TRAJECTORY OPTIONS FOR SOLAR POLAR REGION OBSERVATION MISSION

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Abstract: *A study on the post-HINODE Solar Observation Mission has been started by members in the solar physics community. One candidate of the mission targets on the observation of the solar polar region from the orbit largely inclined with the ecliptic plane. In order to achieve this severe mission target, possible trajectory sequences are investigated considering the application of various trajectory manipulation techniques. Three major trajectory options are listed up to achieve the mission objective. The first option, "SEP option", is characterized by the usage of solar electric propulsion (SEP) combined with the earth gravity assists. The other options are characterized by the usage of planetary gravity assists, and SEP is not used in these options. They are named "Jupiter option" and "Venus option" respectively. The comparison among the trajectory options is discussed in the paper, not only from the aspect of the orbital mechanics, but also from the aspects of the spacecraft design and operation.*

Keywords: *Interplanetary Trajectory Design, Mission Analysis, Out-of-ecliptic Orbit*

1. Introduction

A 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 on FY2017 (Fig. 1) [1].

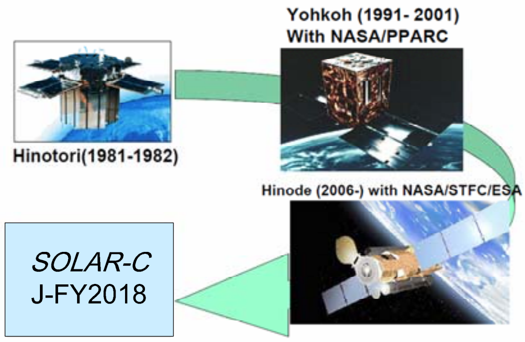
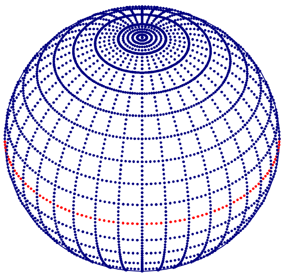
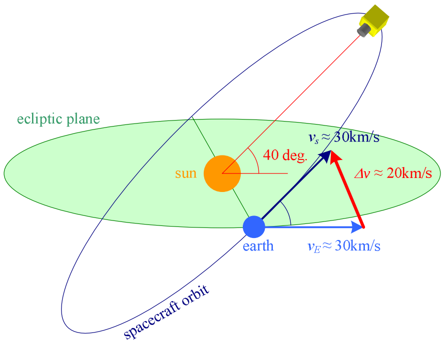


Figure 1. Solar Physics from Space in Japan.

Two possible plans are under discussion for Solar-C mission. One aims at the observation of the polar region of the sun from out-of-ecliptic view point, and the other aims at the high spatial resolution, high throughput observation of the sun from the vicinity of the earth. To focus on the first plan, which is the subject of this paper, it requires the observation from the high latitude point of the sun. The target latitude is tentatively specified as 40° (Fig. 2(a)). 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 of this type. A rough estimate shows that the velocity increment required to inject the spacecraft into this largely inclined orbit is approximately 20km/s (Fig. 2(b)).



(a) View from lat. 40° of the sun



(b) Challenge to lat. 40°

Figure 2. Concept of Solar Polar Region Observation Mission.

A couple of missions were studied in view of injecting a spacecraft into the orbit largely inclined with the ecliptic plane. 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° with respect to the solar equator by way of a number of the earth and Venus gravity assists.

In case of Solar-C, in order to achieve this severe mission target, possible trajectory sequences are investigated considering the application of various trajectory manipulation techniques. The items considered are, the geometric relation (i.e. the tilt of the solar equatorial plane against the ecliptic plane), launcher capacity, planetary gravity assists, and the usage of highly efficient propulsion system. As a result, four major trajectory options are listed up to achieve the mission objective.

The first option is called "SEP option", which is characterized by the usage of the solar electric propulsion (SEP) combined with the earth gravity assists (EGAs) [4][5]. A variation of the SEP option was investigated in [6], which applies an additional Venus gravity assist previous to the sequential EGAs. The other options are characterized by the usage of the planetary gravity assists, and SEP is not used in these options. The second option is called "Jupiter option [7]", which begins by using the Jupiter gravity assist (JGA) to incline the orbit plane largely from the ecliptic plane. What is unique to this option is the usage of EGAs in order to reduce the orbit period after JGA, which enables the frequent observation of the solar polar region. The third option is called "Venus option [8]", which begins with the sequence of Venus gravity assist (VGA) and EGAs to increase the relative velocity (v_∞) to Venus. Then, the sequential VGAs are used to change the direction of $\boldsymbol{v}_\infty$ so as to contribute to incline the orbit.

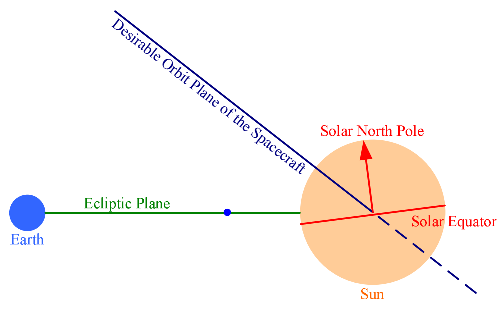
The objective of this paper is to introduce the three trajectory options, and summarize their strength and weakness mainly from the point of the compliance to the mission requirements. The comparison among the trajectory options is made not only from the aspect of the orbital mechanics, but also from the aspects of the spacecraft design and operation. To this objective, first, the mission requirements and presumptions are concretely described in section 2. Then, the concepts and trajectory design results of the four options are shown in section 3. The comparison among the options is summarized in section 4, which is followed by the overview of the paper in section 5.

2. Mission Requirements and Presumptions

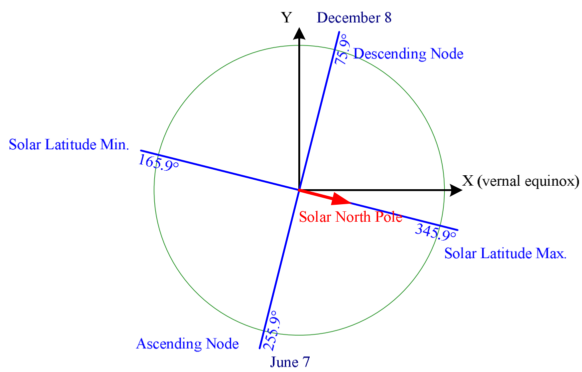
Prior to the discussions of the trajectory options, the mission requirements and the presumptions are summarized in this section.

There are three major requirements imposed from the science mission. The first requirement is on the observation orbit. As is introduced in the previous section, the major requirement of the mission is to observe the polar region of the sun from high solar latitude. The target latitude is tentatively specified as 40°. To observe the sun from the high latitude point, the spacecraft must be placed on the orbit largely inclined with the ecliptic plane. The time frame of the trajectory sequence is constrained as well. To observe the solar polar region at the scientifically important event (i.e. the inversion of the magnetic field at the solar activity maximum), the spacecraft is required to reach the observation orbit by early 2020's. The second requirement is on the mass of the science payload. It is assumed to be larger than 130kg. Though the mass budget of the spacecraft is not explicitly treated in this paper, feasibility of the designed sequence is discussed from this view point as well. The third requirement is on the transmission rate of the scientific data. During the observation period while the spacecraft is in the latitude higher than 30°, the data is expected to be generated continuously onboard in the rate higher than 100kbps. Assuming 8 hours use of the ground station per day, the requirement is interpreted as the down link rate higher than 300kbps. Considering the practical configuration of the communication system (onboard and on the ground), the requirement constrains the distance between the spacecraft and the earth during the observation period.

Through the analyses in the followings, two presumptions are commonly considered. The first presumption is on the launcher. The launcher is assumed to be Japanese H2A heavy launch vehicle equipped with a solid motor upper stage. The launch capacity (i.e. the relation between spacecraft mass and launch energy) is derived assuming the practical settings of the launch site and the launch direction. The specific values used in the analyses are referred to in the section of the trajectory options. The second presumption is on the definition of the solar latitude. The solar latitude is measured from the solar equatorial plane, which is tilted by 7°.25 from the ecliptic plane. The schematics of the geometric relation are shown in Fig. 3. To fully take advantage of this tilt, the top of the orbit (i.e. the point where the solar latitude is the highest) should be in the direction to which the solar rotation axis is tilting (Fig. 3(a)). It is near to the direction of the vernal equinox. To place the top of the orbit in this direction, the ascending/ descending nodes of the orbit should be in the direction perpendicular to this solar rotation axis tilting direction (Fig. 3(b)). In the following trajectory options, planetary gravity assists are used to incline the orbit from the ecliptic plane, and the point of the gravity assists becomes the nodes of the orbit. Therefore, it is desirable to choose the nodes in the direction to fully take advantage of the tilt of the solar equatorial plane when it is permitted.



(a) Solar equatorial plane and desirable orbit plane



(b) Desirable node passage dates

Figure 3. Schematics of Solar Equatorial Plane and Desirable Spacecraft Orbit Plane

3. Overview of Trajectory Options

3.1 SEP Option

Prior to show the baseline sequence of the SEP option, a couple of its variations are introduced briefly. The first method, which is called "Direct Inclining Method (DIM)", is a simple one which uses SEP directly to increase the inclination. The second method uses SEP combined with EGA, which method is called Electric Propulsion Delta-V Earth Gravity Assist (EDVEGA) [9]. The method uses EDVEGA repetitively, and is called "Sequential EDVEGA Method (SEM)". In [4], the two methods were investigated quantitatively, and it was concluded that, DIM is infeasible from the points of the spacecraft's mass budget and the operation time of the ion engine system (IES), whereas SEM is feasible from these points. A slight variation of SEM is also investigated in [6], which applies an additional Venus gravity assist (VGA) prior to the sequential EDVEGA. This method has an advantage in reducing the launch energy drastically compared to the original SEM. However, the use of Venus/earth gravity assists limits the launch opportunity, which makes it difficult to take advantage of the geometrical relation (i.e. the tilt of the solar equatorial plane against the ecliptic plane introduced in section 2). From this point, the usage of VGA is regarded as a back up option.

As a result of the discussion above, SEM is adopted and used to construct the baseline sequence of the SEP option. The basic procedure of SEM is described as follows.

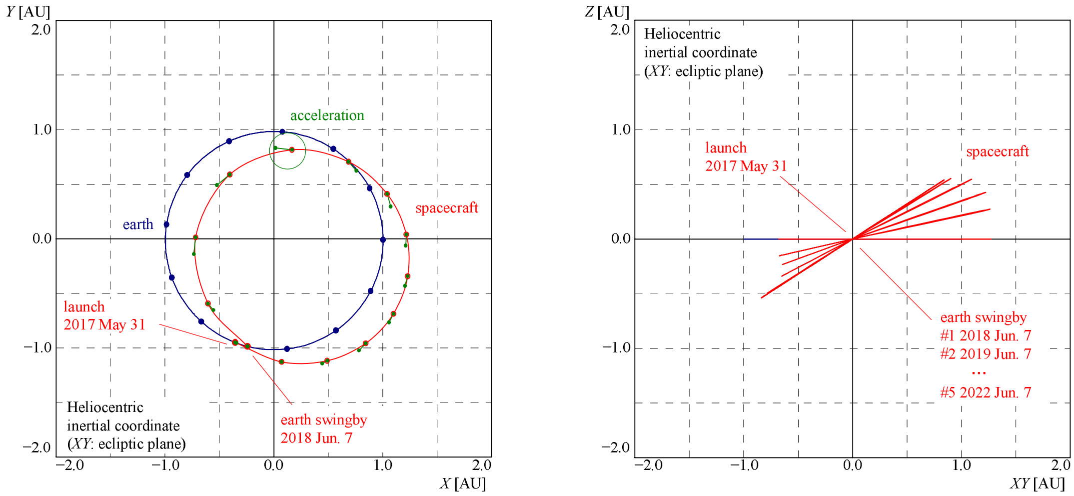
1. The spacecraft is injected into the earth synchronous orbit to re-encounter the earth after one year cruise.
2. During the cruise, SEP is used to maximize the spacecraft's v_∞ to the Earth at the next earth encounter. Note that the thrust does not necessarily increase the inclination by itself. To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

Compared with DIM, this method 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 EGA with negligible cost.

Prior to present the constructed trajectory sequence, assumptions and method which are used in the trajectory design are summarized in the followings. The launch condition is assumed as previously mentioned in the section 2. The initial mass of the spacecraft is assumed to be 1200kg, the launcher is capable of injecting the spacecraft into the escape orbit with v_∞ of 7.3km/s. The launch date is selected so as to take advantage of the tilt of the solar equatorial plane from the ecliptic plane, and they are June 7 or December 8. In the following discussions, June 7 is used as the launch date (see section 2). The second assumption is related to IES. The specific impulse (Isp) of IES is assumed to be 3800s, and the maximum thrust ($F_{\max}$) of IES is assumed to be 120mN. In the trajectory design, the actual thrust available for the maneuver (F) is constrained as

$$F \le \begin{cases} k_{\mathrm{op}} F_{\max} & (r \le r_{\mathrm{E}}) \\ k_{\mathrm{op}} F_{\max} \left(\dfrac{r_{\mathrm{E}}}{r} \right)^2 & (r \ge r_{\mathrm{E}}) \end{cases} \tag{1}$$

where r is the spacecraft's distance from the Sun, and r_{E} is that of the Earth. The lower line means that the available power decreases as the spacecraft's distance from the Sun. k_{op} is the factor to take into account the operation rate of the ion engine, which is assumed to be 0.875. The third assumption is related to the eccentricity (e) of the cruise orbit. From the point of the efficiency to increase v_∞, e 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 trajectory design. Finally mentioned is related to the method used for the trajectory design. The trajectory is designed by each arc which composes the whole sequence. The arc means the part of the trajectory which is bounded by the Earth encounter. 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 departure/arrival time, departure/arrival v_∞ (to the Earth), and the initial mass of the spacecraft are designated as the boundary conditions. The designation of the departure/arrival time is equivalent to that of the spacecraft's position at the departure/arrival. The thrust is constrained as Eq. 1. The problem is directly collocated with a nonlinear programming (DCNLP) [10], and the nonlinear programming (NLP) is solved by the sequential quadratic programming method.



(a) Projected on the ecliptic plane (1st revolution) (b) Side view

Figure 4. Trajectory Profile of SEP Option

Table 1. Sequence of Events of SEP Option

Date	Event	v_∞	i_{SEQ}
2017/05/31	Launch	7.3 km/s	7.2°
2018/06/07	EGA #1	9.8 km/s	7.3° 18.8°
2019/06/07	EGA #2	12.3 km/s	20.2° 25.8°
2020/06/07	EGA #3	14.6 km/s	27.2° 33.0°
2021/06/07	EGA #4	15.7 km/s	34.9° 38.1°
2022/06/07	EGA #5	16.6 km/s	39.8° 40.0°

Figure 4 shows the trajectory profile of the SEP option. In Fig. 4(a), the trajectory of the first revolution is projected on the ecliptic plane. The figure shows that the spacecraft's orbit has eccentricity, and intersects with the ecliptic approximately at the earth's position at the launch. The acceleration vectors indicate that the thrust is used to decelerate the spacecraft at aphelion and to accelerate the spacecraft at perihelion, which result in the increase of v_∞ at the next earth encounter. Figure 4(b) shows the trajectory profile through the sequence. The trajectory is projected on the plane perpendicular to the ascending node direction so that the gradual change of the orbit plane can be displayed obviously. It is observed that the first four cycles have asymmetry resulted from the orbit eccentricity. However, the orbit is finally circularized by EGAs, which results in the symmetry observed in the final orbit (the orbit whose inclination is the highest).

The sequence of events of the SEP option is summarized in Tab. 1. On the table, v_∞ denotes the relative velocity to the earth at EGA, and i_{SEQ} denotes the inclination against the solar equatorial plane. The spacecraft reaches i_{SEQ} of 30° after the 3rd EGA (3 years from the launch), and finally reaches i_{SEQ} of 40° after the 5th EGA (5 years from the launch).

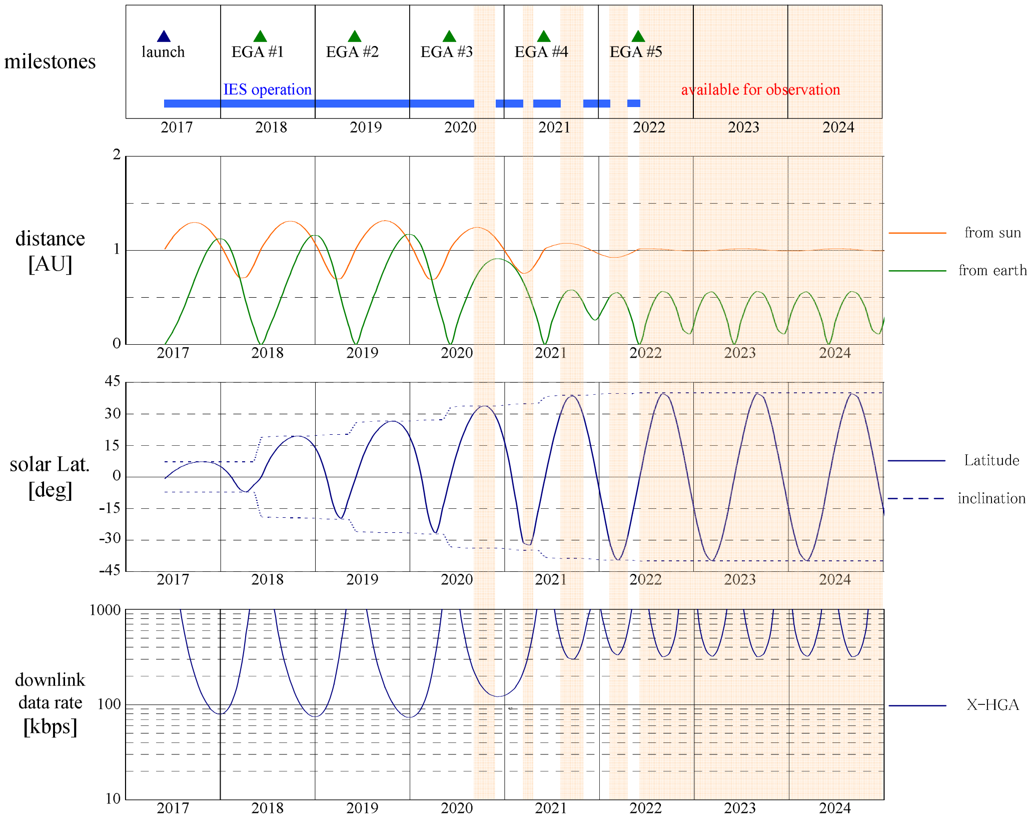


Figure 5. Mission Profile of SEP Option

Figure 5 shows the profiles of some important parameters of the mission. The top chart denotes the points of events and basic operation concept. The three charts below respectively show the profile of the spacecraft's distance from the sun and the earth, its instantaneous solar latitude, and the expected down link rate of the scientific data. In the first three years from the launch, before the spacecraft reaches   of 30°, the spacecraft operation is devoted to increase the inclination. IES is operating most of the time, and silent condition for the scientific observation is basically not guaranteed in this phase. Even in this phase, intermittent suspension of IES is planned for accurate orbit determination, which may be used as occasions of observation. In the fourth year, after EGA #3,   exceeds 30°. From this year, the duration while the spacecraft is in the latitude higher than 30° is allotted as "the observation phase (orange area in the figure)", and the IES operation is intentionally suspended. Even after EGA #3, IES operation is continued while the spacecraft is in the low solar latitude. This IES operation and the following two EGAs contribute to the further increase of the inclination and the circularization of the orbit. Finally, as a result of EGA #5, the spacecraft is injected in to the final observation orbit, the circular one year orbit with   of 40°.

3.2 Jupiter Option

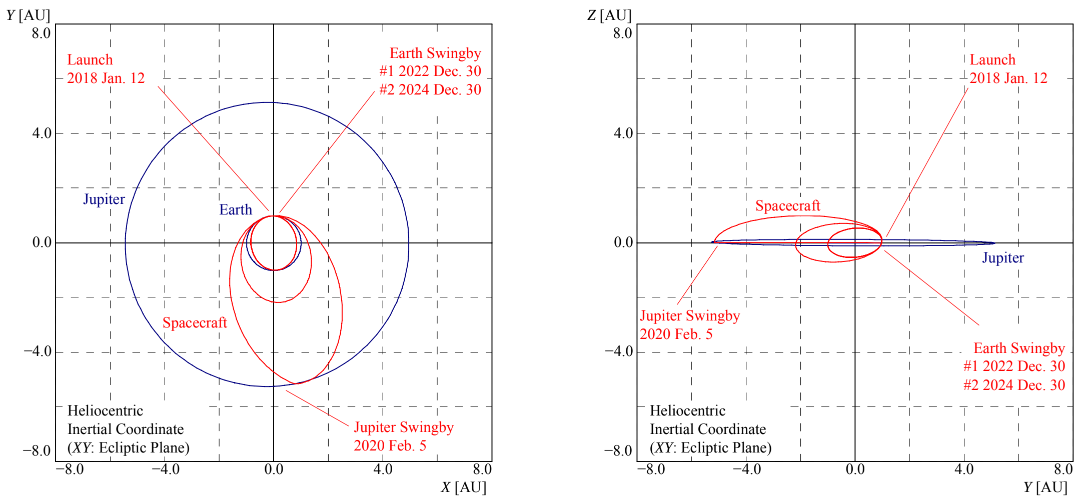
Prior to show the baseline sequence of the Jupiter option, its variation is introduced briefly. It is a simple method which uses only one JGA to incline the orbit from the ecliptic plane. The method is firstly used in Ulysses mission. A serious disadvantage of this method is the long orbit period of the observation orbit (i.e. the orbit after the Jupiter gravity assist), which is typically about 5 years. It results in the long interval between the observations of the solar polar region, which is not acceptable from the scientists who require the frequent observations of the solar polar region.

As a result of the discussion above, in the baseline sequence of the Jupiter option, additional EGAs are used to shorten the orbit period of the observation orbit. The basic procedure of the Jupiter option is described as follows.

1. The spacecraft is injected into the Jupiter transfer orbit to encounter the Jupiter.
2. By JGA, the spacecraft is injected into the earth transfer orbit to re-encounter the earth.
3. By EGA, the orbit period is shortened so as to enable the frequent observations of the solar polar region. The orbit period is selected so that the spacecraft re-encounter the earth again within short interval.
4. By the repetitive use of the step 3, the orbit period is shortened step by step.

As is shown in the followings, by this method, the orbit period is shortened to one year in the end, which enables annual observation of the north/south polar region of the sun.

Prior to present the constructed trajectory sequence, assumptions and method which are used in the trajectory design are summarized in the followings. The launch condition is assumed as previously mentioned in the section 2. The initial mass of the spacecraft is assumed to be 770kg. The launcher is capable of injecting the spacecraft into the escape orbit with v_∞ of 8.9km/s, which is sufficient to inject the spacecraft into the direct transfer orbit to the Jupiter. The trajectory sequence of the earth – Jupiter – earth transfer is constructed by the "trajectory parts connection method" which is exploited by the author [11]. In the method, the trajectory sequence is constructed as a series of Keplerian orbits connected with JGA. The point of the method is to construct the orbits before/after JGA in the form that can be easily connected at JGA. It must be noted that, the sequence, in particular the date of EGA is determined from the geometrical relation between the earth and Jupiter. Therefore, the date of EGA is not necessarily appropriate to take advantage of the tilt of the solar equatorial plane from the ecliptic plane. The sequence of EGAs is constructed by the use of "v_∞ direction diagram" presented in the previous symposium [12].



(a) *XY* view (b) *YZ* view

Figure 6. Trajectory Profile of Jupiter Option

Figure 6 shows the trajectory profile of the Jupiter option. In Fig. 6(a), the trajectory is projected on the ecliptic plane. The figure shows that the size of the orbit get smaller by the sequential use of EGAs, which results in the shorter orbit period of the observation orbit to enable the frequent observation of the solar polar region. Figure 6(b) shows the trajectory viewed from the side. It is observed that the trajectory is largely inclined with the ecliptic plane by JGA, and kept inclined during the phase of sequential EGAs.

Table 2. Sequence of Events of Jupiter Option

Date	Event	v_∞	i_{SEQ}
2018/01/12	Launch	8.9 km/s	7.1°
2020/02/05	JGA	7.3 km/s	31.4°
2022/12/30	EGA #1	17.0 km/s	35.6°
2024/12/30	EGA #2	17.0 km/s	39.5°

The sequence of events of the Jupiter option is summarized in Tab. 2. It takes almost 5 years to complete the earth – Jupiter – earth sequence and re-encounter with the earth. Then, the orbit period is shortened to 2 years by EGA #1, and finally shortened to 1 year by EGA #2. To take a look at i_{SEQ} , the spacecraft reaches i_{SEQ} over 30° after JGA, and finally reaches i_{SEQ} near to 40° by the succeeding two EGAs. As a result of EGA #2, the orbit is almost circularized, and any additional raise of i_{SEQ} cannot be expected by way of ballistic EGA. Obviously, the finally achievable i_{SEQ} is determined from v_∞ at the earth re-encounter. That is to say, higher i_{SEQ} is achievable in the end by adopting larger v_∞ at the earth re-encounter. However, it must be noted that the reduction of the orbit period achieved by a single EGA is limited by v_∞ at EGA. v_∞ of 17.0km/s adopted in this sequence is the maximum v_∞ to reduce the orbit period to 2 years by EGA #1, and to 1 year by EGA #2. That is to say, if larger v_∞ at EGA is adopted, the orbit periods after EGAs get longer, which result in the stretch of the whole transfer sequence. In summary, the finally achievable i_{SEQ} and the the transfer time to the observation orbit are in the relation of trade-off. The sequence adopted here minimizes the transfer time to the observation orbit while satisfying the mission requirement on i_{SEQ} for the most part.

Figure 7 shows the profiles of some important parameters of the mission. The parameters shown in the charts are the same as those in Fig. 5. It is obvious that the first four years of the sequence is devoted to the round trip to the Jupiter. Though the orbit inclination jumps up to 30° by JGA, the solar latitude of the spacecraft position is still low while the spacecraft is far distant from the sun. The distance to the sun constrains the precision of the solar observation, and the distance from the earth limit the transmission rate of the scientific data. There are a couple of opportunities to pass through the high solar latitude region during the transfer sequence. However, the data transmission rate during the passage is still lower than requested. Full scale scientific observation becomes possible when the spacecraft is injected into the final observation orbit after 7 years from the launch. In spite of the weakness in the trajectory sequence to reach the observation orbit, this option has strength in the requirement to the spacecraft design. It does not require advanced technology such as electric propulsion, and it is able to be realized only by using conventional technologies.

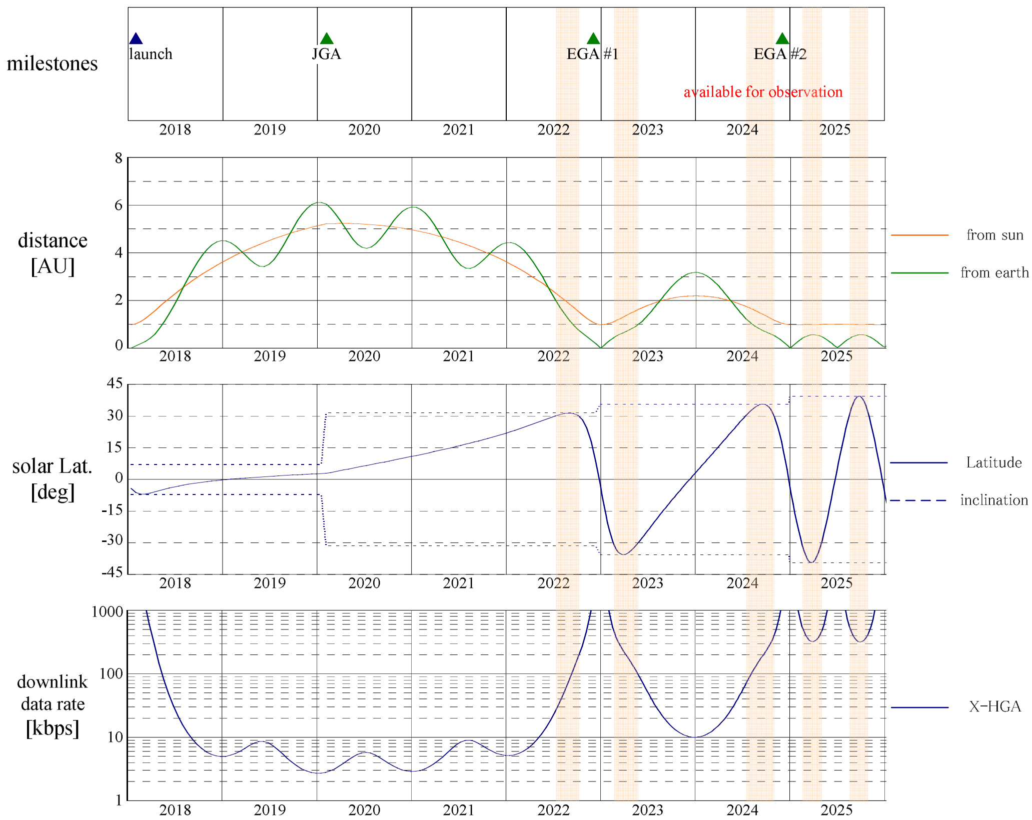


Figure 7. Mission Profile of Jupiter Option

3.3 Venus Option

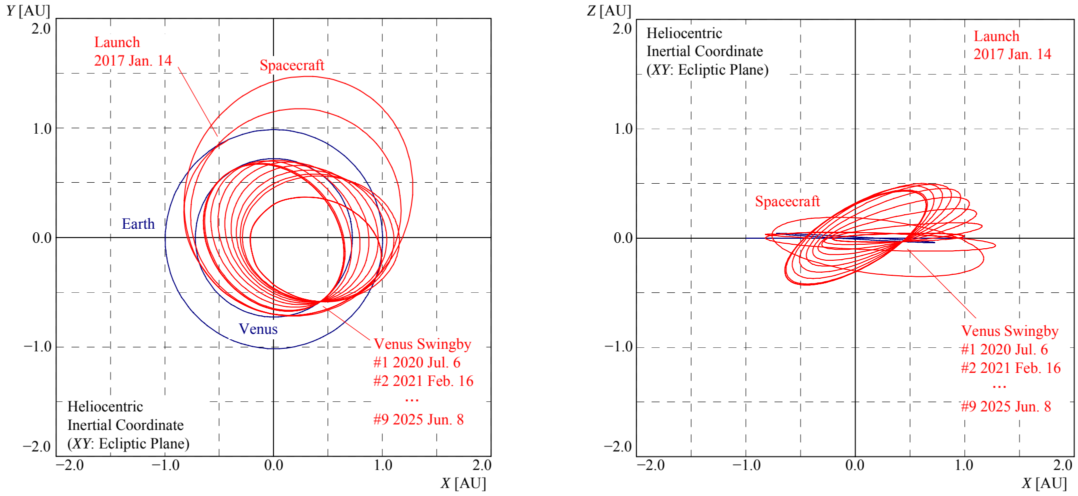
Prior to show the baseline sequence of the Venus option, its variations are introduced briefly. Venus option is characterized by the usage of EGAs and VGAs. Two variations begin with the sequence of EGAs/VGAs to increase $\boldsymbol{v}_\infty$ to the planets. In this phase, the trajectories approximately lie within the ecliptic plane. Then, the sequential planetary gravity assists are used to change the direction of $\boldsymbol{v}_\infty$ so as to contribute to incline the orbit. “Venus-1 Option” uses the Venus for the plane change gravity assists, whereas “Venus-2 Option” uses the earth for this purpose. “Venus-1 Option” achieves the solar latitude of 40° as a result of nine VGAs dedicated for the orbit plane change. On the other hand, in case of “Venus-2 Option”, the limitation of acquirable $\boldsymbol{v}_\infty$ by VGA/EGA sequence constrains the final achievable latitude to 30° .

As a result of the discussion above, in this section, “Venus-1 Option” is introduced as the baseline sequence of the Venus option, since it can achieve the target solar latitude of 40° . The basic procedure of the Venus option is described as follows.

1. The spacecraft is injected into the Venus transfer orbit to encounter the Venus.
2. By VGA, the spacecraft is injected into the earth transfer orbit to re-encounter the earth. The earth – Venus – earth transfer trajectory is designed so that v_∞ at the earth re-encounter becomes larger than that at the launch.
3. By EGAs, the spacecraft is injected into the Venus transfer orbit to re-encounter the earth. The earth – Venus transfer trajectory is designed so as v_∞ at the Venus re-encounter to be larger than that at the first VGA.
4. By VGA, the orbit plane is inclined a little while the orbit period is kept to one Venus year to re-encounter the Venus in short interval.

5. By the repetitive use of the step 4, the orbit is inclined enough to observe the solar polar region. As is shown in the followings, by this method, the target latitude of 40° is achieved in the end.

Prior to present the constructed trajectory sequence, assumptions and method which are used in the trajectory design are summarized in the followings. The launch condition is assumed basically as previously mentioned in the section 2. However, the launch energy required to inject the spacecraft into the initial Venus transfer orbit is far lower than that required for the previous two options ("SEP option" and "Jupiter option"). Therefore, the sufficient launch capacity is achieved without the usage of a solid motor upper stage. The trajectory sequence of the earth – Venus – earth transfer and succeeding earth – earth – Venus transfer are constructed by the "trajectory parts connection method". It must be noted again that, the sequence, in particular the node of sequential VGAs is determined from the geometrical relation between the earth and Venus. Therefore, the positions of nodes are not necessarily appropriate to take advantage of the tilt of the solar equatorial plane from the ecliptic plane.



(a) *XY* view (b) *YZ* view

Figure 8. Trajectory Profile of Venus Option

Figure 8 shows the trajectory profile of the Venus option. Figure 6(a) shows the trajectory projected on the ecliptic plane, and Fig. 6(b) shows the trajectory viewed from the side. It is observed that the orbit is inclined and changes its shape gradually by the repetitive use of VGAs.

Table 3. Sequence of Events of Jupiter Option

Date	Event	v_∞	i_{SEQ}
2017/01/14	Launch	4.9 km/s	–
2017/04/03	VGA #0	10.0 km/s	–
2018/09/01	EGA #1	12.7 km/s	–
2019/09/01	EGA #2	12.7 km/s	–
2020/07/06	VGA #1	22.9 km/s	13.6°
2021/02/16	VGA #2	22.9 km/s	20.9°
2021/09/29	VGA #3	22.9 km/s	27.2°
2022/05/11	VGA #4	22.9 km/s	32.3°
2022/12/22	VGA #5	22.9 km/s	36.0°
2023/08/04	VGA #6	22.9 km/s	38.7°
2024/03/15	VGA #7	22.9 km/s	40.3°
2024/10/26	VGA #8	22.9 km/s	40.9°
2025/06/08	VGA #9	22.9 km/s	40.9°

The sequence of events of the Venus option is summarized in Tab. 3. It takes about 3.5 years to complete the "v_∞ increasing phase", in which v_∞ is finally raised to 22.9km/s respective to Venus by the usage of the sequence of EGAs and VGAs. To take a look at i_{SEQ}, the spacecraft reaches i_{SEQ} over 30° after four VGAs, and finally reaches i_{SEQ} higher than 40° by the succeeding three VGAs.

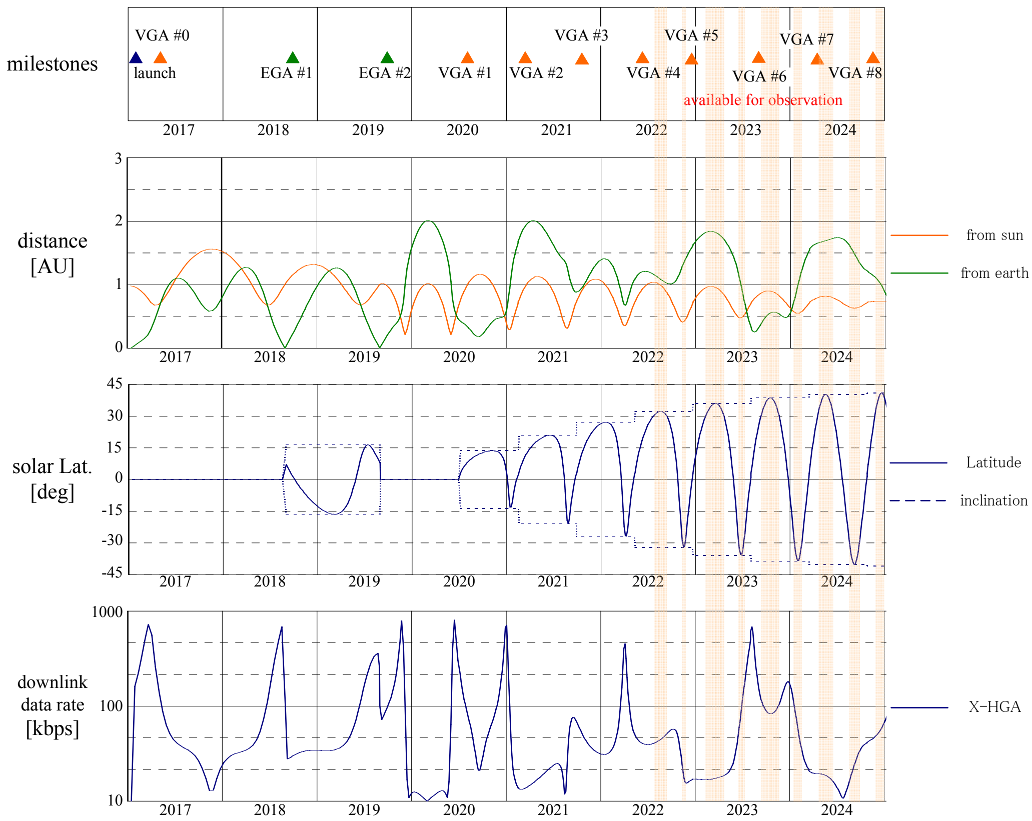


Figure 9. Mission Profile of Venus Option

Figure 9 shows the profiles of some important parameters of the mission. The parameters shown in the charts are the same as those in Fig. 5.

5. Summary

In this paper, trajectory options for the solar polar region observation mission are introduced. The mission is planned as a candidate of the post-HINODE Solar Observation Mission by the members in the solar physics community. The mission requires to inject the spacecraft into the orbit largely inclined with the ecliptic plane. The trajectory options are derived considering the application of various trajectory manipulation techniques to achieve this severe mission target. The three major trajectory options listed up are "SEP option", "Jupiter option", and "Venus option". The concept of the trajectory options are introduced, and their strength and weakness are summarized mainly from the point of the compliance to the mission requirements. The comparison among the trajectory options is made not only from the aspect of the orbital mechanics, but also from the aspects of the spacecraft design and operation. Further work is to be done to compare and prioritize the options to choose the only baseline sequence of the mission.

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