BepiColombo Launch Window Design Based on a Phasing Loop Strategy

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The BepiColombo mission to Mercury includes a lunar swingby before leaving the Earth-Moon system. This paper summarizes the launch window design, which is based on a phasing loop strategy: the spacecraft remains in a highly eccentric Earth orbit for several loops before lunar the gravity assist. To control the luni-solar perturbations, tangential manoeuvres are introduced at the pericentre passages. An optimization tool is used to generate, for each launch date, a numerically integrated trajectory that matches the subsequent heliocentric trajectory while minimizing the total ?V of the manoeuvres. Additional trajectories for a ±3s launcher error in the apogee altitude are calculated for each launch date. Two different types of trajectories are investigated: in the first approach, the apogee altitude of the initial orbit is essentially unconstrained, so that, for most of the launch dates, the initial apogee lies above the Moon distance. In the second approach, the initial apogee is constrained to be below the Moon distance. The results obtained show that, for both options and due to the different effect of the Sun perturbation, the lunar swingby opportunities in June and July 2012 are clearly more favorable than those in August, September and October. Comparing both options, an initial apogee below the Moon distance leads to more possible launch dates per launch window and a smaller DV cost to correct for the launcher dispersion.

I. Introduction

THE BepiColombo interplanetary trajectory combines the use of low thrust (provided by a solar electric propulsion module) with gravity assists of Venus and Mercury[1]. The first part of the trajectory is essentially a classical ΔV Earth gravity assist (ΔVEGA). Instead of a direct launch towards Venus, a lunar swingby is used to inject the spacecraft into a heliocentric orbit with a slightly lower perihelion than the Earth orbital radius. Then a ΔV performed at perihelion (in this case, a low thrust arc) increases the aphelion so that the spacecraft encounters the Earth at a different position and with a higher relative arrival velocity. In this way, it is possible to leave the Earth with the required escape velocity to reach Venus (see Fig. 1). Assuming a Soyuz/Fregat launch, this strategy allows to save around 180 kg of mass compared to the direct launch[1].

The lunar gravity assist is designed to put the spacecraft in the proper heliocentric trajectory gaining as much energy as possible from the Moon. The optimum geometry for the lunar swingby is discussed in detail in Ref. 2. For a launch in 2012, there are five different lunar swingby opportunities, one every month from June to October. In all the cases, the lunar gravity assist takes place in the outgoing leg of the bound orbit, when the Moon is close to its descending node. Table 1 shows the final mass and the swingby details for all the lunar swingby opportunities. The spacecraft leaves the Earth's sphere of influence with a relative escape velocity that lies, approximately, in the ecliptic plane. The direction of the relative velocity is mainly opposite to the Earth velocity (*braking* to reduce the perihelion), with an outward component in June and July and an inward component in September and October. Due to the high flexibility of the low thrust propulsion system, for lunar gravity assists from June to October 2012, the final mass in a 400×12000 km orbit around Mercury differs only by less than 10 kg.

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If the bound orbit inclination is larger than the Moon's orbital plane inclination, for every swingby date there are two solutions with similar energy gain: the Moon can be approached from above or from below its orbital plane[2]. The two possible bound orbits have different right ascension of the ascending node and argument of pericentre but the same orbit size and inclination. This happens for a launch from Baikonur. It is interesting from the point of view of the launch window design because the two types of orbits have different relative inclination with respect to the ecliptic plane. Hence, these orbits are affected by the luni-solar perturbations in a different way. However, in the current mission baseline the spacecraft is to be launched eastward from Kourou (the inclination is 5 deg), so that there exists one single solution for every swingby date (the Moon is approached from below its orbital plane).

The minimum pericentre altitude for the lunar swingby is assumed to be 200 km. The plane of the swingby hyperbola lies close to the Moon's equatorial plane.

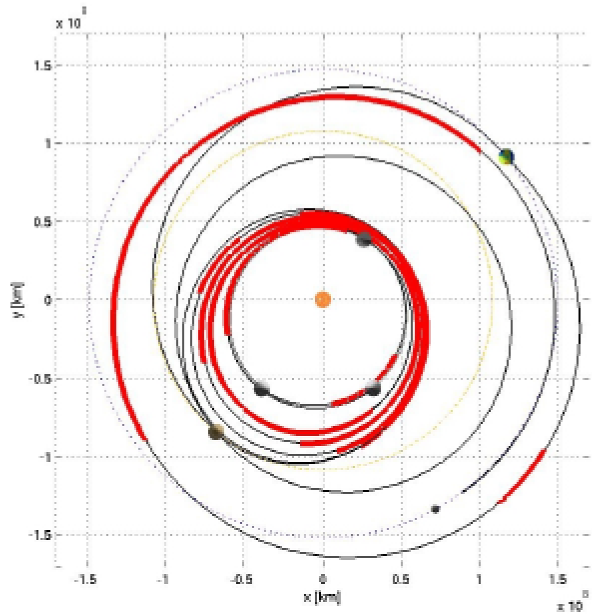

Figure 1. BepiColombo interplanetary trajectory with a lunar gravity assist for a 2012 launch. Thick lines represent thrust arcs.

Table 1. Lunar swingby opportunities in 2012. |V| is the modulus of the spacecraft velocity with respect to the Earth, a **is the angle between the spacecraft velocity and the Sun-Earth direction and** d **is the angle between the spacecraft velocity and the ecliptic plane. The simulations were performed with the tool DITAN[3].**

Lunar swingby date		Final mass in 400×12000 km Mercury orbit	Velocity vector at the exit of the Earth's sphere of influence		
			|V|	α	δ
MJD-2000	Calendar	[kg]	[km/s]	[deg]	[deg]
4560.654	26 Jun 2012	1048.9	1.485	333.0	0.97
4587.896	23 Jul 2012	1056.6	1.461	307.3	0.32
4615.198	20 Aug 2012	1055.1	1.195	270.3	0.26
4642.554	16 Sep 2012	1052.4	1.195	243.2	0.91
4669.970	13 Oct 2012	1049.8	1.349	220.5	3.01

II. Phasing Loop Strategy

To achieve the appropriate velocity vector when leaving the Earth's sphere of influence, the lunar gravity assist must take place at a specific position of the Moon. If the spacecraft were launched directly towards the lunar encounter, the launch window would be constrained to an approximate 2-day period per month. Besides, the correction of the launcher error must be done as quickly as possible and the ΔV cost is high (for a launcher along-track 3σ dispersion of 30 m/s, a correction manoeuvre performed 5 hours after the injection[4] may be as large as 100 m/s). In order to mitigate these problems, a phasing loop strategy is proposed. Rather than launching directly towards the lunar gravity assist, the spacecraft will be injected into a highly elliptical orbit, where it remains for several revolutions (loops) before the lunar swingby. The spacecraft propulsion can be used to adjust the phasing orbits to encounter the Moon with the proper conditions at the right time. The purpose of this strategy is double-fold:

- first, to widen the monthly launch window. Varying slightly the period of the initial orbit according to the launch date and inserting additional loops allows to expand the launch window.
- second, to reduce the ΔV needed to correct for the launcher dispersion. The consecutive perigee passes offer several opportunities to correct for the errors in the launch at very low cost.

This phasing loop strategy has already been used for a number of missions, some of which are listed in Table 2.

Table 2. Summary of missions that included a phasing loop strategy before a lunar gravity assist. The apogee altitude data refer to the actual apogee reached; in the case of Hiten, the planned apogee altitude was 476000 km but the launcher had an underperformance.

Mission	Launch date	Initial apogee altitude [km]
Hiten (Muses-A)	24/01/1990	286183
Geotail	24/07/1992	399941
Nozomi (Planet-B)[5]	04/07/1998	401491
Lunar-A[6]	Planned for 2005	400000
MAP[4]	30/06/2001	292492

An important parameter defining the phasing loop strategy is the apogee altitude of the nominal initial orbit. The initial apogee may be above or below the Moon distance. In the latter case, the launcher performances are higher but the spacecraft must perform a manoeuvre to raise the apogee in order to encounter the Moon with the appropriate relative velocity.

In an early design of the BepiColombo spacecraft, the chemical propulsion module lay below the solar electric propulsion and was therefore not available until the latter was ejected 30 days before the insertion into orbit around Mercury. According to that design, the two only thrust sources that can be used before the lunar swingby are the solar electric propulsion (SEP) and the reaction control system (RCS) thrusters. Both options give only a limited thrust (around 20 N for the RCS thrusters) and are not suited to provide large ΔV (the gravity losses increase rapidly). Consequently, it was decided to select the initial apogee height so as to minimize the ΔV required during the phasing loops to swing by the Moon with the right conditions, rather than selecting it to optimize the final mass.

III. Perturbations in a Highly Eccentric Earth Orbit

Before the lunar swingby, the spacecraft remains for several revolutions (typically 3 to 7) in a highly eccentric Earth orbit (with an apogee altitude ranging from 300000 km to 450000 km). These orbits are strongly affected by the gravitational attractions from the Sun and the Moon. Especially due to the low initial perigee height (200 km), the luni-solar perturbations must be carefully studied.

A. Solar perturbation

The Sun attraction causes periodical changes in the orbital elements. Depending on the direction of the apogee with respect to the Sun, the solar perturbation may cause an increase or a descent in the perigee altitude. The effect is illustrated in Fig. 2. It shows one possible phasing loop trajectory in a rotating coordinate frame with the Earth-Sun direction fixed. If the apogee of the orbit lies in either the first or the third quadrant then the eccentricity of the orbit increases and the pericentre height drops due to the Sun perturbation. On the contrary, if the apogee of the orbit lies in the second or the fourth quadrant then the Sun causes the eccentricity to decrease and, hence, the perigee altitude to increase.

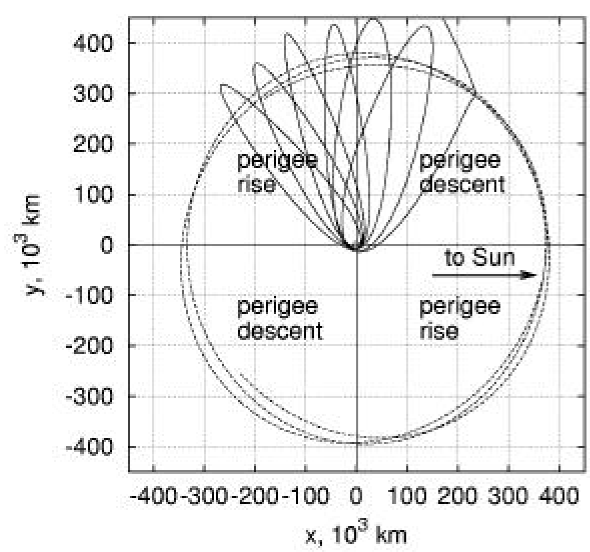

Figure 2. Example of a 6-loop trajectory projected onto the equatorial plane with the Earth-Sun direction fixed. Depending on in which quadrant the apogee of the orbit lies, the Sun attraction has a different effect on the perigee altitude.

The variation in the perigee height depends strongly on the apogee altitude of the initial orbit. For an orbit with a perigee altitude of 200 km and the apogee close to the Moon distance, the perigee height can increase or decrease by as much as 1000 km per revolution, whereas for an initial apogee height lower than 300000 km, the change in the perigee altitude is about 250 km at most. The sign of the effect changes every 90 days (a quarter of a year). The inclination of the orbital plane with respect to the ecliptic also plays a role: the lower the relative inclination, the stronger the effect of the Sun perturbation. The changes in the rest of orbital parameters are not as big as for the eccentricity.

B. Lunar perturbation

The Moon attraction can modify significantly the orbit of the spacecraft during the phasing loops. The changes in the orbital elements depend strongly on the relative geometry of the spacecraft orbit and the Moon and have an approximate frequency of 27 days, a lunar month. Figure 3 illustrates the definition of two angles, θ and γ, which can be used to describe the relative geometry between the spacecraft and the Moon. γ is the angle between the line of apsides of the orbit and the Moon's orbital plane. θ is the angle between the projection of the line of apsides of the orbit on the Moon's orbital plane and the Moon's position when the spacecraft is at pericentre. Note that the orbits that intersect the Moon's orbit and have apogee altitudes around the Moon distance have small values of γ (that is, the line of apsides lies close to the Moon's orbital plane). Then, for small values of γ and a fixed orbit inclination, the lunar perturbation depends only on the value of θ and the apogee height of the orbit.

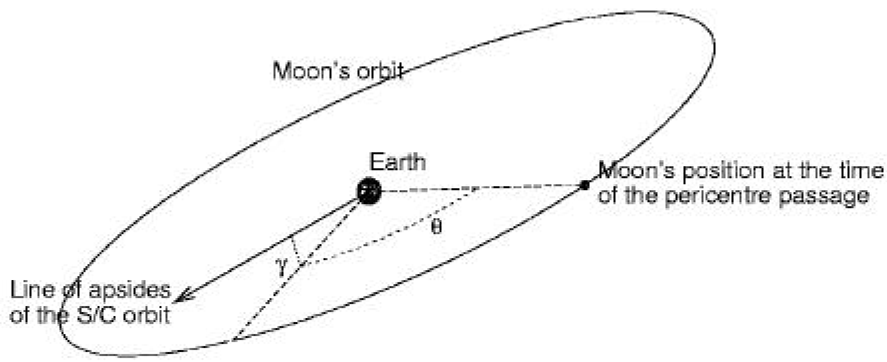

Figure 3. Description of the relative geometry between the spacecraft orbit and the Moon. g **is the angle between the line of apsides of the orbit and the Moon's orbital plane.** q **is the angle between the projection of the line of apsides of the orbit on the Moon's orbital plane and the Moon's position when the spacecraft is at pericentre.**

Figure 4 shows the effects of the Moon perturbation on the orbital elements as a function of θ and the apogee height. Especially critical is the behavior of the perigee altitude. For certain values of θ (approximately from θ = -130 to θ = –30 deg) there is a drastic change in the perigee height. These values of θ correspond to orbits where the spacecraft gets, around its apogee, considerably close to the Moon. These "flybys" are hereafter referred to as a "Moon resonance". For orbits with an apogee altitude below the Moon distance, the minimum distance spacecraft-Moon will happen for θ ≅ –55 deg. For orbits with apogee altitudes above the Moon distance, there is another Moon resonance that happens when the spacecraft crosses the Moon orbit in the incoming leg of the orbit. Around the Moon resonance, the effect of the lunar perturbation is complex but may be explained in the following, simplified way: if the spacecraft is "in front of" the Moon at apogee (leading the Moon), then the lunar attraction is equivalent to a negative ΔV and it causes a perigee descent, whereas if the spacecraft is "behind" the Moon at apogee (trailing the Moon), then the lunar attraction is equivalent to a positive ΔV and the perigee is raised.

For the rest of the dates throughout the month, the effect on the perigee altitude is not so strong but still important, due to the high eccentricity. Indeed, there is another monthly period of around 6 days (θ ranging from roughly from 0 to 100 deg) where the perigee height decreases.

The changes in the apogee height can be as large as 20000 km (with associated changes in the orbital period up to one day) and are important because of the phasing strategy. The orbital plane (the inclination and the ascending node) can rotate as well to a large extent due to the lunar perturbation. To summarize: the lunar perturbation can change completely the initial orbit and its effects depend mainly on the phasing, that is, on the time of the pericentre passage.

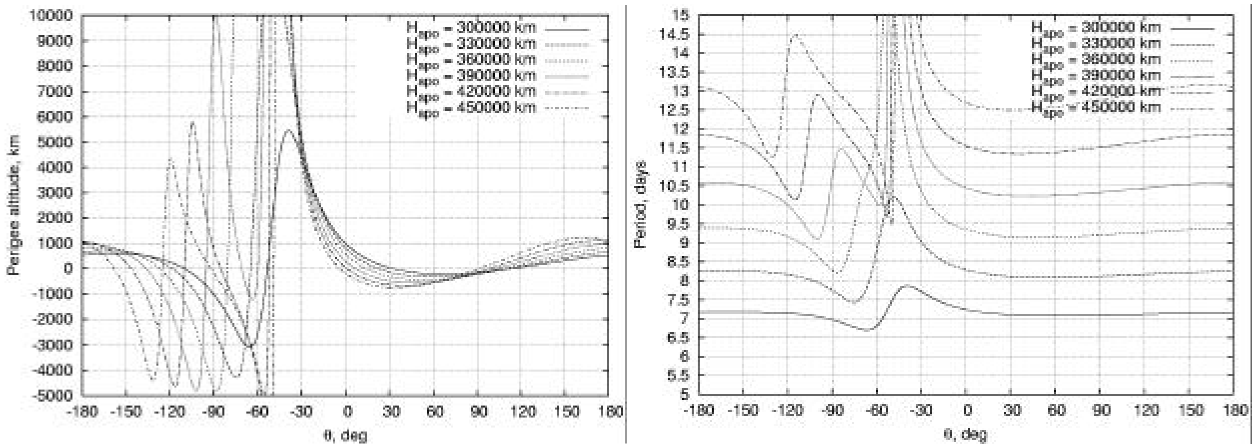

Figure 4. Perigee altitude (left) and period (right) after one orbit due to the Moon perturbation as a function of q. To produce the plots, an initial state vector at pericentre is propagated numerically considering the Moon perturbation for one revolution. The osculating orbital elements at next pericentre are plotted. The initial state vector corresponds to an orbit with perigee height = 200 km, inclination = 5.28 deg, RAAN = -140 deg, argument of pericentre = 138.5 deg and different values of the apogee height as indicated in the plots. The time of the initial pericentre is varied over one lunar month (q ranging from –180 deg to 180 deg).

IV. Numerically Integrated Launch Window

A. Methodology

The previous section demonstrates that the luni-solar perturbations must be taken into account to compute the trajectories. In order to find a new optimal launch date and initial orbit, the complete interplanetary trajectory (until the arrival at Mercury) should be optimized again including the numerical integration of the trajectory inside the Earth's sphere of influence. However, this would increase even more the complexity of the problem to be solved by the optimizer and the computational load. This point is of special importance taking into account that more than 225 different trajectories must be produced to complete the design of the launch window for all the lunar swingby opportunities. Hence, it is necessary to split the problem at a certain point. In the proposed approach, only the part of the trajectory from the launch until the exit of the Earth's sphere of influence is calculated. The matching with the optimal, Keplerian, previously computed, interplanetary trajectory is done imposing constraints on the velocity of the spacecraft when it leaves the Earth-Moon system. The modulus of the velocity vector and its direction are forced to stay within a certain range centered around the optimum value. Table 3 displays the minimum and maximum allowed values for the velocity vector at the exit of the Earth's sphere of influence for the different lunar swingby opportunities. Preliminary simulations performed with DITAN indicate that, for the extreme values of the allowed ranges, the penalty in the final mass delivered into orbit around Mercury is less than 10 kg.

As explained in the previous section, the gravitational attraction from the Sun and the Moon produce complex effects on the spacecraft's orbit. They affect mainly the perigee altitude and the orbital period. In order to maintain a safe perigee height and to control the phasing so that the spacecraft can swing by the Moon at the right time, some manoeuvres are needed during the phasing loops. Because of the sensitivity of the perturbations with respect to the time of the different pericentre passages, the most efficient strategy to produce a trajectory is to control the period of the consecutive orbits with tangential manoeuvres performed at perigee. In addition, a manoeuvre at the first apogee may be necessary to raise the perigee for some launch dates. This sequence of control manoeuvres at apogee and perigee has already been used for some missions that included a phasing loop transfer to a lunar swing-by[4,5].

An optimization problem is used to obtain an optimal, numerically integrated trajectory for each launch date. The details are summarized below:

- An eastward launch from Kourou with Soyuz-2B/Fregat-M is assumed. The assumed launch profile is the so-called "direct ascent", in which there is no coast arc. Hence, the argument of the pericentre of the initial orbit is fixed. The values used for the constant orbital elements of the initial orbit are: perigee height = 200 km, inclination = 5.28 deg, argument of pericentre = 138.5 deg. It is assumed that the Fregat upper stage burn can vary from day to day throughout the launch window. A launch profile with a coast arc in an intermediate parking orbit (giving complete freedom in the argument of pericentre) is currently being investigated.

- The trajectory is propagated forward from the Earth and backward from the Moon. The matching point was chosen close to the apogee of the last complete revolution before the lunar swingby (to ease the numerical problem). From the arrival conditions at the Moon and the swingby parameters (pericentre altitude, inclination of the swingby plane) the trajectory is propagated forward until the spacecraft exits the Earth's sphere of influence. Earth and Moon are treated as point masses, i.e. no J_2 or higher gravitational terms are considered (their effects were shown to be minor[7]).
- A minimum of 3 phasing loops is considered in order to have enough time to correct for the launcher dispersion and target precisely the lunar gravity assist.
- Tangential manoeuvres are allowed at the first apocentre (to raise the pericentre when needed) and at the subsequent pericentre passes (to control the period of the orbit). They are modeled as impulsive ΔV.
- No manoeuvre is allowed at the last pericentre (immediately before the lunar gravity assist) because the errors in the manoeuvre execution may compromise the precise targeting. Since the lunar swingby takes place in the outgoing leg of the bound orbit, the interval between the last pericentre and the gravity assist is typically less than 3 days. This may be too short to perform accurate orbit determination before the final targeting manoeuvre. Note that this constraint would not apply for a lunar swingby in the incoming leg of the bound orbit, which was the optimum geometry for a lunar gravity assist in January 2011[8].
- No out-of-plane component is considered for the manoeuvres because it is more efficient to exploit the lunar perturbation to change the orbital plane as needed by adequately controlling the phasing.
- The minimum allowed perigee altitude is set to 400 km. This is requested in order to ensure the safety of the mission and to avoid large effects of atmospheric drag, which complicates the orbit determination.
- The velocity vector at the exit of the Earth's sphere of influence is constrained as described in Table 3.

Table 3. Assumed constraints on the velocity vector at the exit of the Earth's sphere of influence for the different 2012 lunar swingby opportunities. |V| is the modulus of the velocity. a **is the angle between the velocity and the Sun-Earth direction.** d **is the angle between the velocity and the ecliptic plane.**

		June	July	August	September	October
|V| [km/s]	Min	1.4	1.4	1.15	1.15	1.35
	Max	1.5	1.5	1.25	1.25	1.45
α [deg]	Min	328	302	265	238	215
	Max	338	312	275	248	225
δ [deg]	Min	-4	-4	-4	-4	-4
	Max	4	4	4	4	4

The objective of the optimization is to find, for each launch date, an integrated trajectory that fulfills the constraints while minimizing the total ΔV of the control manoeuvres.

Two different approaches are used to generate the launch window. In the first approach, the apogee altitude of the initial orbit is essentially unconstrained (only limited to be below 475000 km). Most of the trajectories generated in this way have initial apogee altitudes above the Moon distance. In the second approach, the apogee altitude of the initial orbit is constrained to be below 350000 km.

The optimization tool OPRQP is used to generate the trajectories. The large number of variables and constraints involved and the strong non-linearities that appear in the dynamics around the Moon resonances make it a rather complicate problem. As a consequence, the optimizer does not always converge to a solution that fulfills completely all the defined constraints. Moreover, sometimes it cannot jump from one local minimum to another: Even if the number of phasing loops is theoretically free, the final solution has usually the same number of loops than the initial guess. This stresses the importance of having a good understanding of the dynamics in order to generate a proper initial guess.

B. Launcher dispersion

The trajectories obtained according to the methodology described in the previous section are the "nominal" trajectories for each launch date. That is, the launcher is assumed to perform nominally and inject the spacecraft in the required orbit without errors. Actually, the launcher has a certain dispersion and the initial orbit may be different from the nominal. Lacking a model for the Soyuz/Fregat dispersion in highly eccentric Earth orbits, the launcher dispersion matrix assumed for this study is the one provided by Starsem[9] for MARS EXPRESS (which was injected into an escape trajectory).

The effect of the launcher dispersion on the first orbit is illustrated in Table 4, that is obtained by means of a Monte Carlo simulation. It can be seen that the main variation in the initial orbital elements due to the launcher dispersion is produced in the apogee altitude and, as a consequence, in the period of the orbit. The variation in the apogee altitude causes a change in the luni-solar perturbations with respect to the nominal case. This may result in the need to perform a perigee raising manoeuvre at the first apogee that was not needed in the nominal case. In addition, the variation in the period implies that the phasing must be corrected to target the Moon swingby. This is of special importance due to the high sensitivity of the dynamics with respect to the time of the pericentre passage, especially around the Moon resonances.

Table 4. 3σ variation in the nominal orbital elements due to the launcher dispersion. The nominal orbit in this case is 200 $\times$ 400000 km, 5.28 deg inclined.

Apogee altitude [km]	Perigee altitude [km]	Period [days]	Inclination [deg]	RAAN [deg]	Argument of pericentre [deg]
12717	340	0.489	0.087	0.090	0.168

To analyze the effect of the launcher error on the launch window design, a full numerical simulation is performed. For each launch date, the apogee height of the initial orbit is changed according to the $\pm 3\sigma$ launcher dispersion. The rest of orbital elements are kept constant, because the errors in the energy of the orbit are supposed to be the most critical (the validity of this assumption is demonstrated in Ref. 10). Then the control manoeuvres are optimized to get a trajectory that fulfills the constraints. The results are presented together with the nominal trajectories in the next sections.

Note that the ΔV budget for each launch date is not determined by the ΔV requirements of the nominal trajectory, but rather by the maximum ΔV required for the $\pm 3\sigma$ launcher dispersion trajectories. Therefore, in order to minimize the ΔV budget, the nominal trajectory should be the trajectory that minimizes the maximum ΔV of the two trajectories $+3\sigma$ and -3σ. Figure 5 helps to clarify this concept. However, this approach was not adopted as the general method because of the difficulty to implement it and the heavy computational load associated. For some individual cases, though, when one of the trajectories with $\pm 3\sigma$ dispersion in the apogee was clearly worse than the other, the nominal case was recomputed with a different, fixed apogee height.

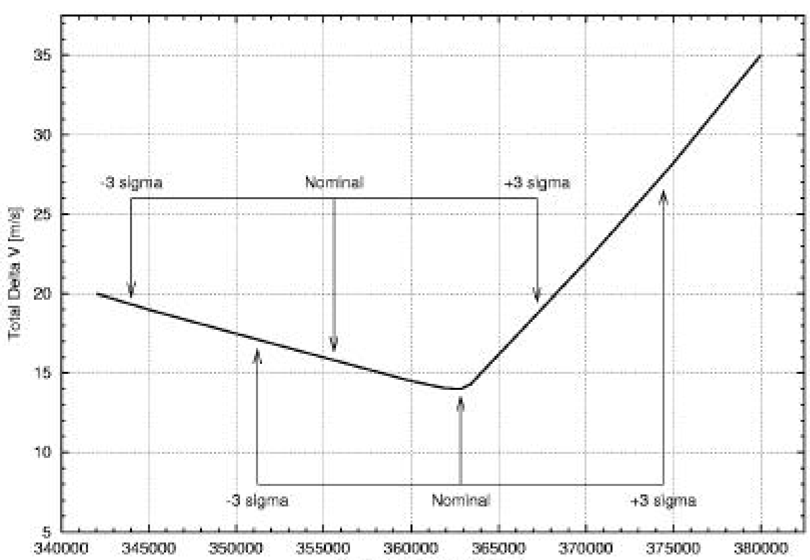

Figure 5. Example of the variation of the total ΔV of the phasing loop trajectory for a certain launch date as a function of the initial apogee height. In the adopted approach, the nominal trajectory is chosen as the trajectory that minimizes the total ΔV. The $\pm 3\sigma$ trajectories are also represented. Because the ΔV budget for each launch date is determined by the maximum ΔV required for the $\pm 3\sigma$ trajectories, in this case it would be better to reduce the initial apogee height of the nominal trajectory.

C. Results with Unconstrained Apogee Height

The launch windows for the different 2012 lunar swingby opportunities obtained with a free initial apogee height are presented in this section. Figures 6 and 7 show the total ΔV and the sequence of pericentre passage times for the launch window corresponding to the July lunar swingby opportunity. The full results (trajectory data in tabular form for all the launch dates and lunar swingby opportunities, from June to October) are included in Ref. 7. Trajectories with launch dates from around 85 to 40 days before the lunar swingby are investigated. The number of complete loops before the gravity assist ranges from 3 to 6, with initial apogee altitude ranging from 300000 to 475000 km. The different launch windows overlap partially: for some launch dates it is possible to find one trajectory with 3 loops targeting to a certain lunar swingby (e.g. June) and another trajectory with 6 loops targeting to a swingby one moth later (e.g. July).

Because the lunar swingby takes place at approximately the same position of the Moon for all the monthly opportunities, the different launch windows look very similar. The differences are caused by the effect of the Sun perturbation (see 8). In the June and July launch windows, the geometry is favorable and the Sun pulls up the perigee for most of the phasing loops. On the contrary, for the August, September and October launch windows, the Sun pulls down the perigee from the first orbit. As a consequence, for most of the launch dates during these months a manoeuvre at first apogee is required to raise the perigee height to the minimum altitude of 400 km. This manoeuvre can be as large as 30 m/s and contributes to increase the ΔV budget significantly.

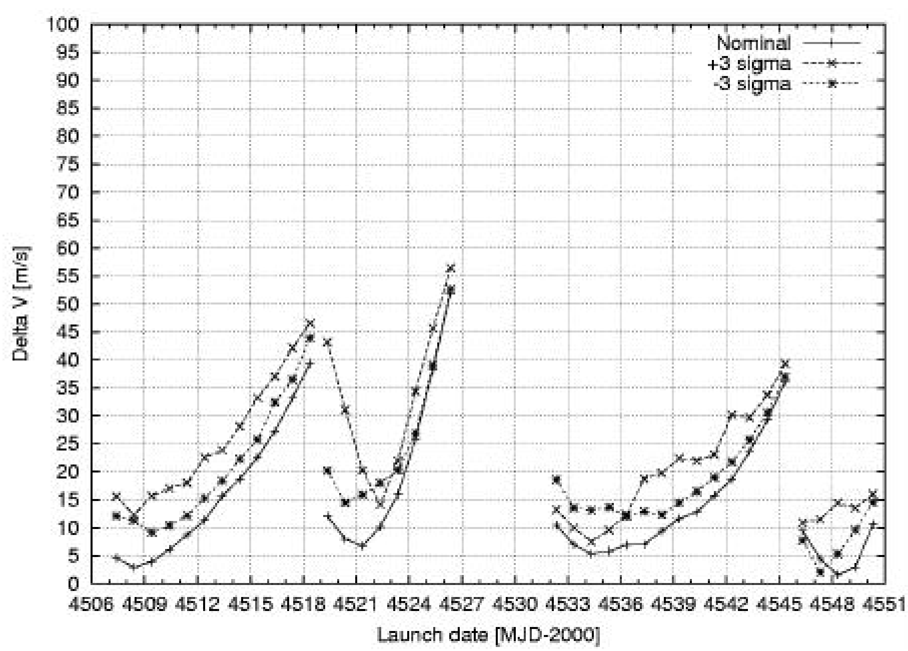

Figure 6. Total ΔV of the different manoeuvres for the nominal case and the cases with $\pm 3\sigma$ dispersion in the apogee altitude (July 2012 lunar swingby opportunity). The trajectories are computed with a free initial apogee height. During the interval from day 4527 to 4532, the spacecraft would pass too close to the Moon at first apogee and no trajectories are found. For a launch in 4546 and 4547, the -3σ dispersion case originally required a very large ΔV. In order to reduce that value, the nominal trajectory was computed again with a higher, fixed apogee height. This is the reason why the new -3σ case requires less ΔV than the nominal.

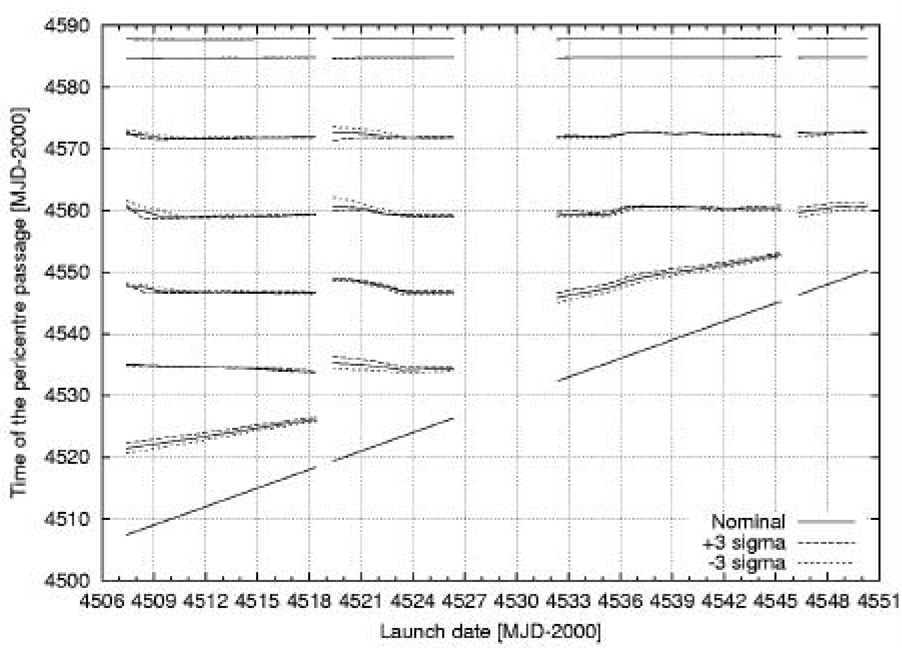

Figure 7. Dates of the pericentre passages and lunar flyby date for the nominal case and the cases with ±3s dispersion in the apogee altitude (July 2012 lunar swingby opportunity). The trajectories are computed with a free initial apogee height. The lunar flyby date is the uppermost horizontal line around 4587 (23 July). There are two intervals of Moon resonance: from day 4527 to 4532 (two months before the swingby) and from day 4553 to 4558 (one month before). The times of the pericentre passages are controlled to avoid those intervals.

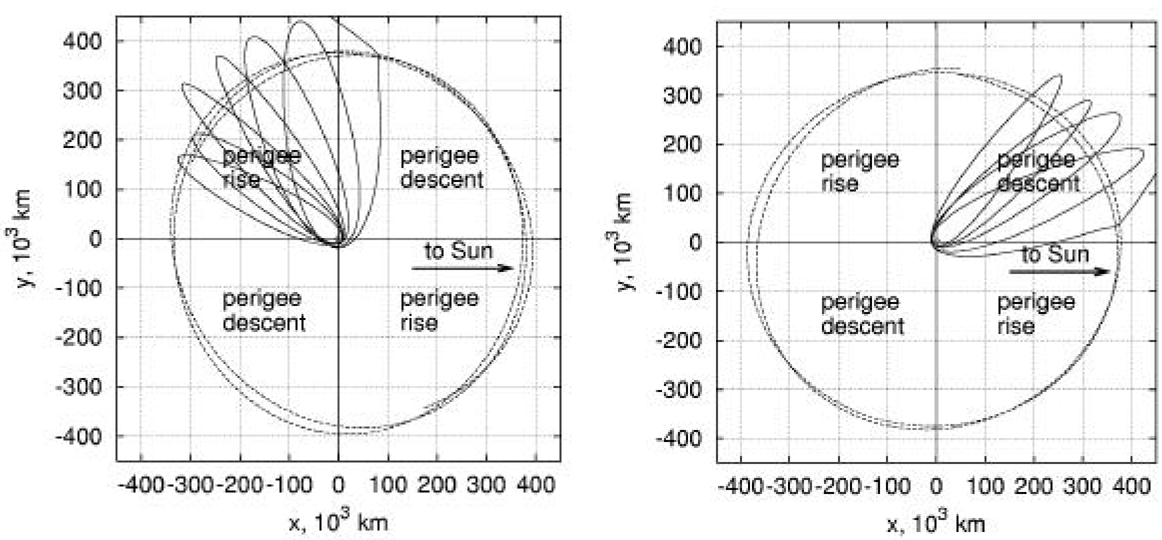

Figure 8. To target the June and July lunar swingby opportunities, the launch dates range from beginning of April to mid-June 2012. During these months, the first loops of the trajectory lie in the second quadrant so that the Sun perturbation increases the perigee height. The figure on the left shows an example of a 6-loop trajectory with launch date 15-Apr-2012. On the contrary, to target the August, September and October lunar swingby opportunities, the launch should be during summer, when the Sun perturbation decreases the perigee height. The figure on the right shows an example of a 4-loop trajectory with launch date 26-Jul-2012.

In addition to the apogee manoeuvres to raise the perigee when needed, some ΔV must be applied at certain pericentre passages to control the orbital period and avoid the negative effects of the Moon resonances. The results obtained help to outline the optimum manoeuvre strategy: the manoeuvres should take place only at the pericentre passages immediately before a Moon resonance (one and two months before the gravity assist). In order to avoid abrupt changes in the orbit, the period of the initial orbit should be carefully selected to control the time of the pericentre passage before the first Moon resonance. At that pericentre passage, a manoeuvre is needed in order to control the pericentre passage time of the next Moon resonance (if there is one) or to adjust the phasing to encounter the Moon at the right time.

The obtained trajectories have one or two pericentre manoeuvres, located at the pericentre passages just before the Moon resonances (see Fig. 9). 3-loop trajectories include a manoeuvre at the first pericentre and 4-loop trajectories a manoeuvre at the second pericentre. 5-loop and 6-loop trajectories need two control manoeuvres (at first and third pericentre in the former case, at second and fourth pericentre in the latter case). The size of the control manoeuvres depends on the launch date. There are certain launch dates for which the trajectories require no control ΔV at all (41, 53 days before the lunar gravity assist for 3- and 4-loop trajectories), but as we launch earlier or later the ΔV increases. Some of the launch dates that correspond to 5-loop trajectories are excluded from the launch window, because the spacecraft would pass too close to the Moon at the first apogee.

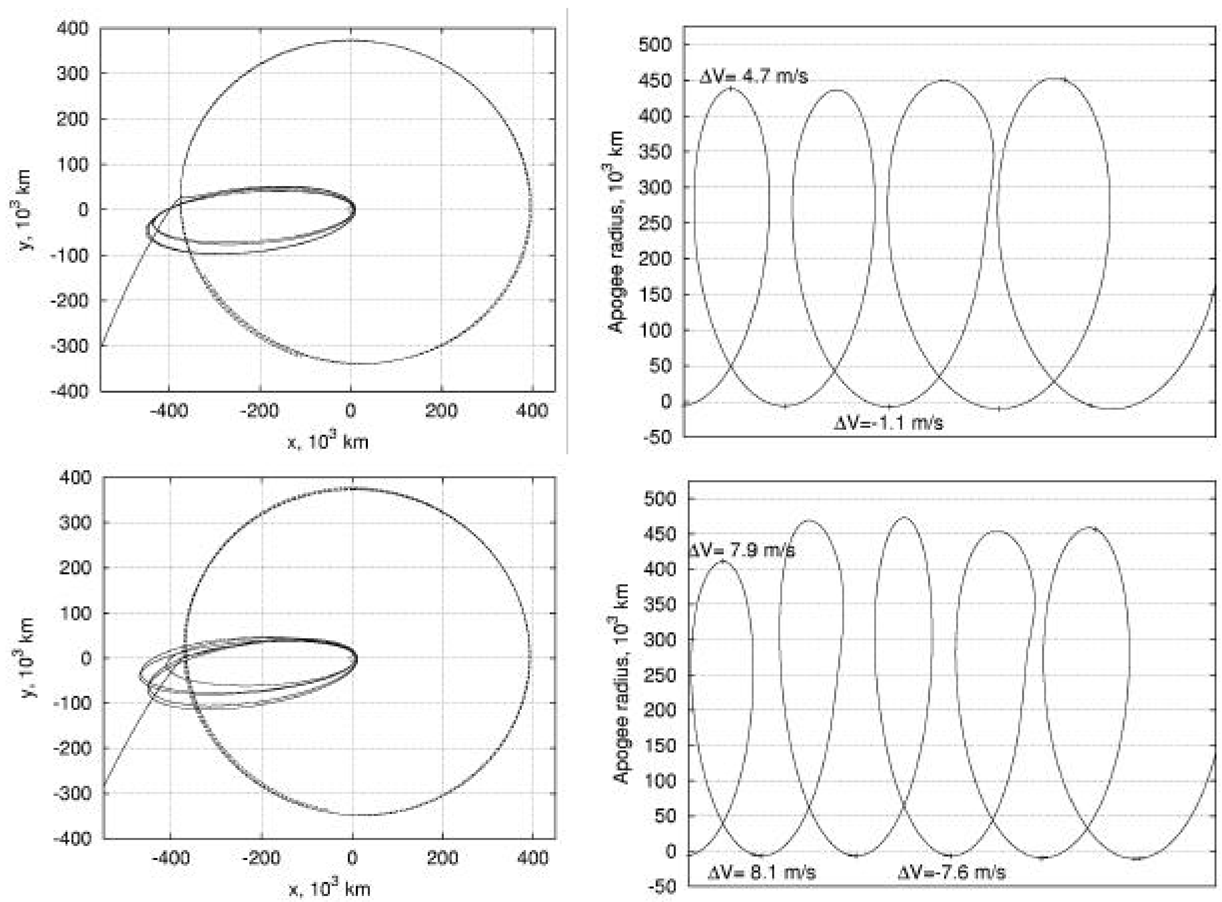

Figure 9. Examples of two numerically integrated trajectories obtained with an unconstrained initial apogee height. The figures above represent a 4-loop trajectory with launch date 1-Jun-2012. The figures below represent a 5-loop trajectory with launch date 8-Aug-2012. In the figures on the left, the coordinate system is inertial and the trajectory is projected onto the Earth equatorial plane. The figures on the right represent schematically the sequence of control manoeuvres and the successive apogee altitudes.

The correction of the launcher dispersion should be done at the first pericentre passage. For most launch dates, around 10 m/s are needed to correct for the worst case (either +3σ or −3σ) dispersion in the apogee height. There are some launch dates, however, where this figure can increase significantly. In particular, when there is a Moon resonance in the second loop, the dynamics are very sensitive with respect to the time of the first pericentre passage and the ±3σ launcher dispersion can be amplified by the lunar perturbation.

Once that the trajectories are generated for all the possible launch dates, each launch window is determined as a set of launch dates with trajectories that fulfill some defined constraints. These constraints are:

- No critical manoeuvre at first apogee: The luni-solar perturbations may cause, depending on the launch date, a reduction in the perigee height so that, after one orbit, the spacecraft would eventually re-enter into the Earth's atmosphere. In these cases, a manoeuvre performed at the first apogee can raise the perigee to the required safe altitude. However, if that manoeuvre fails there is no opportunity to recover and the mission would be lost. In this sense, a *critical* apogee manoeuvre is defined as the apogee manoeuvre whose failure would imply that the next perigee height is below 100 km. It is recommended, hence, to exclude from the launch window the trajectories that, for either the nominal case or the ±3σ dispersion cases, require a critical manoeuvre at apogee.
- Maximum total ΔV: The total ΔV of the nominal trajectory and the ±3σ dispersion cases must be below the ΔV budget for the translunar phase.
- Maximum initial apogee height: The launcher performance for the maximum initial apogee height determines the maximum mass of the spacecraft.

Table 5 shows the number of valid launch dates for each launch window for different values of the constraints (maximum total ΔV equal to 30 m/s or 40 m/s, critical manoeuvre at first apogee allowed or not). For a maximum total ΔV of 40 m/s, excluding the trajectories that require a critical manoeuvre at first apogee causes a significant reduction in the number of valid launch dates.

Table 5. Number of valid launch dates as a function of the defined constraints (maximum allowed total ΔV and critical manoeuvre at first apogee allowed or not). The maximum allowed initial apogee height for all the cases is 460000 km. The launch dates already included in the launch window corresponding to a swingby one month earlier are shown in parentheses.

Lunar swingby	Maximum total ΔV: 30 m/s		Maximum total ΔV: 40 m/s	
	No apogee manoeuvre	Apogee manoeuvre	No apogee manoeuvre	Apogee manoeuvre
June	22	24	26	29
July	10 (+6)	15 (+8)	13 (+8)	19 (+10)
August	4 (+3)	7 (+8)	5 (+6)	16 (+11)
September	4	6	5	9 (+6)
October	4	6	5	18 (+3)
Total	44	58	54	91

D. Results with Apogee Height Below the Moon Distance

In the previous section, the trajectories were computed with an unconstrained initial apogee height. Even when, for certain launch dates, it is possible to find trajectories with negligible ΔV, most of the trajectories do require some ΔV to counteract the reduction of the perigee height caused by the luni-solar perturbation and to avoid the negative effects of the Moon resonances. In addition, the perturbations amplify the launcher injection errors, contributing to increase the ΔV budget.

In this section, an alternative approach to compute the launch window is proposed. Because the effect of the luni-solar perturbation increases with the apogee height, reducing the initial apogee height well below the Moon distance would mean that the orbit is less affected by the gravitational attractions from the Moon and the Sun and, hence, more stable. Of course, in order to encounter the Moon with the proper relative velocity, a manoeuvre to raise the apogee is still needed. But this manoeuvre can take place at the last but one (or last but two) pericentre passage, so that, while the spacecraft is in the high apogee orbit, there are no Moon resonances before the lunar swingby. This strategy will lead to nominal trajectories that require higher ΔV than those previously computed. However, the ΔV to correct for the launcher dispersion will be probably reduced, because the perturbations will not amplify as much the injection errors. In addition (and this point is of special importance), the launcher performance increases significantly for lower apogee heights and this translates in a higher final mass.

To compute the trajectories according to this new approach, the methodology used is the same as before but, in this case, the value of the initial apogee height is constrained to be below 350000 km. Figures 10 and 11 show the total ΔV and the sequence of pericentre passage times for the launch window corresponding to the July lunar swingby opportunity generated with an initial apogee height below the Moon distance.

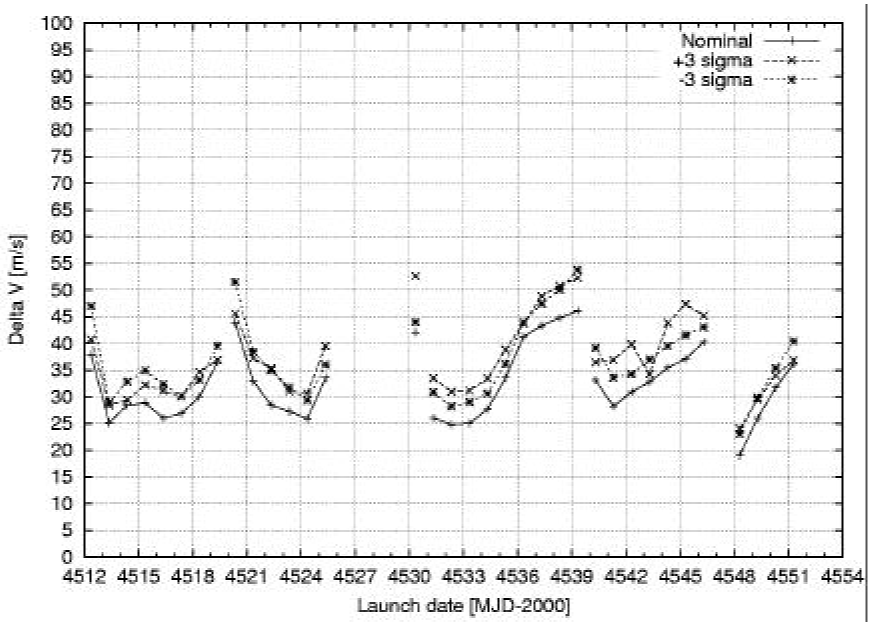

Figure 10. Total ΔV of the different manoeuvres for the nominal case and the cases with $\pm 3\sigma$ dispersion in the apogee altitude (July lunar swingby opportunity). The trajectories are computed with an initial apogee height constrained to be below 350000 km.

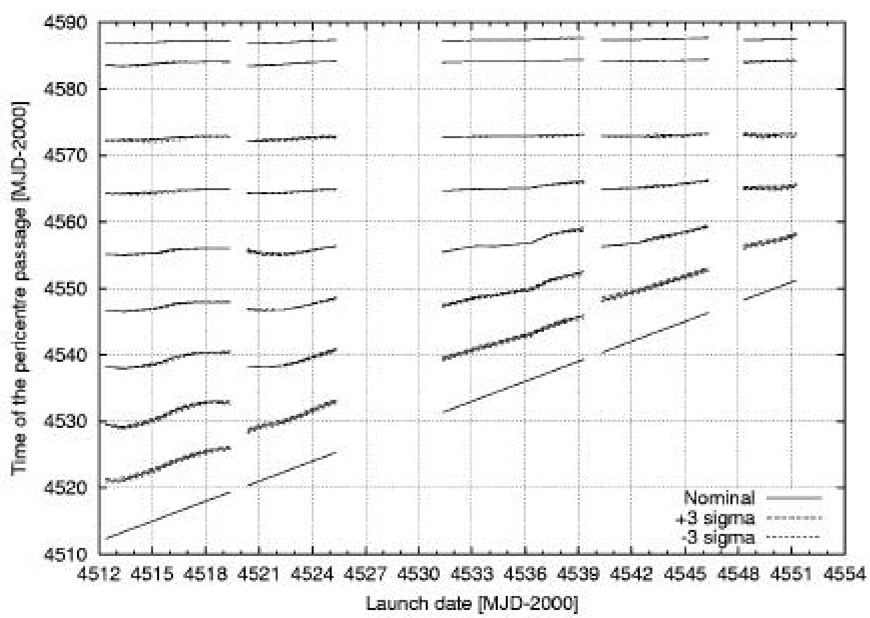

Figure 11. Dates of the pericentre passages and lunar flyby date for the nominal case and the cases with $\pm 3\sigma$ dispersion in the apogee altitude (July lunar swingby opportunity). The trajectories are computed with an initial apogee height constrained to be below 350000 km. The lunar flyby date is the uppermost horizontal line around 4587 (23 July). The time of the final apogee raise manoeuvre is pretty constant.

The trajectories generated in this way have 4 to 8 loops, with initial apogee altitudes ranging from 280000 to 350000 km. The spacecraft remains in that low-apogee orbit until the last but one or last but two pericentre passage. There, a manoeuvre raises the apogee to approximately 415000 km (see examples in 12). Due to the lower apogee, the initial orbit is less perturbed by the luni-solar perturbations and the manoeuvres required to raise the perigee and to avoid the negative effects of the Moon resonances are smaller. For trajectories with 5 loops or less, control manoeuvres before the final apogee raise manoeuvre can be avoided by adequately selecting the initial apogee height. Trajectories with 6 loops or more require one or two manoeuvres to control the time of the pericentre passages. These manoeuvres, however, are less than 10 m/s in all the cases. Indeed, the lunar perturbation can be used to increase the energy of the orbit, thus reducing the final apogee raise manoeuvre.

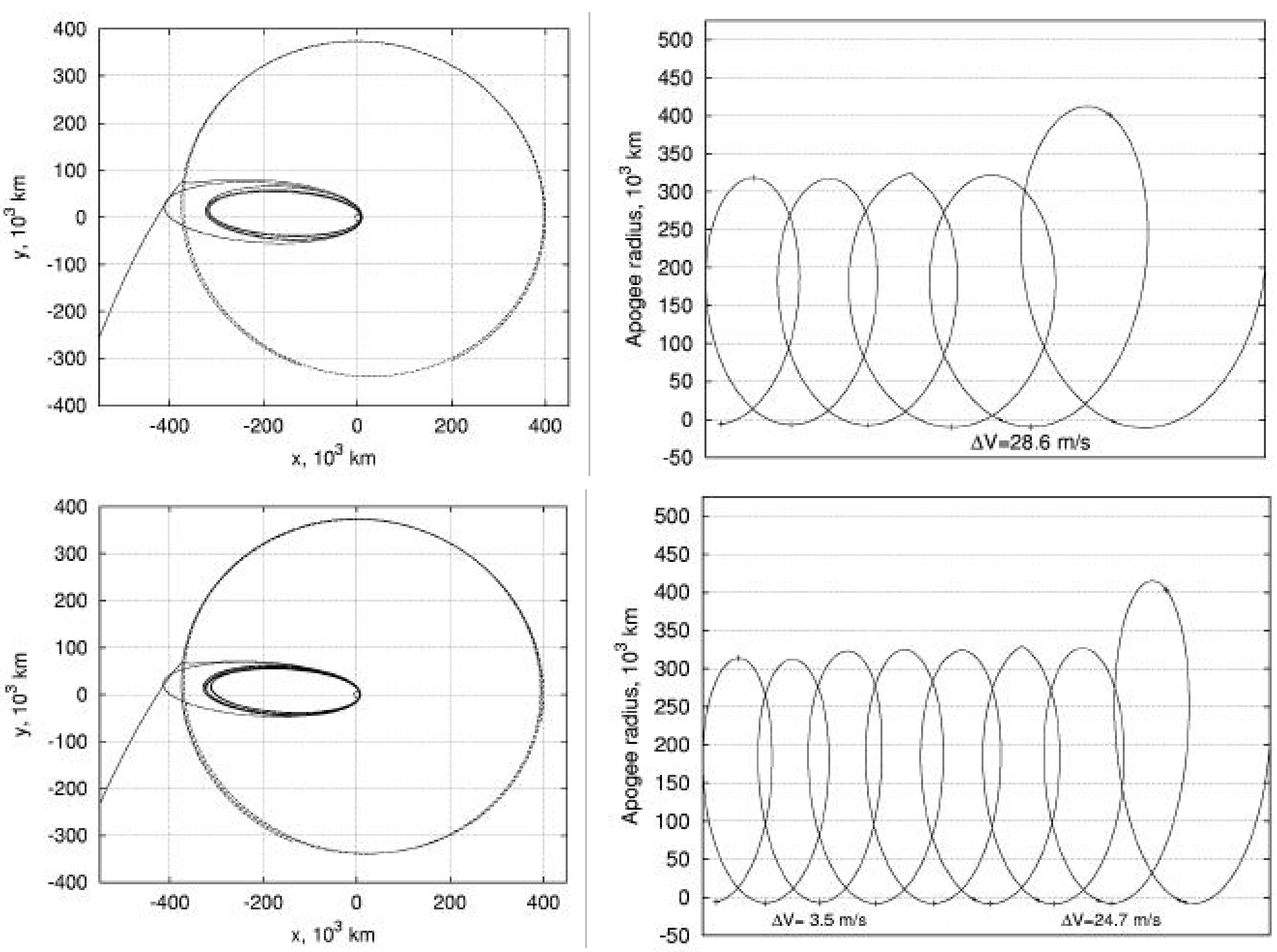

Figure 12. Examples of two numerically integrated trajectories with initial apogee below the Moon distance. The figures above represent a 5-loop trajectory with launch date 11-Jun-2012. The figures below represent a 8-loop trajectory with launch date 14-Jun-2012. In the figures on the left, the coordinate system is inertial and the trajectory is projected onto the Earth equatorial plane. The figures on the right represent schematically the sequence of control manoeuvres and the successive apogee altitudes.

Assuming a maximum total ΔV of 40 m/s and no critical manoeuvre at first apogee, there are more possible launch dates per launch window than in the previous approach (see Table 6). This is due to the fact that the luni-solar perturbations are not so strong as for orbits with apogee above the Moon distance. Hence:

- some launch dates that required a perigee raise manoeuvre at first apogee do not need it anymore with a lower initial apogee height.
- the ΔV to correct for the launcher injection errors is, in general, smaller: Around 5 m/s are sufficient for most launch dates.

In addition, a failure in one of the perigee manoeuvres would not be critical. The spacecraft would stay in a low-apogee orbit, which is relatively stable and cheaper to control compared to an orbit with an apogee above the Moon distance.

Table 6. Number of valid launch dates as a function of the defined constraints (maximum allowed total DV and critical manoeuvre at first apogee allowed or not) for trajectories with initial apogee below the Moon distance. The maximum allowed initial apogee height for all the cases is 335000 km. The launch dates already included in the launch window corresponding to a swingby one month earlier are shown in parentheses.

Lunar swingby	Maximum total ΔV: 40 m/s		Maximum total ΔV: 45 m/s	
	No apogee manoeuvre	Apogee manoeuvre	No apogee manoeuvre	Apogee manoeuvre
June	24	24	29	29
July	14 (+8)	16 (+8)	15 (+9)	18 (+9)
Total	38	40	44	47

V. Auxiliary calculations

A. Gravity losses

The control manoeuvres during the phasing loops were simulated in the previous analysis as impulsive ΔV. If these manoeuvres will be performed with the RCS thrusters, some gravity losses will result. These gravity loses are estimated for the last apogee raise manoeuvre of the trajectories with initial apogee below the Moon distance. This manoeuvre is performed at the penultimate pericentre passage and it raises the apogee typically from 320000 km to some 415000 km. For those conditions, the required impulsive ΔV is 24 m/s. Assuming 88 N thrust, the duration of the manoeuvre is 567 seconds and the gravity losses are only 0.2 m/s. Some launch dates require larger control manoeuvres, but the associated gravity losses are, in all the cases, below 1 m/s.

The possibility to use the solar electric propulsion to perform the control manoeuvres during the translunar phase was also investigated. The idea is to replace the impulsive manoeuvres at perigee by a continuous thrust arc centred around the pericentre. The thrust duration is limited to one complete revolution (thrust beginning and ending at apocentre). Assuming a 200 mN tangential thrust and a 1950 kg spacecraft in a 200 × 380000 km orbit, the maximum achievable change in the orbital period per revolution is limited to 1.5 days, which is equivalent to a 8 m/s impulsive manoeuvre. Therefore, the low thrust provided by the SEP module cannot be used, in general, to perform the control manoeuvres for a phasing loop strategy as the one described in this paper.

B. Eclipses

Basically, two possible types of Earth eclipses may occur while the spacecraft is in the phasing loops:

- long eclipses (up to 10 hours) close to apogee of one of the phasing orbits. This kind of eclipse would only occur for a launch in March. The earliest launch dates in the June launch window are in mid-April. For these dates, the eclipse would occur far away from the apogee (eclipse duration around 90 minutes).
- short eclipses (less than 40 minutes) close to perigee. This kind of eclipse occurs for the final phasing loops of the June and July launch windows and for each orbit of the trajectories within the August, September and October launch windows.

The eclipses for the trajectories with initial apogee height well below the Moon distance are slightly longer than for the trajectories with initial apogee above the Moon distance. In all the cases, though, the maximum duration of the eclipse is less than 40 minutes, except for some early trajectories corresponding to the June lunar swingby.

VI. Conclusion

A phasing loop strategy is proposed in this paper for the 2012 launch window for the BepiColombo trajectory with a lunar gravity assist. Because of the strong effects caused by the luni-solar perturbations on a highly eccentric Earth orbit, a launch window design based on the approximation of Keplerian motion is not valid, not even for preliminary analysis. Numerically integrated trajectories were computed for each launch date for the different 2012 lunar swingby opportunities (from June to October). They match the optimum, previously computed heliocentric trajectory at the exit of the Earth's sphere of influence. In the proposed approach, tangential manoeuvres are introduced at the first apocentre and at the pericentre passages, in order to counteract the effects of the luni-solar perturbations. To take into account the launcher dispersion, trajectories with $\pm 3\sigma$ error in the initial apogee altitude with respect to the nominal launcher performance are also computed for every launch date.

The Sun perturbation is shown to have a different effect on the orbit depending on the direction of the apogee with respect to the Sun. For a lunar swingby close to the descending node of the Moon, the Sun attraction pulls up the perigee for a launch in spring (corresponding to a lunar swingby in June and July), but pulls down the perigee for a launch in summer (corresponding to a lunar swingby in August, September and October). In the latter case, a manoeuvre at first apogee (up to 30 m/s) is needed to raise the perigee to the minimum allowed height (400 km). This manoeuvre increases the total ΔV budget and it is also considered critical, because its failure would cause the loss of the mission. Hence, a lunar swingby in June or July 2012 is clearly more favorable than a swingby in August, September or October.

Passages close to the Moon (referred to as "Moon resonances") one and two months before the lunar swingby can significantly alter the orbit of the spacecraft. To avoid the negative effects of the Moon resonances, the orbital period must be controlled with the pericentre manoeuvres. Performing these manoeuvres only at the pericentre passages immediately before a Moon resonance is found to be the optimal manoeuvre strategy.

Two kinds of trajectories are investigated, which have initial apogees lying, respectively, above or below the Moon distance. If the initial apogee height lies above the Moon distance, it is possible to find trajectories for certain launch dates that require almost no deterministic ΔV before the lunar swingby. However, due to the luni-solar perturbations (whose effects increase with the apogee height), many launch dates need to be excluded from the launch window and the cost to correct for the launcher dispersion is high. If the initial apogee height lies below the Moon distance (ranging from 270000 km to 335000 km), all the trajectories need at least some 25 m/s to raise the apogee for the lunar swingby. However, as the luni-solar perturbations are not so strong as in the previous case, more possible launch dates are obtained and, in general, it is cheaper to correct for the launcher dispersion. In addition, the launcher performances are higher. Hence, this option is recommended. For a maximum ΔV of 40 m/s and no critical manoeuvre at first apogee, this strategy leads to 24 possible launch dates for a lunar swingby in June and 14 extra launch dates for a swingby in July 2012.

Further work should investigate other possible constraints on the trajectories, such as a limited number of flight programs, ground station visibilities, manoeuvres attitude, passages through the radiation belts, etc.

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