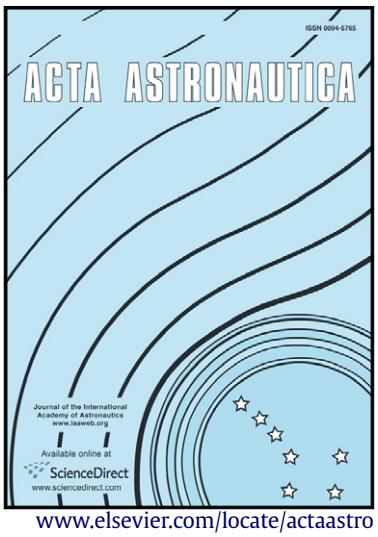

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END-OF-LIFE DISPOSAL CONCEPTS FOR LIBRATION POINT ORBIT AND HIGHLY ELLIPTICAL ORBIT MISSIONS

Camilla Colombo[1], Elisa Maria Alessi[2], Willem van der Weg[3], Stefania Soldini[4], Francesca Letizia[5], Massimo Vetrisano[6], Massimiliano Vasile[7], Alessandro Rossi[8], Markus Landgraf[9]

Libration Point Orbits (LPOs) and Highly Elliptical Orbits (HEOs) are often selected for astrophysics and solar terrestrial missions. No guidelines currently exist for their end-of life. However, as current and future missions are planned to be placed on these orbits, it is a critical aspect to clear these regions at the end of operations to avoid damage to other spacecraft and ensure on-ground safety. This paper presents an analysis of possible disposal strategies for LPO and HEO missions as a result of a European Space Agency study. The dynamical models and the design approach are presented for each disposal option. Five current missions are selected as test cases Herschel, Gaia, SOHO as LPOs, and INTEGRAL and XMM-Newton as HEOs. A trade-off on the disposal options is made considering technical feasibility, as well as the sustainability context.

1 Introduction

Libration Point Orbits (LPOs) and Highly Elliptical Orbits (HEOs) are often selected for astrophysics and solar terrestrial missions as they offer vantage points for the observation of the Earth, the Sun and the Universe. The neighbourhood of the Lagrangian collinear libration points L_1 and L_2 of the Sun – Earth system has been recognised as a vantage location for astrophysics and solar missions since the end of the 70's, with the NASA ISEE-3 mission[1]. Indeed, orbits around L_1 and L_2 are relatively inexpensive to be reached from the Earth and ensure a nearly constant geometry for observation and communication geometry, because the L_1 and L_2 libration orbits always remain close to the Earth at a distance of roughly 1.5 million km. Moreover, L_2 is situated on the Sun – Earth line beyond the Earth and thus it is suitable for highly precise tele-

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scopes requiring great thermal stability. Also, since the Sun, the Earth and the Moon are always behind the spacecraft, L_2 LPO ensure a constant geometry for observation with half of the entire celestial sphere available at all times. On the other hand, highly elliptical orbits about the Earth guarantee long dwelling times at an altitude outside the Earth's radiation belt; therefore, long periods of uninterrupted scientific observation are possible with nearly no background noise from radiations[2].

No guidelines currently exist for LPO and HEO missions' end-of-life; however, as current and future missions are planned to be placed on these orbits, it is a critical aspect to clear these regions at the end of operations[3,4]. In fact, orbits about the Libration point or Earth-centred orbits with very high apogee lie in a highly perturbed environment due to the chaotic behaviour of the multi-body dynamics[5]; moreover, due to their challenging mission requirements, they are characterised by large-size spacecraft. Therefore, the uncontrolled spacecraft on manifold trajectories could re-enter to Earth or cross the Low Earth Orbit (LEO) and Geostationary Earth Orbit (GEO) protected regions. Finally, the end-of-life phase can enhance the science return of the mission and the operational knowledge base.

In this article, an analysis of possible disposal strategies for LPO and HEO missions is presented as a result of a European Space Agency's General Study Programme study[6] within the GreenOPS initiative. End-of-life disposal options are proposed, which exploit the multi-body dynamics in the Earth environment and in the Sun–Earth system perturbed by the effects of solar radiation pressure, the Earth's gravity potential and atmospheric drag. The options analysed in this study are Earth re-entry[7], or injection into a graveyard orbit for HEOs, while spacecraft on LPOs can be disposed through an Earth re-entry[8], or can be injected onto trajectories towards a Moon impact[9], or towards an heliocentric parking orbit, by means of impulsive delta-v manoeuvres[10] or the enhancement of solar radiation pressure with some deployable light-weight reflective surfaces[11]. On the base of the operational cost, complexity and demanding delta-v manoeuvres, some disposal options were preliminary analysed and later discarded such as the HEO disposal through transfer to a LPO or HEO disposal through Moon capture[3].

The article summarises the dynamical models considered for each disposal design: in the case of HEOs the long term variation of the orbit is propagated through semi-analytical techniques[7], considering the interaction of the luni/solar perturbations with the zonal harmonics of the Earth's gravity field. In the case of LPOs the Circular Restricted Three Body Problem[12] (CR3BP) or the full-body dynamics is employed for the Earth re-entry option and the transfer towards the inner or the outer solar system, while the coupled restricted three-body problem[13] is used for the Moon disposal option. The approach to design the proposed transfer trajectories is presented. In order to perform a parametric study, different starting dates and conditions for the disposal are considered, while the manoeuvre is optimised considering the constraints on the available fuel at the end-of-life.

Five ESA missions currently operating on LPO and HEO were selected as test case scenarios: INTEGRAL[2] and XMM-Newton[14] as HEO, Herschel[15], Gaia[16] and SOHO[17] as LPOs. For each mission the disposal strategies are analysed, in terms of optimal window for the disposal manoeuvre, manoeuvre sequences, time of flight and disposal characteristics, such as re-entry conditions or the hyperbolic excess velocity at arrival in case of a Moon impact. In a second step, a high accuracy approach is used for validating the optimised trajectories. Finally, a trade-off is made considering technical feasibility (in terms of the available on-board resources and Δv requirements), as well as the sustainability context. Some general recommendations are drawn in terms of system requirements and mission planning.

2 Selected missions

The selected missions for the detailed analysis of disposal strategies are INTEGRAL (current) and XMM-Newton (current) in the HEO-class, Herschel (past), SOHO (current) and Gaia (current) in the LPO-class.

2.1 Mission constraints

Table 1 summarises the main mission constraints for the selected missions, in terms of available on-board propellant and equivalent Δv-budget, which enables to estimate the propellant at the End-Of-Life (EOL). The available fuel enables a trade-off analysis between the extension of the mission and the feasibility of reliable disposal strategies. It has to be noted that all kinds of manoeuvres are influenced by the instrument lifetime (e.g., batteries, reaction wheels, transponder switches) and components failures. Moreover, the disposal trajectory should be designed considering the pointing constraints (reported in Table 1), due to the thermal and the power subsystem or payload requirements, however this is neglected at the current stage of the study. Finally, other spacecraft parameters relevant for the analysis of disposal trajectories are the reflectivity coefficient c_R (an equivalent reflectivity coefficient c_R^* is computed considering the contribution of all the parts of the spacecraft), and the area-to-mass ratio A/m which define the contribution of the solar radiation pressure and the aerodynamic drag perturbation. Some options of disposal that exploit non-gravitational perturbations, such as the effect of solar radiation pressure, are constrained by the maximum area-to-mass achievable with the current spacecraft configuration or with minor changes to the operational configuration. The overall cross area used to compute the area-to-mass ratio was found by considering the spacecraft spin axis constrains, when applicable, and the spacecraft reflective surfaces. In case of missions around L_1, the projected areas are the spacecraft solar array and the spacecraft bus. On the other hand, for missions around L_2 the projected areas are the spacecraft sunshade and the solar arrays.

Table 1. Missions constraints and spacecraft parameters.

Mission	Type	Dry mass [kg]	Available fuel [kg] (date)	Equivalent Δv [m/s] (date)	Failures	Consumption per year [kg]	Pointing constraints	c_R* BOL	Max A/m EOL [m^2/kg]
SOHO	LPO	1602	108-111 (31/12/2016)	140.8-144.59 (31/12/2016)	loss of gyroscopes	2-1	4.5° < SEV angle < 32°	1.9	0.0196
Herschel	LPO	2800	180 (1/1/2013)	130 (1/1/2013)	Helium finished	4	Constraints due to thermal management and star trackers operation, Sunshade pointing towards the Sun.	1.5	0.0048
Gaia	LPO	1392	5 (EOL)	10 (EOL)	N/A	N/A	Spin axis precessing with an angle of 45°around s/c-Sun line	1.21	0.0585

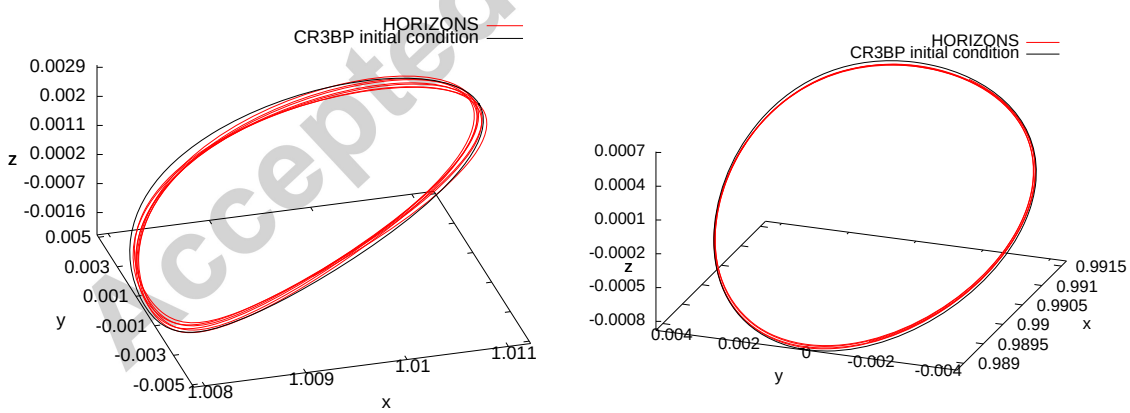
INTEGRAL	HEO	3414	90 (1/6/2013)	59.99	-	8	Telescope never points closer than 15° from the Sun	1.3	0.013
XMM	HEO	3234	47 (1/6/2013)	33.26	Reaction wheel degradation	6	Telescope never points closer than 15° from the Sun	1.1	0.021

2.2 Mission scenarios

The initial conditions considered for SOHO and Herschel, displayed in Table 2 and Figure 1, were selected through comparison with the ephemerides provided by the JPL HORIZONS system[18]. T is the orbital period in adimensional units and C_J the Jacobi constant. In the case of Gaia mission, a Lissajous orbit was computed (see Figure 2) and the corresponding unstable invariant manifold using a Fourier series parameterisation as explained in Ref. 12, in order to match the in-plane and out-of-plane amplitudes.

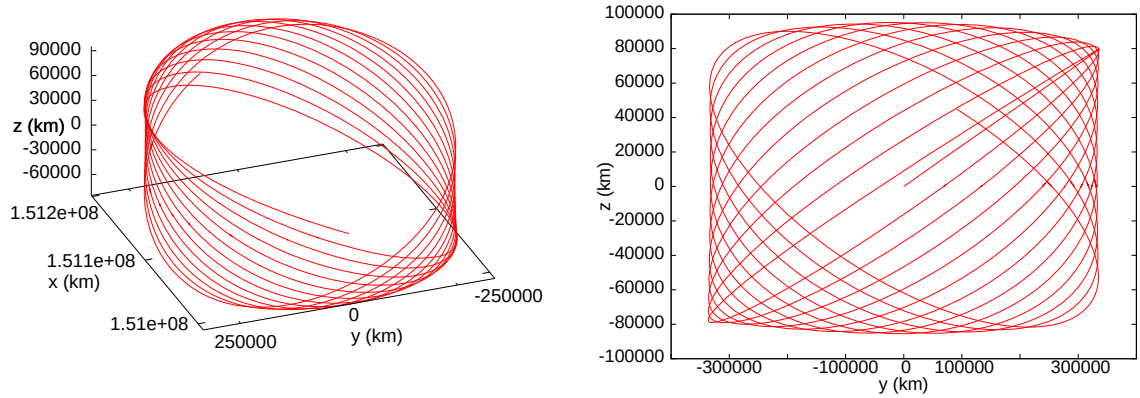
Table 2. Initial conditions selected for simulating the behaviour of SOHO and Herschel. Non-dimensional units, synodic reference system centred at the Sun – Earth + Moon barycentre.

Mission	Orbit	LP	T	x	y	z	v_x	v_y	v_z	C_J
SOHO	Halo South	L_1	3.0595858	0.9888381	0	–0.0008802	0	0.0089580	0	3.0008294
Herschel	Halo North	L_2	3.0947685	1.0111842	0	0.0028010	0	–0.0100059	0	3.0007831



a) **b)**

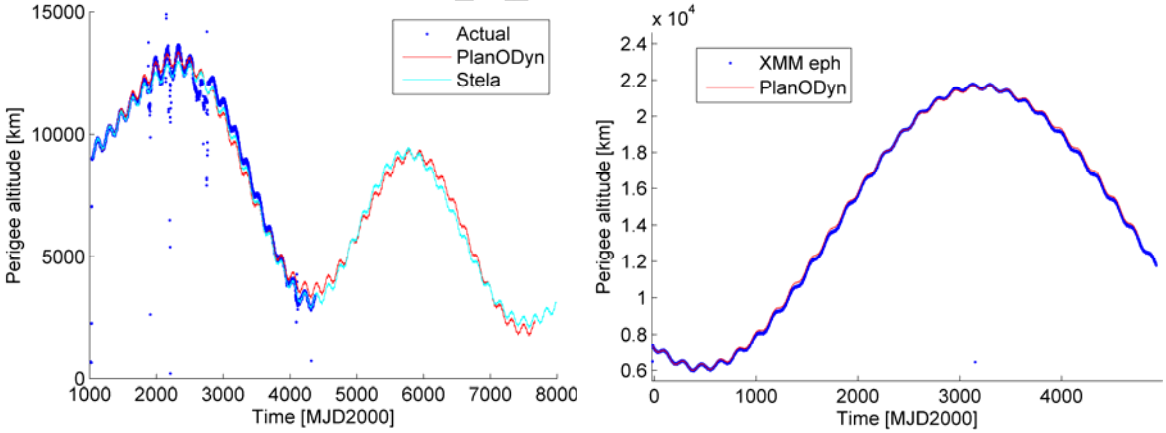
Figure 1. Halo orbit predicted by the JPL HORIZONS system (red) and orbit reproduced in the CR3BP used for the disposal analysis (blue). Non-dimensional units, synodic reference system centred at the Sun – Earth + Moon barycentre. A) Herschel; b) SOHO.



a) **b)**

Figure 2. Lissajous orbit selected to simulate the evolution of Gaia. Synodic reference system centred at the Sun – Earth + Moon barycentre. A) 3D representation and b) *y-z* projection.

The dynamics of HEO with high apogee altitude is mainly influenced by the effect of the third body perturbation due to the gravitational attraction of the Moon and the Sun and the effect of the Earth's oblateness. For the analysis of HEO disposal the suite *PlanODyn* was developed[7], to propagate the Earth-centred dynamics by means of the averaged variation of the orbital elements in Keplerian elements. The perturbations considered are solar radiation pressure, atmospheric drag with exponential model of the atmosphere, zonal harmonics of the Earth's gravity potential up to order 6, third body perturbation of the Sun and the Moon up to degree 4 of the Legendre polynomial. The code was successfully validated against the ephemerides of the INTEGRAL mission from NASA HORIZONS[18], and of the XMM-Newton mission, given by ESA[19] (see Figure 3).



a) **b)**

Figure 3. HEO perigee altitude evolution. Blue: actual ephemerides, Red: propagation with *PlanODyn*, cyan: propagation with STELA software[20]. A) INTEGRAL and b) XMM-Newton.

3 HEO disposal through Earth re-entry

HEO missions can be disposed in a definite way through re-entry into the Earth's atmosphere. This can be achieved by exploiting the natural long-term perturbations to the orbit due to the interaction between luni-solar perturbation and the J_2 effect, as explained in Ref. 7.

The disposal strategy to re-entry is designed considering a single manoeuvre performed during the natural orbit evolution of the spacecraft. The variation in orbital elements Δkep due to an impulsive manoeuvre at a generic time t is computed through Gauss' planetary equations written in finite-difference form as

$$\Delta kep = G\left(kep\left(t_m\right), f_m, \Delta\mathbf{v}\right) \tag{1}$$

where f_m is the true anomaly at which the manoeuvre is given and $\Delta\mathbf{v}$ the velocity change defined by its magnitude, in-plane and out-of-plane angles. The new set of orbital elements after the manoeuvre kep_d

$$kep_d = kep\left(t_m\right) + \Delta kep \tag{2}$$

is propagated for the available interval of time to perform the disposal with *PlanODyn*. Then, the evolution of the perigee altitude $h_p\left(t\right)$, starting from the deviated condition in Eq. (2), is computed and the minimum perigee altitude $h_{p,\min}$ of the perigee history can be determined as the minimum perigee altitude that the spacecraft reaches within the allowed available time span for re-entry $\Delta t_{\text{disposal}}$. Due to the natural oscillations of the orbit because of perturbations, the effect of the disposal manoeuvre will be different depending on the time it is applied. Therefore, different starting dates for the disposal manoeuvre were selected within a wide disposal window. For each initial condition corresponding to a certain time, the manoeuvre magnitude Δv and direction and the point on the orbit where the manoeuvre is performed f_m were determined through global optimisation. Any reached altitude below 50 km is accepted and the total Δv is minimised.

Once a first guess solution was found through global optimisation and considering the averaged dynamics with *PlanODyn,* in a second stage, a high fidelity optimisation is performed. A full dynamical model is considered written in a Cartesian reference frame, centred at the Earth. The high fidelity model includes the higher terms up to the 20th degrees for sectorial and tesseral harmonics of the Earth and Moon gravity field, the gravitational perturbations of the Sun, the Moon and the planets of the solar system, solar radiation pressure, and atmospheric drag with an exponential model of the Earth's atmosphere[6]. Given the fact the model used for designing the re-entry is based on the propagation of the mean orbit and only some perturbations effects are considered, if one performs the re-entry manoeuvre at the nominal time, the orbit could not re-enter because the osculating true anomaly does not correspond to the optimal true anomaly f_m found through global optimisation. For this reason, the solutions computed with *PlanODyn* is then refined to ensure a re-entry also in the high fidelity model. Therefore, the scope in the second stage of the design process is to identify suitable initial conditions, typically finding the instant of time in the neighbourhood of the nominal solution at which a manoeuvre of the same magnitude will offer the most favourable final conditions. A two-step optimisation was performed. The first optimisation step consists in running a broad search in true anomaly with a step of 1 degree to obtain the optimal conditions, which possibly lead to re-entry by applying the nominal Δv manoeuvre computed in Eq. (1). The second optimisation step applies a quasi-Newton method to re-enter the

spacecraft below 150 km by optimising the Δv manoeuvre performed at the identified optimal initial condition.

In the following section only INTEGRAL disposal through Earth re-entry is presented as from the analysis, XMM re-entry resulted unfeasible with the available on-board fuel over a time period of 30 years.

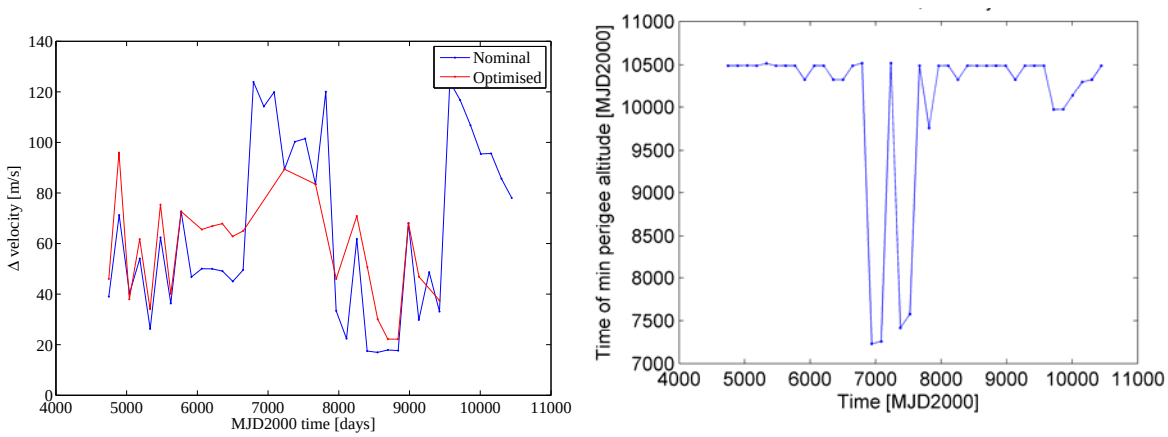
3.1 INTEGRAL disposal through Earth re-entry

INTEGRAL orbit future evolution was predicted until 2029 (see Figure 3). The interval $\Delta t_{\text{disposal}}$ considered for the disposal design is from 2013/01/01 to 2029/01/01. The maximum Δv available for the manoeuvre sequences is estimated to be 61.9 m/s in 2013/01/01 as in Table 1. Within the considered $\Delta t_{\text{disposal}}$, re-entry below 50 km is possible with Δv as little as 40 m/s, as visible in Figure 4 that shows the required Δv for Earth re-entry as function of the time the manoeuvre is performed between 2013/01/01 and 2028/08/07 (blue line). The disposal solutions can be grouped in four families of initial conditions that present similar dynamics behaviour in terms of evolution of orbital elements following the manoeuvre. Indeed, three jumps in the required Δv for re-entry are visible from the blue line in Figure 4a. The best solutions belongs to family 1, which have a re-entry in 2028, with a Δv between 27 and 73 m/s (depending on the year and month the manoeuvre is given between 2013 and the first half of 2018). Family 2 disposal options, instead, need a higher Δv to be given between the second half of 2018 and the first half of 2021 to reach the minimum perigee between 2019 and 2020 (quicker re-entry). Colombo et al. showed the dependences on the different families of re-entry conditions upon the orbital elements with respect to the main disturbing body, i.e., the Moon[21]. The optimal manoeuvre allows increasing the amplitude of the oscillations in anomaly of the perigee measured with respect to the Earth-Moon plane and, therefore, in eccentricity, so that the eccentricity can be increased up to the critical eccentricity

$$e_{\text{crit}} = 1 - \left(R_{\text{Earth}} + h_{p,\ \text{re-entry}}\right)/a$$

where R_{Earth} is the radius of the Earth, a is the semi-major axis and the target perigee altitude $h_{p,\ \text{re-entry}}$ is set to 50 km. In particular, when the nominal orbit eccentricity is low, the optimised re-entry manoeuvre tends to further decrease it; as a consequence, the following long term propagation will reach a higher eccentricity, corresponding to a re-entry. In this case, the manoeuvre is more efficient (i.e., lower Δv is required as in family 1). On the other side, when the nominal eccentricity is high, the re-entry manoeuvre aims at further increasing it. In this case, the required Δv is higher, while the target minimum perigee is reached after a shorter time (Figure 4b). Further details on the optimisation of the re-entry manoeuvre as function of the initial orbital parameters are given in Ref. 7.

In Figure 4a refined solutions with the high fidelity model of the dynamics are shown with a red line. Especially for the best re-entry conditions, there is a very good agreement between the nominal solutions and the refined solutions. There are cases for which the magnitude of the manoeuvres is in the order of 60 m/s, which is the maximum available Δv for INTEGRAL (see Table 1). In the worst cases, instead, an additional Δv in the order of 20-30 m/s with respect to the nominal manoeuvre is required.



a) **b)**

Figure 4. Required Δ*v* to de-orbit INTEGRAL. The blue line refers to the nominal solutions from the global optimisation, the red lines to the optimised solutions.

4 HEO disposal through injection into a graveyard stable orbit

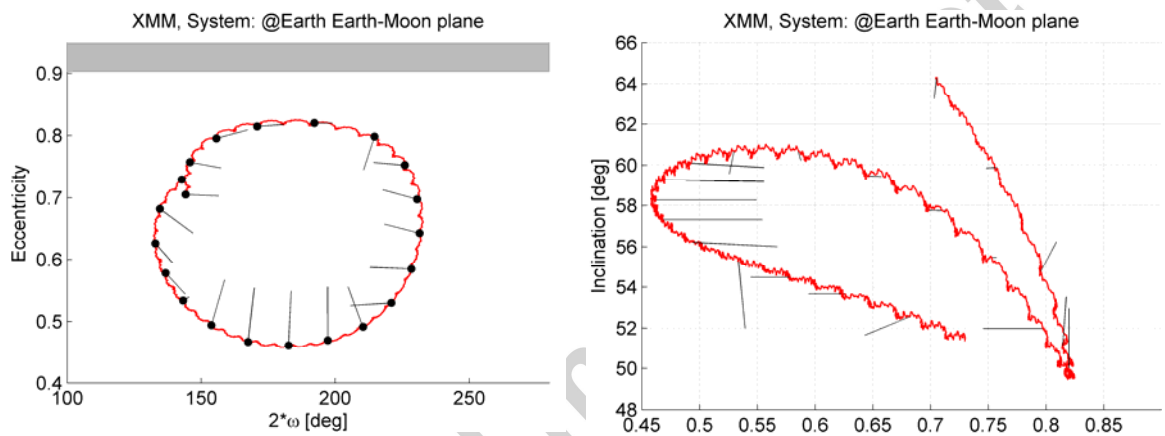
Another option that is investigated for HEO is the transfer into a graveyard orbit. The existence of long-term stable orbits can be investigated, where the evolution of the orbital elements due to natural perturbation is limited. Such orbits can be chosen as graveyard orbits. The design of graveyard orbits is performed with a method similar to the Earth re-entry design. A single manoeuvre is considered, performed during the natural orbit evolution of the spacecraft under the effect of perturbations. Also in this case, the new set of orbital elements Eq. (2) after the manoeuvre are propagated with *PlanODyn*. A graveyard orbit is designed imposing that, after the manoeuvre, the variation of the eccentricity in time stays limited, that is $\Delta e(t) = e(t)_{\max} - e(t)_{\min}$ is minimised, where $e(t)_{\max}$ and $e(t)_{\min}$ are respectively the maximum and minimum eccentricity reached during the natural evolution after the disposal manoeuvre is given. In order to analyse a wide range of disposal dates, different starting dates for the disposal were selected. Since Earth re-entry was shown to be unfeasible with the available on-board propellant, the selected disposal option for XMM is the transfer into a graveyard orbit. This option is not shown for INTEGRAL as, where possible, re-entry has to be preferred to graveyard orbit disposal[6].

4.1 XMM-Newton graveyard orbit disposal

XMM-Newton mission is planned to be disposed after 2016. The interval $\Delta t_{\text{disposal}}$ considered for the disposal design is from 2013/01/01 to 2033/11/25 and for each starting date analysed a graveyard orbit was designed with the requirement to be stable for 30-year period. The maximum Δv available for the manoeuvre sequences is estimated to be 40.5 m/s in 2013/01/01 considering a year consumption of 6 kg, a specific impulse of 235 s and the current propellant mass shown in Table 1. However, in order to keep the general validity of this study to different HEOs, the search for optimal trajectories for disposal was not limited to the on-board propellant; rather, a limit on

the delta-v of three times the available delta-v on-board was considered, which corresponds to 122 m/s on 2013/01/01.

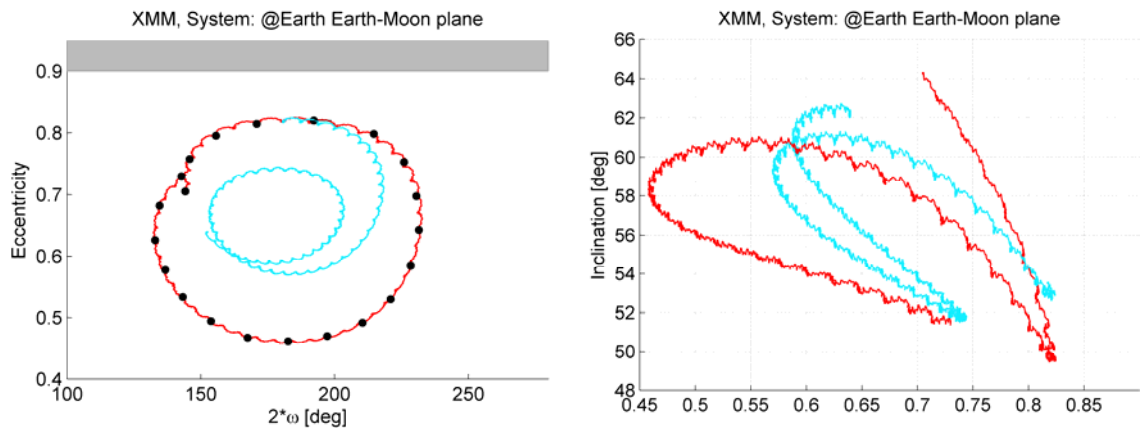
Figure 5 shows the optimal manoeuvre for a transfer into a graveyard orbit. The red line represents the nominal evolution of XMM over $\Delta t_{\text{disposal}}$. Each starting time considered for performing the disposal manoeuvre is indicated by a black circle in Figure 5a. The manoeuvre is represented in the phase space of eccentricity, inclination and anomaly of the pericentre with respect to the Earth-Moon plane. As it can be seen from Figure 5a, the manoeuvre aims at moving the orbit towards the centre of the libration loop in the eccentricity-ω phase space[21]. The magnitude of the manoeuvre is always close to the maximum bound set and, from Figure 5a, it is very clear that a higher available Δv would allow reaching a more stable orbit (i.e., centre of the libration). It has to be noted that, the designed graveyard orbit reduces, at least, the oscillations in eccentricity, preventing the spacecraft from an uncontrolled re-entry within the 30-year period. Indeed, it was verified that, after the manoeuvre is performed, the minimum perigee remains above 4000 km.



a) b)

Figure 5. XMM graveyard disposal manoeuvres between 2013 and 2033: Phase space evolution in the eccentricity-2ω-inclination phase space (Earth-Moon plane) a) 2ω-eccentricity and b) eccentricity-inclination. Red: nominal predicted orbit, black lines: Δv manoeuvres. The shaded area indicates eccentricities beyond the critical eccentricity for re-entry (at perigee altitude of 50 km).

As an example a disposal trajectory is shown, whose manoeuvre is performed on 20/04/2016, which corresponds to the year XMM has been currently extended to[22]. The evolution of the trajectory after the manoeuvre is shown with a cyan line in Figure 6.



a) **b)**

Figure 6. XMM graveyard disposal manoeuvre in 2016: Phase space evolution in the eccentricity-2ω-inclination phase space (Earth-Moon plane) a) 2ω-eccentricity and b) eccentricity-inclination. Red: nominal predicted orbit, black lines: Δv manoeuvre. The light blue line represents the evolution of the orbital elements with respect to the Earth-Moon plane after the disposal manoeuvre is performed for the following 30 years.

5 LPO disposal through Earth re-entry

5.1 Herschel and SOHO disposal through Earth re-entry

The re-entry for Herschel and SOHO LPO missions was designed considering their ephemerides from the HORIZONS system[18], with one-day time step. For Herschel the initial position and velocity were considered from 31/08/2012 to 29/04/2013. For SOHO, the latest available data (01/01/2011 to 01/01/2012) were used to simulate the expected orbit until 15/11/2016, the foreseen end-of-life date for SOHO. For this, SOHO orbit is assumed to be periodic in the synodic CR3BP reference frame with period equal to 178 days.

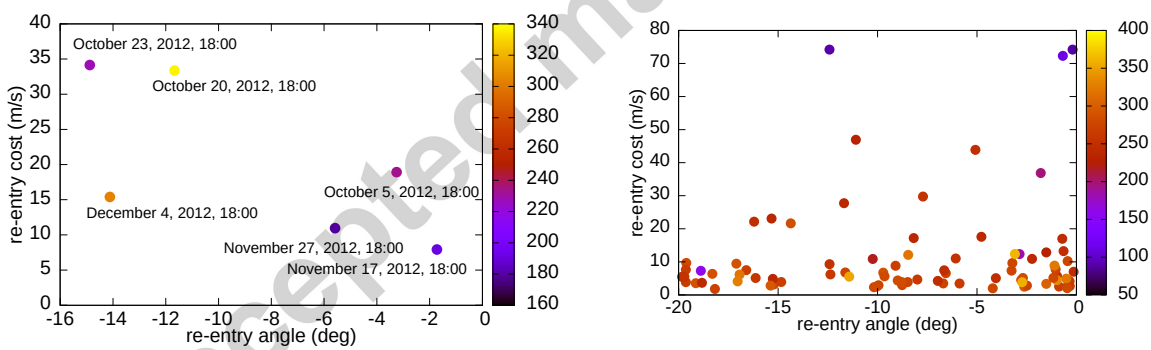
A differential correction procedure is implemented to find the change in the initial velocity such that the spacecraft injects from the actual LPO into the Earth-ward branch of the unstable invariant manifold of the LPO computed in the CRTBP from the initial conditions in Table 2[8]. The actual initial condition for the nominal mission from the real ephemerides is transformed into a non-dimensional synodic Sun – Earth + Moon reference system and it is propagated through the equations of motion of the CR3BP for a time interval between 1 and 30 days. At this point, the spacecraft is expected to inject into the unstable manifold which leads to Earth re-entry. To this end, the variational equations of the CR3BP are propagated together with the equations of motion. A Newton's method is applied to correct the initial velocity of the LPO in the CR3BP frame. In turn, this results in changing the initial velocity of the LPO in the real ephemerides model. The differential procedure targets the initial condition on the unstable manifold (corresponding to the LPO in Table 2) which minimises such manoeuvre. In particular, changing the time of flight to get to the manifold modifies both the required manoeuvre and the point reached on the manifold. In principle, another manoeuvre would be required to join the manifold also in velocity, but in practice this is not needed. Indeed, the CR3BP allows understanding how to move towards the Earth and, in a second stage, the new initial condition obtained through the differential correction

is propagated in a realistic dynamical model accounting for Sun, Earth, Moon and all the planets from Mercury to Pluto, solar radiation pressure, atmospheric drag below an altitude of 2000 km (exponential model), 10 x 10 geopotential if the distance with respect to the centre of the Earth is less than 200000 km. Whenever an orbit gets to an altitude lower than 100 km in less than a year, the re-entry angle is evaluated as

$$\tan\phi = \frac{e\sin f}{1+e\cos f}$$

where e is the eccentricity and f the true anomaly. Several re-entry solutions are obtained and the ones associated with an initial manoeuvre smaller than 150 m/s and a re-entry angle in between 0 and -20 degrees are selected. Note that re-entry angles with higher magnitude are also possible. If the re-entry is designed within the CR3BP dynamical model, then the angle obtained for a given transfer is function of the initial phase of departure from the LPO and the shape of the trajectory; however, when the re-entry is designed in the full model, this correspondence is broken. Two factors seems to be responsible for this: the initial manoeuvre and the solar radiation pressure, which, indeed, can modify significantly the trajectories. The re-entry velocity is always about 11.06 km/s at 100 km of altitude.

The selected feasible solutions for Herschel and SOHO re-entry are shown in Figure 7. In the case of Herschel, no solution takes place in 2013 (i.e., the year of the actual disposal manoeuvre) because a lower limit of the re-entry angle was fixed to -20 degrees. Indeed, a steeper re-entry angle means that the fragmentation of the satellite in atmosphere is less effective and a larger surviving mass is expected[6]. In the case of SOHO, re-entry can take place between 2014 and the end of 2016.



a) b)

Figure 7. Re-entry feasible solutions. A) Herschel mission: the colour bar refers to the total time of flight (days) from the libration point orbit to the Earth. B) SOHO mission: the colour bar indicates the starting date on the LPO for the re-entry (years from J2000).

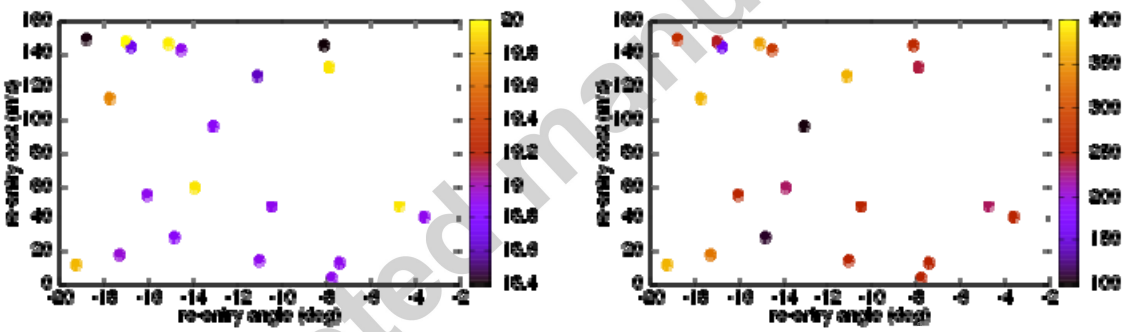
5.2 Gaia disposal through Earth re-entry

The procedure developed for Gaia is slightly different with respect to the one applied for SOHO and Herschel as Gaia cannot re-enter naturally[8], since the minimum distance to the Earth without performing any manoeuvre is about 50000 km. The nominal orbit was assumed to be a Lissajous quasi-periodic orbit (see Figure 2) propagated for about 6 years, to account for the 5.5 years of nominal duration of the mission, plus 6 months to perform the re-entry phase. As at the

time of the study, Gaia had not launched yet, two initial epochs for the first point on the Lissajous orbit were assumed, namely 24/12/2013 and 23/01/2014 to reflect the options for the mission launch.

A differential correction method was applied to compute the manoeuvre which allows the re-entry (rather than inserting the spacecraft into the unstable manifold as in the case of Herschel and SOHO). The equations of motion and the corresponding variational equations describing the full dynamical model are used. The re-entry can take place towards the end of the mission; starting from the LPO point on 28/03/2018, or, in the second launch scenario, about 1 month later. Each state from this epoch on was propagated for 365 days through the full dynamical model. Note that, the presence of other forces apart from the gravitational attraction of Sun and Earth + Moon causes the spacecraft to naturally leave the libration point orbit onto the unstable invariant manifold. A differential correction procedure was applied to each point of the trajectory discretised with 1 day-step, to change the velocity along the tangential direction in order to get to Earth.

Figure 8 shows the optimal solutions in terms of cost and re-entry angle. As expected, the optimal manoeuvres are given in correspondence of a point of the leg of the manifold which represents an apogee of the associated osculating orbit. However, this is not a sufficient condition to obtain a Δv less than 150 m/s. Nevertheless, in some cases the manoeuvre is nearly zero; it turns out that Gaia can arrive to the Earth at no expense by travelling through either a heteroclinic or homoclinic connection to a very high amplitude LPO. Further details on the LPO disposal option though Earth re-entry are given in Ref. 8.



a) **b)**

Figure 8. Optimal solutions for Gaia re-entry in terms of re-entry cost and re-entry angle for $c_R^* A/m = 0.0696$ m²/kg, with initial epoch on the LPO on 24/12/2013. The colour bar reports (a) the initial epoch of the re-entry trajectory (year from J2000) and (b) the total time of transfer (days).

6 LPO disposal towards a Moon impact

Disposal options via lunar surface impact and mission extension by capture into lunar orbit were studied in the coupled restricted three-body problem[13,23].

In order to approximate the trajectory in 4-body dynamics, trajectories legs deriving from the unstable invariant manifolds leaving the LPO in the Sun – (Earth + Moon) CR3BP and the stable manifolds of a LPO around L_2 in the Earth – Moon CR3BP are connected into a single trajectory. The Sun – (Earth + Moon) CR3BP has as primaries the Sun and the Earth – Moon barycentre. Connection between the two models is accomplished via the use of a Poincaré section where the

two phase spaces must intersect. The initial orbital phases α_0^{SE} and α_0^{EM} of both CR3BPs control the geometry of the connection. This can be reduced to a single parameter $\alpha_0 = \alpha_0^{EM} - \alpha_0^{SE}$ as only the relative phasing between Sun-Earth and Earth – Moon systems is necessary[24].

A study was undertaken to create a map of conditions near the L_2 libration point of the Earth-Moon system which can be reached from the Sun – Earth system and lead to a longer duration quasi-periodic orbit about the Moon or an impact on the lunar surface[9]. The resulting set of initial conditions about the Earth-Moon L_2 libration point, their corresponding orbit lifetime, and their category of decay (i.e., Moon impact or lunar capture or exit via libration points) serves as the basis of designing transfers from Sun – Earth libration point orbit towards the Moon. The Sun – Earth LPO unstable manifolds and the trajectory arcs flowing towards chosen lunar target states are computed once and stored. Once this is completed, the transformation of the lunar target state arcs from Earth – Moon to Sun – Earth synodic barycentric reference frame can be quickly performed for the entire domain of the orbital phasing angle α_0. The lowest possible propellant cost was sought for a number of LPOs (see Table 3).

Table 3. Overview of lowest propellant cost solutions expressed in m/s in the coupled CR3BP. For some cases the problem is considered to be planar so the position and velocity along the z axis is neglected.

LPO Type	Capture Δv	Impact Δv
Herschel	N/A	N/A
Herschel Planar	1.5 m/s	2.2 m/s
Gaia	434 m/s	350 m/s
Gaia Planar	N/A	N/A
SOHO	121 m/s	139 m/s
SOHO Planar	10.5 m/s	67.8 m/s

Then, an investigation of the possibility to impact the Moon is pursued based on the actual ephemeris information. Using a full body model, a global optimisation routine is used to introduce a perturbation on the LPO at a given date, the direction and magnitude is selected such that the spacecraft proceeds towards the Moon and impacts upon its surface. It should be noted that although lunar surface impact is achieved here by using a single manoeuvre, a second manoeuvre at a later date will likely be necessary to serve as trajectory correction manoeuvre, or to aim the spacecraft to a particular region on the lunar surface. Note that, with respect to the design in the CR3BP, here a larger manoeuvre is used to depart from the LPO as opposed to a small perturbation in order to bring the spacecraft from the LPO towards the Earth along the flow of the unstable manifold.

In addition to ranking solutions based on characteristics such as time of flight and Δv cost, a metric named the C_3 value may also be used[25], using the orbital elements of the spacecraft state just before impact as:

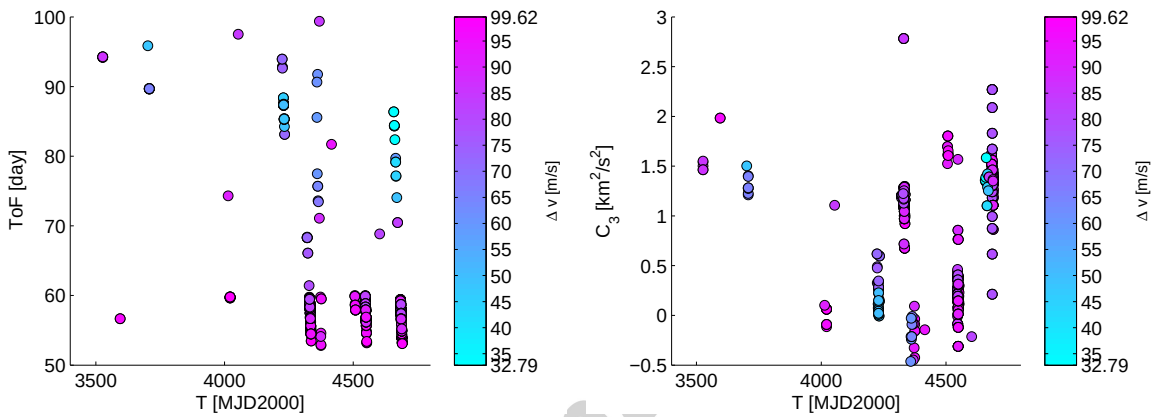
$$\frac{1}{2}C_3 = \frac{v^2}{2} - \frac{\mu_{\text{Moon}}}{r} = -\frac{1}{2}\frac{\mu_{\text{Moon}}^2}{h^2}\left(1-e^2\right) = -\frac{\mu_{\text{Moon}}}{2a} \tag{3}$$

where v and r are, respectively, the spacecraft velocity and position, h is the angular momentum and e the eccentricity. The C_3 value provides an indication of the robustness of the transfer: the lower the value, the more ballistic the capture at the Moon is, and thus the more robust the transfer is in case of contingencies. In the case of missing the lunar surface, a trajectory with low C_3 value will be quasi-captured by the Moon allowing for further small manoeuvres to impact the

spacecraft upon the lunar surface. This value is useful as another parameter to compare one particular transfer with another. Further details on the design strategy of disposal trajectories towards a Moon impact are given in Ref. 9. In the following section the case scenarios of Herschel, Soho ad Gaia are presented.

6.1 Herschel disposal through Moon impact

Herschel disposal through Moon impact was designed from 01/08/2009 to 01/02/2013. An overview of the solutions found for Herschel is shown in Figure 9, which shows the date of departure from the LPO on the *x*-axis and the time of flight and the C_3 value before lunar surface impact on the *y*-axis. Each solution is colour-coded according to the Δv cost in m/s to bring the spacecraft onto its lunar impact trajectory.

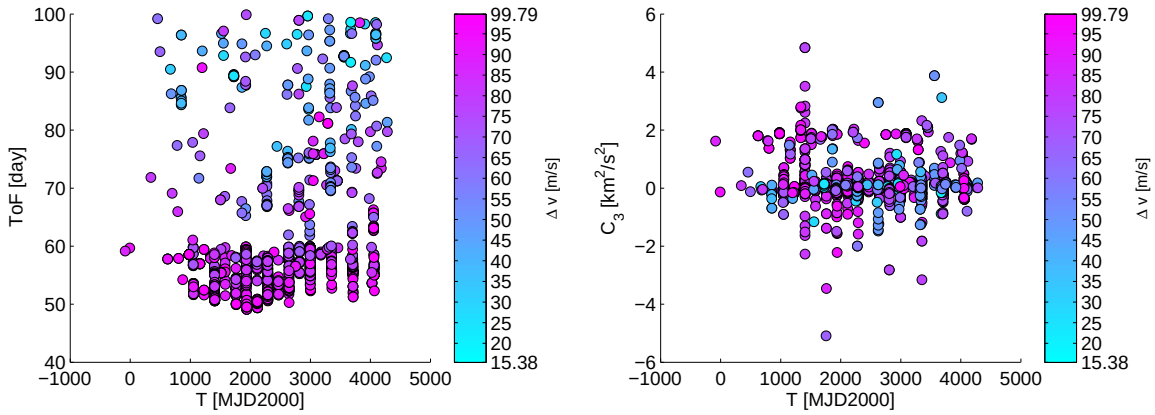


a) **b)**

Figure 9. A) Time of flight and b) C_3 value to impact the Moon as a function of the time of departure from the Herschel LPO in MJD2000. Solutions are colour-coded according to the Δv cost in m/s.

6.2 SOHO disposal through Moon impact

The dates considered for the analysis of disposal manoeuvres for SOHO to Moon impact are from 26/09/1998 to 01/01/2012. Such window does not correspond to the disposal window; however, it is appropriate to study the behaviour of the solution space and the required Δv. Figure 10 shows the solutions for SOHO.

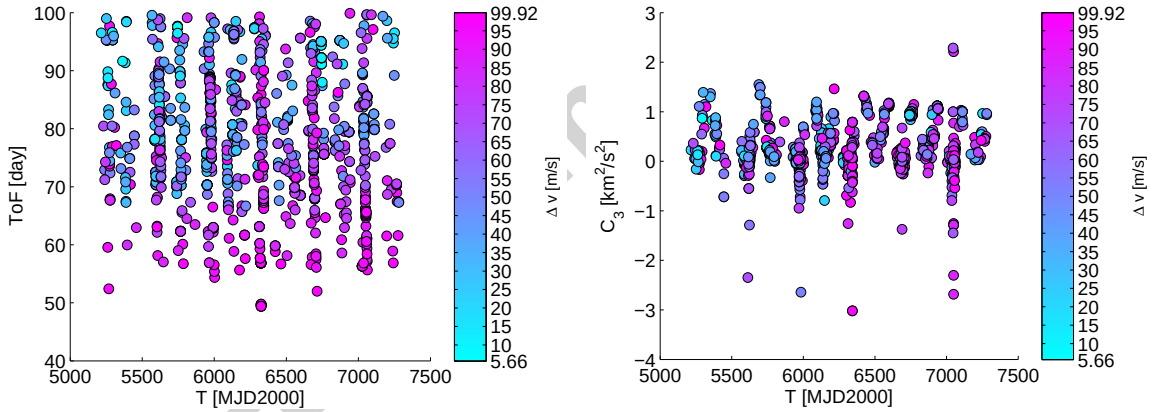


a) b)

Figure 10. A) Time of flight and b) C_3 value to impact the Moon as a function of the time of departure from the SOHO LPO in MJD2000. Solutions are colour-coded according to the Δ*v* cost in m/s.

6.3 Gaia disposal through Moon impact

Solutions for Gaia disposal through Moon impact are shown in Figure 11.



a) b)

Figure 11. A) Time of flight and b) C_3 value to impact the Moon as a function of the time of departure from the Gaia LPO in MJD2000. Solutions are colour-coded according to the Δ*v* cost in m/s.

7 LPO disposal towards an heliocentric parking orbit

Another option is to dispose the spacecraft away from the Earth exploiting the CR3BP dynamics, and then ensuring it does not return to Earth. This strategy was first proposed by Olikara et al[10]. This concept can be effectively explored in the CR3BP, where the energy of spacecraft is directly related to the zero velocity surfaces of the system. If the energy of the spacecraft is brought to the appropriate level, the zero velocity surfaces will be closed around the Earth preventing movement from and to it. If the zero velocity surfaces are closed when the spacecraft is outside the interior region near the Earth the spacecraft will not return to the Earth. The amount of

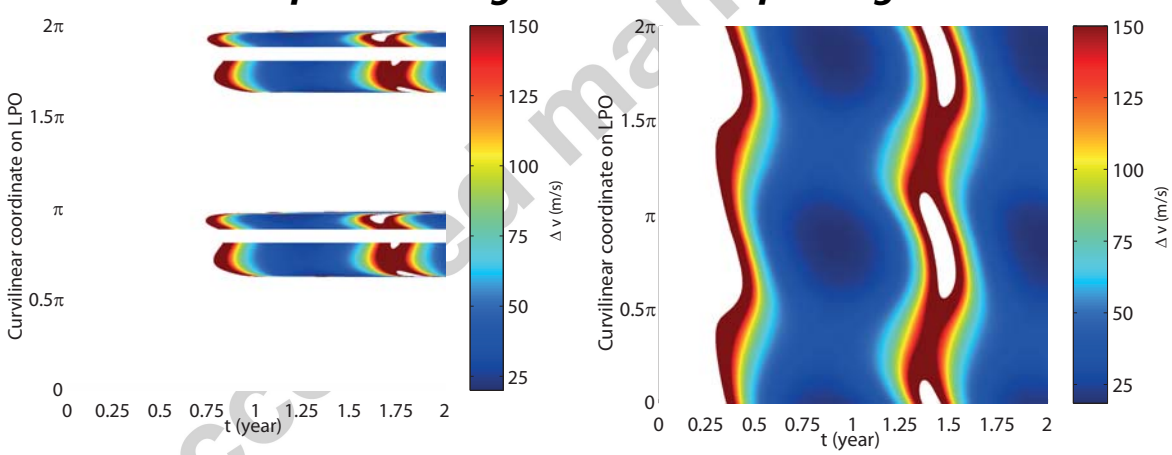
Δv necessary is computed from the Jacobi constant. The spacecraft in its LPO can be placed on one of the unstable manifold legs that flow from the LPO. One branch will lead towards the Sun realm of the Earth-Sun system, while the other towards the outer part of the Earth-Sun system. As the spacecraft moves away from the LPO, the Δv to change the Jacobi constant can be computed at any time along any point of the manifold as $\Delta v_{req} = \left| v_{actual} - v_{req} \right|$ were v_{actual} is the actual velocity of the spacecraft along the manifold and the required velocity can be determined from

$$v_{req}^2 = \left(\dot{x}^2 + \dot{y}^2 + \dot{z}^2 \right) = x^2 + y^2 + 2\left(\frac{1-\mu}{r_S} + \frac{\mu}{r_E} \right) - C_{J\,closed}$$

where x, y, z and $\dot{x}$, $\dot{y}$ and $\dot{z}$ are respectively the positions and velocities in the Sun – Earth + Moon CRTBP with gravitational parameter μ.

This procedure was applied to study SOHO, Herschel, and Gaia disposal. For a period of 2 years after leaving the LPO, the Δv to close the surfaces of Hill is computed for both branches of unstable manifold. In the following sections a value for the Δv can be read for each plot based on the position from where it departed from the LPO (shown on the y-axis) and on the time after having departed the LPO (shown on the x-axis). Some conditions are filtered out due to one of the following reasons: the trajectory approaches the Earth within 60000 km, or portions of a given arc on a manifold may be within the inner region near the Earth (i.e., between the libration points L_1 and L_2).

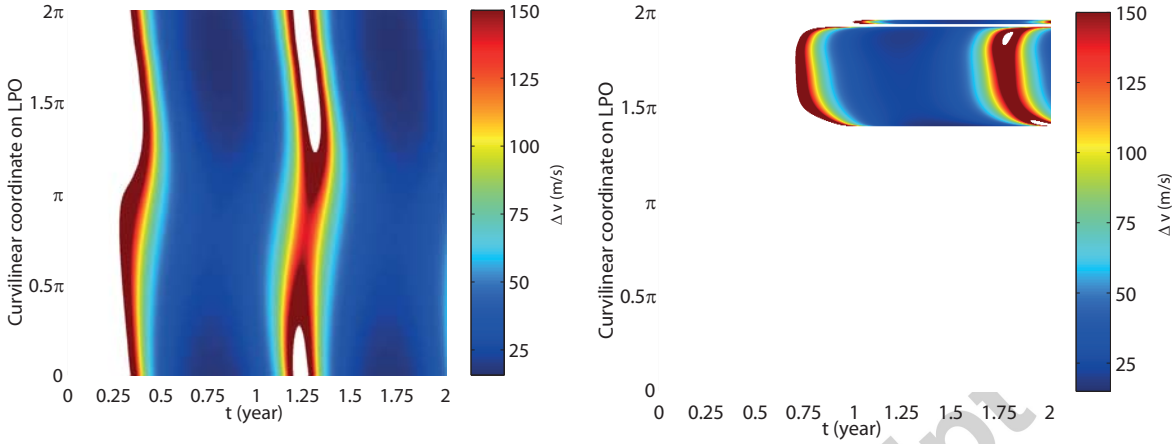
7.1 Herschel disposal through heliocentric parking orbit



a) **b)**

Figure 12. Δv cost for Herschel mission to close the surfaces of Hill as function of the time after departure from the LPO and position of departure on the LPO. Disposal towards a) the inner part of the solar system and b) the outer part of the solar system.

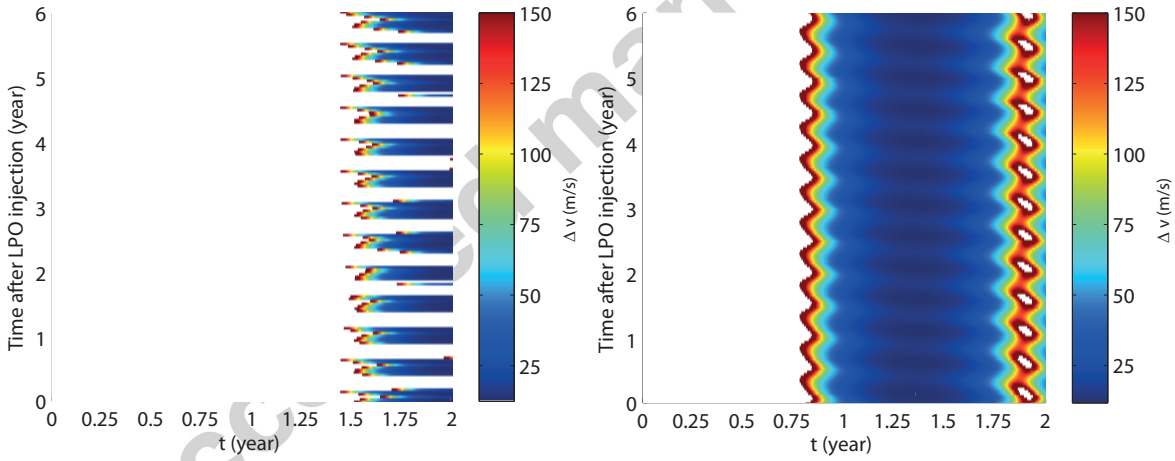
7.2 SOHO disposal through heliocentric parking orbit



a) b)

Figure 13. Δ*v* cost for SOHO mission to close the surfaces of Hill as function of the time after departure from the LPO and position of departure on the LPO. Disposal towards a) the inner part of the solar system and b) the outer part of the solar system.

7.3 Gaia disposal through heliocentric parking orbit



a) b)

Figure 14. Δ*v* cost for Gaia mission to close the surfaces of Hill as function of the time after departure from the LPO and position of departure on the LPO. Disposal towards a) the inner part of the solar system and b) the outer part of the solar system.

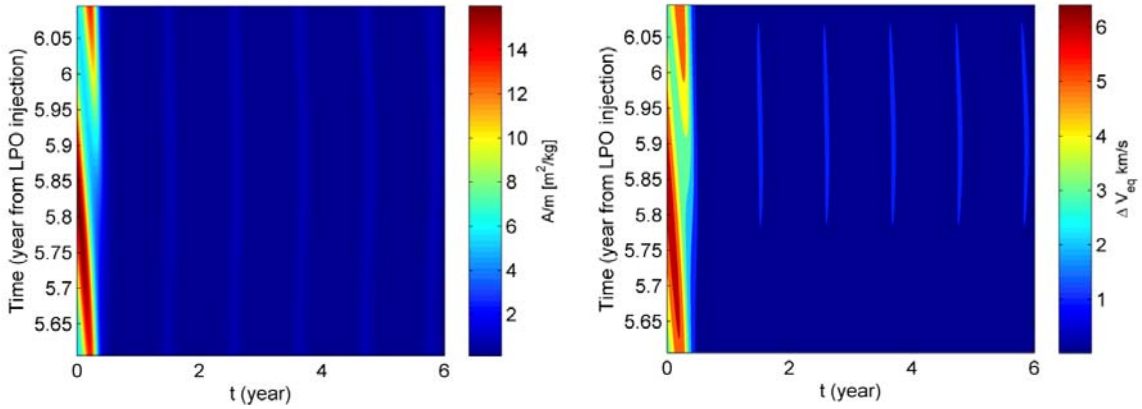
8 LPO disposal towards the outer solar system through solar radiation pressure

In the previous section, it was investigated the LPO End-Of-Life (EOL) disposal which aims to close the Hill's curves to prevent the spacecraft's Earth return. Olikara et al.[10] proposed a similar study that includes a sensitivity analysis on the effectiveness of using the restricted three-body problem as an approximation of the spacecraft's dynamics. The closure of the zero-velocity curves is performed with a traditional Δv manoeuvre and the curves can be closed either at L_1 or in L_2. Thus, the spacecraft can be confined inside the solar system (L_1 closure) or outside the Sun-Earth system (L_2 closure).

In this section, an alternative propulsion option is investigated to perform a quasi-propellant-free manoeuvre that closes the zero-velocity curves enhanced by Solar Radiation Pressure (SRP)[11]. The manoeuvre is performed through the deployment of a sun-pointing and auto-stabilised reflective structure that allows the change in the overall energy of the system. Note that, in this case, the energy is augmented rather than decreased as in the traditional case, because the acceleration of SRP is now included in the dynamics and it is sensitive to the area-to-mass ratio of the spacecraft. Therefore, the deployment of an EOL reflective area changes the shape of the potential function by increasing the energy of the system. The effect of the deployed sun-pointing EOL area is in shifting the position of the collinear Libration points along the *x*-axis, therefore the closure of the Hill's curves occur at the so-called pseudo Libration-point with solar radiation pressure (SL). However, due to the constrains in the acceleration of SRP, the spacecraft is always confined to stay on the right side of the pseudo Libration point; thus, SRP can be exploited only when the LPO disposal is toward the outer Sun – Earth system or, in other words, when the closure is performed at SL_2 [11]. This strategy was studied for Herschel, Gaia and SOHO. After the injection of the spacecraft onto the unstable trajectory, thanks to a Δv manoeuvre (quasi propellant-free strategy) from a starting point of the periodic orbit, the minimum EOL area required to close the Hill's curves in SL_2 is found through numerical optimisation.

Figure 15 shows the results in the case of Gaia in term of required area-to-mass and equivalent Δv. The equivalent Δv is a theoretical value, not achievable with a traditional propulsion, which quantifies changes in the energy of the system[11]. This makes easier to compare this strategy with the traditional propulsion-based strategy. From our analysis, spacecraft's similar to Gaia requires a minimum of 11 m-span (i.e., square flap) of additional area from its original 69 m^2 sunshade to perform the closure at SL_2. Instead, Herschel and SOHO require a delta area of 28.64 m-span and 20.65 m-span, respectively. Ikaros mission demonstrated the deployment of a 20-m span of a squared sail; therefore, spacecraft similar to Gaia can potentially be more likely to use an EOL device.

Current studies are aimed at including the effect of the Earth's eccentricity into the design of the EOL area, in order to determine an area margin that prevent the opening of the zero-velocity curves due to perturbations related to the full-body dynamics[11].



a) b)

Figure 15. Gaia disposal through SRP. A) Required area-to-mass ratio for the closure of the Hill's curves and b) equivalent Δ*v*.

9 HEOs and LPOs disposal strategies trade-off

Based on the analysis on the selected missions, some preliminary guidelines for future and current missions can be drawn as a general output of this study.

- The extension of a mission should be considered in the context of the sustainability of the whole program. HEO and LPO missions are not sustainable without a planned end-of-life disposal as HEO and LPO regions will be selected for other future missions. Moreover, by optimising the disposal phase of the mission, the same or slightly higher mission cost could allow extra output from the scientific and operational point-of view. End-of-life disposal strategies should, therefore, be considered as an extension of the mission.
- On the other hand, it should be considered that the cost of the disposal is not only the additional delta-*v*, but also the cost for operations, and the resources required to maintain the mission team. Moreover, the extension of a mission is subjected to several constraints such as budget and geo-return constraints. Some of them, such as mission constraints and disposal requirements have been taken into account in the present study; indeed, some strategies were discarded based on the operational cost they would require[3].
- Regulations prescribe protected regions for GEO and LEO, and GNNS protected regions are currently under definition[26]. It is important to highlight that space is not divided into definite regions (LPO may naturally transfer to HEO which, during re-entry, could interact with the Medium Earth Orbit MEO and then the LEO environment). Therefore, the different disposals transfers proposed in this study are in strict relation among them as it is possible to transfer from one leg to another.

9.1 HEOs disposal

XMM perigee is much higher than INTEGRAL and this makes more difficult a disposal through re-entry. Predicting the future, it is more likely to expect more missions with high perigee to be outside the radiation belt. This means that more scenarios such as LPOs or high perigee HEOs are expected and the trade-off between a safe environment for observations and a good configuration for disposal should be considered. This issue highlights the importance to plan in

advance the disposal strategy; indeed the exploitation of luni-solar perturbation can decrease the required manoeuvre for re-entry. However, due to the long period for the natural oscillations, the manoeuvre should be planned in advance. Through the gravitational of the Moon, HEOs can also inject to transfer orbits which reaches LPOs in the Earth – Moon and the Earth – Sun system. Therefore, in general, HEO and LPO missions should not be considered as separate classes, rather general guidelines should be adopted. This is also valid for other classes of missions; for example, HEOs, during re-entry interact with the MEO environment. Moreover, other kind of HEOs should be studied, such as Molniya type orbit and disposal orbit for launcher upper stages, which are quite common and have a stronger interaction with the MEO and LEO environment. As disposal strategies, re-entry should be preferred over graveyard orbit injection, as a definite and sustainable solution.

9.2 LPOs disposal

There are several considerations to take into account when choosing between impacting a spacecraft from a LPO upon the Earth's or Moon's surface. For the missions analysed, LPOa transfer towards a Moon impact or an Earth re-entry have similar propellant costs. The former are generally characterised by a shorter time-of-flight than Earth-re-entry options, which may lead to savings in operational cost. Disposal through Moon impact may be more difficult from a navigation point of view; however, specific re-entry angles at the Earth should be targeted so that the last phase of the trajectory is over inhabited zones. Direct Earth re-entry solutions exist; therefore, uncertainties on the ground area can be reduced with respect to re-entry from MEO and LEO.

To reduce operational costs for Moon impact disposal it is advised to use as much propellant as available (whilst still leaving a reserve for trajectory corrections) when leaving the LPO. This not only reduces the total transfer duration, but limits the time spent in the vicinity of the LPO, where motion is more chaotic and harder to control.

Concerning the possibility of a lunar impact, the guideline from this study and the discussion with ESA is to consider this option only if a significant scientific return can be obtained. From one side, it is true that the solar wind would sweep off any dust created by the collision, that GRAIL and LCROSS missions already ended by crashing on sites of special interest to gather new data on the lunar environment and also that the planetary protection policy is not a matter of concern for the Moon. However, on the other hand, a high energy impact should be targeted to avoid the creation of large dimension fragments and the location should be accurately selected, in particular to preserve past missions landing sites. Note that, the LPO missions considered in this study have a higher mass than GRAIL and LCROSS. So from the point of view of operations requirements and effort, the complexity of this strategy resembles the one associated with an Earth's re-entry; however, in the second case, specific re-entry angles and a very short interaction with the LEO/MEO regions should be targeted (the collision probability can be considered anyway to be very low).

Disposal towards the inner or the outer solar system with delta-v manoeuvre to close the Hill regions are generally feasible as they require low delta-v budget. However, the problem of such disposal option is that the spacecraft maintains a 1:1 resonance with the Earth and may return after several years. For this reason, an additional Δv should be taken into account to consider the effect of perturbations on the long-term evolution of the orbit. The option of LPO disposal towards the outer solar system through solar radiation pressure is not feasible for current spacecraft, but could be easily implemented on future spacecraft as it is a no-cost solution, as long as the reflective area on-board the spacecraft can be further extended at the end-of-mission. After the device deployment, the disposal can be completely passive if the devised is self-stabilised to be

Sun-pointing attitude. It is expected that if the device is deployed further from the nominal LPO, the area-to-mass requirements would be decreased[11].

10 Conclusions

This paper proposes a series of end-of-life disposal strategies for Highly Elliptical Orbits and Libration Point Orbits. Some mission scenarios are analysed, namely INTEGRAL and XMM-Newton as HEOs and SOHO, Herschel and Gaia as LPOs. An evaluation of disposal strategies for each of mission scenario is presented. In addition, where possible, a parametric analysis is performed that allows defining optimal disposal strategies as a function of the orbital parameters and the delta-velocity. A further study will aim at analysing the influence of the orbit characteristics and spacecraft parameters on the effectiveness, safety, feasibility and sustainability of the disposal. In light of the objective of sustainability it appears reasonable to postulate a permanent removal of the hardware from the space environment as a main objective for the end of life strategy. For HEO missions this can be achieved by a controlled or semi-controlled re-entry into the Earth atmosphere. For LPO missions, the feasibility of a controlled re-entry to the Earth depends on the operational orbit and the spacecraft capabilities at the EOL. If a re-entry is not possible, a permanent removal from the space environment can be achieved by a lunar impact. If such a disposal is performed in line with a sustainable conduct of avoiding heritage sites and sites of high scientific interest it can be considered more sustainable than the semi-permanent solution of using a parking orbit.

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Highlights

End-of-life guidelines need to be defined for Libration Point Orbit and Highly Elliptical Orbit missions

Possible disposal strategies are presented as a result of a European Space Agency study

The trajectory design exploits the perturbed dynamics in the Earth environment and in the Sun–Earth and Moon system

Five ESA missions are selected: INTEGRAL, XMM-Newton, Herschel, Gaia and SOHO

A trade-off and general guidelines are drawn based on technical feasibility and sustainability

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