Planetary Protection impact on the trajectory design of the MSR-ERO mission

Amedeo Rocchi[(1)], José Mánuel Sanchez Perez [(2)], Lorenz Affentranger [(3)]

[(1)] *GMV at European Space Operations Centre*
Darmstadt, Germany
Email: Amedeo.Rocchi@ext.esa.int

[(2)] *European Space Operations Centre*
Darmstadt, Germany
Email: Jose.Manuel.Sanchez.Perez@esa.int

[(3)] *European Space Research and Technology Centre*
Noordwijk, the Netherlands
Email: Lorenz.Affentranger@esa.int

***Abstract* – This paper addresses planetary protection aspects driving the trajectory design of the Earth Return Orbiter (ERO) mission, the main contribution of ESA to the joint NASA-ESA Mars Sample Return (MSR) campaign. Stringent requirements apply to the return of Mars samples to avoid backward contamination of the Earth's biosphere by potentially hazardous biological material. These requirements have significant impact on spacecraft and trajectory design involving techniques such as trajectory biasing and disposal manoeuvres, counting on spacecraft reliability to ensure disposal at the end of mission and in case of aborted return. The paper describes the design of compliant trajectories and disposal manoeuvre strategies, as well as the approach to verification of the Backward Planetary Protection (BPP) requirements from a trajectory perspective.**

I. INTRODUCTION

Returning samples from Mars is the scientific goal of the MSR campaign being implemented jointly by NASA and ESA. At the time of writing the design of the MSR campaign [1] is composed of the following elements:

1. NASA's Perseverance rover launched in 2020 and already collecting the samples on the surface of Mars and storing them in sample tubes.
2. NASA's Sample Retrieval Lander (SRL), including ESA's Sample Transfer Arm (STA), that will store the sample tubes into the Orbiting Sample (OS).
3. NASA's Mars Ascent Vehicle (MAV) that will deliver the OS capsule to low Mars orbit.
4. ESA's ERO, carrying NASA's Capture, Containment, and Return System (CCRS) including the Earth Entry Vehicle (EEV), that will find and capture the OS in Mars orbit and return it to Earth.
5. A joint Sample Receiving Facility to receive the samples and perform scientific investigations.

According to the mission design as of the end of 2023, ERO is set for launch on an Ariane 64 launcher from Europe's Spaceport in Kourou, French Guiana, on October 2027 and return to Earth with the Mars samples by October 2033, on an overall 6-year mission. A backup launch in October 2028 would still be compatible with the same Earth arrival opportunity.

ERO is a hybrid Electric Propulsion (EP) – Chemical Propulsion (CP) spacecraft with a large 41-kW-class (at 1 AU) 144 m^2 solar array providing the power to operate the solar EP system at Mars. ERO is composed of two modules: the Orbit Insertion Module embarking the high thrust engines and bi-propellant for the Mars orbit insertion and subsequent apoares lowering manoeuvres, and the Return Module (RM) embarking the solar arrays, comms & avionics, EP system and bi-propellant thrusters for attitude control and orbit manoeuvres, as well as the CCRS and sensors suite for the OS capture. To the effect of BPP only the RM and the return trajectory to Earth are relevant.

After the orbiting samples have been launched by the MAV into low Mars orbit, ERO's primary goal is to capture them and return them to the Earth. In the current baseline architecture, the CCRS takes care of capturing, containing and transferring the OS to the EEV for delivery to the Earth atmosphere. ERO will then spiral out of Mars orbit using the EP system and continue using the EP system for about two thirds of the interplanetary Inbound Transfer Phase (ITP) to the Earth. A ballistic trajectory that intersects the Earth orbit is achieved 130 days before arrival on October 11th, 2033.

This paper will focus on methods and results to comply with the BPP requirements for the latest part of the interplanetary trajectory and for the final 30-day EEV Delivery Phase (EDP).

II. BPP REQUIREMENTS

International regulations impose strict limits on the probability of contaminating the Earth with material coming from other celestial bodies (see [2]), in an effort to guarantee Backward Planetary Protection (BPP). Such limits apply in the case of MSR-ERO to both the samples Earth Entry Vehicle (EEV) and the orbiter itself: while the former is bound for Earth re-entry, the latter is employing an Earth avoidance manoeuvre (EAM) to avoid an immediate impact with the Earth after EEV is released. The focus of the current work are the requirements applicable to the orbiter.

COSPAR guidelines and European Cooperation for Space Standardization (ECSS) standards are considered to define mission specific requirements which are then applied for the mission design and operation; the applicable requirement at mission level is the following: the probability of releasing a particle having a size greater than 0.01 μm into Earth biosphere shall be less than 10^{-6} for 100 years after departure from Mars. Applying a conservative approach, the requirement is translated into a maximum Earth impact probability of 10^{-6}; such quantity can be more easily estimated at mission analysis level and becomes the objective to be monitored. The requirement is checked for 100 years after MSR-ERO end-of-life; the 100 years horizon requires keeping into account both the short- and long-term Earth impact probability. Note that in principle also a Moon impact is to be avoided similarly to an Earth one, to guarantee unrestricted Earth-Moon travel; in practise, since the Moon is not approached in any mission phase for the considered architecture, the Moon impact probability is always significantly smaller than the Earth one.

The approach is therefore conservative as the following probabilities are assumed equal to 1:

- ERO probability of being contaminated after being in proximity with the orbiting samples (OS) in Mars Orbit.
- Probability of Mars material present on ERO to be biologically active.
- Probability of spreading biological material from Mars into Earth biosphere if contaminated ERO comes in contact with Earth's biosphere.

The BPP requirements imposed on the ERO mission are driving the spacecraft design, its operations and the trajectory. Concerning the spacecraft a high level of autonomy and failure detection, high reliability and increased redundancy are the major consequence of such strict mission critical BPP requirements; the major consequence is that ERO is designed to be two-faults tolerant (2FT). The operations need also to be robust to contingencies and keep into account all the possible scenarios in which an Earth impact is to be avoided or its probability minimised. The trajectory design shall keep into account the BPP requirements, as modifying the trajectory is often the most effective way to alter the probability of an Earth impact. Even if all these aspects are equally important to respect the imposed requirements, the focus of the current paper is the trajectory part and therefore the other aspects are only briefly described as necessary to give further context or to justify acceptance of non-compliance.

III. MISSION PHASES OF INTEREST

The ERO spacecraft is assumed to carry unsterilised Martian material when departing low Mars orbit; from that moment onwards the BPP requirements apply. However, it is not until ERO has departed Mars's sphere of influence and is well into its Inbound Transfer Phase (ITP), that the Earth impact probability start to differ from exactly zero; the ITP is therefore divided formally in two phases, ITP1 and ITP2, with ITP2 the phase during which BPP considerations drive the operations of the spacecraft. The criterion used to define the entrance in ITP2 is linked to the minimum Earth distance at the next closest approach: for a return infinite velocity at the Earth arrival between 3.5 and 4 km/s, a miss-distance of 2 million km is considered safe. It is verified with simulations that dispersed trajectories above such miss-distance present long-term impact probability always below the required 10^{-6}; below the threshold, the interaction with the Earth gravity field is not negligible and it can change the heliocentric orbit to be in a long resonance with the Earth. The crossing into ITP2 takes place 140 days before arrival, or around 1 week before the end SEP arc that brings ERO on track for the Earth return, with around 100 m/s still to be performed by the electric propulsion system.

At the end of the ITP, ERO enters the Earth Delivery Phase (EDP). During this phase the BPP requirements are extremely relevant as ERO has to first put itself on impact course with Earth by performing an Earth Targeting Manoeuvre (ETM) to deliver the Earth entry Vehicle (EEV) on a high-precision trajectory and then perform an Earth Avoidance Manoeuvre (EAM) to return to a skip trajectory at a safe closest approach altitude. The major events are highlighted in Fig. 1; the timeline includes a final correction manoeuvre (FCM) used to finely navigate the spacecraft to the correct delivery of the EEV.

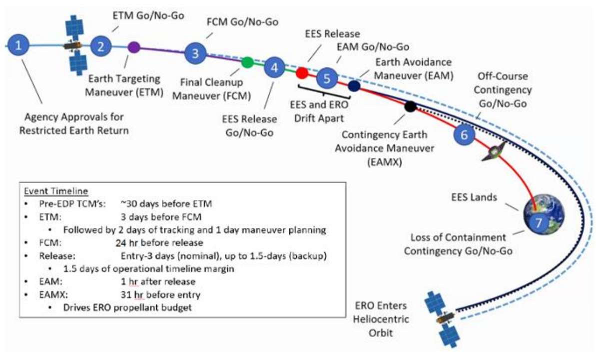


Fig. 1 Earth Delivery Phase scheme with events.

At the end of EDP ERO enters a heliocentric orbit, but the BPP requirements are still relevant: in this case no short-term impact with the Earth are possible but a safe disposal needs to be guaranteed.

IV. APPROACH

In the ideal scenario the trajectory of the ERO shall guarantee that for the whole mission, in case there is an unexpected spacecraft loss, the probability of impacting the Earth remains below the required threshold. This is clearly not possible from a pure trajectory point of view: in the considered architecture, for example, the release of the EEV has to occur while the spacecraft is on impact course with Earth and therefore the spacecraft and its operations have to be designed to guarantee BPP compatibility.

The objective of the trajectory design from a BPP perspective is therefore to reduce the Earth impact probability below the required limit whenever possible and reasonably minimise it otherwise; as reducing the impact probability often requires an additional deltaV/propellant allocation, the right balance is to be found as an iteration of the system design: spacecraft reliability and concept of operations are the other aspects that can be tuned to mitigate the risk and guarantee the BPP requirements are met overall. Focussing on the trajectory design, several aspects have to be considered:

- The relevant mission phase; different phases can require completely different approaches to efficiently reduce the impact probability.
- The 100-year timeframe; to reflect the dynamics that lead to an impact a short- and a long-term can be distinguished; the timeframe before the epoch of the Earth arrival is referred to as short-term, while the remaining is considered long-term; note that the short-term is not relevant after EAM is executed.

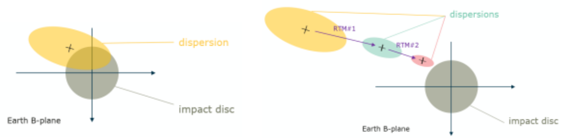
Independently of the aspect to be considered, the estimation of the impact probability is the crucial aspect; in order to have an accurate estimate, massive Monte Carlo propagation simulations are used; this approach is selected instead of more problem informed ones (like the one described in [3]) since it guarantees the accuracy of the results and can be directly compared with the requirements without the need of further approximations to obtain non-optimistic results. Moreover, the Monte Carlo method is agnostic to the problem behaviour and can cope, without loss of accuracy, with the impact probability transitioning to being chaotic (unpredictable with good accuracy with an analytical method). Of course, estimating the impact probability down to 10^{-6} level is challenging from a Monte Carlo perspective for the sheer number of samples required; these can in fact approach several millions to tens of millions per simulation, making the propagation and post-processing of the propagations a challenge. A massively parallel propagation tool called CUDAjectory [4] is however available and makes the simulation tractable; the in-house developed software is based on general purpose graphic processing unit (GPGPU) programming and was developed with planetary protection applications in mind. For reference, CUDAjectory offers an increase of performance from 50 to 100 times with respect to parallel CPU propagators (comparing a single professional-grade GPU vs a 48 cores server-class CPU).

In order to visualise dispersed trajectories, the Earth B-plane at arrival is used. This is convenient because it can give a situation snapshot: given a point on the B-plane, it can be seen whether it will impact the Earth directly (point within the impact radius), and which would be the post-encounter heliocentric period (and if any resonance with the Earth is close by, as a given period isoline is connected to each resonance). The impact plane shown in figures in the current paper is based on the EME2000 reference frame: the T-axis of the B-plane is the projection of Earth South pole direction at J2000 on the plane perpendicular to the arrival v-infinity at the Earth. For visualising the situation after the Earth flyby, especially to design the final disposal, it is also convenient to keep into account the heliocentric period; in this case it can be done directly, as the B-plane is not relevant any longer.

In the following sections the specific approach for the different phases is described, detailing it for both short- and long-term.

A. ITP2

During ITP2, after the SEP arc is concluded, the orbiter is nominally enroute for a closest approach with the Earth, with a minimum target altitude of 1600 km (before the ETM is performed); however, in case the spacecraft is lost during the transfer, a not so remote impact probability exists when accounting for uncertainties in the dynamics and in the initial state dispersion. In order to reduce the short-term impact probability below the required 10^{-6}, the interplanetary trajectory needs to be biased, reaching the arrival conditions in steps: the end of the solar electric propulsion (SEP) arc that brings ERO from Mars to an Earth bound return is anticipated and a series of re-targeting manoeuvres (RTMs) are introduced to progressively bring the spacecraft impact point on the arrival B-plane at the Earth closer. Going closer in steps allows to reduce the perigee radius only when the dispersion mapped to the arrival is small enough to avoid an impact probability higher than the limit. A scheme of why the RTMs are needed is given in Fig. 2.



a) b)

Fig. 2 Scheme of ITP2 a) without and b) with RTMs.

The long-term impact probability in case of spacecraft failure during ITP2 is instead tackled with de-target disposal manoeuvres (DDMs): during the mission, once the spacecraft detects that it is no longer 1FT on its safety critical chain, it waits for ground to take a decision within a limited timeframe, before autonomously performing a pre-loaded DDM to guarantee the impact probability in the long-term is below 10^{-6}. The design of the DDMs to be loaded is the core aspect to be tackled for ITP2 from a BPP perspective.

B. *EDP*

During EDP, between ETM and EAM execution, the short-term impact probability is 100%; therefore, in this situation it is up to the spacecraft autonomy and redundancy capability to ensure that, in case of spacecraft failure, an EAM is always performed to put the spacecraft back on a skip trajectory. The EAM can be pre-loaded and does depend on its execution time; a fine tuning of the EAM can be performed in advance to make sure the long-term return impact probability is minimised. If an EAM that guarantees long-term BPP requirements cannot be found, a further disposal manoeuvre (DM) would be performed after the Earth flyby, once the heliocentric orbit is accurately known and a final correction can be done.

V. ASSUMPTIONS

A. *Scenarios*

The reference scenario considered for the whole analysis is the baseline reported in [5], where it is referenced as ITP33 (ITP with Earth arrival in 2033); the biased trajectory with RTMs is shown in Fig. 3. Three RTMs are assumed during ITP2; operations planning is accounted for in the selection of their schedule, placing in a monthly schedule; the RTMs targeting is constrained with the pericentre altitude at the Earth arrival. The exact constraint is driven by the expected dispersion at Earth arrival, guaranteeing an impact probability below 10^{-6}; the remaining degree of freedoms are used to minimise the overall deltaV.

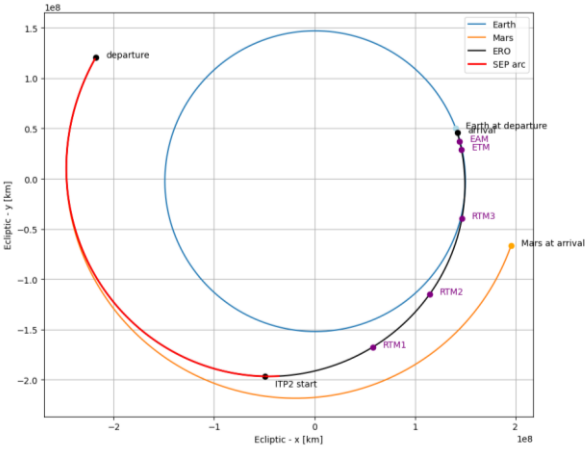


Fig. 3 ITP33 trajectory projected on the ecliptic.

The minimisation of Earth impact probability in the long-term during ITP2 is tackled with DDMs; as only a finite number of DDMs can be loaded on-board for execution in case of return abortion (because spacecraft is not 1FT anymore), the optimal DDM for relatively long spans are studied, up to covering the time between RTMs with a single DDM. It is assumed that 50 to 60 m/s are available to de-target: this budget is the remaining deltaV that in the nominal mission scenario will be used for ETM, EAM and final DM, but is otherwise available if the mission needs to be aborted beforehand (in principle more deltaV will be available before RTMs are executed, but what assumed is conservative).

Regarding EDP, multiple scenarios are considered for the EAM: nominal (occurring ~4 days after ETM), delayed by 1.5 days (using almost all the available timeline margin) and immediately executed after ETM (contingency with EEV delivery aborted). In case of need, a slot for a DM is placed 4 days after the Earth closest approach, to have enough time to do a full orbit determination of the heliocentric orbit achieved; a very precise estimation of the heliocentric orbital period is expected thanks to the observations that are collected while close to the Earth, as they strongly bound the orbit determination solution. An allocation of 15 m/s is considered available for the DM deltaV, derived from extensive analysis of different return scenarios (prograde/retrograde, different arrival years).

B. *Uncertainties considered*

While assessing the BPP requirements and in general while exploring the available trade-space to come up with a compliant trajectory and operation design, uncertainties coming from different source have to be factored in. The uncertainties considered are reported in Tab. 1; note that some are relevant only for a particular phase, such as the parasitic deltaV that is present when the spacecraft is being operated by on-board computer number 2 (OBC2) and in stand-by waiting for DDM execution.

Tab. 1 Uncertainties sources and models.

Uncertainty source	Modelled as
Solar radiation pressure	Reflectivity coefficient uniform distribution
Manoeuvre mis-performance	Mechanisation error in magnitude and direction
State dispersion	Gaussian, from simplified navigation analysis
Parasitic deltaV while on OBC2	Fixed (for DDM attitude acquisition) and time-dependent (for keeping prescribed attitude) gaussian deltaV

The state dispersion at a particular instance in time is computed differently for ITP2 and EDP. For ITP2 it is based on a simplified navigation analysis, for which perfect spacecraft state knowledge is assumed and limited timeframe is simulated. The post EAM state uncertainties in EDP are instead considered as driven by the mechanisation error of EAM itself, which is a good approximation of reality due to the size of the manoeuvre (above 10 m/s in all scenarios).

C. *Simulation setup*

In order to find a complaint trajectory and a strategy that fulfil the BPP requirements, several tools are used. For trajectory optimisation, the reference software for astrodynamics computations in ESOC Flight Dynamics Division, GODOT [6], is used. The already mentioned CUDAjectory is used for all Monte Carlo propagations, whether to estimate impact probabilities, scanning the search space or creating the data for visualisation. While working with different tools, the dynamics used are kept as coherent as possible; in the scenarios examined the following forces are considered acting on the ERO spacecraft:

- Gravity pull from Sun, Earth, Venus, Moon and Jupiter.
- Solar radiation pressure (SRP).

VI. COMPLIANT TRAJECTORY DEFINITION AND REQUIREMENTS VERIFICATION

A. *ITP2*

As described in section IV.A, RTMs are used to avoid the possibility of a direct impact with the Earth (at first perigee); three RTMs are used to progressively lower the pericentre altitude at the Earth: they are assumed to occur 90, 60 and 30 days before arrival. Their targeting and the resulting deltaV is given in Tab. 2.

Tab. 2 Example of RTMs details

Manoeuvre	Epoch [days to arrival]	Closest approach altitude: before and after [km]	DeltaV [m/s]
RTM1	90	160 000 – 40 000	11.8
RTM2	60	40 000 – 5 000	7.6
RTM3	30	5 000 – 1 600	2.2

The deltaV penalty caused by the introduction of the RTMs is less than 30 m/s and is the result of the reduction of the perigee being forced to be executed at a non-optimal location (closer to arrival, where the RTMs are located). The evolution of the impact point on the B-plane, starting from 2 hours before the end of the large SEP arc and ending at ETM, is provided in Fig. 4.

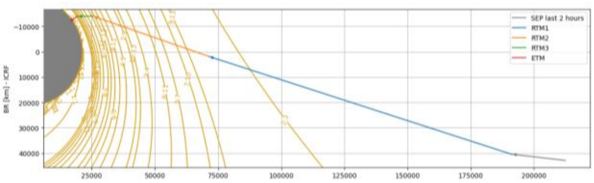


Fig. 4 Impact point evolution in time

With the gradual re-targeting it is made sure that even in case of spacecraft loss no direct impact with the Earth is possible; ERO is however built to be 2FT on its safety critical functions and would perform a safe disposal manoeuvre in case one of these functions would reach 0FT. Such disposal needs to respect BPP requirements and shall therefore minimise the impact probability in the long-term; three strategies have been tested to design DDMs:

i) De-target as much as possible (furthest from B-plane centre).
ii) De-target to area of B-plane in between resonances.
iii) Scan all possible de-target directions and select the most promising.

Strategy i) is the most-straightforward and does, in given circumstances, lower the impact probability before disposal; it is more effective the farther ERO is from the Earth, since the same deltaV can achieve a larger change on the B-plane; the de-target in this case reduces the effect of the Earth flyby, limiting the interaction with Earth also in the following decades. An example of strategy i) is represented in Fig. 5, where several components of the analysis are visualised on the B-plane; red quasi-ellipses represent the area that can be reached with 50 m/s depending on when the DDM is executed (3 are drawn, in case DDM is just before each RTM, with a final one just before ETM); a dashed line is used to connect the B-plane impact point before DDM is executed (red cross) and the selected de-targeting with highest distance (red dot) among the achievable ones; the B-plane is color-coded based on the post-heliocentric period reached and the resonance lines with the Earth are shown in white (resonance ratios are reported as $N_{s/c}$:N_{pl}, number of spacecraft revolutions and planet revolutions respectively); the Earth impact disc corresponds to the greyed area in the centre; finally dispersed shots are plotted in grey (no impact in 100 years) and magenta (impacting sometime within 100 years).

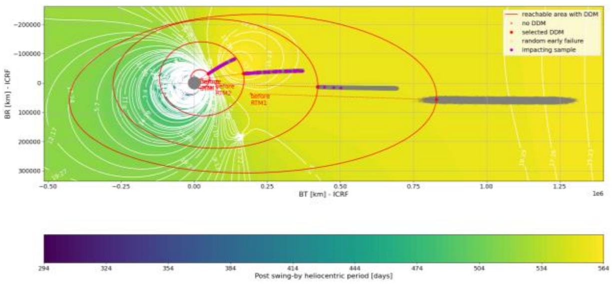


Fig. 5 B-plane situation for farthest de-targeting and random epoch of DDM execution

To obtain Fig. 5, four DDMs are designed a priori with 50 m/s to de-target as much as possible before each RTM and ETM. The dispersed samples are then initialised as follows: a random epoch of execution is selected between the end of the long SEP arc and the ETM; at the randomly selected epochs the closest next DDM is executed as is (three component vector fixed inertially). The resulting initial conditions are propagated forward accounting for an initial uncertainty of the state driven by the mechanisation error of the DDM and the SRP uncertainty. The overview of the situation available in Fig. 5 shows nicely that the strategy of de-targeting as much as possible does lead to low impact probability when the initial impact point is far from the Earth impact disc and more time is available; the closer to the Earth the initial conditions, the closer is also the dispersed cloud of points and the higher is the relative number of impacting shots in 100 years.

A couple of remarks are also needed to clarify some aspects related to Fig. 5:

- Even if the random epoch of failure is sampled continuously between end of SEP arc and ETM, the cloud of dispersed points (grey dots) is not continuous on the B-plane; the interruptions are actually caused by the RTMs, which effectively create a discontinuity in the spacecraft state leading to a separation in the dispersed shots.
- The parasitic deltaV, is not accounted for in the specific reported case.
- Having RTMs in the baseline trajectory also has the beneficial effect that de-targeting at earlier dates is more effective, as the initial impact point is already further away from the Earth.

An alternative to furthest de-targeting is strategy ii): de-targeting in specific B-plane areas in which resonances are further apart; this is especially relevant when de-targeting far away from the Earth impact disc becomes more difficult or out of reach given the limited deltaV available. An example of possible areas of interest is shown in Fig. 6, specifically for a DDM executed after RTM2 and before RTM3.

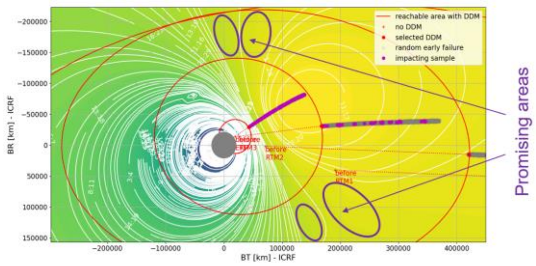


Fig. 6 Example of areas of interest for de-targeting before RTM3

Targeting the lowermost area, right of the 2:3 resonance line, can lower the impact probability substantially, if compared with the previous strategy i); from simulations it is found that the impact probability decreases from 1×10^{-3} to 3×10^{-5}.

Finally, it is also possible to use a brute force approach and probe all possible DDM directions; this, designated strategy iii) before, guarantees finding the DDM design that minimises the impact probability with the Earth in the long term. Such approach is very computationally expensive and is only made possible by the very efficient CUDAjectory software; a test case for disposal between RTM2 and RTM3 is shown in Fig. 7: all the DDMs directions that define the reachable area on the B-plane with a deltaV of 60 m/s are tested with 10 thousands shots each (granularity is such that more than 200 directions are probed in total); the epoch of DDM execution is again random between RTM2 and RTM3.

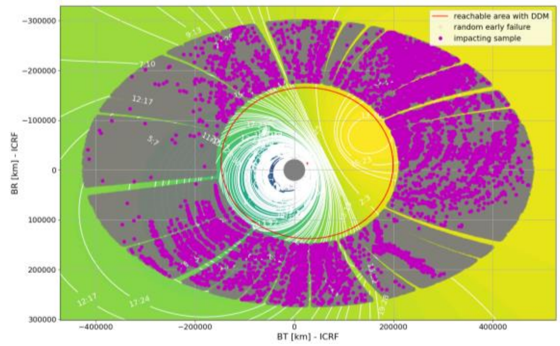


Fig. 7 Full scan of DDMs between RTM2 and RTM3

The simulation reported in Fig. 7 is very useful in visualising at a glance the situation: the impacts are indeed aligned with the resonance lines, confirming the post-arrival heliocentric period is the driver for the impact probability. Alternative promising regions to target to achieve a low impact probability are also evident: other than the ones close to the 2:3 resonance, there is a very broad region of the B-plane left of the Earth impact disc in which the probability of impact is low. The exact reasons for this are still to be studied, but are possibly linked to the Earth perturbation at next close encounters (beyond the sphere of influence and not captured in the simplified resonance mechanism) perturbing the heliocentric orbital period and disposing passively the spacecraft; a similar mechanism was investigated in [7] and further detailed in [8]. Note that targeting that area of the B-plane would require crossing the impact disc of the Earth during the manoeuvre execution, which could require further checks and analysis as the situation would be similar to EDP (direct collision course reached). In the simulation used to create Fig. 7 the parasitic deltaV is accounted for: both the time-independent and time-dependent parts are considered to initialise the state before DDM is executed. The parasitic deltaV leads to an inflation of the dispersed cloud on the B-plane (visible especially in the component perpendicular to the direction in which the grey samples are aligned); additionally, due to the fact that there is always a component in the DDM direction (related to the spacecraft turning to reach the DDM attitude) a bias of the dispersed cloud is present in a direction which is close to radial in the B-plane (this effect is equivalent to the DDM having a slightly bigger size and is the reason why the dispersed samples are detached from the red quasi-ellipse that defines the reachable area in the B-plane if the DDM is executed just before RTM3).

The different de-targeting strategies are tested for different DDM deltaV magnitudes and compared; the results are compiled in Tab. 3, with indicative ranges for the strategy i) and the best strategy overall.

Tab. 3 Impact probabilities for different strategies

DDM execution between	Estimated Earth impact probability		
	Unmitigated	Strategy i)	Best strategy
SEP arc – RTM1	1.2 10^{-4}	< 10^{-5}	< 10^{-5}
RTM1 – RTM2	3.6 10^{-5}	> 10^{-5} and < 10^{-4}	< 10^{-5}
RTM2 – RTM3	3.4 10^{-4}	> 10^{-4} and < 10^{-3}	> 10^{-5} and < 10^{-4}
RTM3 – ETM	8.7 10^{-4}	> 10^{-4} and < 10^{-3}	> 10^{-4} and < 10^{-3}

In Tab. 3 the unmitigated strategy with no DDM execution represents a worst case which can be improved already with the simple furthest de-targeting (strategy i)), which is very effective especially for earlier stages of ITP2; further improvements can be achieved optimising the DDM direction (strategy ii) or iii)). Reaching the required impact probability in line with BPP requirements is possible in several cases (results in Tab. 3 are limited in their accuracy by the number of Monte Carlo shots used); however, after RTM2 is executed, a fixed DDM that achieves a compliant disposal and works independently of the execution time (until the following deterministic manoeuvre) cannot be found: the impact probability can be only minimised.

The fact that no disposal strategy that works for the entire span of time between two deterministic manoeuvres works does not mean that BPP requirements at mission level cannot be respected. First of all, a finer search, with shorter validity for a single DDM can reduce the estimated impact probability at all times in case of spacecraft failure. Finally, the probability of having to execute a DDM during that particular phase of the mission is also not 100%: this chance needs to be factored in and can only be done in the larger context of the full BPP analysis. This is however outside the scope the current work: from a trajectory point of view, finding the reasonable disposal strategy that minimises the Earth impact probability is the ultimate goal.

B. EDP

Since during part of EDP ERO is on a collision course with Earth, the direct impact needs to be mitigated by mean of reliable execution of EAM, even in contingency circumstances. From this perspective the EAM design is rather straightforward, as it just needs to raise the perigee sufficiently to have a safe flyby. The focus of the current work is however the long-term impact probability: the design of EAM shall account for it. The long-term impact probability is even more relevant than in the ITP2 case, as it will be more likely that the mission needs to be disposed after EDP is completed, rather than in case of an undesired mission abortion before EDP.

In case of EAM execution with no mitigation for BPP purposes, the long-term impact probability with Earth is in the order of 10^{-3} to 10^{-4} for the next 100 years. The situation in an example case is provided in Fig. 8 (where an EAM is assumed as contingency immediately following ETM); the post-EAM B-plane dispersion is represented with an equivalent ellipse (5.26-sigma of a 2-dimension Gaussian distribution to capture 1-10^{-6} probability) and the shots that lead to an impact are color-coded based on the impact epoch.

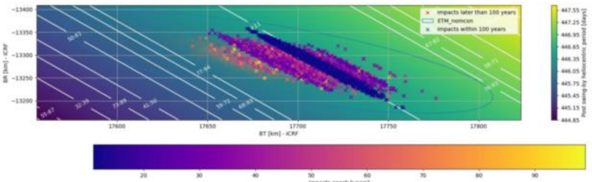


Fig. 8 B-plane dispersion for default EAM.

It can be seen in Fig. 8 that the default design of EAM (based on given perigee altitude target and minimising the deltaV) leads to an unlucky coincidence: in this case the central part of the dispersion is hitting a 9:11 resonance with the Earth and leads to many impacts around 11 years after the Earth close approach (blue crosses). This high impact probability can be mitigated with slightly different perigee altitude target resulting in a 0.5 m/s increase in EAM size: the area with large impact probability (keyhole) close to 9:11 resonance is avoided and the impact probability is reduced by two orders of magnitude (down to ~10^{-5}).

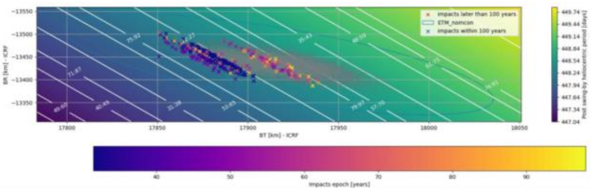


Fig. 9 B-plane dispersion for modified EAM.

In the majority of cases simulated, including different return trajectories, it is found that it is not possible to reduce the impact probability below the required 10^{-6} after EAM; the only solution is therefore to introduce a dedicated DM whenever necessary.

In order to design a DM, it is necessary to accurately know the heliocentric period after the Earth encounter so that the long-term trajectory can be assessed. This can be efficiently done with an orbit determination campaign that includes observations around the closest approach, especially including the ones after it (a gap is assumed in the 4 hours around perigee for the very high angular rates that can exceed the antenna limits); achievable accuracy is below 0.001 days, tested on several ERO return cases.

A major source of uncertainty remains however the long-term SRP one, which cannot be improved with orbit determination, but only partially by stabilizing the long-term attitude after disposal (spinning the satellite around the principal inertia axis and de-pointing the arrays from the Sun). The SRP uncertainty plays in all cases a major role in inflating the dispersion in the long-term and in hitting defined regions on the B-plane characterised by high impact probability (keyholes).

The effectiveness of changing the heliocentric period with a manoeuvre 4 days after the Earth closest approach depends on the reached heliocentric period. For the prograde arrival considered a change of 0.8 days in period can be achieved with the budgeted 15 m/s.

To guide the design of a DM, large Monte Carlo simulations are used: an inflated dispersion at Earth closest approach is used to scan the heliocentric orbital period reachable after the Earth encounter and statistics on the impact probability depending on the period achieved are drawn. Such maps, of which an example is shown in Fig. 10, present in a single plot several elements: the expected dispersion after the closest approach (in green, 4.9-sigma are used to capture the equivalent of $1\text{-}10^{-6}$ probability for 1-dimension Gaussian distribution), the probability of impact in the surrounding heliocentric periods (red line), the resonance lines (in dark yellow, often connected to peaks in impact probability) and the keyholes (shaded in red, they are derived from the impact probability and an applied margin on top to cover for the SRP uncertainty and orbit determination).

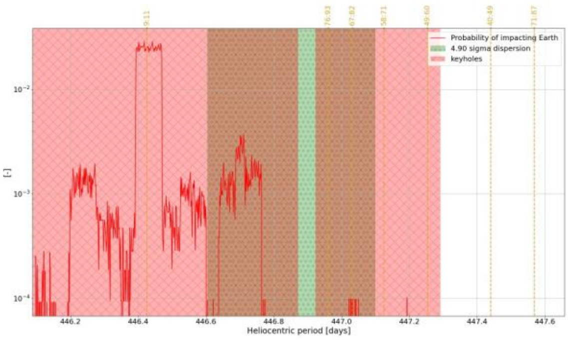


Fig. 10 Example of keyhole map for DM design

Once maps equivalent to the one in Fig. 10 are available, the design of the DM is straightforward: if after the orbit determination it is found that within the possible dispersion cloud (the green area) ERO ended-up in a keyhole (red area), a DM along or against the velocity needs to be executed to bring it to a safe area outside the keyhole; its size can be easily be computed based on the relation of the current state and target difference in orbital period, according to "(1)": ΔT is the orbital period difference required, T is the orbital period, V is the velocity, a is the semi-major axis, μ is the gravitational constant of the Sun, ΔV is the manoeuvre size.

$$\frac{\Delta T}{T} = 3\frac{Va}{\mu}\,\Delta V \tag{1}$$

Using a DM, it is always possible to fine tune the final disposal orbit and guarantee the BPP requirements are respected, even when accounting for execution errors. Finally, note that the biasing of EAM and the use of a DM are non-exclusive: they can be used together to first reduce the long-term impact probability and then ensure the BPP requirements are respected, with a small DM, only when required.

VII. Conclusion

The effects of the BPP requirements on the trajectory design of the ERO mission have been described. The affected mission phases have been tackled separately: during ITP2 a biasing of the trajectory with RTMs allowed to avoid the possibility of a direct Earth impact, while the use of DDMs is effective in lowering the impact probability in the long-term, provided they are designed adequately; during EDP, a tuning of the EAM can improve the long-term impact probability with a small deltaV penalty, while the final disposal can be made compliant in all mission scenarios with a dedicated DM after a final orbit determination is performed after the Earth closest approach. The full verification of planetary protection requirements at mission level remains, however, outside the scope of the current paper as it needs to incorporate aspects beyond mission analysis.

VIII. References

[1] T. Martin-Mur, J. M. Sánchez Pérez, R. Wilson, Z. Olikara, M. Jesick, R. Patel and D. Spencer, "The NASA/ESA Mars Sample Return Campaign," in *Internation Symposium on Space Flight Dynamics*, Darmstadt, 2024.

[2] COSPAR Panel on Planetary Protection, *COSPAR Policy on Planetary Protection*, 2020.

[3] M. S. Wallace, "A Massively Parallel Bayesian Approach to Planetary Protection Trajectory Analysis and Design," in *AAS/AIAA Astrodynamics Specialist Conference*, Vail, 2015.

[4] "CUDAjectory documentation," [Online]. Available: https://midas.io.esa.int/cudajectory_mirror/.

[5] J. M. Sánchez Pérez, A. Rocchi, I. Acedo, M. Van den Broeck, O. Ramírez Torralba and S. Proietti, "ERO Consolidated Report on Mission Analysis," Darmstadt, 2023.

[6] "GODOTPY documentation," [Online]. Available: https://godot.io.esa.int/godotpy/.

[7] A. Boutonnet and A. Rocchi, "Taxonomy of Earth Impacting Trajectories: the Case of JUICE Launcher Upper Stage," in *AIAA Science and Technology Forum and Exposition*, San Diego, 2022.

[8] A. Masat, "Dynamical and computational perspectives on interplanetary orbital uncertainty propagation," Politecnico di Milano, Milano, 2023.