Mission Analysis of HAYABUSA2, the next Japanese Asteroid Exploration Mission

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Abstract

Reported in this paper is the result of the mission analysis of the asteroid explorer mission. Following the results of HAYABUSA, the Japanese asteroid explorer, JAXA has started the study of the next asteroid exploration mission. The mission now under study gives priority on "early" achievement of the sample return from an asteroid with primitive composition. Therefore, the design of the spacecraft follows that of HAYABUSA basically as it is, and the spacecraft is planned to be launched in early 2010s. The objective of the mission analysis is to design a mission sequence, which has launch window in early 2010s, which is feasible by a HAYABUSA-type spacecraft, and whose target asteroid complies with the science objective. The result includes the selection of the target asteroid, the design of nominal mission sequence, and some back up plans.

次期小惑星探査ミッション「はやぶさ 2」のミッション解析

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摘要

「はやぶさ」の ITOKAWA 接近運用の結果を受け、次の小天体探査ミッションについては、他国に先駆けて、できるだけ早い時期に小天体からのサンプルリターンミッションを再実行する可能性を検討することとなった。ミッションは「はやぶさ2」と呼ばれ、探査機は開発期間短縮のために「はやぶさ」の設計をほぼ流用することになる。これにより、開発計画上は2010年度中に探査機を打ち上げられる見通しが立った。以上を受けて、本発表では「はやぶさ2」のミッションデザインの結果を報告する。2010 年に打上げウィンドウがあり、「はやぶさ」同等の性能の探査機により実現可能な、科学的目的に合致する天体からのサンプルリターン、が「はやぶさ2」のミッションデザインの目標となる。発表では、探査対象天体の選定、ノミナルシーケンスの軌道計画、いくつかのバックアッププランについて報告する。

1. Introduction

The Japanese asteroid explorer HAYABUSA launched in 2003 arrived at its target asteroid ITOKAWA in September, 2005. HAYABUSA has made amount of scientific discoveries and technological achievements during its stay at ITOKAWA, and left ITOKAWA in December, 2005. HAYABUSA is on its way back to the Earth. Under this situation, the study of the next asteroid exploration mission has been performed supposing a launch in the first half of the 2010s[1, 2]. The mission studied so far gave priority on the enhancement of the scientific output, and assumed the spacecraft with the larger size and the enhanced capability compared to HAYABUSA. The examples of the missions studied are, the sample return from the multiple asteroids, or from the further asteroids.

However, in actual, because of the programming problems, HAYABUSA failed to get the sample of ITOKAWA in the way originally planned (though there still remains a slight possibility of capturing some fragments of the surface aboard). Following this result, the mission now under study shifts its priority on “early” achievement of the sample return from an asteroid. To realize this objective, the design of the spacecraft follows that of HAYABUSA basically as it is in order to shorten its development time. The spacecraft is called HAYABUSA2, and it is planned to be launched in early 2010s. From the point of the scientific objective, the asteroid with the primitive composition (for example, C-type in spectral type) is supposed to be preferable (ITOKAWA is S-type). Consequently, the objective of the mission analysis is to design a mission sequence, that has launch window in early 2010s, that is feasible by a HAYABUSA-type spacecraft, and whose target asteroid complies with the science objective.

Discussed in the following sections are the selection of target asteroid, the design of nominal mission sequence, and some back up plans.

2. Target Asteroid Selection

Firstly discussed is the target asteroid selection. The target asteroids are selected considering both the scientific objective and the engineering feasibility of the mission. The sample return missions to the various asteroids, assuming the launch in early 2010s, are investigated and evaluated. Original candidates set includes the near Earth asteroids with the absolute magnitude smaller than 20 (i.e. the near earth asteroids not smaller than ITOKAWA).

In order to realize the global survey of the targets and the sequences, the following approach was adopted. The mission sequence is constructed as the series of Keplerian orbits connected with impulsive velocity changes. The dynamical feasibility of the sequence is evaluated quantitatively by the total required Δv to complete the mission. Although the total Δv is required to be compensated for the case of using the electric propulsion, to compare the dynamical feasibility of the sequences, the estimation by the impulsive maneuvers is sufficient. The constructed sequence defines the asteroid to be explored and the opportunity of the mission. Therefore, the sequence is sufficiently definitive to restrict the framework and the scope of the more detailed analysis, and at the same time, gives good initial estimate for it. The global survey of the targets and sequences in the space of the combinations of the ballistic trajectories successfully made clear the mission candidates suited our objective. The detail of the method is described in Ref. 3.

2.1 Results of Sequence Construction

Figure 1 shows the mission sequences obtained which can be achieved with relatively small Δv. 23 sequences to 15 asteroids are shown in the figure with the milestones of the major events in the sequences. The sequences shown in the figure take into account the usage of Electric Delta-VEGA[4] (EDVEGA) technique which was used in HAYABUSA. In case that EDVEGA is applied to the sequence, the spacecraft is launched 1 year (or 1.4 year) prior to the round trip to the asteroids, increases the relative velocity to the earth by the electric propulsion, and directs the velocity by the earth swing-by to transfer to the target asteroid.

Within the asteroids appear in the figure, 1999 JU3, 1989 UQ, 2003 SD220 are supposed to be with the primitive composition and meet our scientific objective.

2.2 Target Asteroid Selection

Within the three asteroids which meet our scientific objective, 1989 UQ and 2003 SD220 are Aten asteroids (the earth

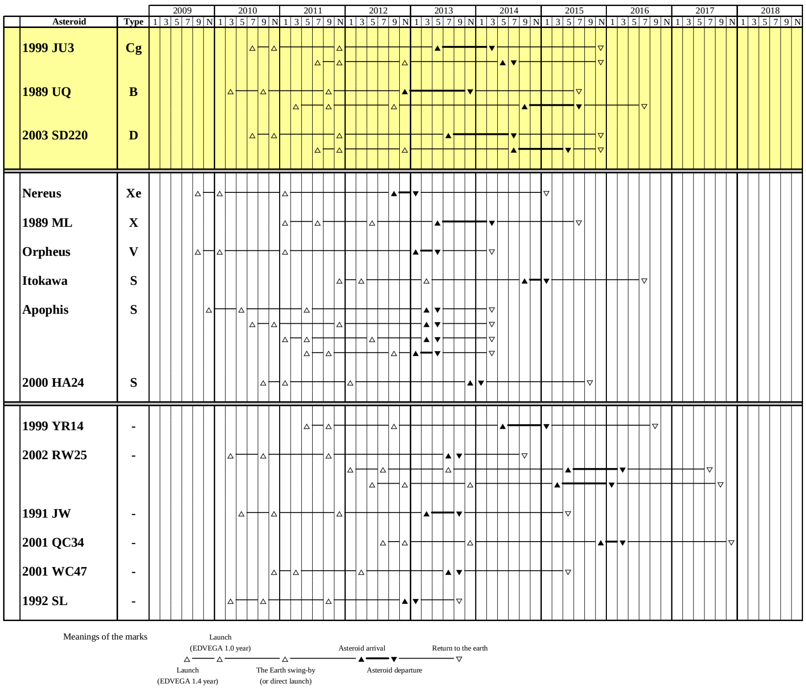


Figure 1 Near Earth Asteroid Sample Return Mission Sequences in Early 2010s

crossing asteroid whose semi major axis is smaller than 1AU). On the other hand, 1999 JU3 belongs to Apollo asteroids (the earth crossing asteroid whose semi major axis is greater than 1AU), whose orbit shape is similar to that of ITOKAWA, the target asteroid of HAYABUSA.

The exploration of the two Aten asteroids has two features which should be recognized as the difference from the exploration of ITOKAWA performed by HAYABUSA. They are,

- The minimum distance from the sun during the mission is approximately 0.7AU, which leads to the severe thermal condition in the hot case. The minimum distance from the sun in case of HAYABUSA is approximately 1AU.
- The angle between the earth and the sun viewed from the spacecraft is possible to be large, which disables the sufficient power supply and the high rate communication link to be compatible simultaneously with the fixed SAP (Solar Array Paddle) and HGA (High Gain Antenna). The angle is basically not so large in case of HAYABUSA, which mainly operates in the outside of the ecliptic.

Either problem is possible to be overcome by the alteration of the spacecraft design or the operation procedure. However, in case of HAYABUSA2, the design of the spacecraft follows that of HAYABUSA basically as it is, in order to shorten its development time. Therefore, there is less merit in choosing Aten asteroids as the mission targets 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 return mission from 1999 JU3 is investigated in detail in the following section.

Table 1 1999 JU3 Sample Return Mission Sequence
(Result of Preliminary Analysis)

Event	Case 1	Case 2
Launch (EDVEGA 1.4 year)	2010 Jul. 10	2011 Jul. 16
(EDVEGA 1.0 year)	2010 Dec. 3	2011 Dec. 8
(Direct Launch)	2011 Dec. 3	2012 Dec. 8
Earth Swing-by (in case of EDVEGA)	2011 Dec. 3	2012 Dec. 8
1999 JU3 Arrival	2013 May 20	2014 May 25
1999 JU3 Departure	2014 Apr. 7	2014 Jul. 27
Earth Arrival	2015 Dec. 12	2015 Dec. 5

3. Mission Analysis of 1999 JU3 Exploration

The two mission sequences to 1999 JU3 shown in Figure 1 are detailed in Table 1. 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 in the two cases, which leads to the shorter stay at the asteroid in Case 2. The total required Δv to complete the mission is not so different between the two cases.

From the view point of the development schedule of the spacecraft, it is supposed to be possible to complete the development and launch the spacecraft in 2010. Therefore, considering the demerit of the short stay in Case2, Case 1 is selected from the two cases. Next, as to the option of the launch schedule, considering the merits of using EDVEGA and the interference of the launch operation with PLANET-C (Japanese Venus explorer, which is planned to be launched in May, 2010), the option of “EDVEGA 1.0 year” is selected from the three options. Consequently, “EDVEGA 1.0 year” option of Case 1 is selected as the baseline sequence of the mission.

Following the result of the selection above, more detailed mission analysis is performed assuming the usage of the electric propulsion. In the analysis, the trajectory is designed by each arc which composes the whole sequence. The arc indicates the part of the trajectory which covers the transfer between the bodies. Each arc (and the thrust control profile) is designed based on the optimal control. The detail of the method is described in Ref. 5.

3.1 Nominal Mission Sequence

Shown in the followings are the results of the mission analysis as to the sample return mission from 1999 JU3. Firstly, the nominal sequence of the mission is shown in

Table 2 1999 JU3 Sample Return Mission Sequence
(Result of the Detailed Analysis)

Event	Date
Launch	2010 Nov. 27
Earth Swing-by	2011 Nov. 27
1999 JU3 Arrival	2013 May 27
1999 JU3 Departure	2013 Dec. 6
Earth Arrival	2015 Dec. 6

Table 2. Note that the dates of the events are modified from those shown in Table 1, since the dates are optimized for the usage of the electric propulsion.

Figure 2 shows the trajectory profile of the mission sequence. In Figure 2(a) to (c), the trajectory is projected on the ecliptic plane of heliocentric inertial coordinate system. The profile is divided into three parts,

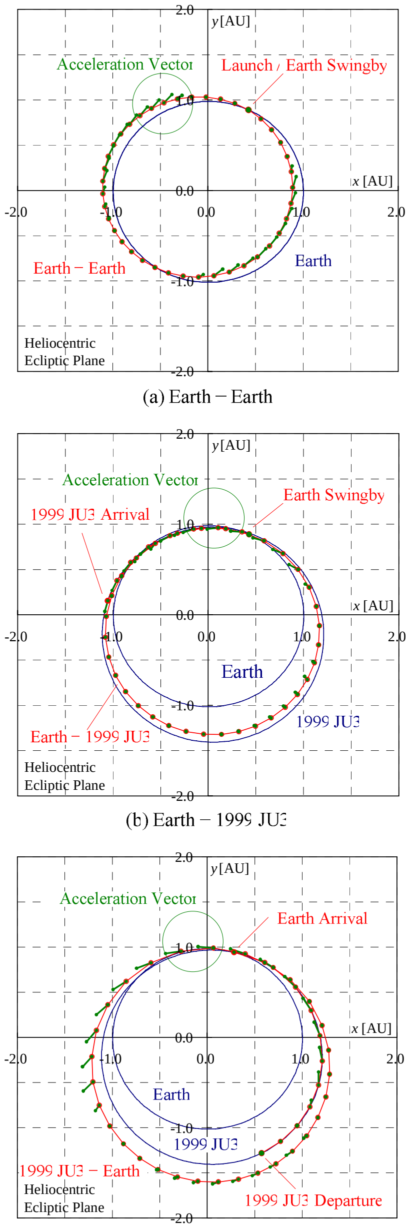
(a) The earth departure to the earth swing-by.
(b) The earth swing-by to 1999 JU3 arrival.
(c) 1999 JU3 departure to the earth arrival

Displayed in the figures are the orbit of the earth and 1999 JU3 (dark thin line), the transfer orbit (light thick line), and the control acceleration vector (sticks at the nodes).

Shown in Figure 2(a) is the trajectory profile in EDVEGA[4] phase. EDVEGA is the orbit control technique which was used in HAYABUSA as well. The spacecraft is launched and directly injected into the interplanetary orbit. The relative velocity to the earth at the injection is 3.12km/s, which corresponds with that of HAYABUSA. The injected orbit is approximately synchronous with the earth orbit, but has small eccentricity. The decelerating maneuver around the aphelion and the accelerating maneuver around the perihelion during one year cruise enlarge the eccentricity of the orbit, which leads to the increase of the relative velocity to the earth at the re-encounter. The relative velocity, which is increased to 4.20km/s, is redirected by the earth swing-by and the spacecraft is injected into the transfer orbit to 1999 JU3. Shown in Figure 2(b) is the trajectory profile in 1999 JU3 transfer phase. After the earth swing-by, the spacecraft is injected into the transfer orbit to 1999 JU3, which is close to the orbit of 1999 JU3. The accelerating maneuver around the perihelion and the orbit plane change maneuver around the aphelion (the acceleration vector of which is not obviously projected on the ecliptic plane) during the cruise for one and a half year adjust the transfer orbit to the orbit of 1999 JU3, which results in the spacecraft's rendezvous with 1999 JU3. After a half year stay at 1999 JU3 for the scientific observation and sampling, the spacecraft departs from 1999 JU3 and return to the earth. Shown in Figure 2(c) is the trajectory profile in the earth return phase. After the departure from 1999 JU3, the accelerating maneuver around the perihelion enlarges the semi major axis of the return orbit. It seems strange at the first glance to go outside of the departure orbit in order to return to the inner destination (i.e. the 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 once transfer to the outer orbit, so that the earth can catch up with the spacecraft within the limited time. Before the arrival at the earth, the decelerating maneuver around the aphelion adjusts the return orbit to encounter the earth. Finally, as a result of the return cruise for two years, the spacecraft arrives at the earth. The relative velocity to the earth at the arrival is 5.16km/s, and the sample retrieval capsule released from the spacecraft directly reenters to the atmosphere.

Figure 3 shows the mass and the thrust profile of the mission sequence. Displayed in the figure are the mass profile (dark solid line, left axis) and the thrust profile (light solid line, right axis). Dashed line denotes the profile of the upper limit of the thrust which takes into account the distance from the sun. The duration where the light solid line is close to the dashed line indicates that the electric propulsion is operated in almost full throttle.

EDVEGA phase from the launch to the earth swing-by includes two segments of maneuvers. The former is the decelerating maneuver around the aphelion, and the latter is the accelerating maneuver around the perihelion (Figure 2(a)). As a result of the optimization, both maneuvers are almost full throttle around their respective optimal points. 1999 JU3 transfer phase from the earth swing-by to 1999 JU3 arrival includes three segments of maneuvers. The first and the second maneuvers are the orbit plane change maneuver around the node, and the third maneuver is the accelerating maneuver around the perihelion (Figure 2(b)). As a result of the



(c) 1999 JU3 – Earth

Figure 2 Trajectory Profile of Nominal Mission Sequence
(Inertial Coordinate System)

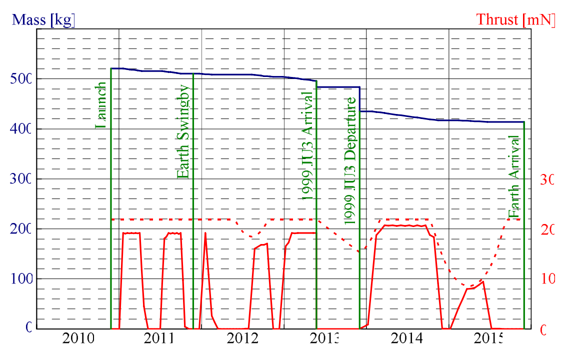


Figure 3 Mass and Thrust Profile of Nominal Mission Sequence

optimization, three maneuvers are almost full throttle around their respective optimal points. At 1999 JU3 arrival, the mass of the spacecraft drops by 12kg, which counts the consumption of the chemical propellant during the EDVEGA phase and 1999 JU3 transfer phase. The spacecraft stays at 1999 JU3 for a half year for the scientific observation and sampling. There is no control acceleration or mass decrease during the stay. However, the mass of the spacecraft drops by 48kg at the end of the stay, which counts the consumption of the chemical propellant during the stay at 1999 JU3. The earth return phase from 1999 JU3 departure to the earth arrival includes two segments of maneuvers. The former is the accelerating maneuver around the perihelion, and the latter is the decelerating maneuver around the aphelion (Figure 2(c)). As a result of the optimization, both maneuvers are almost full throttle around their respective optimal points. What must be pointed out here is that the former segment of the maneuver occupies beyond a half of the orbit around the first perihelion passage with almost full throttle. It indicates that the interruption of the electric propulsion 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 in order that the spacecraft is phased with the earth. Therefore, it cannot be substituted for by the acceleration maneuver at the second perihelion passage just before the earth arrival. Finally, at the earth arrival, the mass of the spacecraft drops by 12kg, which counts the consumption of the chemical propellant during the earth return phase. The total operation time of the electric propulsion for this sequence is estimated to be close to 20000 hours for each ion engine (weighted by the number of the ion engines operating). It largely exceeds the operation result of HAYABUSA (approximately 12000 hours), and it must be recognized as the risk of this sequence.

Figure 4 shows the trajectory profile of the mission sequence projected on the ecliptic plane of the 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 thin line), the trajectory of the spacecraft during the cruise (light thin line) and during the mission phase (during the spacecraft's stay at 1999 JU3) (light thick line).

In the EDVEGA phase from the launch to the earth swing-by, the spacecraft cruises the following side of the earth (about 0.5AU distance in maximum). As a result of the earth swing-by, the spacecraft is injected into the transfer orbit to 1999 JU3. In 1999 JU3 transfer phase, the spacecraft gradually moves to the following side of the earth, because the semi major axis (in other words, the orbital period) of the transfer orbit is larger than that of the earth. After the cruise of one and a half year, the spacecraft arrives at 1999 JU3. At the arrival, the angle between the earth and 1999 JU3 viewed from the sun is approximately 90 deg. The small loop of the trajectory observed around the point of arrival indicates that 1999 JU3 is at its perihelion (the other small loops around the earth departure and around the point (-2.0, 0.0) as well). For a half an year, the spacecraft stays at 1999 JU3 for the scientific observation and sampling, which is called mission phase. What must be pointed out here is that the angle between the sun and the earth viewed from the spacecraft during the mission phase is 18 deg. in minimum (at the end of the mission phase). This angle represents the distance between the direction in which SAP should point at and that in which HGA should point at (i.e. the direction of the sun and the earth viewed from the spacecraft respectively). This angle was small in case of the mission phase of HAYABUSA (in particular at the descent to ITOKAWA for the sampling), which enabled the compatible use of SAP and HGA which were fixed to the spacecraft body pointing at the same direction. This difference may cause the alteration of the spacecraft design or the operation procedure of HAYABUSA2 from those of HAYABUSA. As mentioned before, in case of HAYABUSA2, the design of the spacecraft follows that of HAYABUSA basically as it is, in order to shorten its development time. Therefore, this difference must be recognized as the risk of this sequence. Following the completion of the mission phase, the spacecrafts departs from 1999 JU3 and return to the earth. At the departure, the angle between the earth and 1999 JU3 viewed from the sun is approximately 140 deg., and it is before the conjunction with the earth (again, the mission phase of HAYABUSA was around the conjunction). In the earth return phase, following the perihelion passage (the small loop around the conjunction), the spacecraft gradually approaches the earth from the leading side and arrives at the earth after the cruise for two years. At the beginning of the earth return phase, the angle between the spacecraft and the earth viewed from the sun is approximately 220 deg. (note that the angle here is measured clockwise from the spacecraft to the earth), which is the distance to be caught up with by the earth during the two years of the earth return phase.

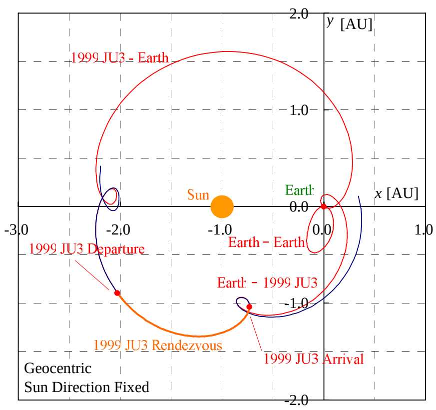


Figure 4 Trajectory Profile of Nominal Mission Sequence (Rotational Coordinate System)

As a result of the analysis, it is shown that the sample return mission from 1999 JU3 is feasible with the nominal mission sequence shown in Table 2. However, as is mentioned in the explanations, the several drawbacks become clear through the analysis. The two major drawbacks of the nominal sequence are summarized as follows.

- The total operation time of the electric propulsion largely exceeds the operation result of HAYABUSA. Additionally, the operation duty of the electric propulsion in the return trip is so high that the interruption of the electric propulsion operation is virtually not allowed.
- The angle between the sun and the earth viewed from the spacecraft during the mission phase is not so small as the case of HAYABUSA, which may cause the alteration of the spacecraft design or the operation procedure from those of HAYABUSA.

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

3.2 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

Table 3 1999 JU3 Sample Return Mission Sequence
(Extension of Return Trip)

Event	Date
Launch	2010 Nov. 27
Earth Swing-by	2011 Nov. 27
1999 JU3 Arrival	2013 May 27
1999 JU3 Departure	2014 Mar. 6
Earth Arrival	2016 Dec. 6

the electric propulsion in the return trip (the first drawback listed on the end of the previous section). However, it becomes clear that it leads to overcome the remaining drawbacks in the result. That is to say, the total operation time of the electric propulsion is reduced, and the angle between the sun and the earth viewed from the spacecraft during the mission phase become small. The new sequence of the mission is shown in Table 3. Although the mission sequence by the arrival at 1999 JU3 is the same as that of the nominal sequence in Table 2, the arrival at the 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 the extension of the mission phase (the stay at 1999 JU3) as well.

Figure 5 shows the thrust profiles in the return trip of the nominal sequence and the new sequence. Displayed in the figure are the thrust profile of the nominal sequence (light solid line), the thrust profile of the new sequence (dark solid line), and the profile of the upper limit of the thrust of the both sequences (dashed lines). Firstly observed is that the duty of the electric propulsion in the first segment of the maneuver (i.e. the maneuver around the perihelion) is obviously relieved in case of the new sequence. It is also true for the second segment of the maneuver (i.e. the decelerating maneuver around the aphelion), whose thrust level is far lower than its upper limit (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 the extension of the thrusting time (with the total amount of the velocity increment conserved), but results from the reduction of 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 the electric propulsion, which is equivalent to the total amount of the velocity increment. Obviously, the area reduces drastically in the new sequence compared to the nominal sequence. The reduction of the total amount of the velocity increment leads to the reduction in the total operation time of the electric propulsion. It is reduced to approximately 13000 hours, which is comparable with the operation result of HAYABUSA.

Figure 6 shows the trajectory profile of the earth return phase of the new mission sequence. The trajectory is projected on the ecliptic plane of heliocentric inertial coordinate

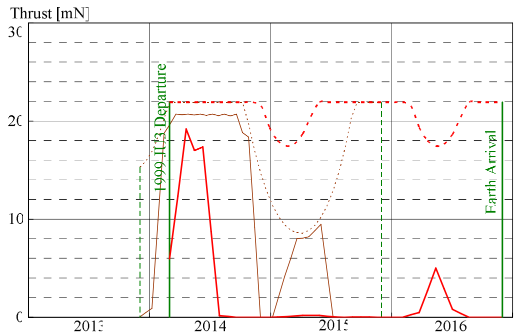


Figure 5 Thrust Profile of Mission Sequence
(Extension of Return Trip)

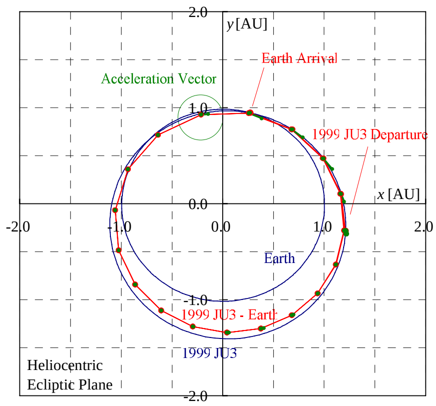


Figure 6 Trajectory Profile of Mission Sequence
(Extension of Return Trip)
(Inertial Coordinate System)

system. Displayed in the figure are the orbit of the earth and 1999 JU3 (dark thin line), the transfer orbit (light thick line), and the control acceleration vector (sticks at the nodes).

The major difference from the trajectory profile of the nominal sequence (Figure 2(c)) is the size (the semi major axis) 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 outer the orbit of 1999 JU3. This fact results in the reduction of the velocity increment in the first segment of the maneuver (Figure 4), which is smaller in case that the required orbit transfer (from the orbit of 1999 JU3 to the return orbit) is smaller. The reason of the alteration of the return orbit can be explained as follows. In the nominal sequence, the elongation between the earth and the spacecraft in the beginning of the earth return phase is 220deg. To reduce the elongation to zero during the cruise of two years, the earth must catch up with the spacecraft in the rate of 110deg./year in average. Considering the orbit rate of the earth (360deg./year), the orbit rate of the spacecraft is required to be 250deg./year in average, which is equivalent to the semi major axis of the return orbit to be 1.28AU. That is to say, in order to return to the earth earlier, the spacecraft must be caught up with by the earth faster, which requires the orbit rate of the spacecraft to be slower, that results in the larger return orbit. On the contrary, in the new sequence, the elongation between the earth and the spacecraft in the beginning of the earth return phase is 180deg. To reduce the elongation to zero during the cruise of two years and nine months, the earth must catch up with the spacecraft in the rate of 65deg./year in average. Considering the orbit rate of the earth, the orbit rate of the spacecraft is required to be 295deg./year in average, which is equivalent to the semi major axis of the return orbit to be 1.14AU, which value close to the semi major axis of 1999 JU3 (1.19AU).

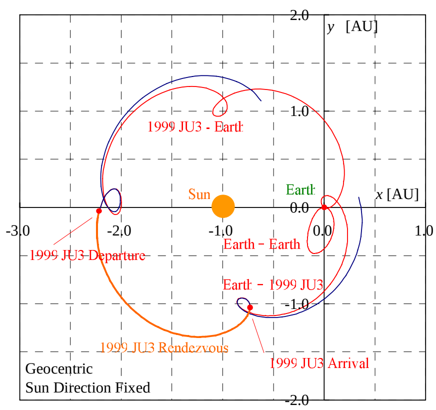


Figure 7 Trajectory Profile of Mission Sequence
(Extension of Return Trip)
(Rotational Coordinate System)

Figure 7 shows the trajectory profile of the new mission sequence projected on the ecliptic plane of the 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 thin line), the trajectory of the spacecraft during the cruise (light thin line) and during the mission phase (during the spacecraft's stay at 1999 JU3) (light thick line).

The major difference from the trajectory profile of the nominal sequence (Figure 4) is the extension of the mission phase. As mentioned before, the relief of the thrust profile in the return trip enables the postponement of the departure from 1999 JU3 by three months. It is expressed in the figure as the extension of the mission phase (light thick line), which reaches the conjunction point with the earth (on *x* axis). It indicates that the angle between the sun and the earth viewed from the spacecraft become small in the end of the mission phase. This is the same situation with the mission phase of HAYABUSA (in particular at the descent to ITOKAWA for sampling), which enables HAYABUSA2 to follow the spacecraft design or the operation procedure of HAYABUSA as it is. The trajectory profile of the earth return phase is similar to that of the nominal sequence (Figure 4). One small difference appeared 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 the study of the next asteroid exploration mission. The mission now under study gives priority on "early" achievement of the sample return from an asteroid with primitive composition. To realize this objective, the design of the spacecraft follows that of HAYABUSA basically as it is in order to shorten its development time. The spacecraft is called HAYABUSA2, and it is planned to be launched in early 2010s. Based on the situation, reported in this paper is the result of the mission analysis of the asteroid explorer mission. The objective of the mission analysis is to design a mission sequence, which has launch window in early 2010s, which is feasible by a HAYABUSA-type spacecraft, and whose target asteroid complies with the science objective. As a result of the investigation, the asteroid 1999 JU3 is selected as the target asteroid and the nominal mission sequence is constructed. However, several drawbacks of the nominal mission sequence become clear through the analysis, and an alternative mission sequence is investigated to overcome the drawbacks. The point of the new sequence is to postpone the earth arrival for one year compared to the nominal mission sequence. As a result of the investigation, the feasibility of the backup plan is confirmed.

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Appendix Mission to"4015 Wilson-Harrington"

A mission following HAYABUSA2 is also investigated, and the result is briefly introduced here. The science interests move to the more primitive bodies, and the target selected here is 4015 Wilson-Harrington (abbreviated as *WH* hereafter), which is supposed as CAT (Comet Asteroid Transfer). *WH* has a unique orbit characteristic as a body in this category, which is (relatively) easy to approach from the earth (r_p =0.99AU, i =2.8deg). A possible mission sequence is shown in Figure A1. A Solar electric propulsion system with six 30mN ion engines is supposed to be used (the power required is 7.5kW at 1AU). Assuming the spacecraft mass of 1500kg, it yields 12μG of acceleration. The sequence assumes the direct launch to the transfer orbit. In case EDVEGA method is to be applied, the launch date should be 1.4 year earlier. The spacecraft arrives at *WH* just after the perihelion. Early departure from *WH* is required for the phasing with the earth in the return trip, which results in the short mission duration (stay at *WH*) of 2 months. v_∞ required in case of the direct launch is 7km/s, and the launch capability of H2B (with an upper stage) for this energy is estimated as 1500kg (*cf.* application of EDVEGA reduces the required vinf to 3km/s, which enables the usage of H2A (with an upper stage) for the same launch mass).

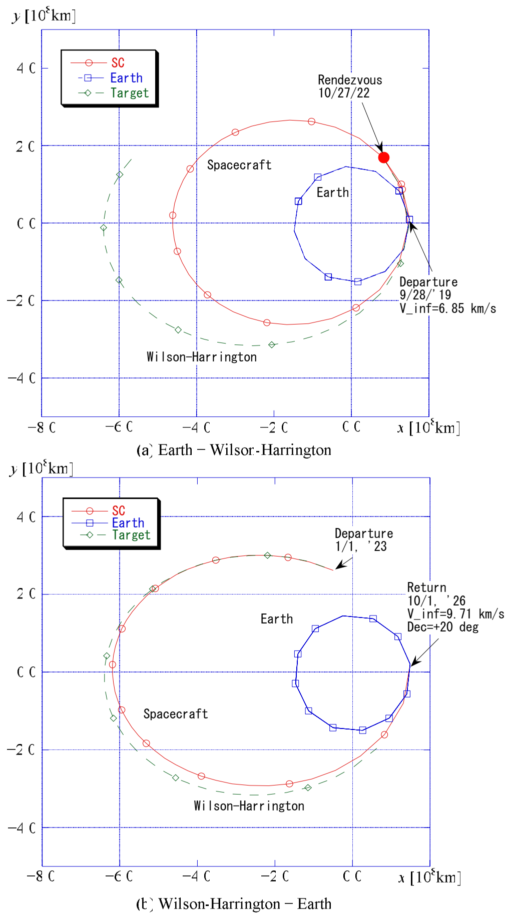

Figure A1 Trajectory Profile of Wilson-Harrington Exploration Mission (Rotational Coordinate System)