Solar Power Sail Trajectory Design for Jovian Trojan Exploration

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A solar power sail, an extended concept of a pure solar photon sail, is one of the most effective way of realizing challenging outer deep-space explorations. Thin-film solar cells attached to the surface of the spin-type solar sail membranes can generate sufficient electric power to drive high efficient ion engines even in the outer planetary region. The orbital control capability of the pure solar sail is too small to accomplish the missions within admissible period. The ion engines which provide much higher orbital control force than the pure solar sail can realize various outer space missions. JAXA has been preparing for a Jovian Trojan asteroid exploration mission via solar power sail. This paper outlines a solar power sail spacecraft under consideration and discusses the trajectory design method and results for the Jovian Trojan asteroid exploration.

Key Words: Solar Power Sail, Jovian Trojan, Low Thrust Trajectory Design

1. Introduction

A solar sail is a space yacht that uses the pressure of sunlight on a large membrane for propulsion. It offers fuel-free space travel and it is considered to be one of the most essential propulsion systems for future deep space exploration. However, the small forces exerted by photons make huge membranes necessary to obtain sufficient propulsive force, but such membranes are difficult to fabricate and deploy. A feasible size membrane provides only small propulsive force which is insufficient for rendezvous and orbiter missions and the small force makes the mission duration very long.

A solar power sail which is an extended concept of the pure solar sail is able to overcome these difficulties. It combines a solar sail with electric power generation capability and high efficient ion engines. Thin flexible solar cells attached on a sail membrane generate electric power that drives highly efficient ion engines. The solar power sail is not fuel free, but it can realize flexible and efficient orbital control capability, even at outer planetary regions of the solar system, without relying on nuclear technology. And it provides high transportation capacity as well because ultra-high specific impulse (Isp) ion engines can be derived by its abundant power supply.

The key technologies for the solar power sail were demonstrated by IKAROS, the solar power demonstration spacecraft.[1), 2)] IKAROS is the world's first deep space solar sail spacecraft launched on May 21, 2010. IKAROS is a spin type sail-craft and its sail membrane wad deployed by its centrifugal force. IKAROS deployed its sail membrane on June 10, 2010 and it was confirmed the camera images (Fig. 1). The sail deployment was the most important mission of IKAROS. Other important technologies like the thin solar cells were also confirmed.

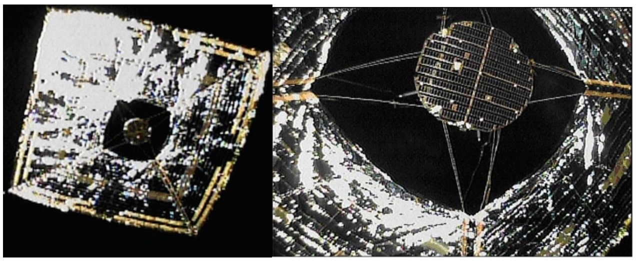

Fig. 1. IKAROS spacecraft. The sail membrane was deployed on June 10, 2010. Its deployable cameras (DCAM) took many images of the sail membrane.

Under above situations, JAXA has been preparing a next outer space exploration mission. Direct exploration of small bodies was inspired by Hayabusa,[3)] and many small body exploration missions like Hayabusa2 and OSIRIS-REx are now being carried out and studied.[4), 5)] Current target bodies of these missions are mainly near-Earth objects (NEOs) and the Mars satellites due to the restrictions of the spacecraft resources and the orbital control capability. However, more primordial bodies like P/D-type asteroids are considered to be worth of investigating. These bodies are mainly located farther away from the Sun and it is difficult for typical chemical spacecraft to explore such target bodies. Especially rendezvous/sample return type missions are difficult to realize because the high amount of delta-V is required. Our solar power sail can be a solution for such challenging explorations.

We are now studying a Jovian Trojan asteroid exploration mission via solar power sail. Jovian Trojan asteroids represent one of the few remaining frontiers in our solar system and may hold fundamental clues to its formation and evolution. We are now performing the conceptual study of the spacecraft. We anticipate its launch in the early-middle 2020's. The spacecraft is a 1500kg class single spinner. The main payload is a 100kg class small lander which collects the asteroid's sample and performs in-situ analysis.

This paper presents an outline of the spacecraft and its design trajectory. The results of the preliminary study were presented at ISTS 2015,[6] but several updates are shown in this paper. The launch year is updated and new target asteroids are searched. The low thrust trajectory is also updated by considering the new configuration of the spacecraft system. This paper also shows some optional missions like an asteroid flyby before the rendezvous with the main target and multiple rendezvous missions.

2. Mission Outline

Figure 2 shows the mission sequence of our Jovian Trojan exploration. The spacecraft is supposed to realize the journey to a Jovian Trojan asteroid around L4/L5 point of the Sun-Jupiter system by using both the Earth and Jupiter gravity assist. It is difficult to inject the spacecraft on the direct transfer orbit to Jupiter because large departure velocity is required. An Electric Delta-V Earth Gravity Assist (EDVEGA) technique is used to obtain the sufficient interplanetary speed toward Jupiter. The Jupiter gravity assist is also very important to reduce the amount of delta-V of the mission. Many Jovian Trojan asteroids have the orbital planes with a large inclination with respect to the ecliptic plane. The Jupiter swing-by can change the orbital plane of the spacecraft. And it can also change the energy of the spacecraft's orbit. As a result, the required delta-V to the Jovian Trojans can be reduced. The direct flight form the Earth to asteroid is possible only when the thrust force of the spacecraft is much larger than our ion engines. However, the required delta-V becomes very large.

After the Jupiter swing-by, the spacecraft heads to the asteroid and it rendezvous with the asteroid. During this phase, spacecraft drives the ion engines to cancel the relative velocity to the asteroid. After arriving at the asteroid, a lander is separated from the spacecraft to collect surface and underground samples and perform in-situ analysis.

We consider that the sample return is an option. In the sample return option, the lander delivers samples to the spacecraft and then spacecraft come back to the Earth by using the Jupiter gravity assist again.

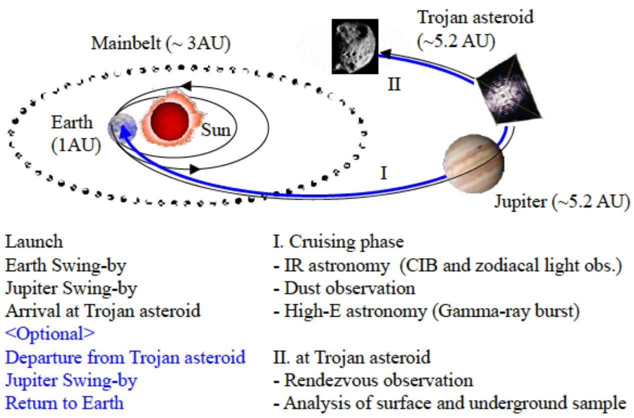

Fig. 2. Mission sequence of the Jovian Trojan exploration. Both the Earth and Jupiter gravity assists are used to reduce the amount of delta-V.

3. Spacecraft Design

3.1 Spacecraft configuration

The preliminary design of the solar power sail-craft has been conducted. Latest configuration of the spacecraft is shown in Fig. 3. It is a single spin type spacecraft which has cylindrical body. A 100 kg class lander is attached to the bottom side of the spacecraft. The power sail and its deployment mechanism are attached to the side panel of the spacecraft. The sail membrane will be deployed and maintained by centrifugal force. The spin rate is about 0.1 rpm.

The sail membrane design is shown in Fig. 4. The membrane is almost entirely covered with the thin solar cells. The length of the side is about 50 m and the weight is less than 200 kg. It can generate sufficient electric power for the spacecraft bus and the ion engine operation in the outer planetary region (5kW at 5.2 AU). It is 10 times larger than that of the solar panels of JUNO (486W at 5.2AU).[7]

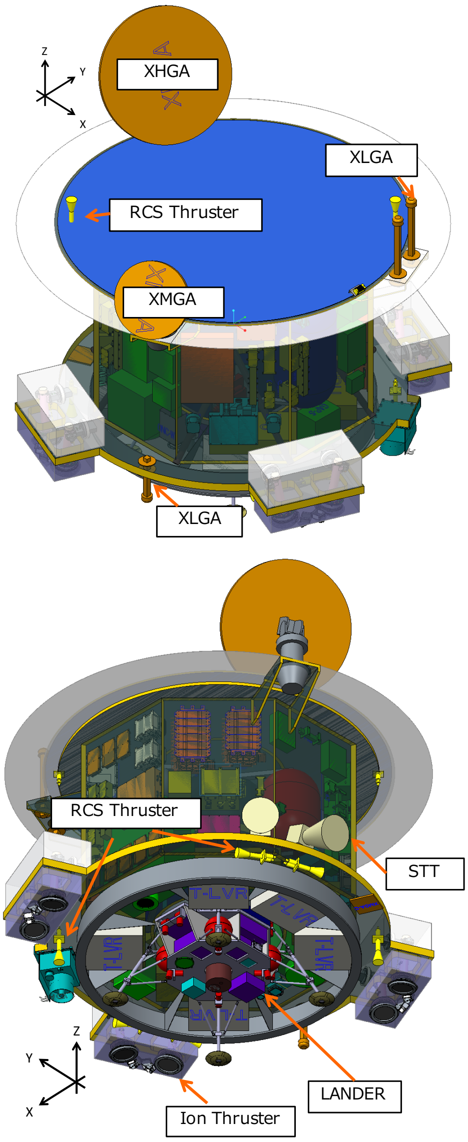

Fig. 3. Spacecraft configuration (without sail deployment mechanism). Main payload is a 100-kg lander attached to the bottom of the spacecraft.

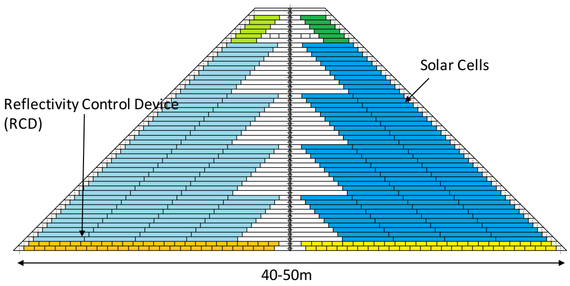

Fig. 4. Design of the solar power sail membrane (1/4 petal). Solar cells are attached to the entire surface of the membrane.

Six ion engines are mounted on the bottom side of the spacecraft. The direction of the thrust towards the sun side (+Z). We considered the ion thrusters on the opposite side in the previous study.[6] However, we have adopted the new configuration shown in Fig. 3 after evaluating the results of the trade-off study. The largest advantage of this configuration is that the time of flight (TOF) to "L4" target bodies can be shortened.

3.2 Attitude control during powered cruise

It is important to control the thrust vector according to the trajectory plan during the powered flight phase. The control of the thrust vector is performed by changing the attitude (the orientation of the spin axis) of the spacecraft because all the ion thrusters of the spacecraft are body-fixed without a gimbal mechanism. The spin axis reorientation of the solar power sail is not easy because the moment of inertial of the system is very large. The spacecraft is equipped with 8 chemical thrusters of reaction control system (RCS), but these RCS thrusters are mainly used for the position control during the proximity operations.

In our solar power sail system, the ion thrusters are used for the attitude control as well as the orbit control during the powered flight phase. Figure 5 explains the spin axis reorientation control with ion thrusters. The reorientation torque is generated by the changing the throttling level of the ion thrusters during a spin motion.

The ion thruster can generate the spin up / down torque as well. The thrust vectors of the ion thrusters are intentionally tilted by several degrees from the spin axis to generates the spin up / down torque.

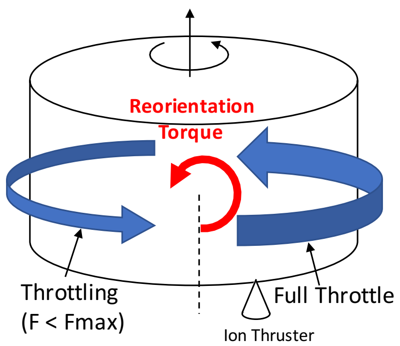

Fig. 5. Spin axis reorientation control with ion thrusters. Thrust force throttling generates the reorientation torque.

4. Target Bodies

Jovian Trojans are asteroid located both in the L4 and L5 equilibrium points of the Sun-Jupiter system. From the science perspective, there is little difference between the L4 and L5 bodies. However, the trajectory to L4 target and L5 target have large difference. As described, we consider that the sample return is an option. If we focus on the outward journey, L4 targets are more attractive than L5. The most important reason is that the TOF from Jupiter to an L4 target is shorter than that to an L5 asteroid. The sun's distance of the flight path to an L4 target is smaller and power generation is greater. It indicates that the thrust force of ion engine is larger and it is possible to reduce the TOF to the asteroid.

A ballistic analysis based on two-body dynamics was carried out to identify candidate targets. As typical Jovian Trojans have large orbital inclinations with respect to the ecliptic, the Jupiter swing-by timing can be fixed at the orbital node (Fig. 6). It means that "the Earth - Jupiter - asteroids" can be designed by considering only two parameters, 1) departure time from the Earth and 2) arrival time at the target body.

Table 1 shows the target candidates selected by the ballistic analysis. Table 2 shows the target candidates with another Jupiter swing-by year. It is very important that several candidates can be found even if we change the Jupiter swing-by year. It indicates that the we can freely choose the launch year.

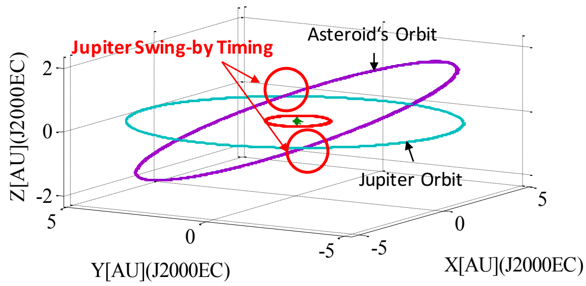

Fig. 6. Orbital node is the best timing of the Jupiter swing-by. It occurs about every six years.

Table 1. Target candidates (Jupiter swing-by: 2026).

Name	SWBY date	H*	TOF_min (J2A)[day]	dV_min (J2A)[m/s]
2001 AT33	2026/6/6	11.5	1622	1658
2001 BJ26	2026/9/20	11.9	1664	1586
2001 DY103	2026/12/22	12.0	1720	927
2007 PB9	2025/10/20	12.6	1643	1304

*) Absolute Magnitude

Table 2. Target candidates (Jupiter swing-by: 2029).

Name	SWBY date	H*	TOF_min (J2A)[day]	dV_min (J2A)[m/s]
2000 FD1	2029/11/14	11.4	1636	1350
2002 FR4	2029/3/28	12.4	1825	1664
2004 HZ11	2029/9/2	12.0	1510	875
2009 SK78	2029/5/16	12.7	1734	1304

5. Trajectory Design

5.1. Formulation

A low-thrust trajectory was designed after identifying the candidate targets. The equations of motion are as follows:

$$\ddot{\mathbf{r}} = -\frac{\mu}{r^3}\mathbf{r} + \mathbf{a}_{SRP} + \mathbf{F}/m \qquad , \qquad (1)$$
$$\dot{m} = -|\mathbf{F}|/I_{sp}g$$

where μ is the heliocentric gravitational constant, $\mathbf{r}$ is the position of the spacecraft, $\mathbf{F}$ is the orbital control force and m is the mass of spacecraft. $\mathbf{a}_{SRP}$ is the solar radiation pressure (SRP) force. We consider a flat plate model.

$$\mathbf{a}_{SRP} = -\frac{p(\mathbf{s}\cdot\mathbf{n})}{m}\left[(C_a + C_d)\mathbf{s} + \left(2(\mathbf{s}\cdot\mathbf{n})C_S + \frac{2}{3}C_d\right)\mathbf{n}\right]A, \quad (2)$$

where p is the solar pressure per unit, $\mathbf{s}$ is the unit vector from the spacecraft to the Sun, $\mathbf{n}$ is the normal vector of the sail membranes, A is the area of the sail membranes. C_a, C_d and C_s are the absorption, the diffuse reflection and the specular reflection constants, respectively. In outer planetary regions, the SRP force is negligible. However, we have to consider it especially in the EDVEGA phase. The control force $|\mathbf{F}|$ has upper limit depending on the spacecraft power generation.

$$|\mathbf{F}| \le f(P_{GEN})$$
$$P = P_0 \frac{1}{(r/AU)^2}(\mathbf{s}\cdot\mathbf{n}) \quad . \quad (3)$$

P is the electric power generation and P_0 is the maximum power generation at one astronautical unit (AU). The sun angle of the spacecraft is limited as follows:

$$\mathrm{acos}(\mathbf{s}\cdot\mathbf{n}) < \beta_{MAX} \ , \quad (4)$$

where β_{MAX} is the maximum sun angle. Two types of objective function were used in this study:

$$J = -m_f \quad (5)$$

and

$$J = t_f \, . \quad (6)$$

Eq. (5) is used to minimize delta-V. It is mainly used in the EDVEGA phase and Jupiter to Asteroid (J2A) phase with fixed arrival date. Eq. (6) is employed to minimize the TOF. It is used for the fastest trajectory design of J2A phase.

5.2. Ion engine

The spacecraft is equipped with 6 ion engine thrusters. The maximum 3 thrusters are driven at the same time. During the EDVEGA phase, the sufficient power generation is possible. It indicates that the maximum thrust force can be realized. On the other hand, the power generation is substantially limited in the outer planetary regions. For the orbital control, it is important to realize the possible largest thrust force in such regions. Figure 7 shows the throttling levels of the ion engine. The thrust force and the required power generation can be finely adjusted by changing the amount of the xenon propellant.

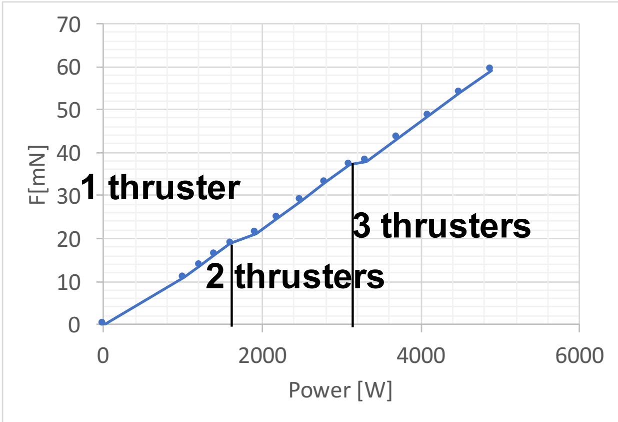

Fig. 7. Power generation vs. thrust force of the ion engine system. Many throttling levels are defined to realize as large force as possible.

5.3. Conditions

Table 3 summarizes the conditions for the low-thrust trajectory design. Thrust force and required power of the ion engine system are shown in Fig. 7. The power generation for the IES is 2600 W at 5.2 AU. Since the spacecraft is a single spinner design, the direction of the thrust is controlled by changing the attitude of the spacecraft. The sun angle should be less than 45 deg. In this paper, the trajectories for the exploration of 2009 SK78, an L4 asteroid are discussed.

Table 3. Conditions for the low-thrust trajectory design.

Spacecraft Mass @ launch	1400 kg
Spacecraft Mass @ Jupiter departue	1350 kg
Spacecraft Mass @ asteroid departure	1150 kg
IES thrust & required power	Fig. 7
IES Isp	Max 7000 sec
IES operation rate	0.8 / 0.85
Power generation for IES	2600W@5.2 AU
Attitude	Sun angle < 45 deg
Target body	2009 SK78

5.4. EDVEGA trajectory

The EDVEGA is very important to increase the departure velocity from the Earth because it is impossible to launch the spacecraft into the Jupiter transfer orbit directly. In this study, a 2-year EDVEGA is assumed. The spacecraft is accelerated by the ion engines around the aphelion and the Earth encounter date is shifted from 2 year. The designed EDVEGA trajectory is shown in Fig. 8. The value of C3 for the departure is assumed to be 31 km2/s2. The final relative velocity at the Earth's encounter is 11.6 km/s. The required delta-V is about 1780m/s. TOF is 670 days. The sun angle history during the EDVEGA phase is shown in Fig. 9. The sun angle is fixed at 45 deg (upper limit) during the powered flight in order to generate the delta-V in tangential direction.

Figure 10 shows the Earth swing-by trajectory. The minimum altitude is about 1750km. The rotation angle between incoming and outgoing velocity vector was about 31 deg. After the Earth swing-by, the spacecraft flies to Jupiter. In this study, the Earth to Jupiter trajectory was calculated as a ballistic orbit and it is not shown in this paper.

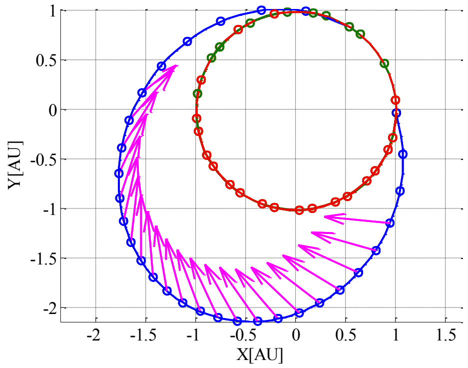

Fig. 8. 2-year EDVEGA trajectory (C3=31). Ion engine is driven around the aphelion.

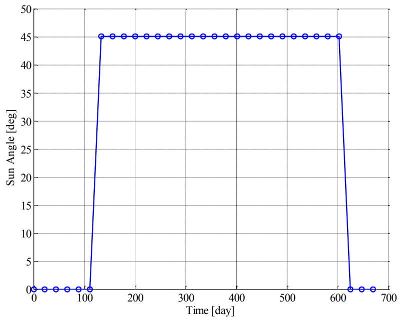

Fig. 9. Sun angle during the EDVEGA phase. It is fixed at the upper limit of the sun angle restriction to maximize the tangential control force.

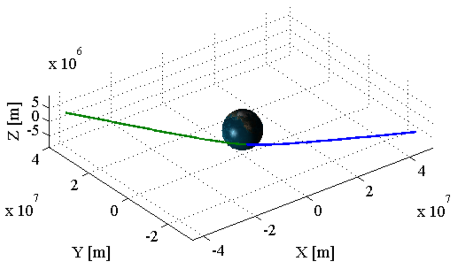

Fig. 10. Earth swing-by trajectory. The swing-by altitude is about 1750km.

5.5. Jupiter to asteroid trajectory

After Jupiter swing-by, the spacecraft heads toward the target asteroid. First the fastest trajectory designed by minimizing the TOF is shown in Fig. 11. Figure 12 shows the history of the control thrust. The spacecraft is always accelerated by the maximum thrust force to minimize the TOF. Table 4 shows the summary of the fastest trajectory to 2009 SK78. The total TOF is about 12 years. The total delta-V is larger than 5000 m/s. It is acceptable for our ion engine system, but the amount of delta-V can be reduced significantly by delaying the arrival date. The trajectory designed for the designated arrival date, Jan. 1, 2038, is shown in Fig. 13. The long coasting which is not seen in the fastest trajectory is appeared. The required delta-V is about 2050 m/s which is much smaller than the delta-V of the fastest trajectory.

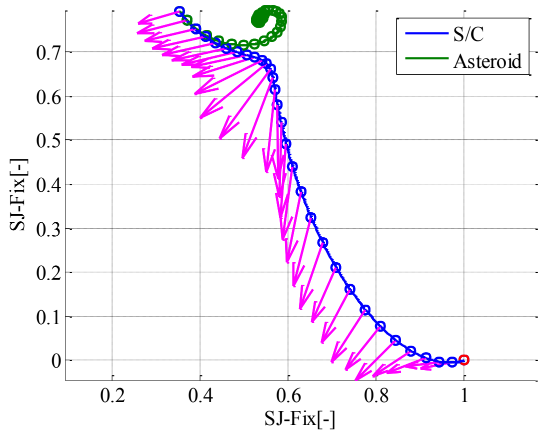

Fig. 11. Fastest trajectory from Jupiter to 2009 SK78 (Sun-Jupiter fixed coordinate).

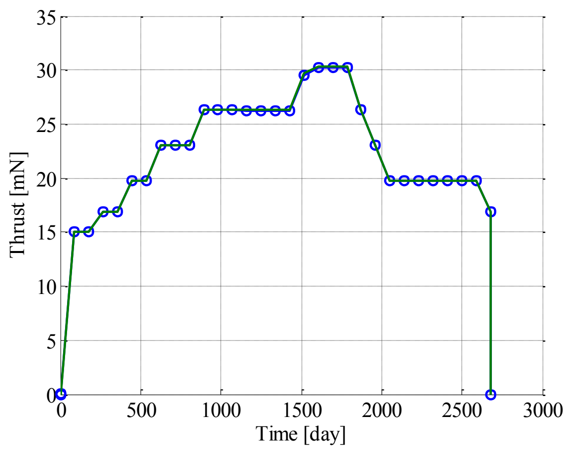

Fig. 12. Thrust force history of the fastest trajectory. The spacecraft is always accelerated by the maximum thrust force.

Table 4. Summary of the fastest trajectory to 2009 SK78.

Phase	Start	End	dV[m/s]
2yr EDVEGA	2024/11/19	2026/09/21	1777
Earth to Jupiter	2026/09/21	2029/05/16	-
Jupiter-to-Asteroid	2029/05/16	2036/09/11	3986
		Total TOF: 11.8 years	

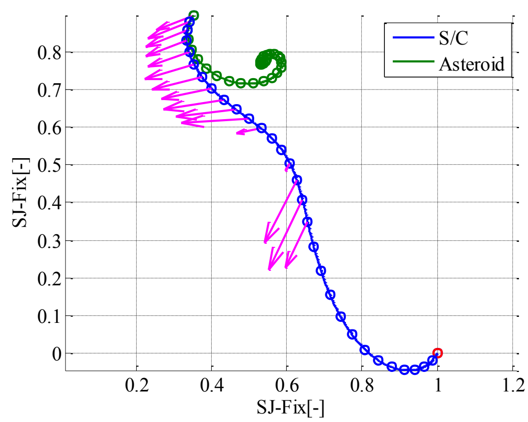

Fig. 13. Trajectory designed by the minimizing delta-V (arrival date: Jan. 1, 2038). It has long coasting time.

5.6. Sample return option

The whole sample return trajectory of 2009 SK78 mission is shown in Fig. 14 and the summary is shown in Table 5. The total mission duration exceeds 30 years. It is difficult to reduce the total TOF because the timing of the Jupiter swing-by occurs about every 6 years. The ion engine with the ultra Isp makes it possible to realize the such challenging sample return mission which requires very large delta-V.

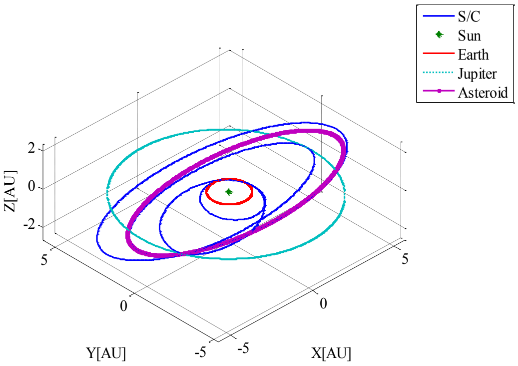

Fig. 14. Overall trajectory of the sample return mission. Jupiter swing-by is also used in the return way.

Table 5. Summary of sample return mission.

Phase	Start	End	dV[m/s]
2yr EDVEGA	2024/11/19	2026/09/21	1777
Earth to Jupiter	2026/09/21	2029/05/16	-
Jupiter-to-Asteroid	2029/05/16	2038/01/01	2049
Proximity Op.	2038/01/01	2039/01/01	-
Asteroid-to-Jupiter	2039/01/01	2053/02/03	2441
Jupiter to Earth	2053/02/03	2055/09/30	-

5.7. Flyby option

Another option is a flyby option. Here we consider the additional asteroid flyby before the rendezvous with the main target. First, we identify the flyby target by searching asteroids which has small distance from the J2A reference trajectory. 2004 XQ185, an outer main-belt asteroid is selected as the flyby target here. Then the low-thrust trajectory is designed. Following one additional constraint is considered:

$$\left|\mathbf{r}_{S/C}(t)-\mathbf{r}_{Ast}(t)\right| \le R, \tag{6}$$

where R is the target flyby distance. $t(0 \le t \le 1)$ is the new parameter which stand for the non-dimensional time. In this example, the flyby occurs on Aug. 4, 2032.

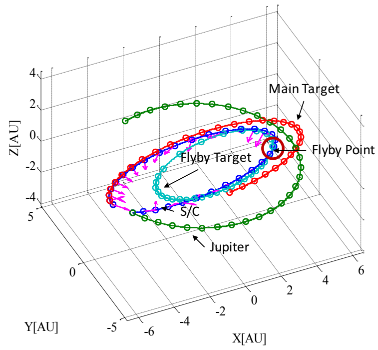

Fig. 15. J2A trajectory with flyby option. Flyby target is 2004 XQ185, an outer main-belt asteroid.

5.8. Multiple rendezvous option

One practical option of our solar power sail mission is a multiple rendezvous option, another rendezvous after the main asteroid arrival taking the advantage of the high orbital control capability. The fastest trajectory from 2009 SK78 to 1996 PS1, the second rendezvous target is shown in Fig. 16. The required delta-V is about 2950 m/s which can be reduced by delaying the arrival date. One more rendezvous can be designed in the similar way if the fuel permitted. 2006 SG371 is a target candidate of the third rendezvous.

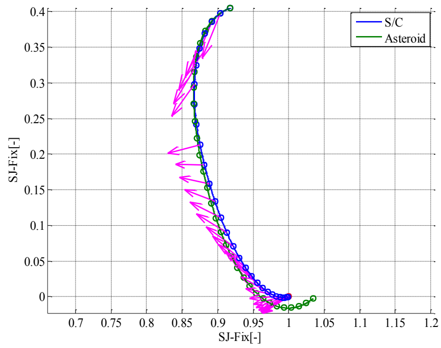

Fig. 16. Trajectory from 2009 SK78 to 1996 PS1. It is the fastest trajectory (delta-V: 2946 m/s)

6. Conclusions

In this paper, we described a Jovian Trojan exploration via solar power sail. The solar power sail, in which a solar sail is combined with a high Isp IES, is an original concept of JAXA and makes it possible to carry a large payload to a Jovian Trojan asteroid. The outline of the solar power sail spacecraft, its trajectory design method, and the trajectory of the nominal one-way mission were presented in this paper. Several practical optional trajectories are also shown.

References

1) Mori, O., Tsuda, Y., Sawada, H., Funase, R., Saiki, T., Yamamoto, T., Yonekura, K., Hoshino, H., Minamino, H., Endo, T., Shirasawa, Y., Mimasu Y., Ono, G. and Kawaguchi, J.: Development and Operation Summary of World's First Solar Power Sail IKAROS, Journal of Space Technology and Science, 27 (2013) pp. 20-37.
2) Tsuda, Y., Mori, O., Funase, R., Sawada, H., Yamamoto, T., Saiki, T., Endo, T., Yonekura, K., Hoshino, H. and Kawaguchi, J.: Achievement of IKAROS—Japanese Deep Space Solar Sail Demonstration Mission, Acta Astronautica, 82 (2013), pp.183-188
3) Kawaguchi, J., Fujiwara, A. and Uesugi, T.: Hayabusa—Its Technology and Science Accomplishment Summary and Hayabusa-2, Acta Astronautica, 62 (2008) pp. 639-647.
4) Tsuda, Y., Yoshikawa, M., Abe, M., Minamino, H. and Nakazawa, S.: System Design of the Hayabusa2 — Asteroid Sample Return Mission to 1999JU3, Acta Astronautica, 91 (2013) pp. 356-362.
5) Berry, K., Sutter, B., May, A., Williams, K., Barbee, B.W., Beckman, M. and Williams, B.: OSIRIS-REx Touch-And-Go (TAG) Mission Design and Analysis, 36th Annual AAS Guidance and Control Conference, AAS 13-095, Breckenridge, CO, 2013.
6) Saiki, T., Mori, O., Kawaguchi, J.: Trajectory Design for Jovian Trojan Asteroid Sample Return Mission, Trans. of JSASS, Space Technology Japan, p. Pd_127-Pd_132, 2016
7) Matousek, S.: The Juno New Frontiers Mission, Acta Astronautica, 61 (2007) pp. 932-939.