Mission Analysis for Sample Retrieval from a Primitive Near-Earth Asteroid

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(Received April 7th, 2009)

Reported in this paper are the results of a mission analysis conducted for an asteroid exploration mission. Following the results obtained from HAYABUSA, a Japanese asteroid explorer, the Japan Aerospace Exploration Agency has started studying the possibility of the next asteroid exploration mission. The mission studied gives priority to the “early” retrieval of a sample from an asteroid with a primitive composition. Therefore, the design of the spacecraft follows that of the HAYABUSA, basically as it is, and the spacecraft is planned to be launched early in the next decade. The objective of the mission analysis is to design a mission sequence which has a launch window early in the next decade, is feasible utilizing a HAYABUSA-type spacecraft, and whose target asteroid complies with the scientific objectives. The results include the selection of the target asteroid, the design of the nominal mission sequence, and the alternative sequence to overcome the drawbacks of the nominal sequence.

Key Words: Mission Analysis, Near-Earth Asteroids, Sample Retrieval Mission

Nomenclature

- $a$: semi-major axis
- EDVEGA: electric propulsion
- Δ$v$: Earth gravity assist
- $F$: thrust
- $F_{\text{max}}$: maximum thrust
- $F_{\text{1AU}}$: reference thrust at 1 AU
- HGA: high-gain antenna
- $r$: distance from Sun
- $r_p$: perihelion radius
- SAP: solar array paddle
- SEP: solar electric propulsion
- $v_\infty$: excessive velocity
- Δ$: velocity increment

1. Introduction

The Japanese asteroid explorer HAYABUSA, launched in 2003, arrived at its target asteroid, ITOKAWA, in September 2005. HAYABUSA made a number of scientific discoveries and technological achievements during its stay at ITOKAWA, and left ITOKAWA in December 2005. Under this situation, a feasibility study of the next asteroid exploration mission has been performed, assuming a launch in the first half of the next decade. The mission studied so far gives priority to the enhancement of scientific output, and assumes a spacecraft with a larger size and enhanced capabilities compared to HAYABUSA. The example missions studied are retrieving samples from multiple asteroids and retrieving samples from asteroids at a greater distance from Earth.

However, in actuality, because of programming problems, HAYABUSA failed to obtain a sample of ITOKAWA in the way originally planned. Following this result, the current mission study shifts its priority to the “early” retrieval of a sample from an asteroid. To realize this objective, the design of the spacecraft basically follows that of the HAYABUSA in order to shorten development time. The spacecraft is called HAYABUSA2, and it is planned to be launched early in the next decade. From the point of scientific objectives, an asteroid with a primitive composition (for example, C-type in spectral type) is preferable (ITOKAWA is S-type).

Discussed in this paper, is the mission analysis of HAYABUSA2. The objective of the mission analysis is to design a mission sequence which has a launch window early in the next decade, is feasible utilizing a HAYABUSA-type spacecraft, and whose target asteroid complies with the scientific objective. Among the conditions of the mission sequence design, the feasibility of utilizing a HAYABUSA-type spacecraft is evaluated quantitatively.

This paper is categorized into the field of asteroid exploration mission analysis. A feature of this paper is the application of solar electric propulsion (SEP) to the asteroid sample retrieval mission. Numerous papers consider applying SEP to an asteroid exploration mission. Many of them are directed to fly-by missions or rendezvous missions, and a limited number of papers deal with a sample retrieval mission. However, a target asteroid is pre-assumed in these two papers, and its
selection process, especially from dynamical feasibility, is not clearly referred to.

Target asteroid selection is another feature of this paper. Some papers consider a global survey of the target asteroid for the sample retrieval mission based on dynamical feasibility, however, discussions are limited to the range of ballistic transfer.

The last feature of this paper discussed is the constraints involved in using the design of the existing spacecraft. Some other papers assume constraints of this kind, but those papers focus on asteroid fly-by or rendezvous missions utilizing ballistic transfer.

In all, the features of this paper are summarized as, the mission analysis of asteroid sample retrieval mission from the target selection process utilizing the existing spacecraft design equipped with SEP.

Following the introduction in this section, the process results of the target asteroid selection are described in Section 2. The mission sequence to the selected target asteroid (1999 JU3) is designed in Section 3. First, the design procedure is briefly introduced (Section 3.2), and the design results of a nominal mission sequence are discussed in Section 3.3. The results show several drawbacks of the sequence, which leads to the proposal of an alternative sequence to overcome the drawbacks (Section 3.4). Finally, the paper is concluded in Section 4.

2. Target Asteroid Selection

This section discusses target asteroid selection. The target asteroids are selected considering both the scientific objectives and dynamical feasibility of the mission. The sample retrieval missions to various asteroids, assuming a launch early in the next decade, are investigated and evaluated.

2.1. Procedure of mission sequence construction

The mission sequences are globally constructed utilizing the “trajectory parts connection method,” which is exploited by an author. In the method, the mission sequence is constructed as a series of Keplerian orbits connected with impulsive velocity changes. The dynamical feasibility of the sequence is evaluated quantitatively utilizing the velocity increment (Δv) required to complete the sequence. Although the compensation of the total Δv is required for the case of using SEP, an estimation utilizing impulsive maneuvers is sufficient to compare the dynamical feasibility of the sequences. The sequence constructed defines the asteroid to be explored and the opportunity of the mission. Therefore, the sequence gives a good initial estimate for the following detailed design considering the use of SEP.

The candidates for the target asteroid are screened from an asteroid database utilizing the three criteria shown in Table 1. The first two criteria, semi-major axis (a) and perihelion radius (r_p), restrict the candidates to near-Earth asteroids, which are expected to be easy to access. The third criterion, the absolute magnitude, restricts the approximate size of the asteroid to be as large or larger than ITOKAWA. As a result, 3,614 asteroids are screened as target asteroid candidates.

The schematic of the mission sequence used in the sequence construction is described in Fig. 1. The round-trip sequence from Earth departure to Earth return is assumed. The seven events during the sequence and the related design parameters are shown in the figure. Listed in the figure are the minimum necessary parameters to construct the sequence, and the parameters inducible from them are not shown in the figure. The ranges of the parameters used in the sequence construction are summarized in Table 2. The ranges are set considering not only excluding the useless (practically infeasible) sequences, but also not dropping the useful sequences from the construction results.

2.2. Results of mission sequence construction

Figure 2 shows the mission sequences that can be achieved with a relatively small Δv. The sequences in the figure satisfy the following two conditions:

Table 2. Ranges of design parameters in sequence construction

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Range</th>
</tr>
</thead>
<tbody>
<tr>
<td>1. Earth departure</td>
<td>2011 Jan. 1 – 2013 Dec. 31 ≤60 km/s</td>
</tr>
<tr>
<td>2. Earth/asteroid transfer</td>
<td>1 month – 2 years ≤2</td>
</tr>
<tr>
<td>3. Asteroid arrival</td>
<td>Δv ≤30 km/s</td>
</tr>
<tr>
<td>4. Asteroid stay</td>
<td>1 month – 1 year</td>
</tr>
<tr>
<td>5. Asteroid departure</td>
<td></td>
</tr>
<tr>
<td>6. Asteroid/Earth transfer</td>
<td>1 month – 2 years ≤2</td>
</tr>
<tr>
<td>7. Earth arrival</td>
<td>2011 Jan. 1 – 2018 Dec. 31 ≤60 km/s</td>
</tr>
</tbody>
</table>
1) Earth departure excessive velocity \( (v_\infty) \leq 4.5\text{km/s} \), and
2) Sum of \( \Delta v \) at asteroid arrival/departure \( \leq 3.0\text{km/s} \).

Actually, there are groups of constructed sequences that share the target asteroid and have a similar schedule. Shown in the figure, are the representatives from such groups. A total of 14 sequences to 8 asteroids are shown in the figure, with the milestones of the major events in the sequences. In addition to the sequences constructed by the procedure above (solid line from Earth swing-by to Earth arrival), the preceding milestones are drawn in the figure (broken line prior to Earth swing-by). These additional milestones assume usage of the electric propulsion delta-v Earth gravity assist (EDVEGA) technique which was used for HAYABUSA. If EDVEGA is applied to the sequence, the spacecraft is launched 1 year (or 1.4 years) prior to the round-trip to the asteroids, increases the relative velocity to the Earth utilizing electric propulsion, and directs the velocity by the Earth swing-by to transfer to the target asteroid.

Within the asteroids appearing in the figure, 1999 JU3 and 1989 UQ are assumed to have the primitive composition and meet our scientific objectives. As to the five asteroids in the lower part of the figure, from Orpheus to 2001 QC34, their spectral types are currently unknown. However, if their spectral types are identified via ground observation, and if they are found to have a primitive composition, they will also meet our scientific objectives.

### 2.3. Target asteroid selection

Within the two asteroids that meet our scientific objectives, 1989 UQ is an Aten asteroid (Earth-crossing asteroid whose \( a \) is smaller than 1AU) (Fig. 3). On the other hand, 1999 JU3 belongs to the Apollo asteroids (Earth-crossing asteroid whose \( a \) is greater than 1AU) and whose orbit shape is similar to that of ITOKAWA, the target asteroid of HAYABUSA.

The exploration of 1989 UQ has two features which should be recognized as different from the exploration of ITOKAWA performed by HAYABUSA. They are:

1) The minimum distance from the Sun during the mission is approximately 0.7AU, which leads to a severe thermal condition. The minimum distance from the Sun in case of HAYABUSA was approximately 1AU.

2) The angle between the Earth and Sun viewed from the spacecraft may be large, which results in an incompatibility of the sufficient power supply and the high-rate communication link with the fixed solar array paddle (SAP) and high-gain antenna (HGA). The angle was basically not so large in the case of HAYABUSA, which mainly operated outside of the ecliptic orbit.

Fig. 3. The orbits of two asteroids that satisfy scientific objectives.
Either problem can be overcome by altering the spacecraft design or operation procedure. However, in the case of HAYABUSA2, the design of the spacecraft basically follows that of HAYABUSA as it is, in order to shorten the development time. Therefore, there is less merit in choosing 1989 UQ, as the mission target instead of 1999 JU3, whose orbit shape is similar to that of ITOKAWA.

Consequently, 1999 JU3 is selected as the primary target asteroid, and the sample retrieval mission from 1999 JU3 is investigated in detail in the next section.

3. Mission to the Asteroid 1999 JU3

3.1. Selection of preliminary mission sequence

The two mission sequences to 1999 JU3, shown in Fig. 2, are detailed in Table 3. In Case 1, the spacecraft departs from the Earth one year earlier than in Case 2, and arrives at 1999 JU3 one year earlier as well. However, the return trip is almost the same for both cases, which leads to a shorter stay at the asteroid in Case 2. The total required $\Delta v$ to complete the mission is not so different between the two cases. Therefore, considering the difference of the short stay in Case 2, Case 1 is selected from the two cases.

Next, there are three options of the launch date for Case 1. Two of the options assume the usage of EDVEGA, and the durations of the EDVEGA phase are different. The remaining option assumes direct injection into the 1999 JU3 transfer trajectory by the launcher. Considering the merits of using EDVEGA and the interference of the launch operation with the other program, the option of “EDVEGA 1.0 year” is selected from the three options.

Consequently, the “EDVEGA 1.0 year” option of Case 1 is selected, and is called “preliminary mission sequence” hereafter (Table 4). It must be noted that this sequence is constructed as the series of Keplerian orbits connected with impulsive velocity changes. Additionally, the sequence in Table 4 requires relatively small $\Delta v$ under this assumption. Though the sequence gives a good initial estimate for the following detailed mission design considering the use of SEP, it is apparent that the sequence is not exactly optimum under the assumption of the detailed design. For this reason, the sequence is to be re-optimized in the following detailed mission design.

3.2. Procedure of detailed mission design

Following the result of the selection above, a more detailed mission design is performed assuming the usage of SEP. The design process is divided into two layers (Fig. 4). They are, “trajectory arc design (inner loop)” and “sequence design (outer loop)”. The details are described in the following.

Explained first is the “trajectory arc design (inner loop)”. The trajectory arc is defined as the part of the trajectory which is bounded by the encounter with the celestial body. Concretely, the arcs composing the sequence are:

1) Earth – Earth (EDVEGA),
2) Earth – 1999 JU3, and
3) 1999 JU3 – Earth.

Each arc (with the thrust control profile) is designed based on an optimal control problem which is formulated as follows: The objective function is to maximize the final mass. The control variable is the thrust vector at each moment, and the magnitude of the thrust ($F$) is constrained as:

$$F \leq \min\left(F_{\text{max}}, \frac{F_{\text{max}}}{r^2}\right)$$

where $r$ is the spacecraft’s distance from the Sun expressed in AU, and $F_{\text{max}}$ is the maximum thrust available. The second item in the parentheses indicates that $F$ is constrained by the available power, which decreases as the spacecraft’s distance from the Sun increases. This is the characteristic of SEP. The following items are designated to fixed values in the trajectory arc design, and treated as the boundary conditions.

1) Departure/arrival time
2) Departure/arrival $v_{\infty}$ (against the celestial body)
3) Spacecraft mass at the time of departure

The design of the departure/arrival time is equivalent to the designation of the spacecraft’s position at the departure/arrival. Optimization of the items is performed in the sequence design (outer loop), described afterwards.

The optimal control problem is directly collocated with a nonlinear programming problem, and the nonlinear
The arrival Earth swing-by date is omitted from the free parameter.

The transfer duration of one year is the local optimum. Thus, the transfer duration is optimized in advance to find that the fixed to the value in Table 5. As for the Earth – Earth arc, treated as the parameters to be optimized:

Table 5. Assumptions for mission sequence design.

<table>
<thead>
<tr>
<th>Item</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Spacecraft mass at launch</td>
<td>520kg</td>
</tr>
<tr>
<td>$v_e$ at launch</td>
<td>3km/s</td>
</tr>
<tr>
<td>Specific impulse of SEP</td>
<td>2900s</td>
</tr>
<tr>
<td>$F_{max}$</td>
<td>21mN</td>
</tr>
<tr>
<td>$F_{IAU}$</td>
<td>28mN</td>
</tr>
</tbody>
</table>

The “sequence design (outer loop)” is explained here. In the sequence design, some of the items that are fixed in the trajectory arc design are parameterized and optimized. The whole sequence is divided into two sub-sequences, and they are designed independently. The designed sub-sequences are simply connected, and compose the whole sequence in the end. The sub-sequences are:

1) Earth – Earth – 1999 JU3, and
2) 1999 JU3 – Earth.

In the design of sub-sequence 1, the following items are treated as the parameters to be optimized:

1a) Earth launch date,
1b) Earth swing-by $v_e$, and
1c) 1999 JU3 arrival date.

As to the other possible tuning items, the launch $v_e$ is fixed to the value in Table 5. As for the Earth – Earth arc, the transfer duration is optimized in advance to find that the transfer duration of one year is the local optimum. Thus, the Earth swing-by date is omitted from the free parameter. The arrival $v_e$ at 1999 JU3 is designated as “0” assuming the rendezvous with the asteroid. Initial mass at the launch is fixed to the value in Table 5. The initial mass of the Earth – 1999 JU3 arc inherits the final mass of the Earth – Earth arc, which is obtained as the result of the Earth – Earth trajectory arc design.

The discrete values of the items around the value obtained in the preliminary mission sequence (Table 4) are prepared. The set of parameters which gives the maximum mass at 1999 JU3 arrival is globally searched, which results in the nominal mission sub-sequence.

Sub-sequence 2 is designed in a similar manner. The parameters to be optimized in the design of sub-sequence 2 are:

2a) 1999 JU3 departure date,
2b) Earth arrival date, and
2c) Earth arrival $v_e$.

The departure $v_e$ at 1999 JU3 is designated as “0” assuming the rendezvous with the asteroid. The initial mass of the 1999 JU3 – Earth arc inherits the final mass of sub-sequence 1.

Finally, the two independently designed sub-sequences are simply connected. Since the constraints as to the stay time at 1999 JU3 are not so strict, and it is sufficiently long in the preliminary design result, the stay time at 1999 JU3 of the connected sequence satisfies the requirement.

3.3. Nominal mission sequence

Reported here are the results of the detailed mission design for the sample retrieval from 1999 JU3. Firstly, the nominal sequence of the mission is shown in Table 6. Note that the dates of the events are modified from those of the preliminary mission sequence (Table 4), since the dates are optimized for the usage of SEP.

Figure 5 shows the trajectory profile of the mission sequence. In Fig. 5(a) to (c), the trajectory is projected on the ecliptic plane of a heliocentric inertial coordinate system. The profile is divided into three parts:

(a) Earth – Earth,
(b) Earth – 1999 JU3, and
(c) 1999 JU3 – Earth.

Displayed in the figures are the orbit of the Earth and 1999 JU3 (dark blue line), the transfer orbit (red line), and the control acceleration vector (green arrows).

The trajectory profile starts with the EDVEGA phase (Fig. 5(a)). The spacecraft is launched and directly injected into an interplanetary orbit. The relative velocity to the Earth at the time of injection is 3km/s, which is approximately the same as that of HAYABUSA (Table 5). The maneuver during the EDVEGA phase increases the relative velocity to the Earth to 4.1km/s. The relative velocity is redirected by the Earth swing-by and the spacecraft is injected into the transfer orbit to 1999 JU3 (Fig. 5(b)). As a result of the maneuver during the 1999 JU3 transfer phase, the spacecraft rendezvous with 1999 JU3. After a half-year stay at 1999 JU3 for scientific observation and sampling (which is called the “mission phase”), the spacecraft departs from 1999 JU3 and returns to Earth (Fig. 5(c)). The relative velocity to the Earth at the time of arrival is 4.2km/s, and the sample retrieval capsule released from the spacecraft directly reenters the atmosphere.

It seems strange, at first glance, to go outside of the departure orbit in order to return to the inner destination (i.e., Earth). However, it is necessary for the spacecraft to be phased with the Earth and encounter the Earth at the arrival date. The spacecraft slows down its velocity by transferring to the outer orbit once, so that the Earth can catch up with the spacecraft within the limited time.

Figure 6 shows the mass and thrust profile of the mission sequence. Displayed in the figure are the mass profile (dark blue line, left axis) and the thrust profile (red solid line, right axis). The red dashed line denotes the profile of the upper limit of the thrust, which is defined by Eq. (1). The duration where the red solid line is on the red dashed line indicates that SEP is operating at full throttle.

The EDVEGA phase includes two segments of

Table6. 1999JU3 sample retrieval mission sequence (Results of detailed analysis).

<table>
<thead>
<tr>
<th>Event</th>
<th>Date</th>
</tr>
</thead>
<tbody>
<tr>
<td>Launch</td>
<td>2010 Dec 2</td>
</tr>
<tr>
<td>Earth swing-by</td>
<td>2011 Dec 3</td>
</tr>
<tr>
<td>1999JU3 arrival</td>
<td>2013 Jun 10</td>
</tr>
<tr>
<td>1999JU3 departure</td>
<td>2013 Dec 9</td>
</tr>
<tr>
<td>Earth arrival</td>
<td>2015 Dec 6</td>
</tr>
</tbody>
</table>
maneuvers to enlarge the eccentricity. They are the decelerating maneuver around the aphelion, and the accelerating maneuver around the perihelion (Fig. 5(a)). The 1999 JU3 transfer phase includes two segments of maneuvers. The maneuver around the perihelion adjusts the orbit period to phase with the Earth, and the maneuver around the aphelion adjusts the final position to encounter with the Earth (Fig. 5(b)). The Earth return phase includes two segments of maneuvers. The maneuver around the perihelion adjusts the orbit period to phase with the Earth, and the maneuver around the aphelion adjusts the final position to encounter with the Earth (Fig. 5(c)).

What must be pointed out in the Earth return phase is that the former segment of the maneuver occupies more than half of the orbit around the first perihelion passage at full throttle. It indicates that interruption of the SEP operation is virtually not allowed in this segment, and it is supposed to be the operational risk of this sequence. It must be noted that the role of the accelerating maneuver at the first perihelion passage is to adjust the orbit period to phase with the Earth. Therefore, it cannot be substituted using an acceleration maneuver at the second perihelion approach just before Earth arrival. The total operation time of SEP for this sequence is estimated to be longer than 20,000 hours for each ion engine (weighted by the number of the ion engines operating). It largely exceeds the operation results of HAYABUSA (approximately 14,000 hours), and it must be recognized as the risk of this sequence.

Figure 7 shows the trajectory profile of the mission sequence projected on the ecliptic plane of a geocentric rotational coordinate system, where the Sun direction viewed from the Earth is fixed at $-x$ direction. Shown in the figure is the trajectory through the sequence. Displayed in the figure are the trajectory of 1999 JU3 (dark blue line), and the trajectory of the spacecraft during the cruise (red line) and during the mission phase (during the spacecraft’s stay at 1999 JU3) (orange line).

In the EDVEGA phase, the spacecraft cruises the leading side of the Earth (a distance of approx. 0.5AU at maximum). In the 1999 JU3 transfer phase, the spacecraft gradually moves to the following side of the Earth, because $a$ (in other words, the orbital period) of the transfer orbit is larger than that of the Earth. Upon arrival, the angle between the Earth and 1999 JU3 viewed from the Sun is approximately 90°. The small loop of the trajectory observed around the point of arrival indicates that 1999 JU3 is at its perihelion (the other small loop around the point (–2.0, 0.0) as well).

What must be pointed out here is that the angle between the Sun and the Earth viewed from the spacecraft during

\[\text{Fig. 6. Mass and thrust profile of a nominal mission sequence.}\]
the mission phase is a minimum of 18° (at the end of the mission phase). This angle represents the distance between the direction SAP should be pointing and that which HGA should be pointing (i.e., the directions of the Sun and Earth viewed from the spacecraft, respectively). This angle was small in the case of the mission phase of HAYABUSA (in particular, during the descent to ITOKAWA for sampling), enabling the compatible use of SAP and HGA, which were fixed to the spacecraft body pointing in the same direction. This difference may require an alteration in the spacecraft design or the operation procedures of HAYABUSA2 as compared to those of HAYABUSA. As mentioned before, in the case of HAYABUSA2, basically the design of the spacecraft follows that of HAYABUSA as it is, in order to shorten development time. Therefore, this difference must be recognized as the risk of this sequence. At departure, the angle between the Earth and 1999 JU3 viewed from the Sun is approximately 140°, and it is before the conjunction with the Earth (again, the mission phase of HAYABUSA was around the conjunction).

In the Earth return phase, the spacecraft gradually approaches the Earth from the leading side and arrives at the Earth. At the beginning of the Earth return phase, the angle between the spacecraft and the Earth viewed from the Sun is approximately 220° (note that the angle here is measured clockwise from the spacecraft to the Earth), which is the distance required to catch up with the Earth during the two-year Earth return phase.

As a result of the analysis, it is shown that the sample retrieval mission from 1999 JU3 is feasible with the nominal mission sequence shown in Table 6. However, as is mentioned in the explanations, several drawbacks become clear through the analysis. The two major drawbacks of the nominal sequence are summarized as follows.

1) The total operation time of SEP largely exceeds the operation results of HAYABUSA. Additionally, the operation duty of the SEP during the return trip is so high that the interruption of SEP operation is virtually not allowed.

2) The angle between the Sun and Earth viewed from the spacecraft during the mission phase is not as small as the case of HAYABUSA, which may require altering the spacecraft design or operation procedures from those used for HAYABUSA.

Investigated in the next section is an alternative sequence to overcome these drawbacks.

3.4. Extension of the return trip

In order to overcome the drawbacks of the nominal mission sequence shown in the previous section, an alternative mission sequence is investigated in this section. The point of the new sequence is to postpone the Earth arrival for one year compared to the nominal mission sequence, which is supposed and seems to contribute mainly to relieve the duty of the SEP in the return trip (the first drawback listed at the end of the previous section).

However, it becomes clear that this leads to overcoming the remaining drawback as a result. That is to say, the total operation time of the SEP is reduced, and the angle between the Sun and Earth viewed from the spacecraft during the mission phase becomes small. The new sequence for the mission is shown in Table 7. Although the mission sequence for the arrival at 1999 JU3 is the same as that of the nominal sequence in Table 6, the arrival at Earth is postponed for one year compared to the nominal sequence. Additionally, owing to the relief of the thrust profile in the return trip, it become possible to postpone the departure from 1999 JU3 for three months, which leads to extending the mission phase (the stay at 1999 JU3) as well.

Figure 8 shows the thrust profiles for the return trip of the alternative mission sequence (red solid line). Also displayed in the figure are the thrust profile of the nominal sequence (brown solid line) and the profile of the upper limit of the thrust for both sequences (dashed lines). First, it can be seen that the duty of the SEP in the first segment of the maneuver (i.e., the maneuver around the perihelion) is obviously relieved in the case of the new sequence. It is

<table>
<thead>
<tr>
<th>Event</th>
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</tr>
</thead>
<tbody>
<tr>
<td>Launch</td>
<td>2010 Dec. 2</td>
</tr>
<tr>
<td>Earth swing-by</td>
<td>2011 Dec. 3</td>
</tr>
<tr>
<td>1999 JU3 arrival</td>
<td>2013 Jun. 10</td>
</tr>
<tr>
<td>1999 JU3 departure</td>
<td>2014 Mar. 10</td>
</tr>
<tr>
<td>Earth arrival</td>
<td>2016 Dec. 3</td>
</tr>
</tbody>
</table>
also true for the second segment of the maneuver (i.e., the decelerating maneuver around the aphelion), the duration for which is far shorter than that of the nominal sequence (note that the time of the second maneuver in the new sequence shifts to the second aphelion passage). What must be pointed out is that, the relief of the duty in the new sequence does not result from extending the thrusting time (with the total amount of the velocity increment conserved), but results from reducing the total amount of the velocity increment. The latter fact becomes clear by comparing the area below the thrust profile of the two sequences. The area represents the accumulation of the acceleration by SEP, which is equivalent to the total amount of the velocity increment. Obviously, the area is drastically reduced in the new sequence compared to the nominal sequence. Reducing the total amount of the velocity increment leads to reduction in the total operation time of the SEP. It is reduced to approximately 15,000 hours, which is not much higher than the operation results of HAYABUSA.

Figure 9 shows the trajectory profile of the Earth return phase of the new mission sequence. The definitions of the items displayed in the figure are the same as those for Fig. 5. The major difference from the trajectory profile of the nominal sequence (Fig. 5(c)) is the size (a) of the return orbit. The return orbit of the new sequence is close to the orbit of 1999 JU3, whereas the return orbit of the nominal sequence is a little outside the orbit of 1999 JU3. This fact results in reducing the velocity increment in the first segment of the maneuver (Fig. 8), which is smaller in the case that the required orbit transfer (from the orbit of 1999 JU3 to the return orbit) is smaller. The reason of the alteration in the return orbit is explained as follows: In the nominal sequence, the elongation between the Earth and spacecraft in the beginning of the Earth return phase is 220°. To reduce the elongation to zero during the two-year journey, the Earth must catch up with the spacecraft at the rate of 110°/year on average. Considering the orbit rate of the Earth (360°/year), the orbit rate of the spacecraft is required to be 250°/year on average, which is equivalent to a of the return orbit to be 1.28AU. That is to say, in order to return to Earth earlier, the spacecraft must catch up with the Earth faster, which requires the orbit rate of the spacecraft to be slower, and that results in a larger return orbit. On the contrary, in the new sequence, the elongation between the Earth and spacecraft at the beginning of the Earth return phase is 180°. To reduce the elongation to zero during two-year, nine-month journey, the Earth must catch up with the spacecraft at the rate of 65°/year on average. Considering the orbit rate of the Earth, the orbit rate of the spacecraft is required to be 295°/year on average, which is equivalent to a of the return orbit to be 1.14AU, the value of which is close to a of 1999 JU3 (1.19AU).

Figure 10 shows the trajectory profile of the new mission sequence projected on the ecliptic plane of a geocentric rotational coordinate system. The definitions of the items displayed in the figure are the same as those of Fig. 7. The major difference from the trajectory profile of the nominal sequence (Fig. 7) is the extension of the mission phase. As mentioned before, the relief of the thrust profile in the return trip enables the postponement of departure from 1999 JU3 by three months. It is expressed in the figure as the extension of the mission phase (thick orange line), which reaches the conjunction point with the Earth (on the x axis). It indicates that the angle between the Sun and Earth viewed from the spacecraft becomes small at the end of the mission phase. This is the same situation as the mission phase of HAYABUSA (in particular, at the time of descent to ITOKAWA for sampling), which enables HAYABUSA2 to follow the spacecraft design or operation procedures of HAYABUSA as they are. The trajectory profile of the Earth return phase is similar to that of the nominal sequence (Fig. 7). One small difference appearing in the new figure is the small loop in the middle of the Earth return trajectory, which expresses the second perihelion passage in the middle of the Earth return phase.

4. Conclusion

Following the results of HAYABUSA, the Japanese asteroid explorer, JAXA has started studying the next
asteroid exploration mission. The mission studied gives priority on the “early” retrieval of samples from an asteroid with a primitive composition. To realize this objective, the design of the spacecraft basically follows that of HAYABUSA as it is in order to shorten the development time. The spacecraft is called HAYABUSA2, and it is planned to be launched early in the next decade. Based on the situation, reported in this paper are the results of a mission analysis for an asteroid exploration mission. The objective of the mission analysis is to design a mission sequence that has launch window early in the next decade, can be feasibly done utilizing a HAYABUSA-type spacecraft, and the target asteroid for which complies with the science objectives.

As a result of the analysis, asteroid 1999 JU3 is selected as the target asteroid and the nominal mission sequence is constructed. That is, the sample retrieval from an asteroid with a primitive composition is feasible utilizing a HAYABUSA-type spacecraft with the launch early in the next decade. This is the first major conclusion of this paper.

However, several drawbacks of the nominal mission sequence became clear through the analysis, and an alternative mission sequence was investigated to overcome the drawbacks. The point of the new sequence is to postpone Earth arrival for one year compared to the nominal mission sequence. The results of the analysis show that the drawbacks in the nominal mission sequence are overcome in the new sequence. That is, this alternative is feasible utilizing a HAYABUSA-type spacecraft, and the target asteroid for which complies with the science objectives.

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References