DESIGN OF LUNAR GRAVITY ASSIST FOR THE BEPICOLOMBO MISSION TO MERCURY

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The BepiColombo mission to Mercury makes use of a lunar gravity-assist to increase the payload mass into the final orbit. The swing-by of the Moon is designed such that the highest velocity in the suitable direction is obtained. Furthermore a phasing loop strategy is investigated, which opens a 5-month launch window. Taking into account the luni-solar perturbations, the launch window has to be closed frequently, but launch opportunities remain from May to September 2012.

INTRODUCTION

This paper presents the lunar gravity assist design and the preliminary launch window for the BepiColombo trajectory. BepiColombo is the ESA cornerstone mission to bring the Magnetospheric Orbiter (MMO) and the Planetary Orbiter (MPO) to Mercury, currently planned for launch in 2012. The name is a recognition of the work performed by the Italian scientist Giuseppe Colombo in the sixties, who explained the 3:2 resonant motion of Mercury[1]. Several options for the interplanetary trajectory are under study. In the most promising options MMO and MPO are launched by a single Soyuz-Fregat. Several gravity-assists as well as solar electric propulsion thrust arcs are needed to reach Mercury with a low approach velocity. The current baseline uses a lunar gravity-assist to increase the launcher performance and thus the payload mass delivered into the target orbit.

DITAN, a tool which implements a direct optimisation approach[2], is used to find the optimal trajectory. In order to take into account swing-bys, which are currently modelled with linked conics, the trajectory is split into several phases, each phase corresponding to a trajectory arc connecting two planets. For each phase, DITAN uses a Finite Elements in Time technique to transcribe differential equations, governing the dynamics of the spacecraft, into a set of algebraic non-linear equations.

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The BepiColombo trajectory combines the use of low thrust (provided by a solar electric propulsion module) with gravity assists of Venus and Mercury. Each of the two probes MMO and MPO is launched into a highly eccentric Earth bound orbit (HEO). Then a lunar gravity assist is used to inject the spacecraft into a heliocentric orbit. An Earth swing-by one year later places the spacecraft into a trajectory towards Venus. After the first Venus swing-by, a 1:1 resonant orbit follows until the second Venus swing-by. Thereafter, 4.5 revolutions with several thrust arcs are needed until the first Mercury swing-by, followed by a 180º coast arc to the second Mercury swing-by. Then some 60-100 day transfer is typically needed to enter Mercury's sphere of influence with low relative velocity and to perform the final orbit insertion manoeuvre using a chemical propulsion module. Figure 1 illustrates the interplanetary trajectory. Similar solutions are found approximately every 20 months (lunar swing-by dates in June 2009, January 2011, July 2012). The current baseline is planned to be launched in July 2012[3].

The mass which can be inserted into a 400 x 12000 km Mercury orbit (MMO orbit) is about 180 kg higher if a lunar swing-by and one year later an Earth swing-by is performed compared to a launch directly towards Venus (assuming a Soyuz/Fregat 2B launch from Baikonur as studied in Campagnola et al.[3]). Another advantage of a lunar swing-by is the high flexibility in the launch date due to the high flexibility of the low-thrust propulsion system.

A launch with the upgraded Soyuz-2B with the Fregat-M upper stage from Baikonur is assumed in this paper. A launch from Kourou has significant advantages in terms of mass at launch[10], but it is not confirmed whether this will be available for BepiColombo. From Kourou, a launch with direct injection into HEO can be considered, although it constrains the argument of the pericentre of the bound orbit (ω_{HEO}) to an interval of about 120º-160º (otherwise significant penalties arise). The maximum launch mass in case of a direct injection is about 50 kg higher when compared to a launch through a parking orbit. Unfortunately, a direct injection into HEO from a launch in Baikonur cannot be considered for BepiColombo because the apogee would be too far in the South (the optimum argument of perigee ω_{HEO} is around 120º and the inclination (i_{HEO}) is 51.8º) to allow an encounter with the Moon. At a true anomaly of 170º, for instance, the spacecraft is 270 000 km below the equator, while the Moon (its inclination is 22º in the period of interest) will not go below -140 000 km.

A simplified solar electric propulsion model is used. The thrust level outside the orbit of Venus is assumed to be 200 mN, whereas inside the orbit of Venus it is 400 mN. The specific impulse of solar electric propulsion (SEP) is fixed to 4600 s. For the swing-bys, the linked conics approximation is used with a minimum swing-by altitude of 200 km for Moon and Mercury and 300 km for Earth and Venus.

TRAJECTORY DESIGN AND OPTIMISATION

The starting point is a near-optimum interplanetary trajectory beginning at the Earth with a relative excess velocity of about 1.4 km/s and ending at Mercury arrival[5]. For this trajectory the "launch" date and the direction of the velocity at the Earth sphere of influence (V_{SoI}) are optimised using DITAN. It is found that the V_{SoI} vector usually has a very low declination with respect to the ecliptic plane and points in a direction mainly opposite to the Earth velocity. This can be explained by looking at the subsequent interplanetary trajectory: the next planetary encounter is again with the Earth, so an inclination with respect to the ecliptic far from 0º is counterproductive. Furthermore, the Earth swing-by shall be used to brake the velocity to reach Venus. The optimum geometry is to arrive at the Earth with an excess velocity vector pointing to the Sun (arriving from outside the Earth orbit). To achieve this the spacecraft needs to brake one year earlier, go inside the Earth orbit (down to 0.85 AU) where it accelerates to go outside the Earth orbit (to 1.15 AU) before finally coming back to the Earth.
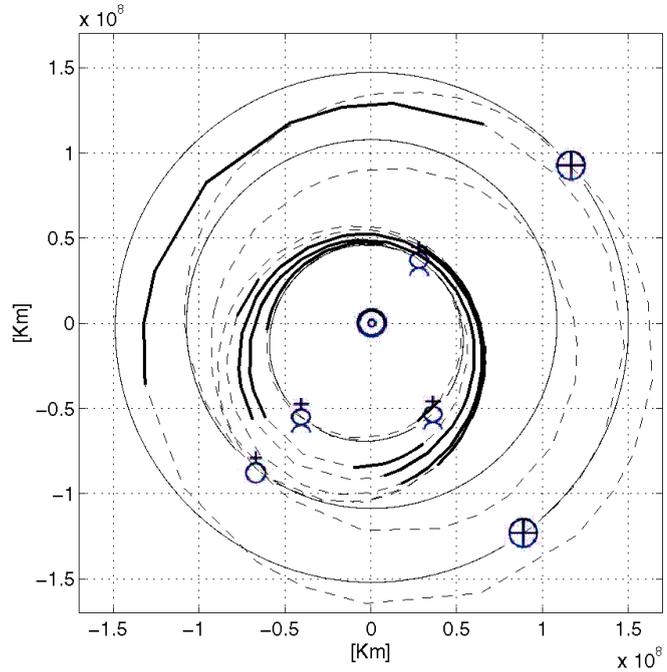
Figure 1: BepiColombo interplanetary trajectory with launch in July 2012 and arrival in November 2016. The thick lines show the thrust arcs.

Now we actually need to find the optimum lunar flyby trajectory to achieve the required excess velocity in the suitable direction. The first step is a detailed analysis of the lunar gravity-assist. The objective is to gain as much energy as possible when leaving the Earth and obtaining at the same time the required outgoing velocity direction.

Lunar swing-by design

DITAN, as any local optimisation programme, finds the optimum in the neighbourhood of a first-guess solution provided by the user. Therefore the aim of this lunar swing-by analysis is to provide the tool with a first-guess solution close to the global optimum.

The probes are launched into a 200x200 circular parking orbit and then injected into the high elliptic orbit. Both orbits are 51.8 degree inclined to maximize the mass at launch[10]. Assuming a Keplerian motion, the apocentre altitude (apo_{HEO}), the right ascen-

sion of the ascending node (Ω_{HEO}), and the argument of perigee (ω_{HEO}) can be varied to find feasible trajectories to the Moon. The effectiveness of the lunar swing-by is now investigated as a function of the position of the Moon and the elements of the initial bound orbit.
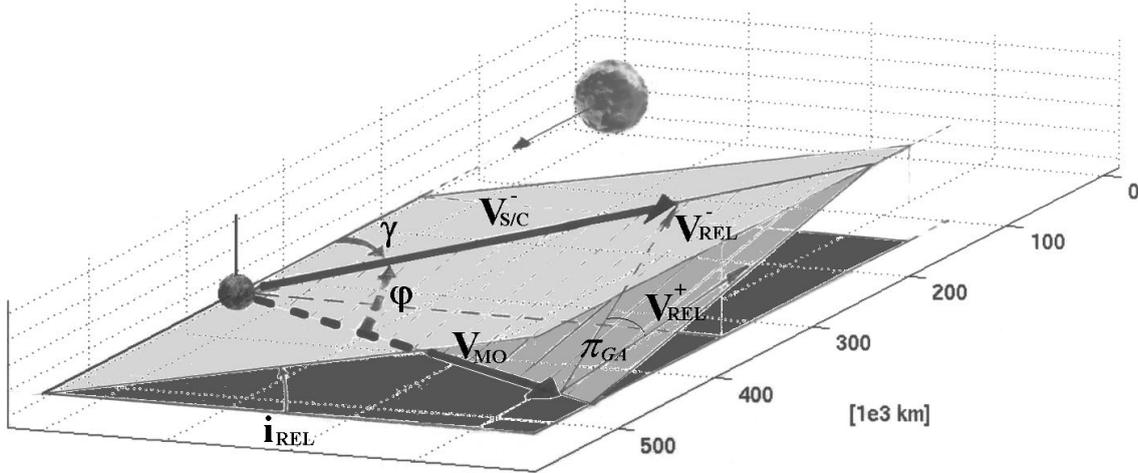
Figure 2: Geometry of a lunar swing-by using linked conics. The black plane is the orbital plane of the Moon. The light grey plane is the orbital plane of the spacecraft before the lunar wing-by. The dark grey plane is the gravity assist plane (π_{GA}). In this plane the incoming and outgoing relative velocity of the swi ng-by are shown.

In fact, for every true anomaly of the Moon (f_{MO}), the maximum V_{SoI} can be determined as a function of the apocentre altitude apo_{HEO} and of the angle φ between the velocity of the spacecraft and the Moon ($V_{S/C}^{-}$ and V_{MO}) at the encounter date (linked conics are used). In the plane π_{GA} that contains these two vectors (see Figure 2), with the x-axis oriented along the velocity vector of the Moon and the z-axis oriented along the cross product $V_{S/C}^{-} \times V_{MO}$, the incoming relative velocity vector is:

$$V_{REL}^{-} = \begin{bmatrix} V_{S/C}^{-}(apo_{HEO}) * \cos(\varphi) - V_{MO}(f_{MO}) \\ V_{S/C}^{-}(apo_{HEO}) * \sin(\varphi) \\ 0 \end{bmatrix} \quad (1)$$

The swing-by is aimed to turn this vector into the plane π_{GA} in order to reduce the angle between V_{REL}^{-} and V_{MO}, and to align them as much as possible. In this way, the outgoing absolute velocity will be maximized. At the same time the escape declination will be reduced, because the velocity vector of the Moon is only some 5 ° inclined with respect to the ecliptic. Note that due to symmetry, positive and negative φ provide the same V_{SoI}. In particular, a positive φ corresponds to an encounter in the incoming leg of the bound orbit. On the contrary, a negative φ corresponds to an encounter in the outgoing leg of the bound orbit.

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Results of this study are shown in Figure 3, where V_{SoI} is represented as function of apo$_{HEO}$ and φ. The swing-by encounter is assumed to happen at the Moon's perigee or apogee. Only positive φ are plotted. It can be seen that a swing-by close to lunar perigee provides the maximum V_{SoI}.
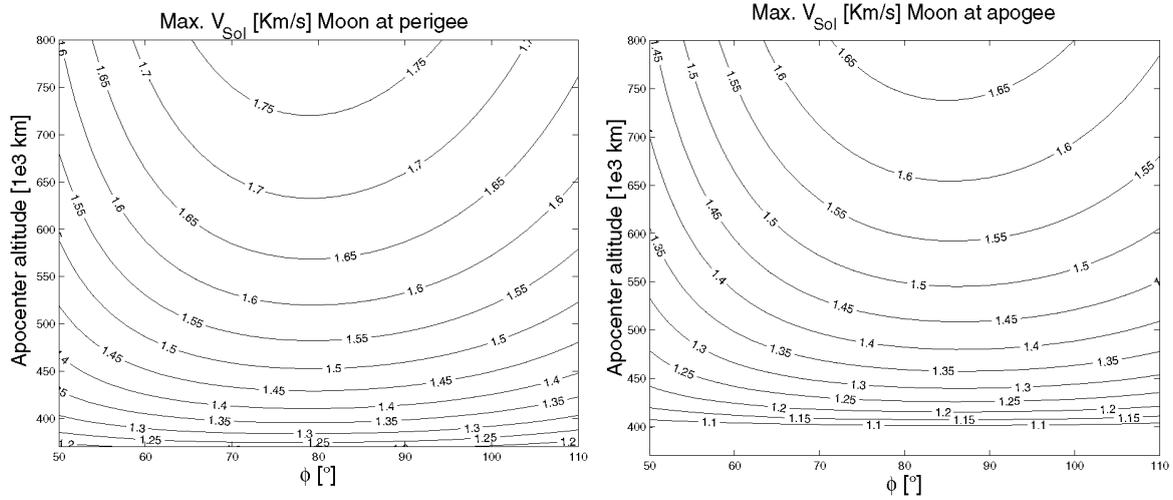
Figure 3: Maximum V_{SoI} achievable as a function of the angle φ and the apogee altitude of the initial bound orbit. In the picture on the left, the lunar swing-by occurs when the Moon is at perigee. In the picture on the right, the lunar swing-by occurs when the Moon is at apogee.

With this overview of the lunar swing-by geometry, the areas of convergence for the optimisation problem can be localized. As a first conclusion, the velocity at the sphere of influence strongly depends on the true anomaly of the Moon. Therefore local maxima for V_{SoI} and hence for the payload mass are expected once every lunar month, when the swing-by takes place when the Moon is close to its perigee.

Now a swing-by in the incoming or in the outgoing leg of the HEO can be chosen to target V_{SoI} into the required direction. Given f_{MO} and apo$_{HEO}$, $r=p/(1+e\cos(f_{S/C}))$ can be solved to find $\cos(f_{S/C})$, where $f_{S/C}$ is the true anomaly of the spacecraft right before the swing-by. This explains why two solutions are possible ($\pm f_{S/C}$), as suggested previously by analysing the geometry. If the optimum $f_{S/C}$ is not 180°, the two possible cases usually belong to two different areas of convergence. Before starting the optimisation programme it must be analysed which of the two cases (swing-by in the incoming or outgoing leg) provides the better velocity direction needed for the subsequent heliocentric trajectory.

Finally, there is another trade-off to be made. For every swing-by date, the Moon can be approached from above or from below its orbital plane. The two possible bound orbits have different (ω_{HEO}, Ω_{HEO}), but similar φ and therefore also similar energy gain[5]. In fact, φ is very much linked to the angle γ (see Figure 2) which is usually very small for highly eccentric orbits like the ones assumed for this paper (unless the encounter occurs extremely close to the apogee).

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It is worth mentioning that in case of a launch from Kourou, where the inclination of the HEO is 5°, the Moon can be encountered only coming from below close at the descending node of the Moon orbit, or coming from above at the ascending node of the Moon orbit. This implies the existence of only one local maximum, having fixed the other parameters.

Trajectory options with a lunar flyby in 2012

For a launch in 2012, the preliminary optimisation suggests a launch date around summertime. Figure 4 shows the Earth-Sun and the Earth velocity directions as well as the positions of the Moon at perigee for the same period. All vectors are shown in the ecliptic, Earth centred reference frame.
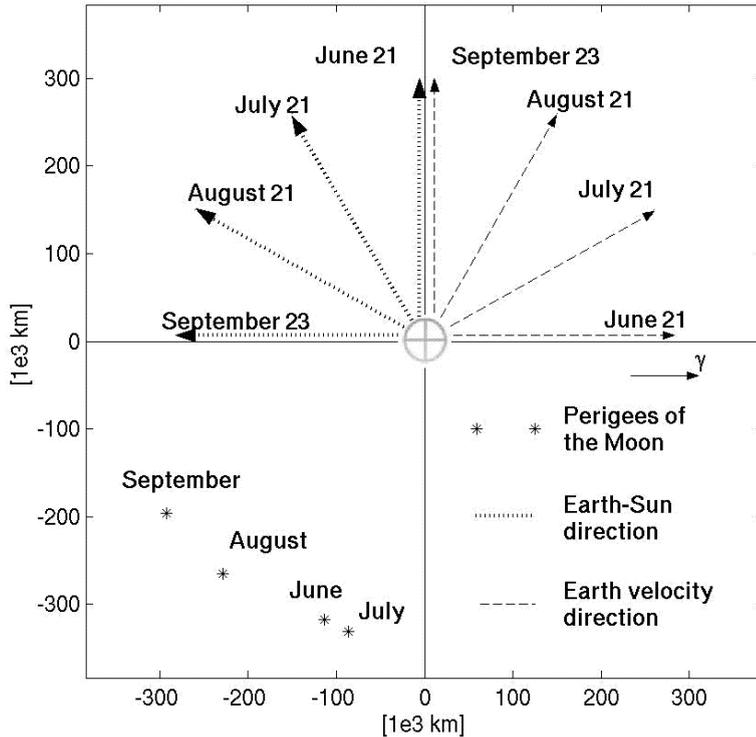
Figure 4: Perigees of the Moon, Earth-Sun and Earth veloc-ity directions in summer 2012 are shown.

The Moon passes through its perigee always in the third quadrant. That is where the flyby must take place. Furthermore, in summertime the Earth to Sun vector is located in the second quadrant. Correspondingly, the required direction of V_{SoI} (with a component opposite to the Earth velocity) is mainly in the third quadrant (but partially also in the second and in the fourth). To satisfy this condition best, it is found that the swing-by should occur in the outgoing leg of the HEO.

Then two similar orbits 200x400000 km altitude, with inclination of 51.8° but with two different values for Ω_{HEO} and ω_{HEO}, are calculated to meet the Moon from above or from below its orbital plane. The swing-by occurs at the outgoing leg of the bound orbit and within or close to the third quadrant. These orbits as well as the hyperbolic escape trajectory following the lunar swing-by are added at the beginning of the direct-escape interplanetary trajectory optimised before. Several months are considered for the launch. For every opportunity, the same orbital parameters are adopted for the first-guess escape trajectory.

The trajectories are then optimised, and the results are presented here. In all the cases the lunar swing-by takes place close to the descending node, satisfying most of the conditions used to generate the first-guess escape trajectory. Thanks to the high flexibility of the low thrust propulsion system, for Moon swing-bys between June and October 2012 (i.e. 5 different swing-by opportunities), the final mass in MMO orbit differs only by less than 10 kg as calculated with the optimisation software DITAN.
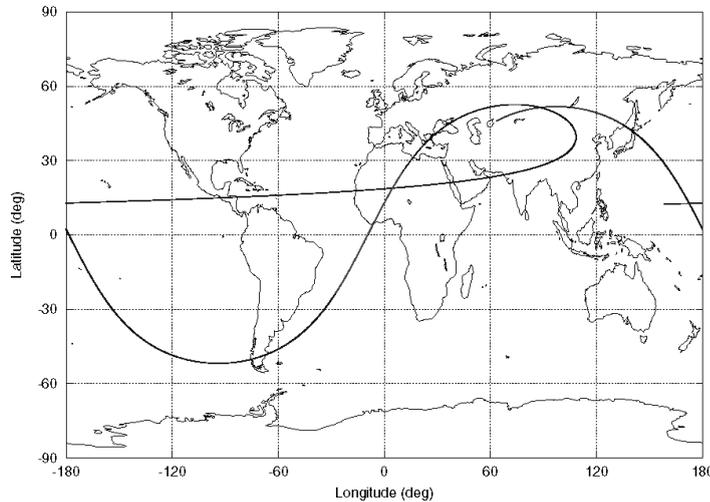
Figure 5: Ground track of a "North-Bound escape" trajectory. BepiColombo is separated in a 200 x 450000 km orbit with W_{HEO} and w_{HEO} close to 0°.

As explained above, for each month there are two options. In option "$\omega_{HEO}0$" we have a North-bound "escape" manoeuvre from the 200 km circular parking orbit, where the FREGAT burn takes place close to Africa like in the MarsExpress case. The ground-track of the trajectory is shown in Figure 5. The FREGAT is burning when crossing the equator from South to North. The end of the burn is over Africa and leaves the spacecraft in a highly eccentric orbit with an argument of perigee close to zero degrees. And since the lunar swing-by takes place close to the descending node of the orbit of the Moon which is close to right ascension of 180°, the ascending node of BepiColombo's lunar encounter orbit is close to zero degrees. Hence in option "$\omega_{HEO}0$" the ascending node and argument of perigee are both close to zero.

Figure 6: Ground track of a "South-Bound escape" trajectory. BepiColombo is separated in a 200 x 450000 km orbit with W_{HEO} and w_{HEO} close to 180°.

In option "$\omega_{HEO}180$" we have a South-bound "escape" manoeuvre from the 200 km circular parking orbit, where the FREGAT burn takes place over the Pacific Ocean as shown in Figure 6. The resulting highly eccentric orbit has an argument of perigee of 180º. The ascending node is also close to 180º (again the lunar swing-by takes place close to a right ascension of 180º), hence it's called option "$\omega_{HEO}180$".

In both options the overall results in terms of final mass in the target orbit around Mercury are nearly identical. The North bound escape option was finally chosen, because of the better ground station coverage (identical to the MarsExpress mission scenario).

Table 1 shows the final mass and the lunar swing-by details of these 10 trajectories, where α is the angle between the spacecraft velocity and the Earth-Sun direction and δ is the declination of the velocity vector (the angle between the spacecraft velocity and the ecliptic plane). As previously mentioned, the maximum difference in the target mass is only 8 kg. This can be explained by the fact that the SEP system allows to "jump" onto the optimum trajectory from different initial starting points with little penalty.

Table 1
MASS IN MMO ORBIT FOR TEN DIFFERENT SCENARIOS

Lunar Flyby Date		ω and w	Final Mass in MMO orbit	Conditions at the exit of the Earth's sphere of influence		
				V_{SoI}	a	d
(MJD-2000)	(Calendar)	(°)	(Kg)	(Km/s)	(°)	(°)
4558.877	24 Jun 2012	0	939.5	1.433	131.644	-0.471
4559.520	25 Jun 2012	180	939.8	1.440	131.458	0.762
4587.678	23 Jul 2012	0	943.9	1.461	124.095	-0.127
4588.330	24 Jul 2012	180	943.8	1.456	124.255	0.526
4615.410	20 Aug 2012	0	940.5	1.222	94.2357	0.278
4616.250	21 Aug 2012	180	940.6	1.247	98.8584	0.226
4640.108	14 Sep 2012	0	941.9	1.453	43.8326	-0.500
4640.742	14 Sep 2012	180	942.2	1.462	43.9932	0.917
4668.657	12 Oct 2012	0	937.9	1.474	31.1037	-1.088
4669.241	13 Oct 2012	180	935.9	1.465	30.1371	-1.645

Figure 7 shows the 5 trajectories for the different lunar swing-by dates (the "$\omega_{HEO}0$" options are always shown, the difference to the "$\omega_{HEO}180$" options would hardly be noticeable on this scale). Except for a lunar swing-by in October 2012 all trajectories are very close to each other (see the figure on the right). The same trajectories are plotted in a different coordinate system where the small differences can be better seen (see the figure on the left). At solar longitude of about 120º ("January") the spacecraft needs to be closest to the Sun. Therefore, if the lunar swing-by is in June or July the spacecraft has to leave the Earth-Moon system with an outward velocity component. In August,

the spacecraft is expelled nearly exactly in the direction of the Earth velocity, whereas in September and October the spacecraft will orbit immediately inside the Earth orbit. In all 5 cases the spacecraft will cross the Earth orbit at a solar longitude of about 240° ("May"), before having the Earth swing-by at a solar longitude of 30° to 45° (between 24 October and 10 November 2013). Table 1 also shows that the declination of the velocity vector is always close to zero after the lunar swing-by, i.e. the lunar swing-by is also used to deflect the spacecraft into the ecliptic plane.
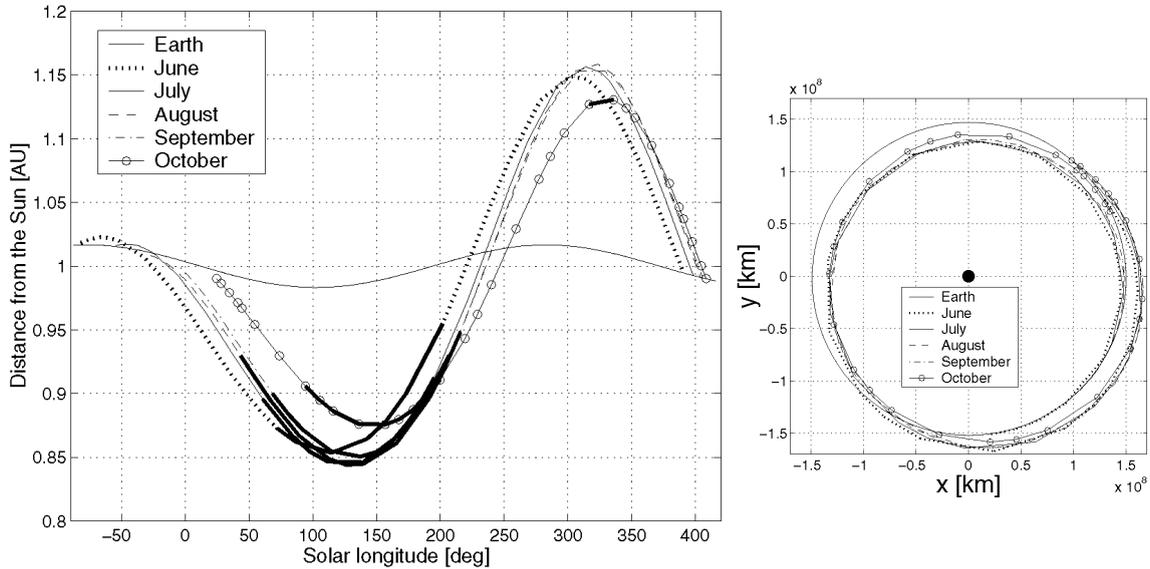
Figure 7: First part (until Earth swing-by) of the optimum transfer trajectories for lunar swing-by dates between June and October 2012 (w$_{HEO}$0 options). The picture on the left shows the heliocentric distance. Thrust arcs are indicated by thick lines. On the right, the trajectories are shown in the ecliptic plane.

PHASING LOOP STRATEGY AND PRELIMINARY LAUNCH WINDOW

As shown previously, in order 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 (close to its descending node for the 2012 opportunities). 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 quick 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 may be as large as 100 m/s)[6]. In order to mitigate this problem,

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a phasing loop strategy is proposed. The idea is, rather than to launch directly towards the lunar gravity assist, to inject the spacecraft into a highly elliptical orbit and to stay there for several revolutions (loops) before the lunar swing-by. The spacecraft propulsion system 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:

1. To widen the monthly launch window. Varying slightly the period of the initial orbit according to the launch date and inserting additional loops allows expanding the launch window to roughly 30 days.

2. 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. Among them, the Japanese missions Hiten, Geotail and Nozomi[7, 8] and the MAP (Microwave Anisotropy Probe) mission from NASA[9]. Figure 8 shows a sketch of a phasing loop trajectory with 4 loops.
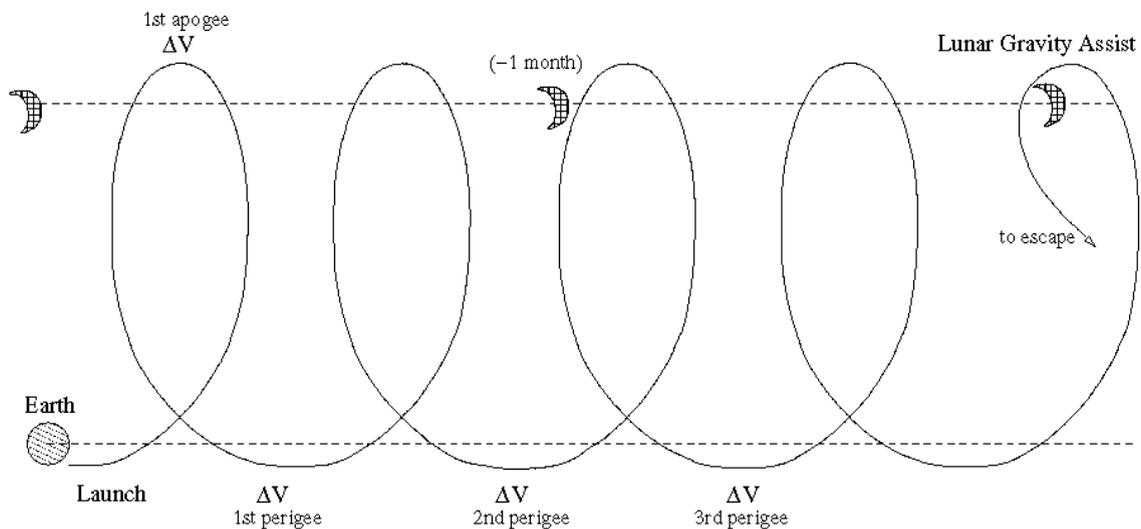
Figure 8: Schematic representation of the phasing loop strategy with 4 orbits before the lunar swing-by.

Preliminary design of the launch window

The preliminary design of the launch window is done using the approximation of Keplerian orbits and assuming no manoeuvres before the lunar encounter. Hence, only the final leg of the phasing loop trajectory (from the last perigee to the lunar gravity assist) is considered. The example of launch window presented hereafter was calculated assuming a lunar swing-by in 2011. The nominal initial orbit, obtained as a result of the optimisation of the mass delivered into orbit around Mercury, is described in Table 2.

Table 2
ORBITAL ELEMENTS OF THE INITIAL ORBIT FOR A LUNAR SWING-BY IN 2011

Final Perigee passage (MJD-2000)	Apogee altitude (km)	Perigee altitude (km)	Orbital period (days)	Inclination (°)	W (°)	w (°)	Lunar flyby date (MJD-2000)
4033.0085	406700	200	11.1	51.8	-1.730	2.194	4041.8236

From the nominal initial orbit, the launch window is expanded in the following way:

1.   4 loops are added before the final perigee passage to obtain the trajectory for the nominal launch date (3-Dec-2010).

2.   For earlier launch dates, the apogee height is increased gradually to a maximum of 450000 km, so that the spacecraft arrives at the Moon with similar arrival conditions (slightly higher approach velocity, because of the higher energy of the orbit).

3.   For later launch dates (one or two days later), postponing the lunar gravity assist date is considered (that is, the apogee height of the nominal initial orbit is kept constant but the line of apsides changes).

As a result, 4-loop trajectories are obtained for 11 consecutive launch dates. A 30-day launch window is easily built by adding or subtracting one complete revolution to the trajectories already computed. In order to ensure enough time for the correction of the launcher dispersion and the precise targeting of the lunar swing-by, a minimum of 3 loops is assumed.
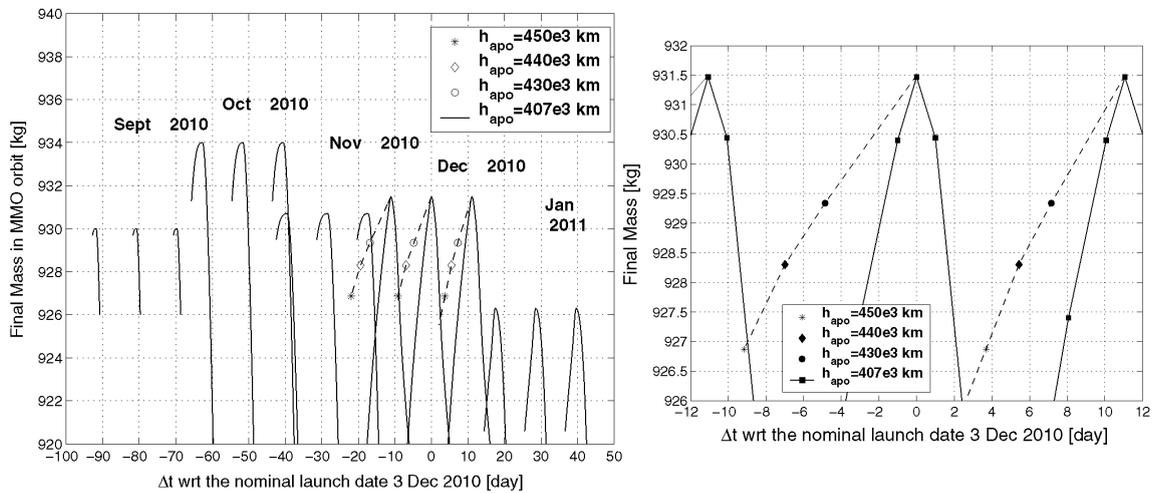
Figure 9: BepiColombo preliminary launch window for lunar swing-by in 2011: general view (left) and detail (right). The trajectories were obtained assuming Keplerian orbits. For the days not represented the final mass can be extrapolated.

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The same process can be applied to the lunar gravity assist opportunities found in the previous and following months. Thus, a 140-day long launch window is defined with little penalty in the final mass delivered into orbit around Mercury.

Figure 9 shows the final mass for a 24- and 140- day launch window. The Figure on the right shows a 24-day launch window around the nominal launch date. The left peak refers to a 5-loop trajectory, the central peak refers to a 4-loop trajectory and the right peak refers to a 3-loop trajectory. The dotted lines show the final mass for different initial apogee altitudes.

PERTURBATION EFFECTS AND NUMERICALLY INTEGRATED TRAJECTORY

The Keplerian approximation is useful to get a first feeling of the problem and to define the theoretical launch window, but the gravitational attractions from the Sun and the Moon are very strong in a highly eccentric orbit such as the one considered here (apogee height close to the Moon distance) and must be taken into account.
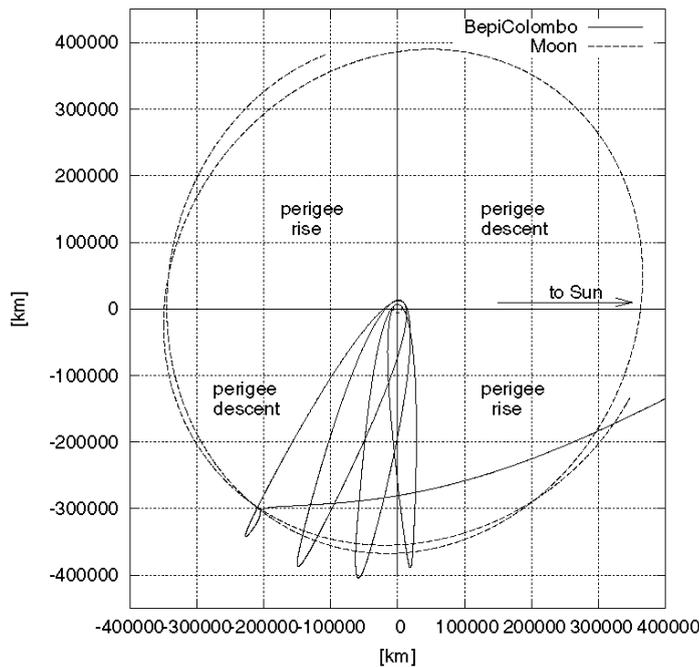
Figure 10: Projection into the equatorial plane with the Earth-Sun direction fixed of a trajectory with 3 phasing loops.

The Sun attraction causes periodical changes in the orbital elements. The changes in the eccentricity are especially important due to the low perigee altitude. Depending on the relative geometry, the Sun perturbation may cause an increase or a descent in the perigee altitude. The effect is illustrated in Figure 10 that represents a phasing loop trajectory in a rotating coordinate frame 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.

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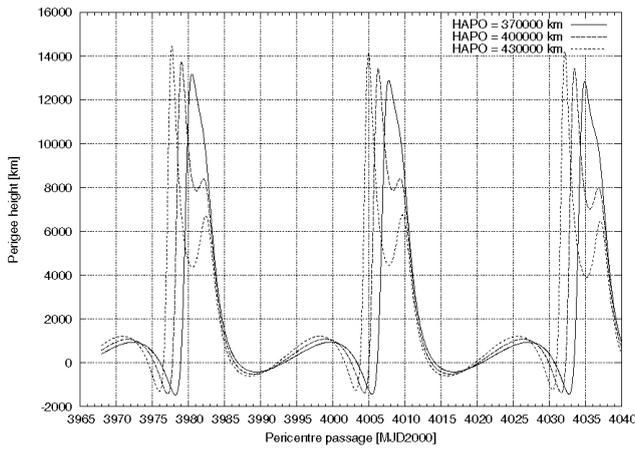
Figure 11: Perigee altitude after one orbit as a function of the time of the pericentre passage due to the Moon perturbation. The perigee altitude of the initial orbit is 200 km.

Also 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 and the Moon and have an approximate frequency of 28 days, the orbital period of the Moon (see Figure 11). Obviously, the closest the spacecraft is with respect to the Moon, the strongest the perturbation. Especially critical is a period of around 7 days that takes place monthly when the Moon lines up with the line of apsides of the orbit (which is referred to as a Moon resonance). The lunar perturbation becomes then particularly strong and significantly modifies the initial orbit. But even when the spacecraft is relatively far from the Moon, the perturbation can cause a significant decrease or increase in the perigee altitude.

In order to analyse in more detail the effects of the luni-solar perturbations, a set of trajectories around the nominal launch date is numerically integrated[4]. Constraints on the lunar swing-by date and the conditions at the exit of the Earth's sphere of influence are introduced to ensure that the trajectories match with the optimum heliocentric trajectory.
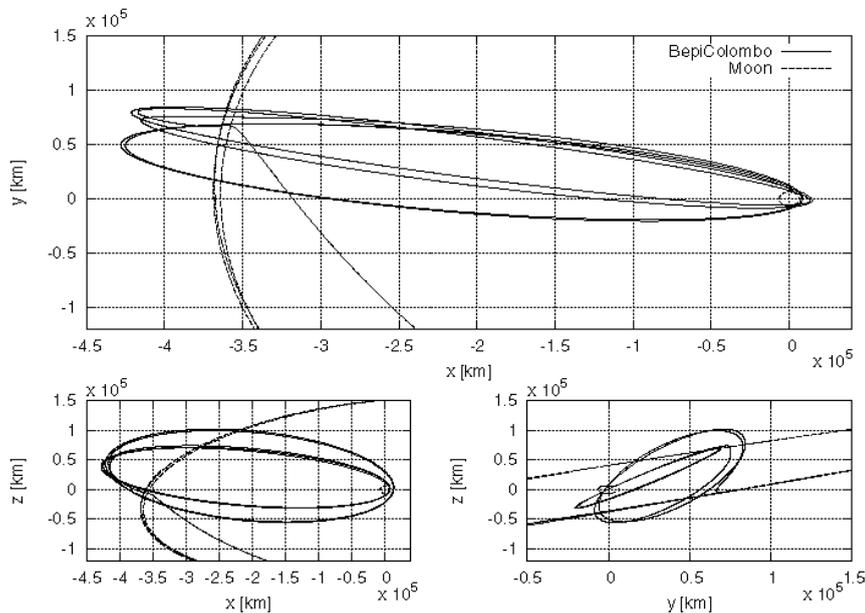
**Figure 12: Example of a 4-loop numerically integrated trajectory[4].
The launch date is 28 November 2010.**

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An example of a numerically integrated 4-loop trajectory (for a lunar swing-by in January 2011) can be seen in Figure 12. The luni-solar perturbations cause significant changes in the initial orbit (especially around the Moon resonance) so that some chemical manoeuvres are needed to encounter the Moon at the proper time and with the required arrival velocity. Moreover, for some launch dates the perturbations are so strong that the launch window needs to be closed in order to prevent that the spacecraft re-enters into the Earth atmosphere. A detailed analysis of the launch window design (including the correction of the launcher dispersion) will be presented in Corral et al.[11]. There it is shown that when the Sun is in the "wrong" quadrant and causes the perigee height to decrease, a launch is only possible during two 4-day intervals per month. This happens for the lunar swing-by opportunities in August and September 2012. On the contrary, if the Sun is in the "right" quadrant (like in June, July and October 2012), up to 14 launch days per month are found. Similar results are obtained for the lunar swing-by opportunity in January 2011.

The current baseline scenario foresees a launch of BepiColombo on 5 May 2012 with a lunar flyby on 24 July after 6 phasing orbits. A launch delay of up to 4 months can be accommodated with the design of the launch window as presented in this paper.

CONCLUSIONS

In this paper the lunar gravity assist design and the preliminary launch window for ESA's BepiColombo mission to Mercury are presented.

A simplified model, which assumes linked conics, is used to investigate the effectiveness of the lunar swing-by, in terms of providing the maximum excess velocity in the suitable direction. Furthermore different possible geometries for the lunar swing-by are presented.

The most promising ones for a lunar swing-by in summer 2012 are optimised, together with the interplanetary trajectory until the insertion into the target orbit at Mercury. Ten different scenarios are optimised for lunar swing-bys between June and October. The final mass in MMO differs only by less then 10 kg.

A phasing loop strategy is presented which widens the launch window of every scenario. A thirty days launch window is obtained with small variations in the period of the initial orbit and with the insertion of additional loops. Considering several monthly opportunities, a 140-day launch window is achieved.

Finally, the effects of the luni-solar perturbations are considered. During favourable months when the Sun is pulling up the perigee of the phasing orbit about 14 launch days remain. During unfavourable months (like August and September 2012) only short periods of launch opportunities (about two 4-day intervals per month) remain.

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