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Solar–Electric Planetary Missions with an Initial Out-of-Ecliptic Thrust Phase

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A solar–electric propulsion mission profile with Earth gravity assist is considered that differs from more conventional mission designs by letting the thrust phase preceding the Earth swingby be inclined with respect to the ecliptic. The thrust acceleration is oriented normal to the orbit plane thus adding inclination without increasing the 1-astronomical unit solar distance. The out-of-ecliptic mission phase serves to accumulate a major relative velocity increase, directed normal to the spacecraft's orbit plane as thrust time progresses. This effect is essential to gaining greater effectiveness of the subsequent Earth swingby maneuver that converts this relative velocity component to the desired outbound direction, for example, in missions to Jupiter or other outer planets. This mission profile considered here offers considerable flexibility in the selection of launch dates and simplifies a trade between mission duration and solar–electric power. An initial inbound thrust phase in outbound missions is avoided, and thereby, the spacecraft design is simplified. The various characteristics of the modified mission profile that differ from the conventional profile are described, and the respective advantages that are due to these modifications are shown. It is shown that the out-of-ecliptic thrust phase, in combination with Earth swingby, offers considerably greater flexibility in the overall mission profile design, which in turn relieves launch date constraints and simplifies thrust vector control requirements. These factors have not been sufficiently explored as yet in previous contributions to the electric–propulsion mission and system design literature.

Nomenclature

a_{av}	=	average thrust acceleration between beginning and end of the total solar–electric propulsion (SEP) thrust phases
I_{sp}	=	electric thruster specific impulse, s
m_0	=	initial spacecraft mass, kg
n	=	total number of days of pre-Earth swingby (ESB) thrust required to reach desired relative velocity at ESB
P	=	SEP power used for propulsion, varies with mission phase, kW
Q	=	number of pre-ESB thrust phase intervals
R_j	=	Jupiter orbit radius, AU
r_p	=	perigee radius, km
r_0	=	Earth radius, km
T	=	SEP thrust force, N
V_{Earth}	=	Earth's velocity in solar orbit, km/s
V_{rel}	=	spacecraft relative velocity (magnitude) with respect to Earth, km/s
$V_{rel,1}$, $V_{rel,2}$	=	spacecraft relative velocity, before and after ESB, respectively, km/s
V_∞	=	spacecraft asymptotic velocity, at departure, km/s
η	=	SEP thrust efficiency, %
ϕ	=	ESB angle change of relative velocity vector

Introduction

WITH the advent of solar-electric propulsion (SEP) as a practical option for planetary missions,[1–4] it is worth reexamining the mission profile leading to Earth swingby, which was first proposed in 1970.[5] This mission mode includes an extended thrust phase, preceding the Earth encounter, that increases the heliocentric orbit inclination continuously by thrusting in a direction normal to the orbit plane, while maintaining an essentially constant 1-astronomical unit (AU) distance from the sun. The main objective is to accumulate enough relative velocity with respect to Earth, at a fixed power level, during the initial, out-of-ecliptic thrust phase and thereby minimize the subsequent propulsion requirements after Earth swingby (ESB). This is desirable particularly in outer-planet missions where the available SEP output decreases rapidly with solar distance. More conventional SEP mission profiles with Earth gravity assist, as described in Refs. 2 and 3, achieve the desired preencounter velocity gain by a flight path that remains entirely in the ecliptic plane.

The initial out-of-the-ecliptic thrust phase offers Earth encounter options every 6 months, at successive ascending and descending nodes. At any of these encounters, if within close enough proximity, the gravity assist from Earth can be used to convert the relative velocity vector into a direction parallel (or nearly parallel) to the ecliptic, as shown in Fig. 1 for the case of an outbound mission. Missions to the outer or inner planets, and also to comets or asteroids, can effectively utilize this technique.

One of the principal advantages of this mission mode is to permit a choice of the initial thrust phase duration, for example, 6, 12, 18, ... months, which facilitates trading the thrust magnitude and, therefore, the required amount of solar array power, against the total thrust time, before and after Earth swingby. A mission that would require 10 kW of propulsion power, using a 1-year initial thrust phase, can thus be performed with only about 5-kW propulsion power by lengthening this thrust phase to 2 years. This allows significant solar array, propulsion system, and overall spacecraft size and mass reductions and corresponding cost savings. Also, the required launch vehicle capability may thereby be reduced, and additional cost savings can be realized. Otherwise, the solar array and propulsion system mass reductions also can be used to increase the payload mass. By comparison, the more conventional SEP mission mode, where the initial, pre-Earth-encounter thrust phase remains in the ecliptic plane, imposes more stringent limits on the variation of thrust duration and, hence, on the trade between propulsion power and total flight time.

The modified mission mode permits gaining sufficient mission energy by including an initial thrust phase followed by Earth gravity assist without requiring an additional swingby of another planet. Thus, launch date constraints can be avoided or mitigated that are based on the combined synodic period of that planet and Earth and that of the target planet such as Jupiter or Saturn. In this respect,

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the conventional SEP-mission mode that uses an initial, in-ecliptic thrust phase also can provide this advantage.

The paper gives a detailed description of the modified SEP-mission profile and compares it with the conventional mission profile that remains in the ecliptic plane before the ESB. The greater mission design flexibility and the increased range of time vs power and propulsion performance trades, as well as various other advantages inherent in the proposed mission mode, will be evaluated in quantitative terms.

Characteristics of the Out-of-Ecliptic Mission Profile

In the out-of-ecliptic initial pre-ESB thrust phase (Fig. 1), the thrust acceleration, always directed normal to the sun line, continuously increases the orbit inclination relative to the ecliptic while the solar distance of 1 AU remains constant. The thrust direction is reversed at the antinode, that is, 90 deg after Earth departure, to further increase the orbit inclination. At the descending node, 6 months after launch, the relative velocity gained during this thrust phase is converted parallel (or nearly parallel) to the Earth's heliocentric velocity by passing Earth at close range. This velocity conversion is shown at the bottom of the figure by the angle change ϕ between the arrival and departure hyperbolic excess velocity vectors $\boldsymbol{V}_{\mathrm{rel},1}$ and $\boldsymbol{V}_{\mathrm{rel},2}$ and their effect on the heliocentric velocities $\boldsymbol{V}_1$ and $\boldsymbol{V}_2$. The amount of the swingby angle change will be discussed in a later section. Spacecraft orientation reversals between the thrust phases in the ascending and descending parts of the inclined orbit and at the beginning of the post-ESB thrust phase also are shown in Fig. 1.

The geometry depicted here shows only one 6-month out-of-ecliptic thrust phase. Generally, it is advantageous to extend this thrust phase by 6 or 12 months, that is to the second or third Earth encounter opportunity. Thus, a higher relative velocity is accumulated, which shortens the required post-ESB thrust phase and limits the loss of propulsion power due to the increasing distance from the sun. Insertion of a coast phase of several weeks between thrust periods in the ascending and descending orbit segments is appropriate. Thrusting during this interval would primarily affect only the nodal position, first in the forward and then in the backward direction while increasing the orbit inclination only slightly, so that propellant use becomes inefficient.

Extending the initial thrust phase tends to reduce the required acceleration level and, hence, the SEP thrust and the solar array power required to accomplish the mission. Because the out-of-plane trajectory has the same orbit period as Earth, there will be Earth-encounter opportunities at each node, that is, every 6 months, if the Earth and spacecraft orbits are exactly circular. Thus, there will be considerable flexibility in mission design, timing, and acceleration requirements, a principal advantage offered by the proposed mission profile. Actually, the Earth's orbit and the inclined spacecraft orbit both have the same slight eccentricity because no thrust component is applied that would change the spacecraft in-orbit velocity profile from that of Earth. Thus, the nodes can always be made to coincide with near-Earth passages, even if not at precisely 6-month intervals. If the desired Earth swingby is to be postponed to a subsequent node crossing, any close approaches at the preceding encounters must be avoided to minimize Earth gravity effects on the spacecraft trajectory. This only requires minor guidance corrections before and after each passage.

Trade of Thrust Level and Thrust Duration Before ESB

The accumulation of relative (heliocentric) velocity with respect to Earth in the pre-ESB thrust phase depends on the thrust acceleration level, on the inclination change effectiveness, and on the thrust phase duration. Because the inclination change due to a given out-of-plane acceleration decreases in proportion with the cosine of the central angle, as measured from the nodal crossing, it is best to stop thrusting before this thrust effectiveness goes to zero, at the antinodes. Figure 2 shows the average thrust effectiveness for cutoff

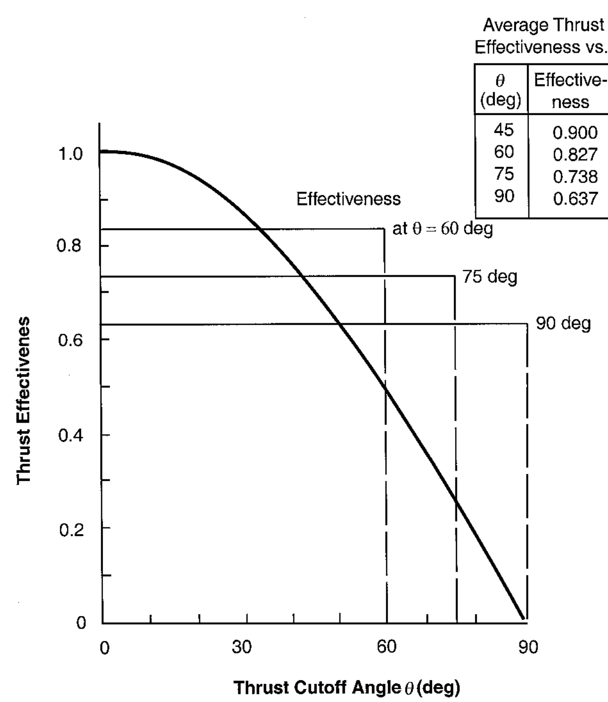


Fig. 2 Average thrust effectiveness for several cutoff angles.

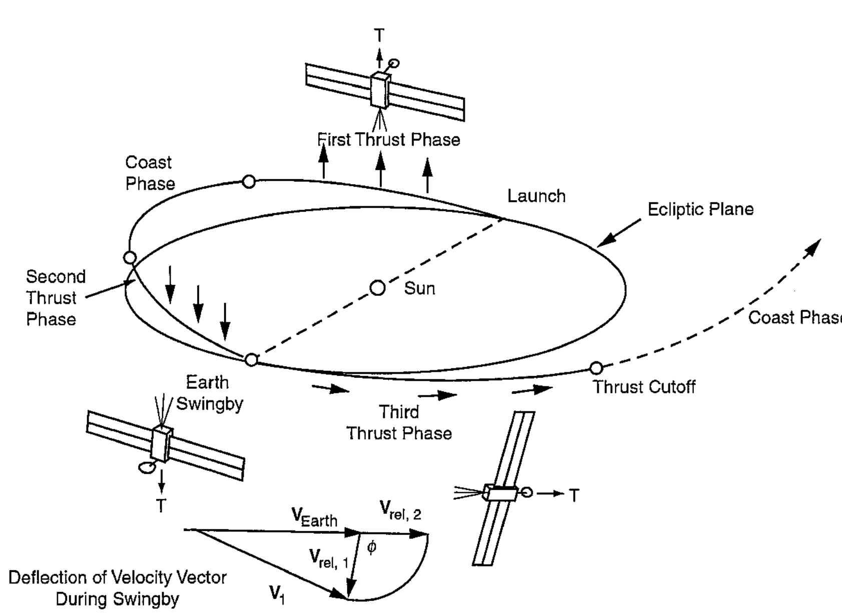


Fig. 1 Outer planet mission with ESB: mission profile and thrust pointing requirements.

angles of 60, 75, and 90 deg after the nodal crossing. The decrease of the average thrust effectiveness as function of cutoff angle is tabulated in Fig. 2. The average thrust acceleration actually increases slowly with propellant expenditure. (As an approximation, a pre-ESB propellant expenditure of 16% is reflected by an 8% increase in the average acceleration levels used in the data to be given.)

For performance comparisons of the modified mission mode with the more conventional SEP mission to Jupiter described in Ref. 2 (Fig. 3), the same power, initial mass, and electric propulsion characteristics at 1 AU are assumed for both mission types: initial mass $m_0 = 583$ kg, SEP power $P = 3.25$ kW, specific impulse $I_{sp} = 3500$ s, and efficiency $\eta = 60\%$.

For the assumed SEP characteristics, the thrust force T is 0.114 N in accordance with the equation $T = 2.040(P\eta/I_{sp})$. It remains constant during the pre-ESB thrust phase at the 1-AU solar distance. The average effective acceleration a_{av} is obtained by taking into account the reduction in thrust effectiveness and the slight decrease in spacecraft mass, as discussed earlier. The number of days, n, of thrusting to reach a desired relative velocity V_{rel} at ESB for three values of the thrust cutoff angle is $n = 1.40 \times 10^{-5} V_{rel}/a_{av}$ for 60-deg thrust cutoff angle, $1.57 \times 10^{-5} V_{rel}/a_{av}$ for 75-deg thrust cutoff angle, and $1.82 \times 10^{-5} V_{rel}/a_{av}$ for 90-deg thrust cutoff angle.

The corresponding number of 6-month thrust phase intervals Q before ESB for two values of relative velocity (5 and 6 km/s) at swingby is shown in Fig. 4 as function of SEP input power, ranging from 2 to 6 kW. Also shown is the effect of using 60-, 75-, and 90-deg thrust cutoff angles, which corresponds to Q value differences of 0.25–0.5 units for the range of SEP power shown. These data illustrate the trade between power and propulsion phase duration that is offered by the modified mission profile. In the reference mission (Fig. 3), the corresponding pre-ESB thrust duration is 317 days, a time interval that cannot be varied as conveniently.

Figure 4 shows that for three thrust phase intervals (1.5-year thrust phase duration) P ranges from 2.5 to 2.9 kW to reach 5 km/s relative velocity. For four periods (2 years) P ranges from 1.9 to 2.2 kW. The corresponding values for reaching 6 km/s relative velocity are 3.1–3.5 kW for three periods, and 2.3–2.6 kW for four periods. The upper and lower values correspond to 60- and 90-deg thrust cutoff angles, respectively. The extra half year of the pre-ESB phase saves about 0.6–0.7 kW of power in the 5-km/s case and 0.8–0.9 kW in the 6-km/s case. The corresponding propellant expenditure is 88 kg for both $V_{rel} = 5$ and 6 km/s, with a 54-deg thrust cutoff angle in both cases. The much more propellant consuming but power saving use of a 90-deg cutoff angle would require 122 and 147 kg of propellant, respectively, for these two relative velocities.

Of practical interest are thrust accelerations of 20–50 μg ($\approx 2 \times 10^{-4}$–5×10^{-4} m/s^2) typically used in interplanetary SEP missions. The preencounter thrust durations to reach $V_{rel} = 6$ km/s at these acceleration levels range from 340 to 140 days, corresponding to pre-ESB thrust phases of 12 and 6 months. Lower accelerations would require extending the pre-ESB phase to Earth encounters 18 or 24 months after launch. Some flexibility is gained by lengthening or shortening the coast phases between thrust cutoff and resumption near the inclined-orbit antinodes. However, a low-thrust cutoff angle is generally preferred to maximize the thrust efficiency.

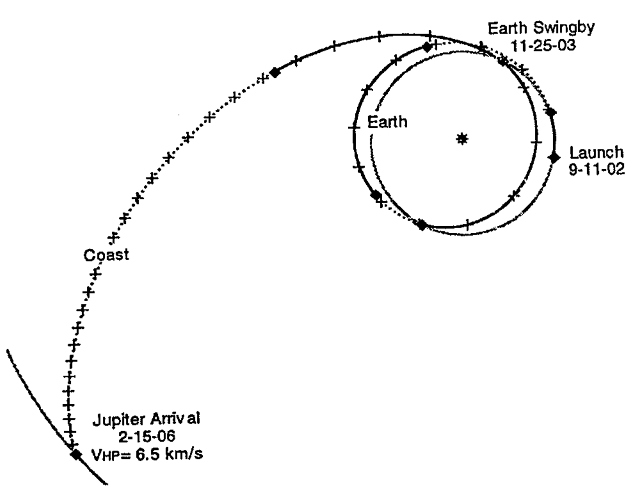


Fig. 3 SEP Earth gravity assist Jupiter orbiter trajectory with in-ecliptic pre-ESB thrust phase (from Ref. 2), + indicate 30-day intervals.

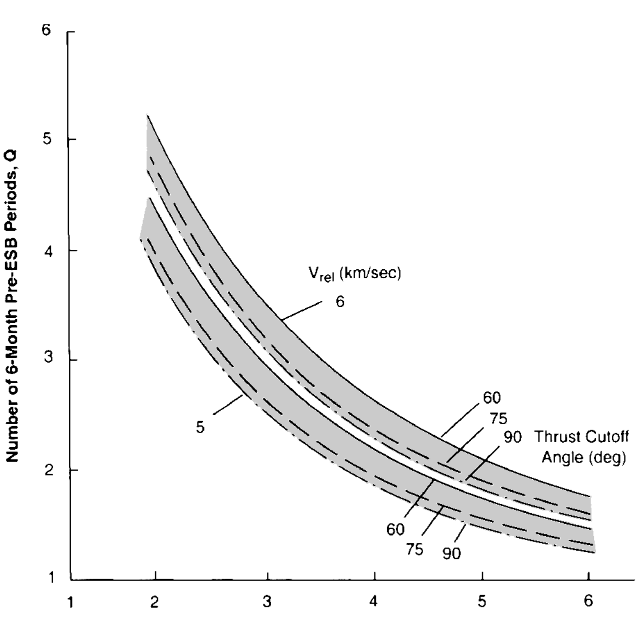
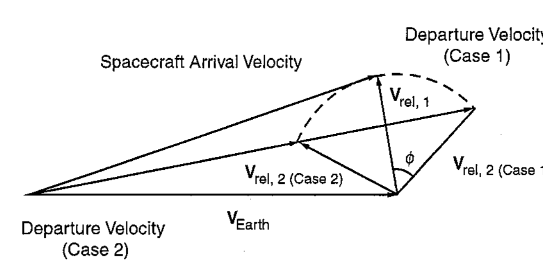


Fig. 4 Number of pre-ESB periods vs SEP power.

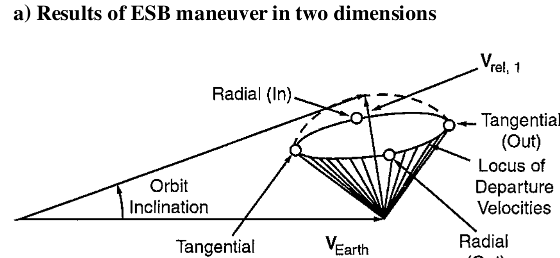
ESB Maneuver Characteristics and Implications

The vector diagrams in Fig. 5 show details of relative velocity changes at Earth swingby. Figure 5a shows deflection angles ϕ of the relative velocity vector in posigrade or retrograde directions, with an increase or decrease of the heliocentric velocity vector, for outbound or inbound missions, respectively. Other options are possible, as indicated in a three-dimensional version of this velocity diagram (Fig. 5b). Figure 5 shows that any outgoing relative velocity vectors $\boldsymbol{V}_{rel,2}$, deflected by the angle ϕ from the incoming relative velocity vector $\boldsymbol{V}_{rel,1}$ lie on the surface of a cone with half-angle ϕ. The axis of this cone is parallel to $\boldsymbol{V}_{rel,1}$. (For clarity of illustration, a half-cone angle much smaller than 90 deg is shown here.) Actually a deflection of about 90 deg would be required to attain a post-ESB trajectory in or nearly in the ecliptic plane.

Mission types of principal interest are those to the outer planets, to the inner planets, and to asteroids and comets, which often have highly eccentric orbits and also significant orbit inclination angles. The latter targets would require a swingby maneuver with



a) Results of ESB maneuver in two dimensions



b) Results of ESB maneuver in three dimensions

Fig. 5 Velocity vector diagrams of ESB.

the outgoing $\boldsymbol{V}_{\rm rel}$ that may include a radially inward- or outward-pointing component.

The Earth encounter geometry must be selected carefully to achieve the desired postencounter mission profile and to gain the maximum energy advantage from the swingby maneuver. With Earth's comparatively small gravity only providing moderate velocity deflections during swingby, the choice of encounter conditions for achieving a given mission objective is actually quite narrow. Figure 6 shows the variation of the deflection angle ϕ as function of the $V_{\rm rel}$ magnitude and the closest approach distance r_p (expressed nondimensionally in terms of Earth radius r_0). With increasing values of $V_{\rm rel}$ and r_p, this angle decreases rapidly. For purposes of this discussion, we assume an upper limit of ϕ at approach distances close to $r_p/r_0 = 1$. In past ballistic ESB missions such as the Galileo Jupiter orbiter mission, the closest approach has been as low as about 1.05, that is, only 300-km altitude. The angle ϕ decreases to 90 deg at $V_{\rm rel} = 5$ km/s and to 60 deg at $V_{\rm rel} = 8$ km/s. Beyond this point, the effectiveness of the swingby diminishes rapidly, and the conversion of the velocity gained during the pre-ESB thrust phase into a usable velocity increment is significantly reduced. For example at $V_{\rm rel} = 8$ km/s, where $\phi = 60$ deg, the reduction would be 13.4%, and the resulting relative velocity in departure direction is reduced to 7.1 km/s. This places an upper bound on the magnitude of the actual velocity increment to be gained during the pre-ESB phase.

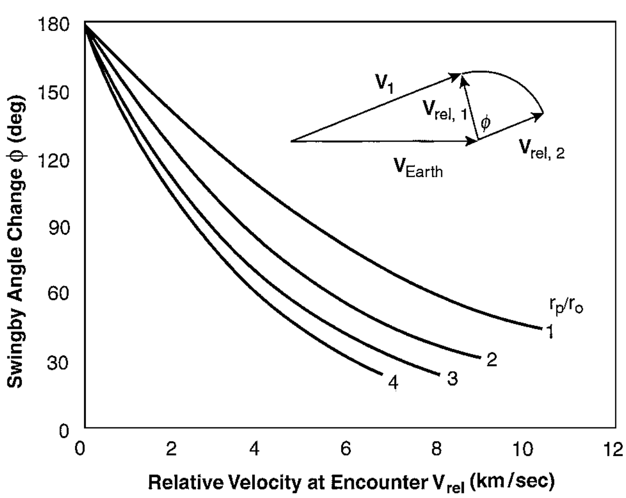


Fig. 6 Angle deflection of relative velocity vector.

The decrease in rotation ϕ of the relative velocity vector at large ESB velocities leads to an incomplete conversion of the trajectory into the ecliptic plane. However, there is a mitigating consideration: a few degrees of residual inclination of the V_2 vector departing from Earth actually can be quite acceptable. The resulting slightly inclined transfer orbit often is consistent with reaching the target object at a location slightly above or below the ecliptic. This condition can occur in outbound missions, for example, to Jupiter or Saturn, if the total transfer angle is close to 180 deg. A rotation angle ϕ somewhat smaller than 90 deg will be required in these cases.

Jupiter Flyby Mission Examples

Two examples of Jupiter flyby missions via ESB were derived on the basis of the pre-ESB thrust phase performance data discussed earlier. The post-ESB trajectories in the ecliptic plane were obtained with the thrust vector pointing along the velocity vector. The thrust cutoff point was set at the time when the subsequent ballistic transfer would reach Jupiter ($R_j = 5.20$ AU) at its aphelion. The sample trajectories, one of which is shown in Fig. 7, assume 5- and 6-km/s Earth departure velocities after swingby. The corresponding reference Jupiter mission trajectory using ESB but no other planetary gravity assist, and remaining in the ecliptic during the pre-ESB phase, is that shown in Fig. 3. Identical electric propulsion characteristics and initial input power, 3.25 kW at 1 AU, are used in these examples. The efficiency variation with solar distance of a silicon solar array (see Ref. 2) is included in these data.

For integrating the post-ESB trajectories shown in Fig. 7, a custom-designed version of Microcosm's high-precision orbit propagator (HPOP) was employed in a form adapted to heliocentric SEP missions. (The HPOP code, described in Ref. 6, was originally developed for use in Earth orbital missions.)

Table 1 lists relevant characteristics of the two sample transfers, based on the same power and thrust levels at 1 AU and the same initial spacecraft mass. The data shown include the thrust phase durations and transfer times, as well as the propellant mass, the spacecraft mass at burnout, and characteristics of the pre- and post-ESB phases. Also listed for comparison are the corresponding characteristics of the reference trajectory shown in Fig. 3. In the cases considered

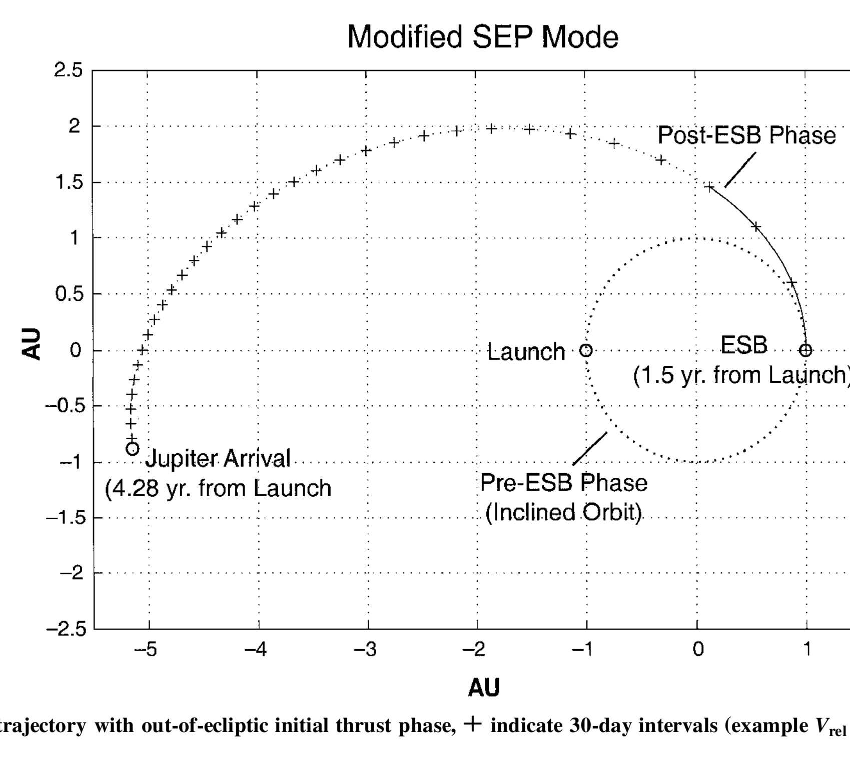


Fig. 7 Jupiter mission trajectory with out-of-ecliptic initial thrust phase, + indicate 30-day intervals (example $V_{\rm rel}$ = 6 km/s at end of pre-ESB phase).

here, the assumed launch vehicle is the Medlite/Delta 7326. For the larger Delta 7925 launch vehicle and the larger initial spacecraft mass and SEP power level, also considered in Ref. 2, the differences in mission characteristics would be comparatively larger. However, emphasis is placed here on the smaller-size and lower-cost mission type that is of timely interest in space exploration planning.

The two examples of the modified mission profile, both using a 1.5-year (548 days) pre-ESB phase, require practically the same total propellant mass, although the thrust cutoff angle, that is, 48 deg, is somewhat smaller in the case of $V_{rel} = 5.0$ km/s and therefore leads to a slightly smaller pre-ESB propellant requirement. If the pre-ESB phase of the modified mission profile is shortened to 1 year, a larger thrust cutoff angle will be required, and the total propellant consumption increases by 15 kg or 14%. The advantage gained would be a shortening of the total trip time by 183 days or 11%.

The overall performance of the two cases of the modified mission mode listed in Table 1 is very similar to that of the conventional mission mode, with equal total propellant consumption and final gross spacecraft mass. However, the advantages inherent in the modified mode, particularly the simpler propulsion subsystem design and the less demanding thermal control being required here reflect in a substantially greater net payload mass and lower development cost than the conventional mission mode, as will be further discussed in the next section.

Table 1 Comparison of modified and conventional mission modes in Jupiter mission example

Item	Modified mode V_{rel}[a] 5 km/s	Modified mode V_{rel}[a] 6 km/s	Conventional mode (Fig. 3)
Thrust force (1 AU), mN	114	114	114
Initial S/C mass, kg	583	583	583
Pre-ESB			
Thrust time, days	281	349	317
Thrust cutoff angle, deg	48	57	——
Phase duration, days	548	548	440
Propellant mass, kg	66	81	——[b]
S/C mass at ESB, kg	517	502	——[b]
Post-ESB			
Thrust time, days	190	90	153
Transfer time, days	1044	1015	810
Thrust at cutoff, mN	28	57	25
Sun distance at cutoff, AU	2.125	1.388	2.20
Power at cutoff, kW	0.904	1.890	0.850
Propellant mass, kg	36	22	——[b]
S/C mass at cutoff, kg	481	480	480
Total mission			
Total thrust time, days	471	439	470
Total trip time, days	1592	1563	1250
Total propellant mass, kg	102	103	103
Final S/C mass, kg	481	480	480

[a] V_{rel} at ESB. [b] Data unavailable from source.

Advantages of the Out-of-Ecliptic Initial Thrust Phase

As already mentioned, some of the principal advantages of using the proposed mission profile include the following. 1) There is flexibility of the pre-ESB thrust phase duration. 2) Acceleration levels and, hence, propulsion power requirements vs pre-ESB thrust duration may be conveniently traded. 3) In outbound missions, there is avoidance of an initial thrust phase at solar distances below 1 AU. This avoids extra thermal control requirements for the spacecraft and extra propulsion subsystem capacity above that available at 1-AU solar distance. 4) Full solar array power is utilized at 1-AU solar distance, limiting the decrease of propulsion power after ESB in outbound missions. 5) Shorter post-ESB thrust duration, ending at lower distance from sun, is accommodated within SEP engine throttling range. Therefore, it does not require a larger number of thrusters that would be rated at a lower nominal power level. This reflects in significant cost and weight savings. 6) There is a simpler thrust-orientation and control profile and simpler navigation and guidance during the pre-ESB phase compared with the conventional (in the ecliptic) pre-ESB mission mode. 7) Combined launch of several spacecraft with different target destinations is made possible by using different ESB encounter conditions, for example, accommodating both inbound and outbound post-ESB trajectories. Table 2 lists these and other benefits associated with the proposed mission profile. The corresponding parameters and conditions characterizing the more conventional mission profile also are indicated for comparison, where applicable.

As a specific example, consider the Jupiter flyby mission described earlier and compare its pre-ESB thrust phase with that of the reference mission (Fig. 3), which initially involves an outbound phase to a maximum solar distance of 1.2 AU and subsequently an inbound phase to a minimum distance of 0.75 AU. This minimum solar distance requires added thermal protection against nearly twice the thermal loads at Earth. The modified mission profile avoids this design requirement. The addition of extra thrusters to conform with the higher available propulsion power available during the inbound excursion is avoided. The modified mission mode allows a simple trade between a greater pre-ESB thrust phase duration and lower required power levels, with Earth encounter options occurring every 6 months. The additional benefit of less thrust level throttling required in this mission mode will be explained.

Under the assumption of 3.38 kW reference solar array power, as in Ref. 2, with 3.25 kW used for propulsion, a SEP specific mass of 40–50 kg/kW and a solar array specific mass of 8–12 kg/kW, the solar array and SEP propulsion subsystem mass together account for about 180 kg, or nearly 40% of the 480-kg dry mass at launch. Reducing this by a factor of two by doubling the pre-ESB thrust duration would not only allow major savings in spacecraft and subsystem cost, but also would permit an increase in payload instrument mass by at least 80 kg, if the same spacecraft mass were launched by the assumed launch vehicle. The projected spacecraft cost savings, however, would be offset in part by the cost of added mission duration. Additional analysis of the savings that might be

Table 2 Advantages of modified SEP mission mode

Advantages	Details	Constraints of conventional mode
Flexible pre-ESB phase duration	Multiples of 6 months	Typically 12–15 months
Adjustable launch window	6, 12, 18, ... months before ESB date Accommodates launch date preferences	12–15 months before ESB date
Easy SEP power and thrust level trade vs pre-ESB phase duration	Result of pre-ESB thrust time flexibility Allows size, mass, cost reductions of propulsion subsystem S/C mass reductions may allow smaller launch vehicle	Not as flexible in changing pre-ESB phase duration
Avoids $r < 1$ AU before ESB in outbound missions	No extra thermal protection needed No added thrusters needed	Goes to 0.75 AU (requires these features) Needs added thrusters
Avoids $r > 1$ AU before ESB	Uses full 1 AU power throughout pre-ESB phase	Goes to 1.2 AU, reduced SEP power
Simplified navigation, guidance, and control	Thrust vector always normal to sun line, parallel to Earth line Earth encounters occur every 6 months	More complex thrust orientations Must guide carefully to Earth for swingby
Accommodates post-ESB inclined transfer orbit to comets and asteroids	ESB angle change selected for this purpose	Limited. Requires extra ΔV to leave ecliptic

accomplished with different spacecraft sizes and power requirements would be warranted, but is beyond the scope of this paper.

As shown in Table 1, the total thrust duration is reduced from 471 to 439 days in the case of 6-km/s ESB relative velocity for the modified mission mode compared with the conventional mode. Current estimates project a 10,000-h, or 417-day, average lifetime of ion thrusters. This lifetime is exceeded considerably in the case of the conventional mission mode (470 days), whereas the modified mode (for the case of $V_{\mathrm{rel}} = 6$ km/s) requires a thrust duration that is lower by about 30 days.

In the reference mission, the inbound excursion to 0.75 AU increases the maximum power by 80% to 5.20 kW, and the silicon solar array efficiency is reduced to 85% at this distance, according to the data in Ref. 2. Added thrusters are needed in that mission to take advantage of the higher power level.

In the reference mission mode, the SEP input power is reduced to 0.85 kW (about 25% of the 3.25 kW at 1 AU) during the outbound thrust phase. This necessitates carrying multiple thrusters rated at about one-half of the 1-AU input power level, to permit sequential thruster cutoff rather than requiring excessive throttling. In the modified mission mode, the corresponding power reduction to 1.89 kW (58%) in the 6-km/s relative-velocity case is within a reasonable throttling range. The total number of thrusters in the modified mode, therefore, will be smaller than in the reference mission mode. However, these assessments apply only to outbound missions such as the one being specifically examined here. Other mission types listed in Refs. 2 and 3 require further comparison and evaluation.

Note that the modified mission mode could be subject to a launch-azimuth payload penalty of up to about 5% if the departure direction of the launch asymptote is pointed normal to the ecliptic plane. However, this penalty can be avoided by using a launch asymptote direction that is less than 90 deg to the ecliptic, typically 40–55 deg, depending on the required launch date. As the asymptotic velocity V_∞ is very small, that is, less than 0.8 km/s in the example considered here, this angular change from the ideal departure direction can be readily corrected during the initial months of thrusting and at a negligible increase in the total thrust impulse.

Another option, avoiding the departure angle discrepancy, would be to use a parabolic departure, at $V_\infty = 0$, with a slightly higher initial payload mass and requiring a small extra ΔV increment during the SEP thrust phase. This simplifies the initial thrust direction control procedure. The small increase in the initial gross spacecraft mass is partly used for the greater propellant mass dictated by the slight increase in total low-thrust ΔV before or after Earth swingby. On the whole, there is considerable flexibility in dealing with the launch velocity direction requirements of the modified mission mode, without causing adverse effects on performance characteristics.

An interesting aspect of the pre-ESB phase in the modified mission mode is the departure to an angle of 9.5–11 deg inclination out of the ecliptic, depending on the desired relative velocity at ESB. This offers the opportunity of carrying piggyback payload instruments for astrophysical observations at out-of-ecliptic distances up 25 to 30×10^6 km, at 1-AU solar distance, and always at the same celestial longitude as Earth, an objective of considerable space-science interest, for example, for solar observations. An early paper by Breakwell and Gillespie[7] advocates an out-of-the-ecliptic excursion of this kind specifically for astrophysical observations, but it does not make reference to the SEP propulsion issues discussed here.

Conclusions

Compared with conventional SEP mission profiles, the modified profile described here offers greater flexibility for trading SEP power against increased thrust time and, thus, permits significant solar array and spacecraft size, mass, and cost reductions. Another advantage is the avoidance of an inbound excursion of the pre-ESB trajectory in outer-planet missions that require extra thermal protection and added SEP propulsion capacity for using the increased power level available during that phase. Navigation, guidance, and thrust control become much simpler in the initial 1-AU thrust phase, with the full thrust always directed normal to the spacecraft orbit plane. The repeated Earth encounter options provide launch date flexibility and may alleviate launch date priority conflicts with other missions that are more critically constrained in their specific launch opportunities. A major advantage inherent in the modified mission mode is the reduction of system development cost and subsystem mass due to the simpler design requirements afforded by the out-of-ecliptic pre-ESB phase. A quantitative comparison, in the case of a Jupiter mission, with the more conventional reference mission profile does not show a specific performance advantage of the modified mission profile in terms of propellant mass reduction. However, the net spacecraft mass tends to be greater for the modified mission profile, because of the system and subsystem design simplifications that have been pointed out earlier. The proposed technique is of practical interest in missions to outer or inner planets and also to asteroids and comets, where an inclined post-ESB transfer trajectory is appropriate.

Additional study is recommended to define fully the details and the promising aspects of the proposed SEP mission profile. Some of the potential limitations inherent in the modified mission mode should be more fully investigated. The mission characteristics and performance benefits vary with target locations (inner vs outer planets, out-of ecliptic targets) and with mission objectives at destination, such as flyby vs orbit insertion or landing. Only a transfer mission to Jupiter has been considered in this paper. Comparison with various other mission types described in the literature will be important.

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