Re-entry strategies to comply with Space Debris Mitigation guidelines

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1 INTRODUCTION

The rising concern of space debris has driven guidelines to limit and reduce their creation in the future. One of the key requirements lies in the removal of space objects from protected regions within 25 years following the end of their operational life (EoL). Orbits below 2000km altitude are of particular concern, as they remain a favoured region for many types of space missions, it is getting increasingly easy to access them for a growing number of space-fairing nations, and they will soon welcome several mega-constellations.

Complying with the Space Debris Mitigation (SDM) guidelines requires changes in the design of the satellite in many aspects, and noticeably in the propulsion system. Indeed, the propulsion system used to de-orbit the spacecraft will drive the type of re-entry achievable: controlled or un-controlled. The choice between these two options relies on the casualty risk: un-controlled re-entry is allowed only with a risk below 10^{-4} and controlled re-entry is usually the preferred option when the risk lies above this threshold.

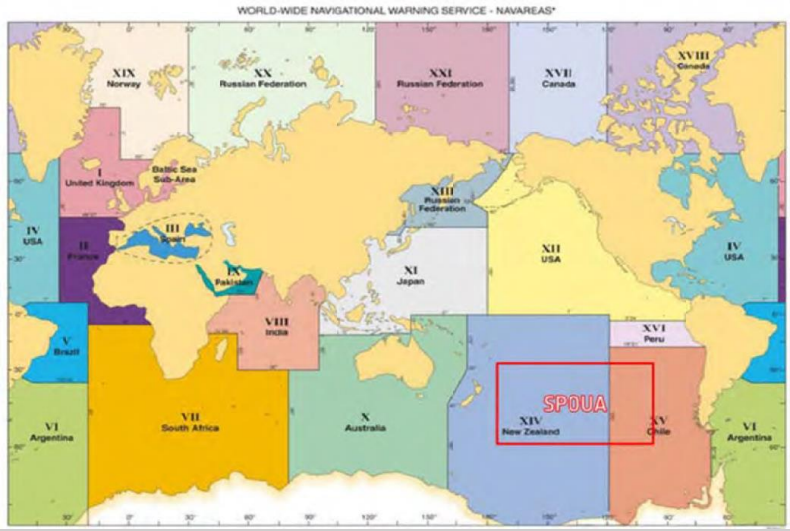


Fig. 1. Example of targeted regions when performing a controlled re-entry, the SPOUA (South Pacific Ocean Uninhabited Area) ensures an extremely low casualty risk for space debris

Performing controlled re-entry has many advantages but remains challenging. Because of the complex and changing interactions between the atmosphere and the debris, the longer the debris stays in the atmosphere, the more difficult it is to predict accurately the fallout area on-ground. Therefore, in order to perform controlled re-entry, the debris has to enter the atmosphere with a steep angle, making the predictions for the fallout area more accurate.

A controlled re-entry is feasible with propulsion systems having powerful enough thrust, meaning that electrical propulsion systems are not sufficient and would require an additional extra system. Different combinations of systems have been studied, considering classical options – monopropellant in blowdown mode or actively pressurized, Hall Effect Thrusters and Arcjets – to evaluate the impact on the overall system from a mass and cost point of view, while considering other constraining factors, such as acceleration, temperature and pressure changing inside the tanks, miscellaneous thruster constraints, etc.

This study aimed to provide ESA fast and simplified guidelines when it comes to choosing the most relevant combination of propulsion systems for a given mission and payload, as well as quickly evaluating the impact of these options on the overall mission.

2 STUDY CASE

2.1 Assumptions

Orbital characteristics

LEOs are the most populated orbits and the most suitable for controlled re-entry at EoL. Fig. 2. shows the spatial density in LEO, showing a particularly crowded region around 800km altitude. This will be our study case: a satellite in Sun-Synchronous Orbit (SSO) at 800km.

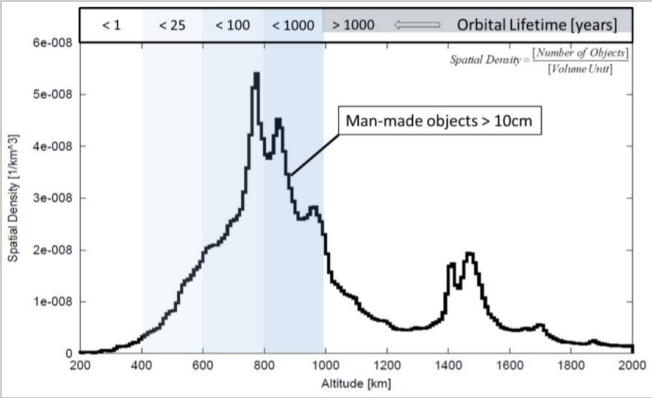


Fig. 2. Spatial density on man-made objects bigger than 10cm, and orbital lifetime

Propulsion systems

Controlled re-entry requires high thrust engines for the last burn (the one before the satellite enters the atmosphere). A classical approach is to use monopropellant hydrazine-based thrusters. However, they are not particularly efficient and a controlled re-entry would, consequently, require a significant amount of propellant, increasing mass and cost for the entire mission. A way to mitigate this impact is by having on board two propulsion systems (summary presented Fig. 3.):

- an efficient one (likely electric-based) for the potential orbit raising, nominal mission and orbit lowering at EoL
- a high-thrust engine (likely monopropellant) for the final burn

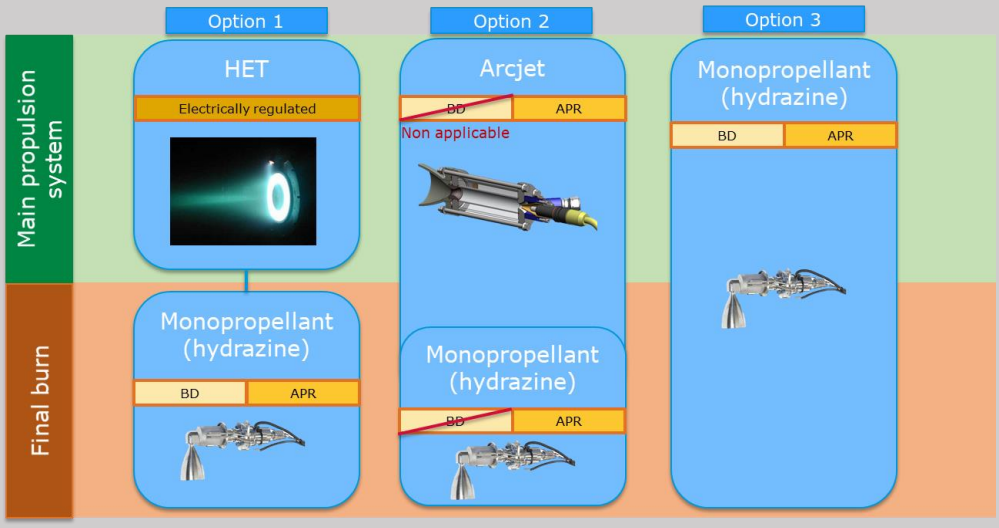


Fig. 3. Simplified summary of propulsion systems architecture

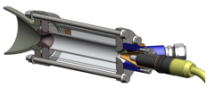
For the purpose of this study, only three different propulsion systems have been chosen, HET (Hall Effect Thruster), Arcjet and monopropellant (hydrazine-based), the characteristics of which are summarized in Table 1. These technologies have been selected for the following reasons:

- *Monopropellant (hydrazine-based)*: classical chemical propulsion system with high thrust necessary for final burn during controlled re-entry
- *HET (Xenon-based)*: highly efficient electric propulsion system
- *Arcjet (hydrazine-based)*: propulsion system showing a good compromise between efficiency and thrust level. Being based on hydrazine, an hybrid version is possible with common parts being used for both Arcjet and monopropellant thrusters.

Concerning the pressurization systems:

- HET has its own system, electrically regulated
- Arcjet can only be pressurized with active pressure regulation (APR) system
- Monopropellant thruster can use APR or BD (blowdown)

Table 1. HET, Arcjet and monopropellant propulsion systems with their respective characteristics

	HET	Arcjet	Monopropellant		
Model	SAFRAN: PPS-1350-G	Lockheed Martin: Aerojet MR-510	ArianeGroup		
			1N	20N	400N
Propellant	**Xenon**	**Hydrazine**	**Hydrazine**		
Pressurization	**Electrically regulated**	**APR****	**BD*/APR****		
Thrust	≤ 0.090 N	≤ 0.250 N	[0.320; 1.1] N	[7.9; 24.6] N	[120; 420] N
I_{sp}	**1660 s**	**600 s**	**[200; 223] s**	**[222; 230] s**	**[212; 220] s**
					

Scenarios

Two scenarios have been investigated, using Vega-C or Soyuz-ST to inject the satellite at 300km or 800km altitude. Should it be at 300km, an orbit raising would then be necessary with the satellite's main propulsion system up to 800km ($\Delta v = 274$ m/s). Mass-wise, it could be the most relevant choice in certain cases, as studied later in this study. Payload masses available in these scenarios are presented Table 2.

Table 2. Launchers' capacity with respect to altitude injection

Altitude	Vega C	Soyuz ST
800 km	2080 kg	4430 kg
300 km	2550 kg	4710 kg

Then, the main propulsion system is used during operational lifetime ($\Delta v \approx 100$ m/s) and perigee lowering at EoL down to 430 km ($\Delta v = 99$ m/s). Finally, the high thrust propulsion system, which could be the main propulsion system or a dedicated one, performs a final burn ($\Delta v = 124$ m/s).

On a side note, the altitude raising and lowering have been chosen to limit gravity and drag losses, but they can both be subject to discussion.

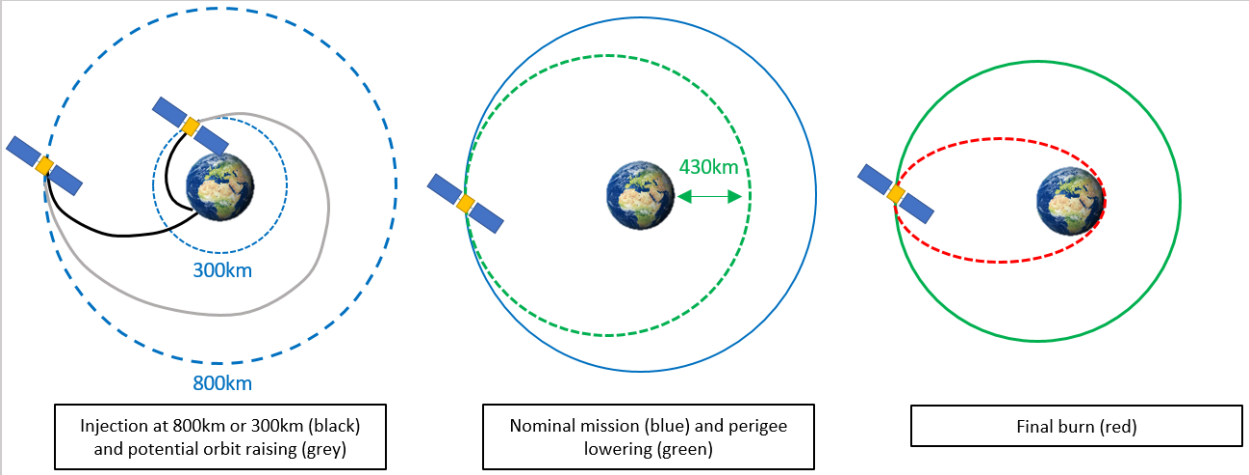


Fig. 4. Studied satellite mission's phases

2.2 Constraints

Filling ratio

The filling ratio α might not change much for APR systems, trying to optimize the amount of propellant usable, while considering some losses - few percentages of left-overs remaining in tank at EoL and small part of the tank allocated for pressurant gas at BoL (beginning of life). In case of BD mode similar left-over values are expected, however at BoL, α is more restricted. Indeed, temperature and pressure will drop over propellant consumption, and depending on α and the thermodynamic transformation, they might end up too low to maintain hydrazine in liquid form or to be within the thrusters working range. Consequently, α is lower when using BD mode over APR systems.

Overall, the differences are:

- *Option 1*: the monopropellant system is solely used during a single continuous final burn. Comparison with high fidelity calculations showed that the resulting transformation can be approximated as isentropic adiabatic when using a BD system. Adding constraints on pressure and temperature resulted in a poor maximum acceptable filling ratio of 34%.
- *Option 2*: Arcjet requires a regulated pressure feeding the thruster, meaning that an APR is necessary in any case.
- *Option 3*: with an APR system, as explained, no specific constraint is applied to α. In BD mode, the phases of orbit raising, nominal mission and perigee lowering can be considered to be done in several slow burns, allowing to keep a constant high temperature. The last phase of final burn is isentropic adiabatic, as mentioned previously, however it occurs with a tank already partially emptied. Overall, those two different consecutive transformations constrain an initial filling ratio of 74% in our scenarios.

Thrust level

One can adjust thrust level by choosing different types and number of monopropellant thrusters – typically with 1N, 20N and 400N models. The configuration selected is constrained by the acceleration and is twofold:

- The acceleration should be low enough, not to break any structure (threshold typically set to 0.04g).
- However, it should be sufficient to reduce the gravity losses and to allow entering the atmosphere with a steep enough trajectory and perform a controlled re-entry.

Combining those two constraints gives the feasibility range, within which one can perform a controlled re-entry. Fig. 5. hereunder is one example of feasibility range achievable in a particular thrusters configuration.

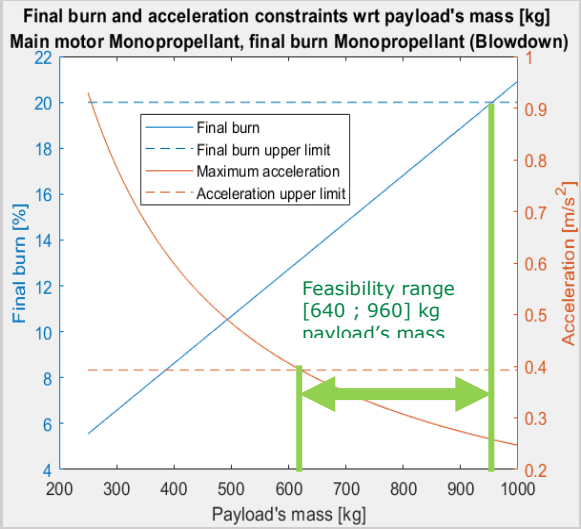


Fig. 5. Acceleration level and final burn duration with respect to satellite's mass, in case of having Option 3 as propulsion system architecture (with 12 x 20N thrusters)

Other constraints

Other restrictions apply when choosing the propulsion system configuration and should be taken into account, such as the number of cycles, the maximum single and total burn duration, the maximum propellant throughput acceptable, etc. They have been considered later on in our case study.

3 RESULTS

3.1 Feasibility range

As mentioned earlier, the acceleration undergone by the satellite is constrained and ruled by the type and number of thrusters chosen for the final burn. One example feasibility study has already been presented Fig. 5. If considering solely one type of thruster per satellite (1N, 20N or 400N), from 1 to 12 units, one can compile all the scenarios considered in this study into one single graph. Fig. 6. is the result of this compilation, helping the reader to easily and quickly assess the type and number of thrusters possible for a given scenario.

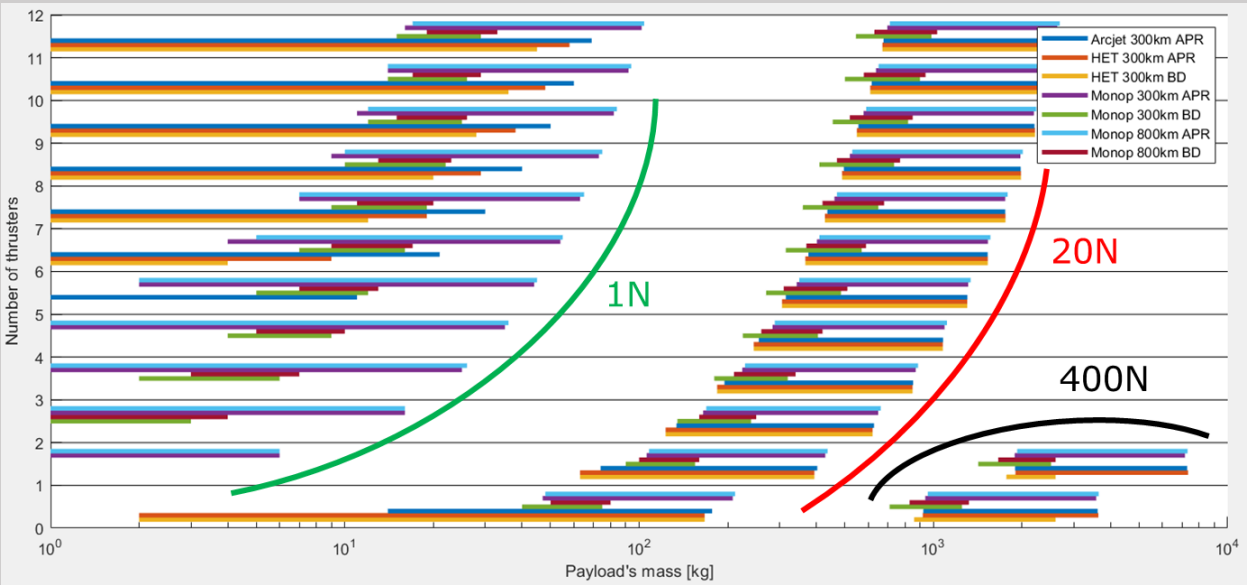


Fig. 6. Feasibility range considering 1N, 20N and 400N thruster configurations for spacecraft up to 10 000 kg

3.2 Mass budget

In this section, an estimation of the overall mass of the propulsion system is evaluated, considering:

- Propulsion system selected (Option 1, 2 or 3)
- Injection altitude (300km or 800km)
- Pressurisation system (APR or BD)
- Number and type of monopropellant thrusters (from 1 to 12 units of 1N, 20N and 400N thrusters)
- Payload's mass

In order to simplify the study and decouple potential dependencies between subsystem, Fig. 7. presents the assumed simplified decomposition of the spacecraft:

- "S/C – P/S" is meant as the mass of satellite without the propulsion system(s), later called m_{PL}
- "Main P/S" is the main propulsion system (for orbit raising, nominal operations and perigee lowering), later called $m_{main\ prop.\ syst.}$
- "FB P/S" is the propulsion system for the final burn, later called $m_{final\ burn}$

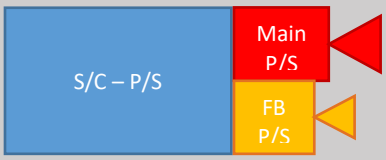


Fig. 7. Decomposition of the spacecraft masses

One a side note, it can be mentioned that "Main P/S" and "FB P/S" can be the same system (when having a unique monopropellant P/S) or at least sharing some components (Arcjet and monopropellant P/S can share common tanks, pipes, etc.).

In Fig.8. is presented one example of results obtained with a configuration of 2 × 400N monopropellant thrusters, for the different scenarios presented above.

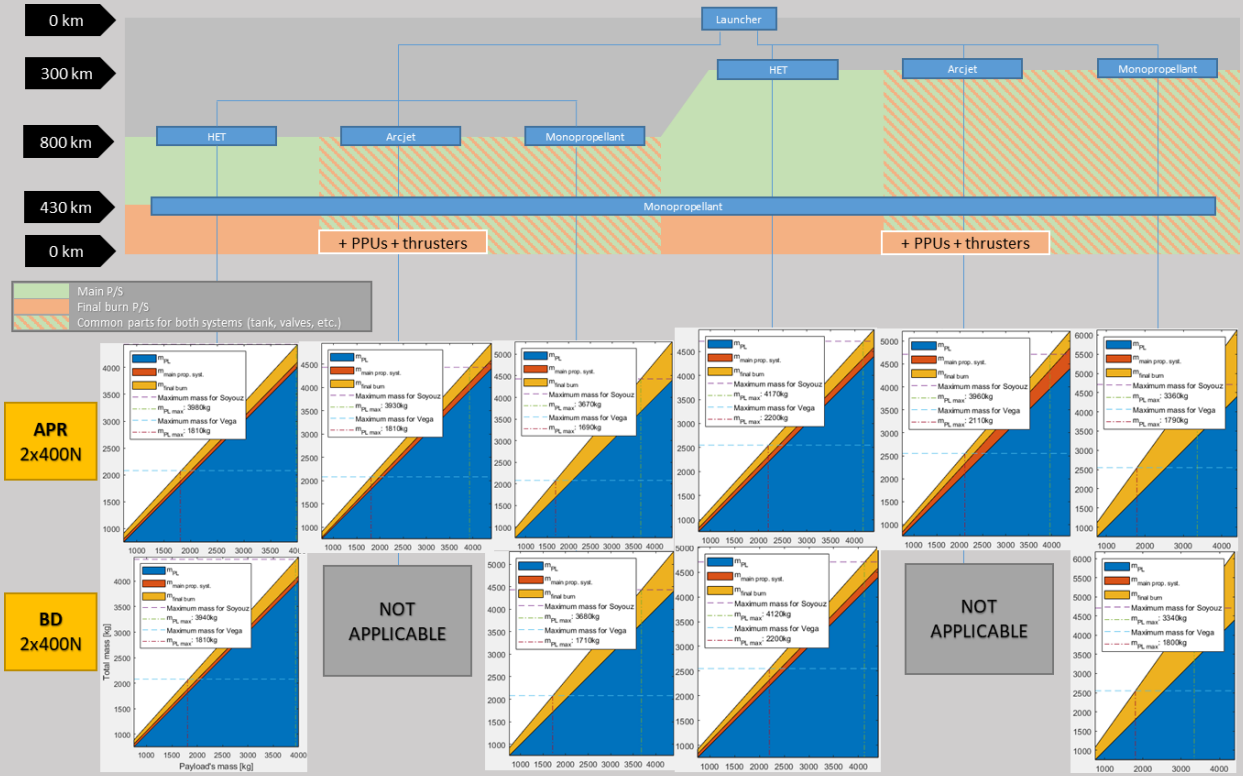


Fig. 7. Mass budget obtained with a 2 × 400N configuration monopropellant thrusters

In these graphs, one can identify the additional mass required to perform a typical SSO mission while a controlled re-entry at EoL is foreseen. The payload capacity of Soyouz-ST and Vega-C is also included.

Injection altitude and P/S configuration

Mass-wise, general conclusions can be drawn such as:

- In general, option 1 being injected at 300km has always shown to be the best configuration in order to minimize the P/S mass
- When considering options 1 and 2, it is always better mass-wise to inject the satellite at 300km and perform an altitude raising with the satellite main propulsion system
- With option 3, injecting at 300km is preferable for small satellites and at 800km for heavy satellites, though the difference remains not significant

Pressurization system

APR is in general more suitable for heavy spacecraft, as the extra mass coming from the pressurisation system is negligible compared to the overall mass and because it is, at one point, compensated by the higher efficiency of the engines. This is particularly noticeable for Option 3 at 300km and 800km, while for the other scenarios it is also highly depending on the other parameters (P/S configuration, injection altitude, type and number of thrusters, etc.).

Other considerations

The following aspects not dealt with in detail within this study must be considered when choosing the propulsion system:

- Option 1 is the best solution mass-wise, however, HET has a low thrust level, implying more manoeuvres, longer exposure to space debris and micrometeoroids, adding operation complexity, etc.
- Option 1 presents also two independent propulsion systems, adding complexity to the overall system
- Option 2 is an alternative proposing a compromise between efficiency (high I_{sp}) and thrust, while having a less complex hybrid P/S

3.3 Costs

In this section, two aspects of the costs are considered: those related to manufacturing (both first assembly and recurrent assembly) and the ones related to launch (considering Soyouz-ST and Vega-C launchers). Indeed, efficient engines may be more expensive, but their mass gain can be high enough to compensate the launch cost.

As a result, the overall cost is highly dependent on the study case. As an example, one configuration is chosen in this study to illustrate the costs differences: a 1500kg spacecraft, with 2 redundant chains of 4 × 20N monopropellant thruster, injected at 300km, having either Option 1 (HET and monopropellant with APR system, the most efficient mass-wise) or Option 3 (single monopropellant system in blowdown mode, the simplest and thus cheapest version). The differences in masses and costs are summarised Table 3.

Table 3. Cost and mass differences of a 1500kg spacecraft, injected at 300km, with 4 × 20N monopropellant thrusters, considering Option 3 in BD mode as baseline

	Monopropellant pressurisation system	Overall spacecraft mass	Spacecraft cost		Spacecraft cost (including launcher)	
			First assembly	Recurrent assembly	First assembly	Recurrent assembly
Option 3	BD	Baseline				
	APR	+1%	+74%	+49%	+12%	+6%
Option 2	APR	+11%	+219%	+183%	+23%	+7%
Option 1	BD	+14%	+283%	+224%	+29%	+8%
	APR	+15%	+350%	+260%	+38%	+10%

Despite the significant mass gain, having the simplest Option 3 solution seems overall more advantageous cost-wise. This result is highly dependent on the study case and should be reiterated for each scenario considered.

Moreover, one should keep in mind other aspects which are not taken into account in this study, but could have an impact on the cost. For instance, having a lighter spacecraft allows to embed a heavier – potential – second payload, which can change the repartition of the cost of the overall launch. It can also imply using smaller, cheaper, more accessible, launchers. On the negative side, low thrust engines may require more expensive operation costs due to longer manoeuvers.

4 CONCLUSION

Guaranteeing sustainable and safe space activities – both in orbit and on ground – requires the compliance by space actors of common well-proven guidelines. Among those recommendations, the deorbiting within 25 years after EoL with a limited casualty risk on ground is one critical part. Despite its significant impact for the satellite, a controlled re-entry might be necessary for a casualty risk reaching the 10^{-4} threshold. Its impacts mass and cost-wise have been presented in this study with alternative propulsion system options. Nevertheless, one should keep in mind the other aspects which were not presented in this study but may have significant importance, such as the additional complexity (on the platform and operations), the reliability, accommodation constraints, etc. Finally, other means are possible and should be studied in order to reduce the casualty risk, thus avoiding the need to perform a controlled re-entry, typically by improving the spacecraft demisability (finding alternative materials with lower melting points, exposing critical components, allowing early break-up of the satellite, etc.).

5 REFERENCES

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