A Study on the Trajectory Design of the post-HINODE Solar Observation Mission

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Abstract

The study on the post- HINODE Solar Observation Mission has been started by members in the Solar physics community. One mission candidate targets on the observation of the high latitude region of the Sun, which requires the space observatory (spacecraft) to be injected into the orbit largely inclined with the ecliptic plane. Reported in this paper are the trajectory design results for this orbit transfer, which contains a sequential application of ED-VEGA procedure.

次期太陽観測ミッションの軌道計画についての一考察

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摘要

「ひので」に続く太陽観測ミッションの検討が開始されている。そのミッション案の一つに、太陽の高緯度領域の観測を目的として、黄道面から大きく傾いた軌道に探査機を投入しようというものがある。目標は太陽緯度45度であり、そこに到達するために必要な増速量は単純計算で20km/sを超える。本発表では、複数回のEDVEGAを利用する案を中心に、この目標に到達する軌道計画の検討結果を報告する。

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 FY2016 (Fig. 1).

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 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 degree (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. Two options are investigated to achieve this objective. The first option is the usage of the Solar electric propulsion (SEP) combined with Earth swing-by, which option is discussed in this paper. The second option is the usage of the Jupiter swing-by, which option is discussed in Ref. 2.

2. Trajectory to High Solar Latitude by SEP

Reported in this paper are the trajectory design results for the orbit transfer to the high Solar latitude by use of SEP. Two cases are considered for the comparison. The first case, which is called “Case 1”, is a simple strategy which uses SEP to increase the inclination. The Earth swing-by is not used in this case. The second case, which is called “Case 2”, is a strategy which uses SEP combined with the Earth swing-by. The method is called Electric Propulsion Delta-v Earth Gravity Assist (EDVEGA), which is world’s firstly demonstrated by Japanese asteroid explorer HAYABUSA.

Prior to start the discussions as to the individual cases,
the assumptions and method which are used commonly in both cases are summarized in the followings.

The first assumption is related to the launch condition. The launcher assumed is the Japanese H2A heavy launch vehicle equipped with a solid motor upper stage. The initial mass of the spacecraft is assumed to be 1200kg. Assuming the practical settings of the launch site and the launch direction, the launcher is capable of injecting the spacecraft into the escape orbit with the excessive velocity ($v_\infty$) of 7.7km/s. The launch date is selected so as to take advantage of the not negligible tilt (7.25deg.) of the Solar equatorial plane from the ecliptic plane. To place the top of the orbit in this direction, the ascending and descending nodes of the orbit should be in the direction perpendicular to this Solar North pole tilting direction. The spacecraft should be launched on the date when the Earth is at these ascending or descending nodes, and they are June 7 and December 8 (Fig. 3(b)). In the following discussions, June 7 is used as the nominal launch date.

The second assumption is related to SEP. The specific impulse (ISP) of SEP is assumed to be 3000s, and the maximum thrust ($F_{\text{max}}$) of SEP is assumed to be 150mN. In the analysis, the actual thrust available for the maneuver ($F$) is constrained as

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

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.

Finally mentioned is related to the method used for the trajectory design[^4]. 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 $v_\infty$ (to the Earth), and the initial mass of the spacecraft are designated as the boundary conditions.
strategy is simpler. It has an advantage in that the distance to the Sun is kept approximately to 1AU, which ease the thermal design of the spacecraft.

Fig. 4 shows the trajectory profile of the first cycle of Case 1. In Fig. 4(a) and 4(b), the trajectory is projected on XY plane (the ecliptic plane) and XZ plane respectively of the heliocentric inertial coordinate system. Displayed in the figures are the ecliptic (the dark line), the trajectory of the spacecraft (light line), and the control acceleration vector (sticks at the nodes). Fig. 4(a) shows that the distance from the Sun is kept approximately to 1AU. Fig. 4(b) shows that the acceleration vector is approximately perpendicular to the orbit plane.

Fig. 5 shows the trajectory profile through the sequence of Case 1. In Fig. 5, 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. Fig. 5 shows that the orbit is gradually being inclined, while the orbit radius is kept approximately at 1AU.

The trajectory sequence of Case 1 is summarized in Table 1. Various values at/between the events (launch and the Earth encounter) are listed on the table. The listed values at the events are, the mass of the spacecraft \( m \), the relative velocity to the Earth \( v_e \), the inclination against the Solar equatorial plane \( \Delta v_e \), and the operation time of the ion engine system \( \tau_{\text{unit}} \). As to \( \tau_{\text{unit}} \), two values “total” and “unit” are noted. The “total” means the operation time regardless of the number of operating engines. The effect of \( k_{\text{op}} \) is included. As shown in Eq. (1), when \( r > r_{\text{op}} \), 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 the “total” operation time, which is noted as “unit” on the table.

Three points must be pointed out on Table 1. The first, the target \( \Delta \varepsilon_{\text{SEQ}} \) of 45deg. is achieved by 5 years from the launch. The second, the spacecraft is injected into the Earth synchronous orbit so that the distance to the Earth is kept in the limited range through the sequence. Launch \( v_e \) is directed to maximize the inclination under this constraint. The third, during the cruise, SEP is used directly to increase the inclination. That is, the thrust direction is basically normal to the orbit plane at the moment.

2.1 Case 1: Simple Inclination Increase Strategy

The basic sequence of this strategy is described as follows.

1) The spacecraft is injected into the Earth synchronous orbit so that the distance to the Earth is kept in the limited range through the sequence. Launch \( v_e \) is directed to maximize the inclination under this constraint.

2) During the cruise, SEP is used directly to increase the inclination. That is, the thrust direction is basically normal to the orbit plane at the moment.

3) By the repetitive use of the step 2, the inclination is increased step by step.

Compared with the following strategy (Case 2), this strategy is simpler. It has an advantage in that the distance to the Sun is kept approximately to 1AU, which ease the thermal design of the spacecraft.

Table 1. Trajectory Sequence of Case 1

<table>
<thead>
<tr>
<th>Date</th>
<th>Event</th>
<th>( m )</th>
<th>( v_e )</th>
<th>( \alpha )</th>
<th>( \delta )</th>
<th>( \ell_{\text{SEQ}} )</th>
<th>( \Delta v_e )</th>
<th>( \tau_{\text{tot}} / \text{unit} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>2010/06/07</td>
<td>Launch</td>
<td>1290.0 kg</td>
<td>7.7 km/s</td>
<td>-180.0 deg.</td>
<td>82.5 deg.</td>
<td>22.3 deg.</td>
<td>2.1 km/s</td>
<td>6452 / 6379 h</td>
</tr>
<tr>
<td>2011/06/07</td>
<td>Earth Encounter #1</td>
<td>1086.5 kg</td>
<td>9.8 km/s</td>
<td>-180.0 deg.</td>
<td>80.4 deg.</td>
<td>26.5 deg.</td>
<td>2.3 km/s</td>
<td>6402 / 6327 h</td>
</tr>
<tr>
<td>2012/06/07</td>
<td>Earth Encounter #2</td>
<td>973.2 kg</td>
<td>12.1 km/s</td>
<td>-180.0 deg.</td>
<td>78.1 deg.</td>
<td>31.0 deg.</td>
<td>2.6 km/s</td>
<td>6523 / 6450 h</td>
</tr>
<tr>
<td>2013/06/07</td>
<td>Earth Encounter #3</td>
<td>858.6 kg</td>
<td>14.7 km/s</td>
<td>-180.0 deg.</td>
<td>75.5 deg.</td>
<td>36.3 deg.</td>
<td>3.0 km/s</td>
<td>6794 / 6722 h</td>
</tr>
<tr>
<td>2014/06/07</td>
<td>Earth Encounter #4</td>
<td>739.6 kg</td>
<td>17.7 km/s</td>
<td>-180.0 deg.</td>
<td>72.4 deg.</td>
<td>42.4 deg.</td>
<td>1.3 km/s</td>
<td>2037 / 2008 h</td>
</tr>
<tr>
<td>2015/06/07</td>
<td>Earth Encounter #5</td>
<td>702.7 kg</td>
<td>19.0 km/s</td>
<td>-180.0 deg.</td>
<td>71.1 deg.</td>
<td>45.0 deg.</td>
<td>0.0 km/s</td>
<td>0 / 0 h</td>
</tr>
<tr>
<td>2016/06/07</td>
<td>Earth Encounter #6</td>
<td>702.7 kg</td>
<td>19.0 km/s</td>
<td>178.6 deg.</td>
<td>71.1 deg.</td>
<td>45.0 deg.</td>
<td>0.0 km/s</td>
<td>0 / 0 h</td>
</tr>
</tbody>
</table>
launch. That is, from the point of the orbit design, on condition that 5 years is acceptable, it is possible to reach the Solar latitude 45deg. by Case 1. The second, mass at the end of the sequence is approximately 700kg. In other words, 500kg of the propellant is required to complete this sequence. The spacecraft mass budget is estimated under the condition that the total mass of 1200kg including the propellant mass of 500kg. The result shows the minus margin of approximately 50kg, which means that Case 1 is not feasible from the point of the brief spacecraft design. The third, the sum of IES \( t_{\text{ies}} \) through the sequence is approximately 28000h, which far exceeds the operation result of HAYABUSA. This is a serious concern on Case 1 from the point of the ion engine design. To summarize, Case 1 is concluded as infeasible from the latter two points listed above. The use of the more powerful launcher or the tuning of the ion engine specification may improve the situation a little but it seems not to be able to overturn the conclusion.

2.2 Case 2 : EDVEGA Approach

The basic sequence of this strategy 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 \( v_\infty \) at the next Earth encounter. Note that the thrust does not necessarily increase the inclination by itself. To enhance the efficiency to increase \( v_\infty \), an elliptic orbit is used for the cruise orbit.
3) By the Earth swing-by, the direction of \( v_\infty \) 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 the previous strategy (Case 1), this strategy has an advantage in that the thrust is used more efficiently to increase \( v_\infty \). The efficiently increased \( v_\infty \) contributes to the inclination increase as a result of the Earth swing-by with negligible cost.

From the point of the efficiency to increase \( v_\infty \), 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.

The launch \( v_\infty \) is designed so as to satisfy the aforementioned requirements and constraints. In the beginning, \( v_\infty \) is set to 7.7km/s to fully utilize the capacity of the assumed launcher. The remaining variables are the direction of \( v_\infty \), that is, \( \alpha \) and \( \delta \). Shown in Fig.7 are the contours of the various parameters plotted on \( \alpha \delta \) plane. First, the practical settings of the launch site and the launch direction limit the possible range of \( v_\infty \) declination. The permissible area from this point lies between the curves of the maximum/minimum \( v_\infty \) declination. Second, the spacecraft is required to be injected into the Earth synchronous orbit to re-encounter the Earth after one year cruise. \( v_\infty \) direction must be on the curves of the Earth synchronous orbit (light colored dots) for this purpose. Third, \( e \) should be close to 0.3 considering the balance between \( v_\infty \) increase efficiency and...
the spacecraft’s thermal design. Considering the increase of $e$ by the maneuver during the cruise, the points on the curves of $e = 0.25$ (dark colored dots) should be selected as $v_\infty$ direction. Consequently, two directions (solid and vacant circles) turn out to satisfy these conditions. In this analysis, $(\alpha, \delta) = (-98.1 \text{ deg}, 21.8 \text{ deg})$ (the left solid circle) is selected as the launch $v_\infty$ direction.

Fig. 8 shows the trajectory profile of the first cycle of Case 2. The definition of the coordinate system and plots are the same as those of Fig. 4. Fig. 8(a) shows that the spacecraft’s orbit has eccentricity, and intersects with the ecliptic at the Earth’s position at the launch. The acceleration vectors projected in $XY$ and $XZ$ plane (Fig. 8(a) and (b) respectively) indicate that the vectors lie approximately in the orbit plane to increase $v_\infty$ at the next Earth encounter.

The Earth swing-by condition (which is equivalent to the post swing-by $v_\infty$) is designed so as to satisfy the aforementioned requirements and constraints. In the beginning, $v_\infty$ is set to 10.6km/s which is provided as the trajectory design result of the first cycle. The remaining variables are the direction of $v_\infty$, that is, $\alpha$ and $\delta$. Shown in Fig.9 are the contours of various parameters plotted on $\alpha \delta$ plane. First, the perigee altitude ($h_p$) at the swing-by must be positive so that the spacecraft does not crash into the Earth. The permissible area from this point is surrounded by the curve of $h_p = 0$. Second, the spacecraft is required to be injected into the Earth synchronous orbit after the swing-by to re-encounter the Earth after one year cruise. $v_\infty$ direction must be on the curves of the Earth synchronous orbit (light colored dots) for this purpose. Third, $e$ should be close to 0.3 considering the balance between $v_\infty$ increase efficiency and the spacecraft’s thermal design. Considering the increase of $e$ by the maneuver during the cruise, the points on the curves of $e = 0.25$ (dark colored dots) should be selected as $v_\infty$ direction. Consequently, the direction
(α, δ) = (−104.1 deg., 43.1 deg.) (solid circle) turns out to satisfy these conditions as the post swing-by direction. Fig. 10 shows the trajectory profile through the sequence of Case 2. The definition of the coordinate system is the same as that of Fig. 5. Being different from Case 1, it is observed that the first four cycles whose inclination is low have asymmetry resulted from the orbit eccentricity. However, the orbit is finally circularized by the Earth swing-by, which results in the symmetry observed in the final orbit (the orbit whose inclination is the highest) shown in Fig. 10.

The trajectory sequence of Case 2 is summarized in Table 2. The definitions of the listed values are the same with those of Table 1.

Three points must be pointed out on Table 2. The first, the target SEQ of 45 deg. is achieved by 5 years from the launch. That is, from the point of the orbit design, on condition that 5 years is acceptable, it is possible to reach the Solar latitude 45 deg. by Case 2. The second, m at the end of the sequence is 834 kg, which is improved evidently compared to Case 1. It results in the spacecraft mass budget with the plus margin of approximately 80 kg, which means that Case 2 is feasible from the point of the brief spacecraft design. The third, the sum of \( t_{\text{IES}} \) through the sequence is approximately 20000 h, which is also improved evidently compared to Case 1. Though it is still longer than the operation results of HAYABUSA, it is supposed to be within the acceptable range. To summarize, Case 2 is concluded as feasible at least from these points listed above. The use of the more powerful launcher, the tuning of the ion engine specification, or the use of the Venus swing-by may improve the situation further.

3. Summary

Reported in this paper are the trajectory design results for the orbit transfer to the high Solar latitude by use of SEP. Two cases are investigated. They are, a simple strategy which uses SEP to increase the inclination (Case 1), and a strategy which uses SEP combined with the Earth swing-by (Case 2). The results show that Case 1 is infeasible from the points of the mass budget and the ion engine operation time, whereas Case 2 is feasible from these points.

References


Table 2. Trajectory Sequence of Case 2

<table>
<thead>
<tr>
<th>Date</th>
<th>Event</th>
<th>m (kg)</th>
<th>( v_\infty ) (km/s)</th>
<th>( \alpha ) (deg.)</th>
<th>( \delta ) (deg.)</th>
<th>( i_{\text{SEQ}} ) (deg.)</th>
<th>( v_\infty ) (km/s)</th>
<th>( t_{\text{IES}} ) (total / unit)</th>
</tr>
</thead>
<tbody>
<tr>
<td>2010/06/07</td>
<td>Launch</td>
<td>1200.0</td>
<td>7.7</td>
<td>-98.1</td>
<td>21.8</td>
<td>13.0</td>
<td>2.9</td>
<td>6088 / 5016 h</td>
</tr>
<tr>
<td>2011/06/07</td>
<td>Swing-by #1</td>
<td>1090.8</td>
<td>10.6</td>
<td>-94.3</td>
<td>17.3</td>
<td>13.6</td>
<td>1.1</td>
<td>6182 / 5113 h</td>
</tr>
<tr>
<td>2012/06/07</td>
<td>Swing-by #2</td>
<td>1016.0</td>
<td>13.4</td>
<td>-104.1</td>
<td>43.1</td>
<td>22.1</td>
<td>2.8</td>
<td>6122 / 5157 h</td>
</tr>
<tr>
<td>2013/06/07</td>
<td>Swing-by #3</td>
<td>923.1</td>
<td>16.2</td>
<td>-99.9</td>
<td>36.7</td>
<td>23.5</td>
<td>2.8</td>
<td>5850 / 5121 h</td>
</tr>
<tr>
<td>2014/06/07</td>
<td>Swing-by #4</td>
<td>834.3</td>
<td>19.0</td>
<td>-112.4</td>
<td>53.9</td>
<td>29.6</td>
<td>2.8</td>
<td>5449 / 4893 h</td>
</tr>
<tr>
<td>2015/06/07</td>
<td>Swing-by #5</td>
<td>834.3</td>
<td>19.5</td>
<td>-107.0</td>
<td>59.2</td>
<td>32.0</td>
<td>2.8</td>
<td>5449 / 4893 h</td>
</tr>
<tr>
<td>2016/06/07</td>
<td>Swing-by #6</td>
<td>834.3</td>
<td>19.1</td>
<td>-117.4</td>
<td>64.3</td>
<td>43.6</td>
<td>0.5</td>
<td>0 / 0 h</td>
</tr>
</tbody>
</table>