Trajectory to Largely Inclined out of Ecliptic Orbit by way of Electric Propulsion and Venus/Earth Gravity Assists

By Yasuhiro KAWAKATSU¹⁾, Hitoshi KUNINAKA¹⁾ and Kazutaka NISHIYAMA¹⁾

¹⁾ Japan Aerospace Exploration Agency, The Institute of Space and Astronautical Science, Sagami-hara, Japan

(Received July 21st, 2009)

The study on the post- HINODE Solar Observation Mission has been started by members in the Solar physics community. One candidate of the mission targets is the observation of the high latitude region of the Sun, which requires the injection of the space observatory (spacecraft) into the orbit largely inclined with the ecliptic plane. Reported in this paper is the trajectory design result for this orbit transfer by way of the Solar electric propulsion and the Venus/Earth gravity assists. The sequence is divided into two phases, the Venus Earth Gravity Assist (VEGA) phase and the sequential Electric Propulsion Delta-V Earth Gravity Assist (EDVEGA) phase. The designed trajectory through the sequence is provided, and it is compared with the trajectory solely using the sequential EDVEGA strategy.

Key Words: Trajectory Design, Electric Propulsion, Gravity Assist

Nomenclature

DCNLP	: direct collocation with nonlinear programming
e	: eccentricity
EDVEGA	: electric propulsion delta-v Earth gravity assist
F	: actual thrust of SEP available for maneuver
$F_{\max}$	: maximum thrust of SEP
h_p	: perigee altitude
i_{SEQ}	: inclination against Solar equatorial plane
I_{sp}	: specific impulse
k_{op}	: operation rate of ion engine
m	: mass of spacecraft
NLP	: nonlinear programming
r	: spacecraft's distance from the Sun
r_E	: Earth's distance from the Sun
SEP	: Solar electric propulsion
t_{IEStotal}	: operation time of ion engine system
t_{IESunit}	: operation time of ion engine averaged per unit
VEGA	: Venus Earth gravity assist
v_{∞}	: excessive velocity, or the magnitude of $\mathbf{v}_{\infty}$
$\mathbf{v}_{\infty}$	: relative velocity vector to the Earth
α	: angle to define direction of $\mathbf{v}_{\infty}$ (see Fig. 5)
δ	: angle to define direction of $\mathbf{v}_{\infty}$ (see Fig. 5)
Δv_{∞}	: increment of v_{∞}

1. Introduction

The Japanese Solar observation satellite “HINODE” was launched in 2006. HINODE has achieved many new scientific discoveries which lead to the progress of the Solar physics. At this stage, the study on the post-HINODE Solar observation mission has been started by members in the Solar physics community. The mission is coded as “Solar-C” whose launch is targeted in FY2016 (Fig. 1).¹⁾

Fig. 1. Solar physics from space in Japan.

Two possible plans are under discussion for Solar-C mission. The first plan, which is called “Plan A”, aims at out-of-ecliptic magnetic/X-ray and helioseismic observations of the polar and the equatorial regions to investigate properties of polar region, meridional flow and magnetic structure inside the Sun down to the bottom of the convection zone. The second plan, which is called “Plan B”, aims at high spatial resolution, high throughput, high cadence spectroscopic (polarimetric) and X-ray observations seamlessly from photosphere to corona to investigate magnetism of the Sun and its role in heating and dynamism of Solar atmosphere.

Focusing on the Plan A, it requires the observation from the high latitude point of the Sun. The target latitude is tentatively specified as 45 degrees (Fig. 2). To observe the Sun from the high latitude point, the space observatory (spacecraft) must be on the orbit largely inclined with the ecliptic plane. It is not an easy task to inject the spacecraft into the orbit largely inclined with the ecliptic plane. A rough estimate shows that the velocity increment required to inject the spacecraft into this largely inclined orbit exceeds 20km/s.

Fig. 2. View from the high Solar latitude.

A couple of missions were studied in view of injecting a spacecraft into the orbit largely inclined with the ecliptic plane.^{2, 3)} The study on Solar Polar Orbiter,²⁾ though it is a technology reference study, shows the feasibility to inject a spacecraft into the Solar polar orbit by way of Solar sail propulsion. Solar Orbiter,³⁾ which is proposed in ESA Cosmic Vision program, plans to achieve the perihelion distance as low as 0.23 AU and the orbit inclination larger than 30 degrees with respect to the Solar equator by way of a number of the Earth and Venus gravity assists.

In case of Solar-C, considering its mission requirements and technology readiness, two options are investigated to achieve this objective. The first option is called “SEP option”, which is characterized by the usage of the Solar electric propulsion (SEP) combined with the planetary gravity assists. Discussed in this paper is a variation of SEP option. The second option is called “Jupiter option”, which is characterized by the usage of the Jupiter swingby. Jupiter option is not treated in this paper, but its result can be found in Reference 4.

On the SEP option, firstly studied⁵⁾ is the strategy which uses SEP combined with the Earth swingby. The method is called Electric Propulsion Delta-V Earth Gravity Assist (EDVEGA), which is world's firstly demonstrated by Japanese asteroid explorer HAYABUSA.⁶⁾ The study result shows that the sequential application of EDVEGA is more efficient than the simple strategy which uses SEP directly to increase the inclination. And the result suggests that the application of the sequential EDVEGA strategy is necessary for the Solar-C Plan-A mission feasibility.

In the sequential EDVEGA strategy, the eccentricity (e) of the cruise orbit is kept to be approximately 0.3 considering both the efficiency to increase v_∞ and the thermal design of the spacecraft. The resulting perihelion distance of the cruise orbit is 0.7, which is just inside the Venus orbit. This is the trigger of this study.

The objective of this study is to investigate the possibility of applying additional Venus swingby to the sequential EDVEGA strategy, and evaluate its effect quantitatively.

In this new strategy, the sequence is divided into two phases. The first is the Venus Earth Gravity Assist (VEGA) phase and the second is the Sequential EDVEGA phase. Following the introduction in this section, the trajectory design method and its result of each phase are discussed in

Section 2 and 3 respectively. The obtained results are integrated and compose the whole sequence, which is compared with the result of the previous study⁵⁾ which solely used the sequential EDVEGA strategy. Finally, the study is concluded in Section 5.

2. Trajectory Design of VEGA Phase

2.1. VEGA strategy

The basic strategy of VEGA is described as follows.

1. The spacecraft is injected into the Venus transfer orbit to encounter Venus.
2. By the Venus swing-by, spacecraft is injected into the Earth transfer orbit to encounter the Earth.
3. The Earth – Venus – Earth trajectory is designed so as the excessive velocity (v_∞) at the Earth re-encounter to be larger than v_∞ at the launch.

Prior to the detailed trajectory design considering the use of SEP, the VEGA sequence is constructed based on the ballistic transfer assumption.

2.2. VEGA sequence design

VEGA sequences are globally constructed by the “trajectory parts connection method” which is exploited by the author.⁷⁾ In the method, the mission sequence is constructed as the series of Keplerian orbits connected with the Venus swingby. The point of the method is to construct the orbits before/after the swingby in the form that can be easily connected at the swingby.

The dynamical feasibility of the sequence is verified by checking various sequence design constraints, and the effectiveness of the sequence is evaluated quantitatively by v_∞ at the launch and the Earth swingby. Although the compensation of v_∞ is required for the case of using SEP, the estimation by the impulsive maneuvers is sufficient to compare the dynamical feasibility and the effectiveness of the sequences. The constructed sequence defines the dates of major milestones and boundary conditions. Therefore, the sequence gives good initial estimate for the following detailed design considering the use of SEP.

The schematic of VEGA sequence is described in Fig. 3. VEGA sequence is the round trip from the launch, through the Venus swingby, to the Earth swingby. Shown in the figure are the five events during the sequence, and the related design parameters (in black) and design results (in red). The design parameters are the minimum necessary parameters to construct the sequence, and the design results are the values

Fig. 3. Schematic of VEGA sequence design.

Table 1. Ranges of VEGA sequence design parameters.

Parameter	Range
2. Earth/Venus transfer • <i>transfer time</i> • <i>number of rotations</i>	≤ 2 years ≤ 2
3. Venus swingby • <i>date</i> • v_{∞}	2017 Jan. 1 – 2019 Dec. 31 ≤ 15.0 km/s
4. Asteroid/Earth transfer • <i>transfer time</i> • <i>number of rotations</i>	≤ 2 years ≤ 2

Table 2. Constraints on VEGA sequence design results.

No.	Constraints
1.	Launch date from 2017 Jan. 1 to 2017 Dec. 31.
2.	Launch $v_{\infty} < 7.7$ km/s.
3.	Launch v_{∞} declination within acceptable range.
4.	Venus swingby altitude > 0 km.
5.	Earth swingby $v_{\infty} > 7.7$ km/s.
6.	Time from Launch to Earth swingby ≤ 2 years.
7.	Launch $v_{\infty} < \text{Earth swingby } v_{\infty}$.
8.	Solar distance through the sequence > 0.7 AU.

induced from designed sequence. The ranges of the design parameters used in the sequence construction are summarized in Table 1. The ranges are set considering not only to exclude the useless (practically infeasible) sequences but also not to drop the useful sequences from the construction results. On the two design parameters, the date and v_{∞} of the Venus swingby, the sets of the discrete values are used for the sequences construction, whose resolutions are 7 days and 1 km/s respectively.

From the constructed VEGA sequences, the possible sequences are selected considering the constraints specific to Solar-C (Table 2). The selected sequences are shown on Table 3. Note that the sequences constructed here are limited to those whose Venus swingby date and v_{∞} are the discrete values with the resolutions given above. Therefore, in actual, there supposed to be other possible sequences around the sequences selected here. Eight sequences are shown on the table, with specific values at the launch, the Venus swingby,

Table 4. Selected VEGA sequence.

Event	Date	v_{∞}
Launch	2017 Jan. 5	4.121 km/s
Venus swingby	2017 Apr. 9	8.000 km/s
Earth swingby	2018 Aug. 27	11.587 km/s

and the Earth swingby.

Among the possible sequences, the sequence 1 achieves the maximum v_{∞} increment between the launch and the Earth swingby. The sequence 1 is selected, and used as a baseline VEGA sequence in the following detailed trajectory design (Table 4).

2.3. Trajectory design of VEGA with SEP

Following the result of the selection above, more detailed mission design is performed assuming the usage of SEP.

In the baseline sequence shown on Table 4, it takes only three months from the launch to the Venus swingby. Considering the practical operation of SEP, it is assumed not to use SEP in this phase short after the launch. The ballistic trajectory already obtained in the sequence design is used as the nominal trajectory from the launch to the Venus swingby. Then, the trajectory between the Venus swingby and the Earth swingby is designed in the following manner considering the usage of SEP.

The Venus – Earth trajectory is designed based on an optimal control problem which is formulated as follows. The objective function is to maximize the final mass. The state variables are the position, velocity and the mass of the space craft. The distance from the Sun (r) is always constrained to be larger than 0.7 AU, so that the thermal environment of the spacecraft not to be too severe. The control variable is the thrust vector at each moment, and the magnitude of the thrust (F) is constrained as

$$F \leq \begin{cases} k_{\text{op}} F_{\text{max}} & (r \leq r_E) \\ k_{\text{op}} F_{\text{max}} \left(\frac{r_E}{r} \right)^2 & (r \geq r_E) \end{cases} \quad (1)$$

where r_E is the Earth's mean distance from the Sun (i.e. 1 AU), F_{max} is the maximum thrust available, and k_{op} is the factor to take into account the operation rate of SEP. The lower line of Eq. (1) means that F is constrained by the available power, which decreases as the spacecraft's distance from the Sun. It is the characteristics of SEP.

Table 3. Constructed VEGA sequences which satisfy all the constraints.

No.	Launch			Venus swingby			Earth Swingby	
	Date	v_{∞} [km/s]	v_{∞} dec. [deg]	Date	v_{∞} [km/s]	Altitude [km]	Date	v_{∞} [km/s]
1.	2017 Jan. 5	4.121	22.2	2017 Apr. 9	8.000	4008	2018 Aug. 27	11.587
2.	2017 Jan. 18	3.973	14.5	2017 Apr. 16	7.000	5366	2018 Aug. 29	11.122
3.	2017 Jan. 31	4.619	4.8	2017 Apr. 23	6.000	4120	2018 Aug. 24	10.623
4.	2017 Feb. 13	6.660	4.8	2017 Apr. 16	8.000	8694	2018 Sep. 6	11.661
5.	2017 Dec. 2	7.658	-10.6	2019 Jan. 13	5.000	32568	2019 Nov. 28	9.219
6.	2017 Dec. 2	7.658	-10.6	2019 Jan. 13	5.000	10617	2019 Nov. 14	8.961
7.	2017 Dec. 13	7.537	-10.4	2019 Jan. 20	5.000	23638	2019 Dec. 8	9.384
8.	2017 Dec. 13	7.537	-10.4	2019 Jan. 20	5.000	9938	2019 Nov. 27	9.198

Table 5. Assumptions on detailed trajectory design.

Item	Value
Spacecraft mass at launch	1200 kg
Specific impulse of SEP	3000 s
$F_{\max}$	120 mN
k_{op}	0.875

The following items are designated to fixed values when solving the optimal control problem, and treated as the boundary conditions.

- 1) Time of the Venus/Earth swingby.
- 2) v_{∞} at the Venus/Earth swingby.
- 3) Initial spacecraft mass.

The fixed values used are those obtained as a result of VEGA sequence design in the previous step, except for the Earth swingby v_{∞} , which is increased to 12.0km/s in this trajectory design. It must be noted that there remains a space of further optimization by relaxing the parameters those fixed in this study.

On the selection of the Earth swingby v_{∞} , a larger v_{∞} is possible from the point of the trajectory design of the VEGA phase (and it may seem better from the point of the whole sequence design). However, the larger v_{∞} constrains the deflection angle at the Earth swingby, which makes difficult to design the Earth swingby conditions to satisfy the post swingby constraints. This result in the selection of 12km/s as the Earth swingby v_{∞} .

The optimal control problem is directly collocated with a nonlinear programming problem,⁸⁾ and the nonlinear programming problem is solved by the sequential quadratic programming method. The major assumptions on the trajectory design are listed in Table 5.

Fig. 4 shows the trajectory profile of VEGA phase. The trajectory is projected on XY plane (the ecliptic plane) of the heliocentric inertial coordinate system. Displayed in the figures are the orbits of the Earth and the Venus (dark blue lines), the trajectory of the spacecraft (red line), and the control acceleration vector (green sticks at the nodes). Since the increment of v_{∞} achieved by the contribution of SEP is

Fig. 4. Trajectory of VEGA phase.

small in this phase (substantially 11.6km/s to 12.0km/s), the control acceleration vector is not so apparent in the figure.

3. Trajectory Design of Sequential EDVEGA Phase

3.1. Sequential EDVEGA strategy

Following the VEGA phase, the sequential use of EDVEGA increases the spacecraft's v_{∞} (against the Earth) up to the value enough to achieve the mission target.

The basic sequence of the sequential EDVEGA strategy is described as follows.⁵⁾

- 1) By the Earth swingby, the spacecraft is injected into the Earth synchronous orbit to re-encounter the Earth after one year cruise.
- 2) During the cruise, SEP is used to maximize v_{∞} at the next Earth encounter. Note that the thrust does not necessarily increase the inclination by itself. To enhance the efficiency to increase v_{∞} , an elliptic orbit is used for the cruise orbit.
- 3) By the Earth swingby, the direction of v_{∞} is changed to contribute to the inclination increase.
- 4) By the repetitive use of the steps 1 to 3, the inclination is increased step by step.

Compared with the simple strategy which uses SEP to increase the inclination, this strategy has an advantage in that the thrust is used more efficiently to increase v_{∞} . The efficiently increased v_{∞} contributes to the inclination increase as a result of the Earth swingby with negligible cost.

From the point of the efficiency to increase v_{∞} , the eccentricity (e) of the cruise orbit had better be larger. However, considering the difficulties in the thermal design of the spacecraft, e is constrained to be less than 0.3 in the analysis.

3.2. Design of the Earth swingby condition

The Earth swingby condition (which is equivalent to the post swingby v_{∞}) is designed so as to satisfy the aforementioned requirements and constraints. In the beginning, v_{∞} is set to 12.0km/s which is provided as the trajectory design result of VEGA phase. The remaining variables are the direction of v_{∞} , that is, α and δ whose definitions are shown in Fig. 5. Described in Fig. 6 are the contours of various parameters at the first Earth swingby plotted on $\alpha\delta$ plane. First, the perigee altitude (h_p) at the swingby must be positive so that the spacecraft does not crash into the Earth. The permissible area from this point is

Fig. 5. Definition of α and δ .

Fig. 6. First Earth swingby condition design.

surrounded by the green curve of $h_p = 0$. Second, the spacecraft is required to be injected into the Earth synchronous orbit after the swingby to re-encounter the Earth after one year cruise. $\mathbf{v}_\infty$ direction must be on the curves of the Earth synchronous orbit (red dots) for this purpose. Third, e should be close to 0.3 considering the balance between $\mathbf{v}_\infty$ increase efficiency and the spacecraft's thermal design. The points on the curves of 0.3 (dark blue dots) should be selected as $\mathbf{v}_\infty$ direction. Consequently, the direction

$$(\alpha, \delta) = (-105.0\text{deg.}, -39.0\text{deg.}) \quad (2)$$

the red solid circle in the figure, turns out to satisfy these conditions as the post swing-by $\mathbf{v}_\infty$ direction.

3.3. Trajectory design of EDVEGA

The method used in EDVEGA trajectory design is the same with that used in the Venus – Earth trajectory design, with the simple replacement of the term “Venus” to “Earth”. $\mathbf{v}_\infty$ at each Earth swingby is set so that the required total increment of $\mathbf{v}_\infty$ (11.1km/s) is achieved in five EDVEGA cycles.

Fig. 7 shows the trajectory profile of the first cycle of EDVEGA phase. The definition of the coordinate system and plots are the same as those of Fig. 4. Fig. 7(a) shows that the spacecraft's orbit has eccentricity, and intersects with the ecliptic at the Earth's position at the Earth swingby. The acceleration vectors projected in XY and XZ plane (Fig. 7(a) and (b) respectively) indicate that the vectors are not perpendicular to the orbit plane, but inclined to the orbit plane to increase $\mathbf{v}_\infty$ at the next Earth encounter.

Fig. 8 shows the trajectory profile through the sequence, from VEGA phase to the sequential EDVEGA phase. In Fig. 8, the trajectory is projected on the plane perpendicular to the ascending node direction so that the change of the orbit plane can be displayed obviously. It is observed that the first five cycles of EDVEGA whose inclination is low have asymmetry resulted from the orbit eccentricity. However, the orbit is finally circularized by the Earth swingby, which

Fig. 7. Trajectory of EDVEGA phase (the first cycle).

Fig. 8. Trajectory of the sequence (out of ecliptic plane).

results in the symmetry observed in the final orbit (the orbit whose inclination is the highest) shown in Fig. 8.

The trajectory sequence is summarized in Table 6. Various values at/between the events (launch and the Venus/Earth swingby) are listed on the table. The listed values at the events are, the mass of the spacecraft (m), pre/post swingby velocity (v_∞ , α , δ), the inclination against the Solar equatorial plane (i_{SEQ}). The listed values between the events are, the increment of v_∞ between the events (Δv_∞), and the operation time of the ion engine system in two definitions ($t_{\text{IES total}}$ and $t_{\text{IES unit}}$). $t_{\text{IES total}}$ means the operation time regardless of the number of operating engines. The effect of k_{op} is included. As shown in Eq. (1), when $r \geq r_E$, the available thrust is constrained due to the decrease of the available power. That is, not all of the ion engines are operating in this area. Considering this effect, the operation time per unit can be decreased compared to $t_{\text{IES total}}$, which is noted as $t_{\text{IES unit}}$ on the table.

Fig. 9. Comparison of sequence profile.

4. Comparison with Simple EDVEGA Strategy

In this section, the trajectory design result is compared with the result of previous study⁵⁾ which solely used the sequential EDVEGA strategy to evaluate quantitatively the effect of applying VEGA,

Fig. 9 shows the profiles of (m, i_{SEQ}) from the launch to the end of the sequence. The result of this study (named “VEGA+EDVEGA” hereafter) is shown in the brown line, and the profile of the result of the previous study (named “EDVEGA” hereafter) is shown in the green line. Each plot indicates the point of the Earth swingby, labeled with its date and v_∞ .

The first observation is about the launch v_∞ . The required launch v_∞ of VEGA+EDVEGA sequence (4.1 km/s) is much smaller than that of EDVEGA sequence (7.7 km/s). This indicates that, assuming the same spacecraft to be launched, VEGA+EDVEGA sequence can be achieved with relatively smaller launcher. This is an obvious merit of VEGA+EDVEGA sequence.

Table 6. Summary of the sequence.

Date	Event	m	v_∞	α	δ	i_{SEQ}	Δv_∞	$t_{\text{IES total}}$	$t_{\text{IES unit}}$
2017/01/05	Launch	1200.0 kg	4.1 km/s	−178.0 deg.	16.4 deg.	5.1 deg.	7.9 km/s	—	—
2017/04/09	Venus swingby	1200.0 kg	8.0 km/s	84.5 deg.	7.5 deg.	5.1 deg.		755 h	530 h
				46.4 deg.	20.9 deg.	7.3 deg.			
2018/08/27	Earth swingby #1	1192.4 kg	12.0 km/s	−94.6 deg.	−7.5 deg.	7.3 deg.	2.2 km/s	6126 h	5261 h
				−105.0 deg.	−39.0 deg.	15.7 deg.			
2019/08/28	Earth swingby #2	1115.7 kg	14.2 km/s	−102.3 deg.	−35.8 deg.	17.1 deg.			
				−113.7 deg.	−53.6 deg.	23.2 deg.			
2020/08/28	Earth swingby #3	1034.0 kg	16.4 km/s	−109.6 deg.	−49.3 deg.	25.0 deg.	2.2 km/s	6166 h	5457 h
				−121.4 deg.	−58.1 deg.	28.4 deg.			
2021/08/27	Earth swingby #4	954.3 kg	18.6 km/s	−116.5 deg.	−54.2 deg.	30.6 deg.			
				−128.0 deg.	−59.5 deg.	33.2 deg.			
2022/08/27	Earth swingby #5	879.0 kg	20.8 km/s	−122.8 deg.	−56.1 deg.	35.7 deg.	2.2 km/s	5800 h	5382 h
				−143.5 deg.	−64.3 deg.	39.1 deg.			
2023/08/28	Earth swingby #6	800.3 kg	23.0 km/s	−136.6 deg.	−61.5 deg.	42.3 deg.			
				−161.7 deg.	−65.9 deg.	44.7 deg.			
2024/08/27	Earth swingby #7	800.3 kg	23.0 km/s	−161.7 deg.	−65.9 deg.	44.7 deg.	—	—	—
				−180.0 deg.	−67.2 deg.	45.0 deg.

Fig. 10. Tilt of Solar equator and sensitivity of node position.

The second observation is about the trend of (m, i_{SEQ}) profile. In view of the mission objective, the more preferable is the sequence which achieves high i_{SEQ} with less fuel consumption, which is equivalent to the line in the right – upper side in the figure. To observe the two lines in the figure from this point, VEGA+EDVEGA sequence leads the other in the beginning, but is caught up in the middle, and is finally overtaken. It is true that VEGA+EDVEGA sequence achieves 12.0km/s of v_∞ with little fuel consumption, which is approximately 4km/s larger than that achieved by EDVEGA sequence at the launch. This is what observed in the beginning of the sequence. However, required v_∞ to achieve i_{SEQ} of 45 deg. are different between the two sequences, and VEGA+EDVEGA sequence requires 4km/s larger v_∞ than EDVEGA sequence. This is the reason of what observed in the latter of the sequence.

The reason of the difference in the required v_∞ is clarified in Fig. 10. There is not negligible tilt (7.3 deg.) of the Solar equatorial plane from the ecliptic plane. Fig. 10(a) shows the geometrical relation between the Solar equatorial plane and the orbit's node direction of the two sequences. In case of EDVEGA sequence, the launch date is selected to take advantage of this tilt. On the other hand, in case of VEGA+EDVEGA sequence, the node of the orbit (i.e. the Earth swingby position) is determined from the VEGA sequence, which depends mainly on the geometrical relation between the Earth and the Venus, but nothing to do with the Solar equatorial plane. Fig. 10(b) shows the relation between

this geometry and the required v_∞ . It is obvious that the tilt of the Solar equatorial plane is not negligible, and the positioning of the orbit's node is an important factor.

Of course, this is not a general demerit of the VEGA+EDVEGA sequence, but the defect of this specific sequence selected in this study. There may be opportunities that satisfy both the good positioning of the orbit's node and large v_∞ increment by VEGA sequence. Further investigation must be done on this point.

5. Conclusion

Reported in this paper is a design result of the trajectory to inject a spacecraft into the orbit largely inclined with the ecliptic plane by way of SEP and the Venus/Earth gravity assists. The design methods of the sequence and trajectory are introduced, and their effectiveness are shown by the design results. The existence of the design results which satisfy the given constraints indicates the feasibility of VEGA+EDVEGA sequence. The effectiveness of this sequence is evaluated by comparing the selected baseline sequence with the simple sequential EDVEGA sequence constructed in the previous study. Two major conclusions are obtained from this comparison. First, the required launch v_∞ of VEGA+EDVEGA sequence can be much smaller than that of EDVEGA sequence. Second, the required v_∞ to be obtained by SEP is strongly effected by the positioning of the orbit's node, which should be considered in selecting the VEGA sequence.

Reference

- 1) Sakao, T. and Solar-C Study Team,: Overview of the next Solar Observation Mission following 'Hinode', *The 8th ISAS Space Science Symposium*, 2008, S2-18.
- 2) Macdonald, M., Hughes, G., McInnes, C., et al. : Solar Polar Orbiter: A Solar Sail Technology Reference Study, *Journal of Spacecraft and Rockets*, **43** (2006), pp. 960-972.
- 3) ESA Science & Technology : Solar Orbiter, <http://sci.esa.int/solarorbiter>.
- 4) Atchison, J., Morimoto, M., Kawakatsu, Y., et al. : A Study on the Trajectory Design of the post-Hinode Solar Observation Mission (Part 2), *The 18th Workshop on Astrodynamics and Flight Mechanics*, 2008, B-4.
- 5) Kawakatsu, Y., Kuninaka, H. and Nishiyama, K.: Trajectory to the Orbit Largely Inclined with the Ecliptic Plane by way of Electric Propulsion Delta-V Earth Gravity Assist, *19th AAS/AIAA Space Flight Mechanics Meeting*, 2009, AAS 09-152.
- 6) Kawaguchi, J.: Solar Electric Propulsion Leverage: Electric Delta-VEGA (EDVEGA) Scheme and its Application, *Proceedings of 10th Work-shop on Astrodynamics and Flight Mechanics*, 2000, pp. 113-122.
- 7) Kawakatsu Y.: Near-Earth Asteroids Sample Return Mission Opportunities in Early 2010s, *Space Technology Japan, the Japan Society for Aeronautical and Space Sciences*, **6** (2007), pp. 87-96.
- 8) Enright, P. J. and Conway, B. A.: Optimal Finite Thrust Spacecraft Trajectories Using Collocation and Nonlinear Programming, *Journal of Guidance and Control*, **14** (1991), pp. 981-985.