa2. We put four (three nominal and one backup) 10mN-class ion engines as the baseline for the Hayabusa2 mission. Though the thrust level is 20% larger than that of Hayabusa, this configuration is supposed to be readily available technology.
Figure 1 shows the two-impulse (Lambert) solution of the NEOs with the known spectrum type. 1999JU3 is the only asteroid we found as the C-type asteroid reachable with the realistic ΔV. 1999JU3 is suitable for our target in terms of asteroid size, rotation period and Sun-Earth-asteroid geometry during the mission phase (i.e. suitable for landing). The major parameters of 1999JU3 are shown in Table 1.

3. Incorporating Impactor Mission

One of the novel and distinctive mission objectives defined in the early stage of the Hayabusa2 mission scenario planning is to create an artificial crater on the surface of the asteroid to know the sub-surface information of the asteroid. To meet this objective, three types of impactor system were studied (Fig.2).
(1) Impact After Rendezvous (IAR) Sequence: Two independent (and full GNC capability) spacecraft, main spacecraft and impactor spacecraft, are launched by an identical launch vehicle. The impactor spacecraft flies a ballistic path to impact 1999JU3 at about 3km/s.

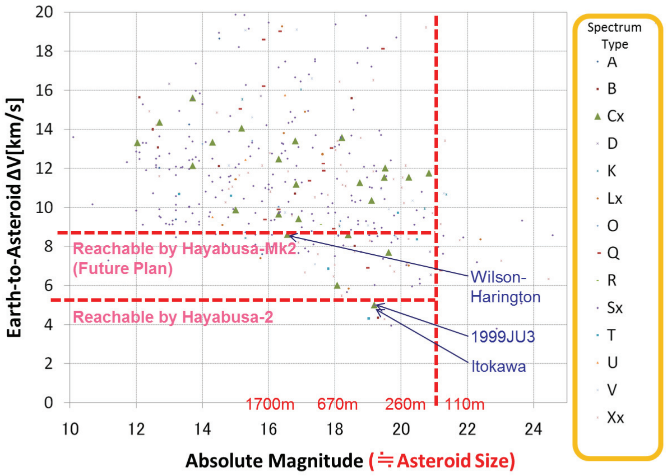


Figure 1. One-way Delta-V for various Near-Earth Asteroids based on Lambert solution

Rotation Period	7.6hr
Diameter	0.922±0.048km
Aspect Ratio	1.3:1.1:1.0
Geometric Albedo	0.063±0.006
Magnitude(H)	18.82±0.021
Slope Parameter(G)	0.110±0.007
Perihelion/Apohelion	0.85/1.4AU
Spectrum Type	Cg

Table 1. Parameters of 1999JU3

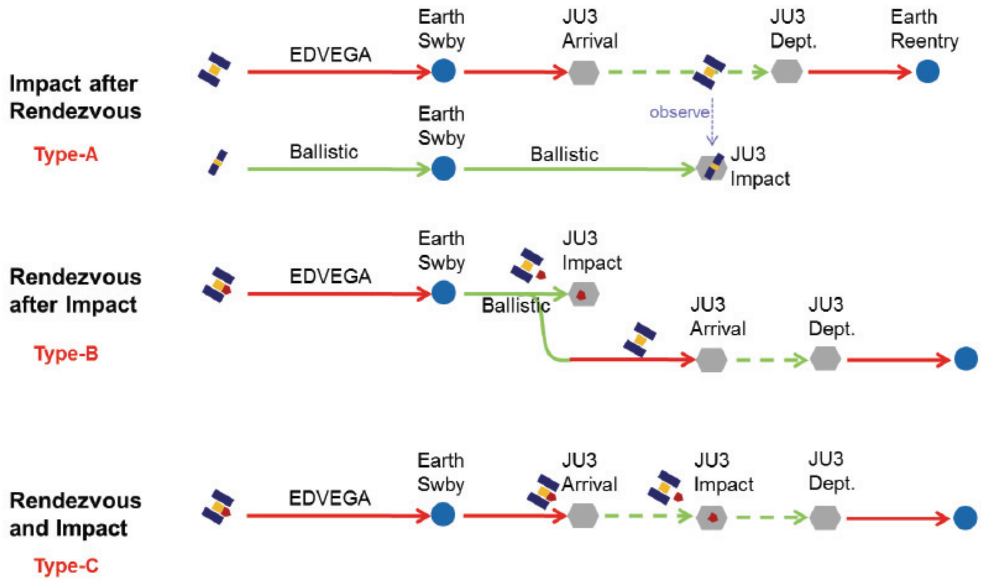


Figure 2. Three options for imcorporating Impactor mission into Hayabusa2 mission scenario.

(2) Rendezvous After Impact (RAI) Sequence: The impactor is equipped on the main spacecraft. The main spacecraft fly-bys the asteroid at 3km/s first, and releases the impactor at the fly-by to create the crater. The main spacecraft once escapes from the asteroid impact course and then by activating ion engine, it makes rendezvous with the asteroid. The impactor requires an ability of terminal GNC for the impact.

(3) Rendezvous And Impact (R&I) Sequence: The impactor is equipped on the main spacecraft. The main spacecraft directly rendezvous with the asteroid. The impactor is released from the main spacecraft at the proximity of the asteroid. It immediately accelerates up to 2km/s and impacts the asteroid. The impacor requires very rapid acceleration but GNC capability (Main spacecraft is responsible for all the targeting accuracy).

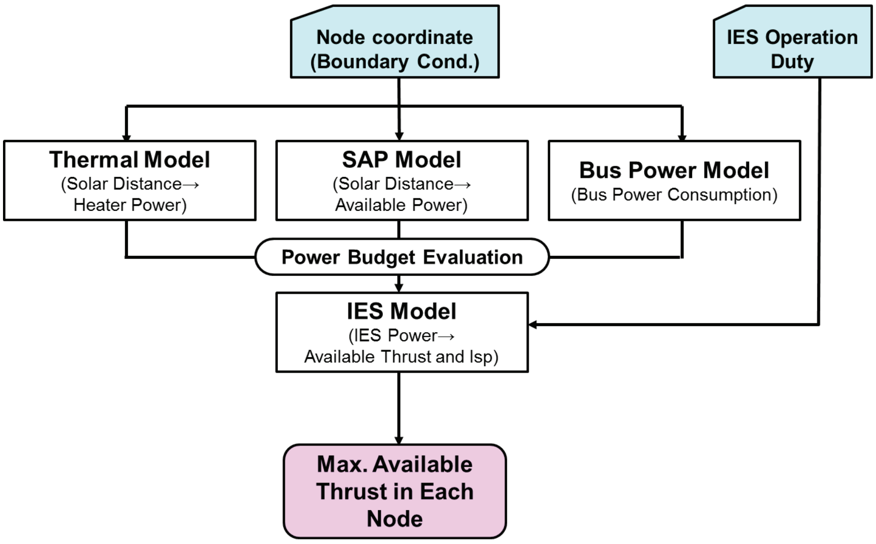


Figure 3. Spacecraft Model for Trajectory Deisgn

The R&I Sequence was finally chosen after the trade-off between scientific merit, technological impact and cost.

5. Trajectory Design for Hayabusa2 Mission

5.1. Trajectory Optimization

The trajectory sequence for the R&I mission is as follows; ①EDVEGA phase (Earth to Earth), ②Transfer phase (Earth to 1999JU3 phase), ③Mission Phase (1999JU3 proximity operation), ④ Return Phase (1999JU3 to Earth). We applied the low thrust trajectory design independently for the phase ①② and ④. The continuity between the phases is satisfied by imposing boundary conditions (position and velocity) of the optimization method. The optimality of the choice of the boundary condition is checked by parametrically varying the boundary condition parameters.
The trajectory design is based on the following optimization problem

$$\min f(x) \text{ such that} \begin{cases} c(x) \leq 0 \\ ceq(x) = 0 \end{cases} \tag{1}$$

where $f(x)$ is an objective function (ΔV or fuel consumption), $c(x)$ is the state inequality constraints (ion engine performance, attitude, launch conditions, etc.), $ceq(x)$ is the state equality constraints (equation of motion, initial and final velocity and position).
The Nonlinear Sequential Quadratic Programming (NLSQP) is applied to solve the problem Eq.1. The trajectory path is divided into short nodes, whose boundary state is treated as the state variables of the NLSQP. The spacecraft performance is defined as Fig.3. The available ion engine thrust and specific impulse are derived for each node and iteration strep of the optimization by following the flow shown in Fig.3.

5.2. EDVEGA Phase Design

The objective of the Electric Delta-V Earth Gravity Assist (EDVEGA) phase for the Hayabusa2 mission is to connect to the transfer phase via Earth swing-by while ①enlarging the operational margin of the ion engine (as the EDVEGA phase has long non-operation duration of the ion engine), ②separating the launch constraints and asteroid reachability constraints (which in turns leads to the enlargement of the launch window), ③providing back up windows.

The basic property of EDVEGA can be derived by formulating the following Hill's equation around the Sun and the Earth as the coordinate origin

$$\ddot{x} = 2n\dot{y} + 3n^2 x + a_x,\ \ \ddot{y} + 2n\dot{x} = a_y,\ \ddot{z} + n^2 z = a_z. \tag{2}$$

where x, y, z is the radial, in-track and orbital plane-normal position, n is the natural frequency of the orbit, a_x, a_y, a_z is the acceleration along x, y and z direction. The auto-return trajectory imposes that

$$x(0) = y(0) = z(0) = 0,\ \ a_x(t) = a_y(t) = a_z(t) \equiv 0\,(\text{const.}) \tag{3}$$

and hence from Eq.(2) and (3) we get

$$\begin{gathered}\begin{pmatrix} x(t_\mathrm{f}) \\ y(t_\mathrm{f}) \end{pmatrix} = \begin{pmatrix} \frac{1}{n}\sin nt_\mathrm{f} & \frac{2}{n}(1-\cos nt_\mathrm{f}) \\ -\frac{2}{n}(1-\cos nt_\mathrm{f}) & \left(-3t + \frac{4}{n}\sin nt_\mathrm{f}\right) \end{pmatrix} \begin{pmatrix} \dot{x}(0) \\ \dot{y}(0) \end{pmatrix}, \\ z(t_\mathrm{f}) = \frac{\dot{z}(0)}{n}\sin nt_\mathrm{f}. \end{gathered} \tag{4}$$

where t_f is the time when the spacecraft returns back to the Earth. So as for Eq.4 to be insensitive to the initial velocity, the matrix in Eq.4 must be singular. Thus the in-plane solution is

$$\begin{gathered}\sin\left(\frac{nt_\mathrm{f}}{2}\right) = 0 \ \text{ or } \ \frac{3}{4}\left(\frac{nt_\mathrm{f}}{2}\right) = \tan\left(\frac{nt_\mathrm{f}}{2}\right) \\ \rightarrow \frac{nt_\mathrm{f}}{2\pi} = 0,\ 1,\ 1.4067,\ 2,\ \cdots.[\text{year}]. \end{gathered} \tag{5}$$

and the outer-plane solution is

$$\begin{gathered}\sin(nt_\mathrm{f}) = 0 \\ \rightarrow \frac{nt_\mathrm{f}}{2\pi} = 0,\ 0.5,\ 1,\ 1.5,\ \cdots[\text{year}]. \end{gathered} \tag{6}$$

The EDVEGA trajectory refers to these auto-return trajectories as the reference solution (initial guess) of the optimization. More detail discussion about EDVEGA is found in Reference [2].

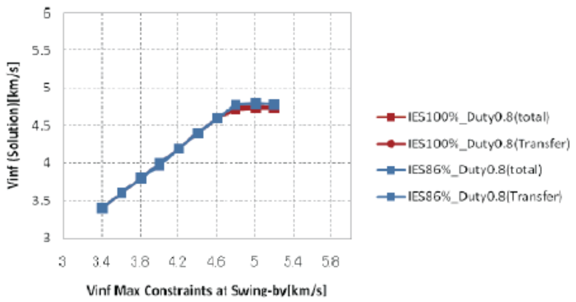
The first four shortest flight duration EDVEGA solutions (0year / 0.5year / 1year / 1.4year) are investigated in the Hayubusa2 trajectory design. Note that 0year option is the one without EDVEGA and therefore referred to as "Direct transfer" option.

Once an EDVEGA option is chosen, the design parameters for the EDVEGA phase is the initial injection V-infinity ΔV_{inj}, the swing-by V-infinity ΔV_{inf} and the ion engine operation profile.

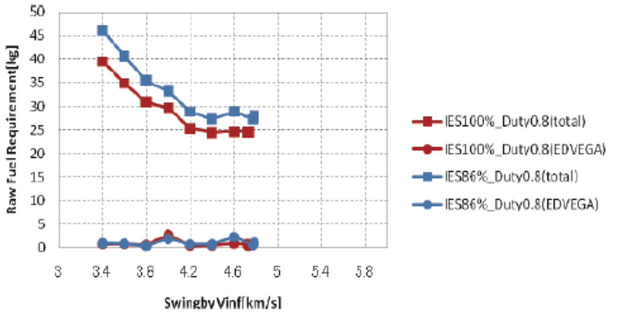
ΔV_{inf} should be chosen so as to have the identical V-infinity with the initial velocity of the transfer phase.

Figure 4 shows the results of the trajectory optimizations with the swing-by V-infinity as the parameters (Max. ΔVinf constraints). It is found from this analysis that the choice of ΔV_{inf} =4.4km/s provides the minimum fuel trajectory.

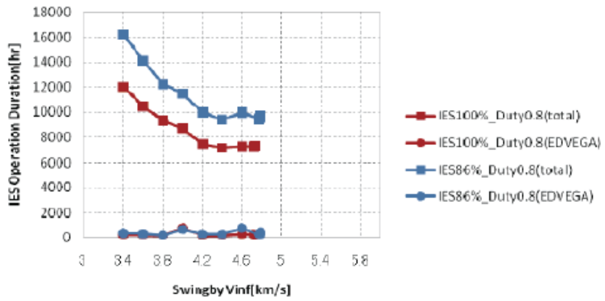
If we can expect sufficient injection performance for the launch vehicle, no ion engine operation is required in the EDVEGA phase, which is the case with the H2A launch vehicle. Consequently, we have ΔV_{inj}=ΔV_{inf} =4.4km/s as the baseline of the Hayabusa2 trajectory.



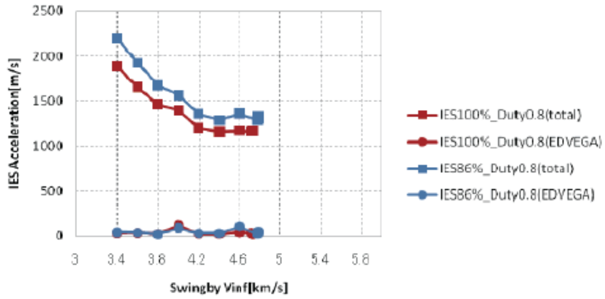
(a)Vinf constraints vs optimized Vinf solution



(b)Swing-by Vinf vs fuel consuption

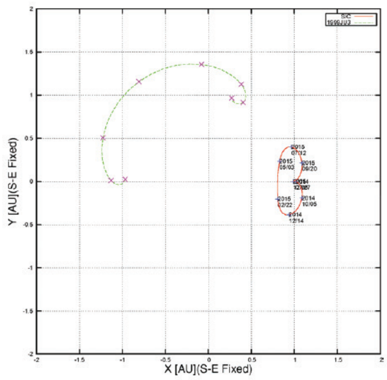


(c)Swing-by Vinf vs ion engine duty

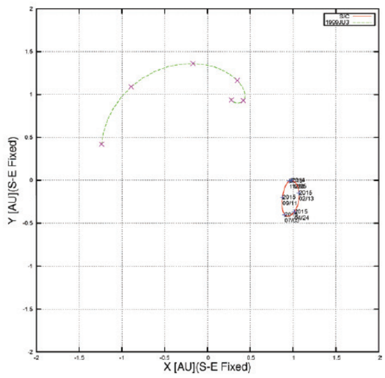


(d)Swing-by Vinf vs total impulse

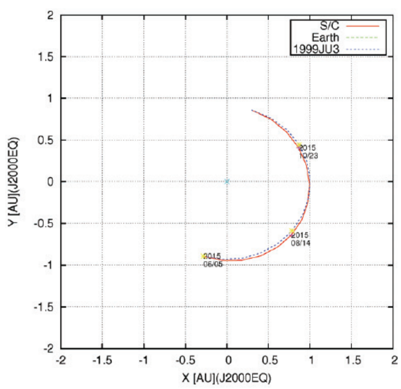
Figure 4. Optimality analysis of boundary conditions for EDVEGA and Transfer phases.



(a)1.4 year EDVEGA(Outbound-first)



(b) 1 year EDVEGA(Outbound-first)



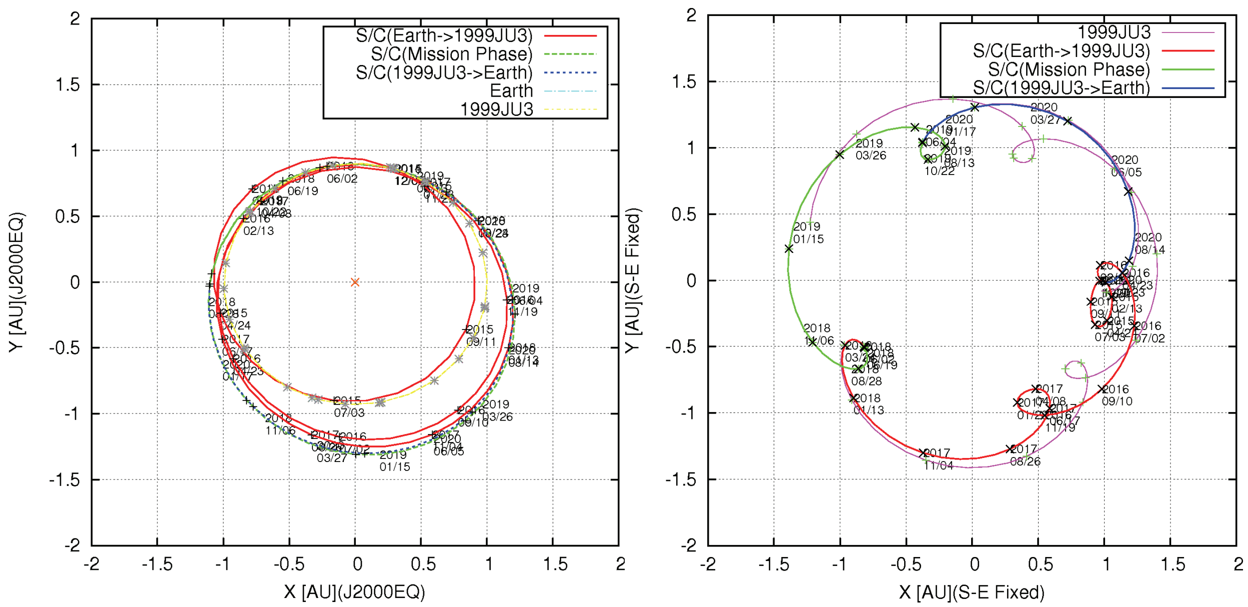
(c) 0.5year EDVEGA (North-first)

(a)(b) are drawn in the Sun-Earth line fixed heliocentric J2000EC., (c) is drawn in heliocentric inertial (J2000EQ)

Figure 5. Three EDVEGA trajectory options.

5.2. Incorporating Realistic Constraints

As trajectories subject to the Hill's equation are symmetry about x and z axes, each EDVEGA option has two symmetrical solutions for each departure/arrival date combination. For example, 1.4 year and 1 year EDVEGA trajectories both have outbound-first and inbound-first solutions. 0.5 year EDVEGA has north-first and south-first solutions. This flexibility is used to meet the realistic constraints of the mission and the spacecraft configuration, such as the swing-by altitude limit, antenna usage, launch condition, etc. After the trade-off study, consequently, the outbound-



(a) Heliocentric inertial (J2000EQ), (b) Sun-Earth line fixed heliocentric J2000EC.

Figure 5. Three EDVEGA trajectory options.

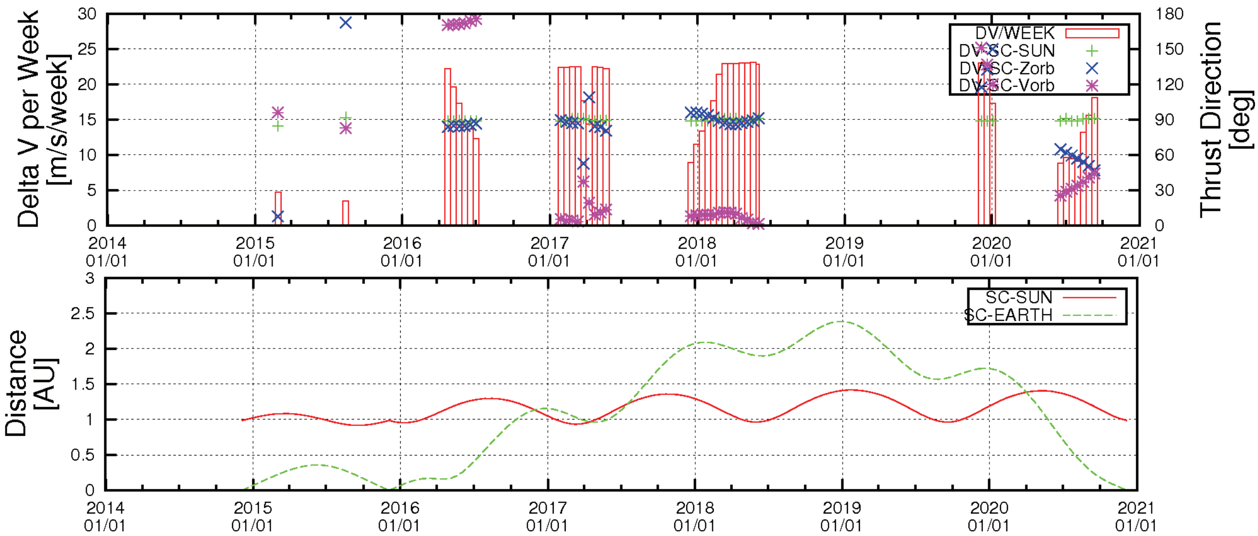


Figure6. Ion engine operation profile and Sun, Earth distance for a whole flight duration.

first solution is chosen for 1.4 year and 1 year EDVEGA, and the north-first solution is chosen for 0.5 year EDVEGA. These three EDVEGA trajectories are shown in Fig.6.

5.3. Trajectory Design Result

Based on the discussion in the previous sections, the nominal trajectory of the Hayabusa2 mission has been designed as shown in Fig. 6, 7. Taken into account the spacecraft development schedule, the nominal launch is in 2014 with the 1 year EDVEGA phase. The Earth swing-by is

in December 2015. It arrives at 1999JU3 in June 2018, stays for about 1.5 year, and leaves in December 2019. The Earth reentry is in December 2020.
Three back-up windows can be prepared by taking advantage of the EDVEGA property. 0.5 year and direct transfer options can be taken as the back-up, all of which are to be connected to the identical transfer trajectory as depicted in Fig.6, 7.

6. Conclusions

This paper described the trajectory design of the next generation sample-return mission Hayabusa2. Trajectory design is one of the most important processes in the early mission design. This paper shows how the trajectory design activity contributes to the creation of the mission scenario. As the result of the whole NEOs survey, the C-type asteroid 1999JU3 has been chosen as the target of the mission. The resulting trajectory has 1.5 year asteroid proximity operation duration, and enables us to have four launch windows in between 2014 and 2015. The paper also briefly described how the realistic constraints from the spacecraft configuration and operational considerations are incorporated into the design of the trajectory.

References

[1] Yoshikawa, M., Hayabusa2 Project Team, "Outline of the Next Asteroid Sample Return Mission of Japan - Hayaubsa-2 Outline of the Next Asteroid Sample Return Mission of Japan - Hayabusa-2", 28th International Symposium on Space Technology and Science, 2011-k-19, 2011.

[2] Tsuda, Y., Yoshikawa, M., Minamino, N., Nakazawa, S., "System Design of Hayabusa2 – Asteroid Sample Return Mission to 1999JU3", 63rd International Astronautical Congress, IAC-12-A.3.4.5, Naples, Italy, 2012.

[3] Kawaguchi, J, "Solar Electric Propulsion Leverage: Electric Delta-VEGA (EDVEGA) Scheme and its Application", AAS 01-213, AAS/AIAA Space Flight Mechanics Meeting, Santa Barbara, California, Feb. 11-14, 2001.