THE IMPACT OF SPACE DEBRIS MITIGATION REQUIREMENTS ON MISSION DESIGN CHOICES: AN OVERVIEW FROM ESA CLEAN SPACE

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ABSTRACT

Space debris mitigation has been identified as a vital step to reduce and possibly stop the space debris population rapid growth that has been taking place in the last years. To optimise and enhance current technologies and to unfold innovative solutions related to the End-Of-Life (EOL) of satellites, the ESA Clean Space Office undertook and supported various studies within the CleanSat programme, which involves a collaboration with the European Large System Integrators (LSIs). A brief definition and overview of the main activities related to EOL that are currently being carried out within the ESA Clean Space Office will be given: Passivation, Controlled and Uncontrolled Re-entry, Design for Demise, Deorbit Manoeuvres and Deorbit Passive Devices. A compilation of lessons learnt and concurred proposed solutions deriving from years of collaborative systems and technology studies will be presented.

Keywords: Clean Space; space debris; mitigation; CleanSat; design for demise; passivation; re-entry strategies; casualty risk; disposal, end of life.

1. INTRODUCTION

The amount of space debris has been growing steadily since the 1960s, with some instantaneous rises due to collisions or explosions, as reported by [28]. To invert this trend, two strategies need to be concurrently implemented: space debris mitigation and space debris remediation. The former acts on the source of debris, decreasing the probability that functional spacecraft will become derelict after or during the operational lifetime. Instead, the latter aims at alleviating the collision risk deriving from debris that are already in orbit.

The need to apply the aforementioned countermeasures to the growth of the debris population required first of all the definition of commonly accepted and agreed upon Space Debris Mitigation (SDM) requirements. This is why the ISO standard 24113 [37] on Space Debris Mitigation (SDM) Requirements was first published. This document was last updated in 2019. The importance of regulation supporting SDM and the success of post-mission disposal has been analysed in various papers and publications, such as [36] and [41].

Clearly, the compliance with the SDM requirements has multiple significant implications for the architecture of space missions, because it can directly impact the spacecraft design, mission cost and development time. Indeed, those requirements highlighted new knowledge gaps and technology needs that ESA and other institutions have been trying to identify and solve. That is why in 2013 ESA decided to establish CleanSat, which is a collaborative initiative that became specialised in the optimisation and enhancement of current technologies and in the unfolding of innovative solutions related to a safe End-Of-Life (EOL) of satellites. During the last decade CleanSat laid the foundation for a strong cooperation between ESA and industry, particularly with the European Large System Integrators (LSI), such as Airbus Defence and Space, Thales Alenia Space and in earlier phases also OHB. The LSIs not only carry out studies on their own, but also offer their support and feedback to other activities. As they might be the end users of newly developed technologies, they guide and encourage design choices of smaller institutions depending on internally identified integration and innovation needs.

Therefore, CleanSat has become a cornerstone in this domain and has lead to the development of invaluable knowledge about the logical processes that relate high-level SDM requirements to design choices and new technologies development for ESA missions.

The following provides a definition and overview of the main activities related to EOL in Low Earth Orbit (LEO) that are currently being carried out within the ESA Clean Space Office will be given: Uncontrolled Re-entry, Passivation, Deorbit Manoeuvres and Deorbit Passive Devices, Design for Demise and Controlled Re-entry. A compilation of lessons learnt and proposed solutions de-

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rived from years of systems and technology studies will be presented. Subsequently, an overview of system and subsystem level impact of several of the aforementioned requirements will be given, alongside with the main drivers of research on new technologies.

2. SPACE DEBRIS MITIGATION REQUIREMENTS

Space debris pose a significantly higher threat in the zones where they have been accumulating the most over the years: the Low Earth Orbit and the Geostationary Earth Orbit (GEO). This paper will be focused on technologies that apply mostly to LEO, which includes all altitudes below 2000 km.

The SDM requirements constitute the core of the [37] and have been adopted by ESA with [25] and [27]. The main ones that are related to the technologies developed within the CleanSat initiative are summarised in the following:

- Space systems operating in the LEO protected region shall be disposed of by re-entry into the Earth's atmosphere within 25 years after the end of their operational phase, if they are capable of collision avoidance manoeuvres, or after their orbit injection epoch, if they have no capability to perform collision avoidance manoeuvres.
- A spacecraft or launch vehicle orbital stage, for which a controlled re-entry has not been planned, shall be passivated (i.e. permanently depleting, irreversibly deactivating, or making safe all on board sources of stored energy capable of causing an accidental break up) in a safe and controlled manner before the end of life.
- For space systems that are disposed of by re-entry, the prime contractor shall perform an analysis to determine the characteristics of fragments surviving to ground impact, and assess the total casualty risk to the population on ground assuming an uncontrolled re-entry. To understand how the casualty risk is usually evaluated, it is possible to refer to [26], [35] and [29].
- In case the total casualty risk is larger than 10^{-4} , uncontrolled re-entry is not allowed. Instead, a controlled re-entry must be performed such that the impact footprint can be ensured over an uninhabited area, with sufficient clearance of landmasses and traffic routes.
- The probability of successful disposal of a spacecraft or launch vehicle orbital stage shall be at least 0.9 through to the end of life.

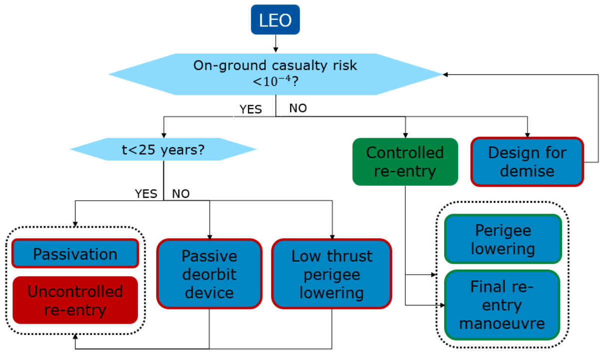

Figure 1. EOL operations decision-making process.

3. TECHNOLOGY AREAS

When planning the EOL manoeuvres and procedures of an ESA spacecraft, the operators and engineers need to go through the decision-making process shown in Fig. 1. First of all, it is necessary to assess the on-ground casualty risk deriving from the re-entry of the spacecraft. If this value is lower than the threshold, 10^{-4}, it is possible to perform a so-called uncontrolled re-entry, but there is another condition that needs to be checked. If the time needed to deorbit after the EOL is less than 25 years, passivation of the propulsion and power system can be performed and the spacecraft can be left to passively decay and finally be disposed through an uncontrolled re-entry. If this is not the case, the decay can be accelerated with a deorbit device or the altitude can be lowered with a propulsive manoeuvre. After this is achieved, passivation procedures shall be implemented. Lastly, the spacecraft will naturally decay with an uncontrolled re-entry. On the other hand, if the casualty risk is higher than the threshold, two options are available. The first is to implement the so-called Design for Demise (D4D), which means to design the spacecraft and its equipment to maximise its ablation during the re-entry phase. The objective of these design modifications is to reduce the casualty risk to be able to perform an uncontrolled re-entry. If this is not possible, according to the requirements, a controlled re-entry is mandatory. Often, to reduce the thrust level required to perform such a manoeuvre in a short time and to reduce the propellant required for the final burn, the perigee is first lowered with a separated manoeuvre.

The structure of the following sections will follow the decision-making process. As shown in Fig. 1, first all the technologies associated to Uncontrolled Re-entry (red outline) will be explained. Afterwards, all the ones related to Controlled Re-entry (green outline) will be analysed and detailed.

3.1. Uncontrolled Re-entry

Any spacecraft that is orbiting the Earth at an altitude low enough to experience the effects of the atmospheric

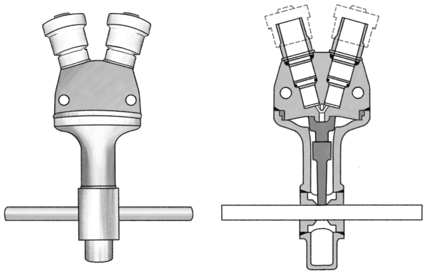

Figure 2. Illustration of a pyrovalve.

drag will eventually naturally decay into the Earth's atmosphere. Of course, this re-entry strategy is always the preferred one, because it has lower system impacts than the alternative, and it is therefore the easiest and the cheapest.

3.1.1. Passivation

Passivating a spacecraft after its end of operation is becoming more and more a common practice among spacecraft operators, both for LEO and GEO platforms. Not performing passivation is indeed, as reported by [28], one of the main sources of new debris due to on-board stored energy, and has already generated many in-orbit explosions in the past. Usually, passivation operations follow the same order: first, propulsion passivation is achieved, then other sources of on-board stored energy (*e.g.* batteries and reaction wheels) are removed or made safe, while at last the final power passivation is performed.

In general, the main challenge for any kind of passivation is the added risk of erroneous activation that results from the passivation device itself. Indeed, if passivation was performed during the operational lifetime of the spacecraft, it could result in a complete loss of the mission. The most used mitigation strategy for this risk is a double safety barrier for the activation of any passivation device, and, when possible, reversibility.

A challenge that has been identified by [47] in this field is the difficulty of performing a thermal analysis of the spacecraft after the EOL. Indeed, temperature is one of the key parameters that may trigger an in-orbit break-up, which means that such an analysis could be greatly beneficial for the development of passivation solutions and for the unfolding and investigation of new technologies. More in general, all environmental aspects (e.g. micrometeoroids, surfaces optical properties, aging) are extremely difficult to predict and simulate after EOL and there is no agreed approach to assess their validity.

Propulsion system Performing propulsion passivation corresponds to depleting all the sources of energy that are left in this system after EOL. In particular, propellant residuals have been identified to be the main risk. Indeed, the residuals could self-ignite, vaporise or dissociate if left in the tank, increasing the internal pressure and eventually causing an explosion. Two events were identified to be the typical origin of these phenomena (see [47]): over-temperature, which for example can lead to a thermal runaway for hydrazine systems, or a hyper-velocity impact of a debris or micrometeoroid into the propellant tanks.

This is mostly a risk for hypergolic propellants, like hydrazine and other monopropellants, which are the most used in LEO. Among the few exceptions, there are small satellites, that may use cold gas, and recent constellations that are increasingly choosing Electric Propulsion (EP). Bipropellant systems are rarely used in LEO.

In general, passivation of the propellant left inside the system can be achieved through two main means: a dedicated passivation valve (*e.g.* micro-perforator, pyrovalve, Shape Memory Alloy (SMA) valve) or depletion burns. Two examples of valves can be visualised in Fig. 2 and 3.

In principle, passivating cold gas propulsion systems is not a challenging task, as it only requires to open the thrusters in a controlled way to let out the pressurized gas. EP systems can also be depleted fully through the thrusters. However, due to the small mass flow of the thrusters embedded in the system, this operation can be long, and thus costly. This is why the usage of a dedicated passivation valve can be beneficial. Lastly, common monopropulsion systems can be passivated with both the aforementioned methods. Nevertheless, special care is needed due to the usage of high energetic propellants and the design of the thrusters.

Considering a commonly used hydrazine propulsion system (an example of the schematic is available at [56]), it includes a diaphragm tank, the feeding lines and the thruster, sometimes separated in different branches. Since the tank normally has a diaphragm implemented to separate the pressurant gas from the propellant, the depletion of all stored energy, including as well the pressurant, would require that both sides of the system are passivated. This can be avoided if the diaphragm allows the pressurant gas to flow through. A dedicated permeability assessment, done in conjunction with the tank supplier, could then be sufficient to prove that passivation would be achieved. Otherwise, implementation of a dedicated passivation valve on the pressurant side would be required.

For the propellant side of the system, one way to achieve passivation is to deplete through the thruster. Depending on the system design and accordingly the thrust class, the usage of the thrusters has to happen in a robust Attitude Orbit Control System (AOCS) mode, because firing will generate a ΔV, which could have a detrimental effect on the spacecraft dynamics, if not accurately controlled in

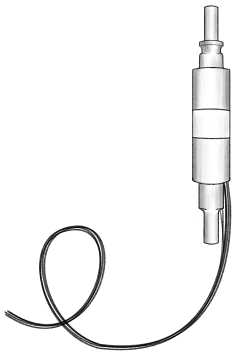

Figure 3. Illustration of a shape memory alloy valve.

direction. Additionally, the propellant supply will not be balanced anymore, due to the asymmetric depletion process within the system and the different thrust generated by the thruster due to insufficient propellant feeding (*e.g.* bubbles). Furthermore, the thrusters are normally qualified to a higher inlet pressure, leading potentially to unexpected behaviour during the depletion. Besides this qualification concerns, the time required for passivation through the thruster can be quite long, according to the propellant tank size, and therefore possibly costly from the operational point of view. Lastly, since it is complicated to quantify the amount of propellant left within the system at EOL, it is difficult to state that the propulsion system is fully passivated. Leaving some residuals inside the tank could make the passivation incomplete and could lead to an explosion during the re-entry of the spacecraft. On top of this, a clear and unique definition of the maximum amount of fuel that could be considered safe to leave in the tank is still to be agreed upon.

On the other hand, the design of a dedicated passivation valve, even if only considered for the pressurant part, has to take into account the material compatibility with the propellant. An additional challenge is the fact that the valve needs to be qualified to last for many years in the space environment and then reliably work at EOL. Moreover, even common features, like safety for ground personnel (*e.g.* leakage, handling of explosives) has to be ensured. In recent years, some valves were developed for this purpose already. One example is the microperforator, which is already qualified and is using a pyrotechnic device for activation. Within CleanSat, a trade-off between pyrotechnic devices and SMA valves was carried out by [4] and [10]. This assessment revealed that the lifetime of pyrotechnic devices is mostly driven by the explosives, while this does not apply to the SMA valves. Moreover, these SMA valves require a low amount of power, have a simple electrical driver and have no mass penalty with respect to the pyrovalves. On top of this, due to the avoidance of explosives, there are no legal constraints related to handling and testing of those, which instead are common for pyrotechnic devices. Moreover, the lower actuation energy implies significantly reduced

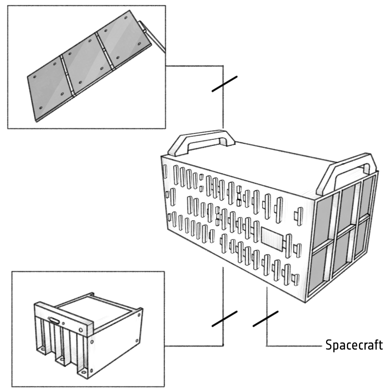

Figure 4. Illustration representing power passivation: the solar arrays (top left) and the batteries (bottom left) are linked to the Power Conditioning and Distribution Unit (centre), which in turn is linked to the spacecraft. All links must be severed.

shock loads on local structure and hydraulic self-induced shock or adiabatic compression in liquid lines. Lastly, the SMA valves have an activation time of around 15 minutes (typical heater power for a spacecraft application is supposed), while pyrotechnic devices take 10^{-3} seconds to act. Higher activation times are an advantage when performing passivation, because in case of an unwanted activation this gives the operators a larger time margin to stop and reverse the passivation process. For all these reasons, SMA valves are currently being further investigated separately by both ArianeGroup and Arquimea, within the CleanSat framework.

To conclude, as it was already anticipated, a dedicated assessment by means of a thermal analysis of the passivation procedure is an important step that is often overlooked. Local increase of temperature and pressure can happen during passivation operations, and it is known that this can result in ignition and explosion of the residuals. This holds for hydrazine, but even more for specific alternative propellants, such as the newly developed green propellants.

Power system In current spacecraft, all the energy of the power system is stored in batteries. As identified and investigated by [14], a so-called thermal runaway, which corresponds to a positive temperature feedback that can lead to an explosion, can be induced by over-temperature, internal or external short-circuit, overcharge or any structural damage (*e.g.* deriving from an hyper-velocity impact). Concerning the temperature, further studies conducted by [49] have determined that the temperatures within the batteries range from -50° C to 80° C in LEO missions. A battery failure mode that was tested and that did not result in any safety issue for the passivation was the over-discharging of the battery. Of course, this holds unless the battery is recharged again afterwards.

Mitigation strategies were identified by both [1] and [14] for each of the aforementioned issues. First of all, discharging the battery to the lowest possible state of charge and lowest possible voltage is already beneficial to ensure that no charge is stored. The battery should contain a dedicated and isolated passivation circuit through which it can be disconnected from the main power system and be discharged through bleed resistors. The switch connecting the battery to the passivation circuit should be as close as possible to the power source. Also, developing safer batteries (*e.g.* solid electrolyte, casings, inter-cells material) or placing them internally within the satellite would reduce the risk of explosion linked to hyper-velocity impacts. However, the former would entail higher development costs, while the latter directly contradicts the common practice of mounting the batteries internally directly on the spacecraft structure, for them to be in direct contact with the radiator. An interesting system level mitigation strategy that was proposed was to spin the spacecraft to avoid the battery overheating, due to long sun exposure. However, as it was already mentioned, the thermal analysis of the spacecraft for the post-EOL phase is not systematically performed. Possibly, this will be implemented in the future.

There are three main viable options for power passivation, according to its definition. Firstly, the batteries can be discharged and their link with the main bus can be severed, by isolating or short-circuiting them. Secondly, the Solar Arrays (SA) can also be isolated from the main bus. In these two cases, it is either possible to open the link between batteries or SA and the main bus, or to short-circuit the SA to ground or the batteries with bleeding resistors. In this last case, the batteries need to be first disconnected from the main bus. A visualisation of power passivation is shown in Fig. 4.

Lastly, it can also be sufficient to make the power storage safe, by shielding or re-designing the batteries. An interesting solution that has been proposed in [15] is the usage of solid state batteries, where the flammable and volatile liquid electrolyte would be substituted by a solid one. In this case, preliminary tests that simulated various battery failure modes were promising, as they did not cause any gas emission, fire or explosion.

According to [30], it is preferable to isolate the SA rather than the batteries, because the latter require bigger devices and solutions that are specific depending on the application. Moreover, by isolating the SA the spacecraft bus does not receive any more power, while the SA would keep providing power to the Power Conditioning and Distribution Unit (PCDU), if only the batteries are isolated. Therefore, this would represent a less robust solution.

The PCDU can manage the power generated from the SA using two different architectures: Series Switching Shunt Regulation (S3R) and Maximum Power Point Tracking (MPPT). The former is usually employed in missions that include a Solar Array Driving Assembly, which guarantees that all the cells of the SA are pointing towards the sun, and therefore work at the same temperature and with the same illumination conditions. The latter is used in missions with high power requirements, because it allows to extract the maximum possible power from the SA, but also when bigger variations in illumination and temperature conditions are expected. However, a PCDU based on S3R is simpler and cheaper than one based on MPPT. Some of the solutions that were investigated within the CleanSat framework by [30] and [57] are specific for one of these two architectures, while others can be applied to both. Permanent activation of the MOSFETs to short-circuit the SA is a passivation solution applicable only to S3R, while the usage of MOSFETs to isolate the SA from the main bus, and galvanic insulation, are suitable only for MPPT. Electro-mechanical devices and relays can be applied to both.

The studies that were performed did not result in a clearly preferred solution, even if relays were identified as a good compromise solution, because they are applicable in all cases. The passivation method selection depends on the spacecraft, taking into account the solar array interface (S3R or MPPT) and the platform size.

For this reason, the chosen way forward was the parallel development of various passivation technologies, for both architectures. Indeed, within CleanSat, TAS Belgium is currently working on the qualification of power relays that could be placed in parallel to the SA for S3R or MPPT architectures, power MOSFETs in addition to low level latched relays to isolate the SA for MPPT architecture and galvanic isolation of the SA using the transformer implemented in the DC/DC converter of the MPPT architecture.

3.1.2. Deorbit Manoeuvre

A way to ensure compliance with the requirement stating that a spacecraft shall re-enter within 25 years after its EOL, is to perform a deorbit manoeuvre to lower the perigee before passivation. Indeed, depending on various variables, such as their mass and drag surface, it is estimated that objects orbiting at altitudes between 500-600 km will naturally decay in the Earth's atmosphere within 25 years. Therefore, by lowering a spacecraft's altitude with a propulsive manoeuvre, an operator can make sure that it will comply with the SDM requirement. Of course, performing such a manoeuvre requires that the tank capacity is sufficient to fulfil the nominal mission and EOL operations. This manoeuvre can be performed with low or high thrust propulsion systems. The former will be further analysed in Subsection 3.2.1 and the latter in Subsection 3.2.2.

3.1.3. Passive deorbit devices

These technologies are usually adopted when the mission is not complying with the requirement that states that they can take up to maximum 25 years to deorbit. This requirement is even stricter for the spacecraft that do not have a propulsion system, because, given the fact that they cannot perform collision avoidance manoeuvre, the 25 years count starts from their orbit injection epoch.

Most of the passive deorbit devices exploit the interaction with the Earth's atmosphere, ionosphere or magnetosphere, to generate forces that accelerate the re-entry of space objects. For example, drag sails increase the surface of the spacecraft to enhance the atmospheric drag acting on them. For this reason, such devices only work when the spacecraft's mass and altitude are such that the force generated is strong enough to have a significant effect. In the case of the drag sail, if the altitude is too high, increasing the surface of the spacecraft will have negligible effects due to the extremely low air density. In general, they are simple systems which have a limited footprint on the host spacecraft. Their cost is also low and they are designed to be scalable.

Of course, passive deorbit devices are particularly attractive for spacecraft without any form of propulsion, that are required to re-enter within 25 years after their injection, because they cannot perform collision avoidance manoeuvres. Instead, to be competitive with respect to the main alternative, which is to perform a deorbit manoeuvre using the on-board propulsion system, passive deorbit devices should ideally have an overall mass and volume lower than the ones corresponding to the amount of fuel that would otherwise be needed to deorbit the spacecraft.

Some challenges linked to passive deorbit devices are micrometeoroids impacts, that can decrease their performances, and the effect of the Atomic Oxygen (ATOX) environment, in which they spend many years. Moreover, on one hand, uncontrolled tumbling of the host spacecraft can dramatically jeopardize the deployment of passive devices, which need to be pointed in a specific direction to be effective. On the other hand, the deployment itself can also be the source of undesirable tumbling motion of the host spacecraft.

Lastly, it is important to point out that for all passive devices, autonomous deployment is considered to be a great asset because it would imply compliance with the SDM requirement even in the case of a failure of the host spacecraft. To check the status of the satellite, a watchdog system can be implemented, taking into account that its interface with the host system is often non-trivial. Usually, the deployment of the passive device is commanded from ground.

Drag augmentation devices These devices aim at increasing the effective drag area of the satellite, to enhance the drag force, slowing down the satellite and accelerating its decay. Among the various options that have been investigated, there have been both sails and inflatable devices.

An example of the former is given in [48] and [52], while Fig. 5 illustrates the concept. In this case, the drag augmentation is achieved through a lightweight membrane supported by rigid booms, while the deployment is performed using stored spring energy. A challenge that is important to take into account when designing a sail is cold welding. Indeed, two objects that stay in contact for long times in a very cold environment, such as outer space, might weld. Moreover, one of the main drawbacks of this technology is that by increasing the surface of the spacecraft it also increases its probability of collision with debris. Another drawback is that drag sails need to be taken into account already during the spacecraft design, to ensure that deployment clearance is achieved.

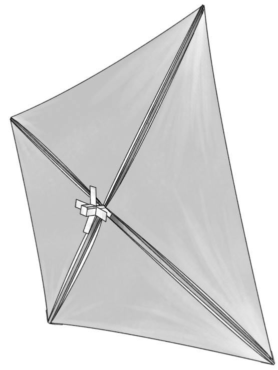

Figure 5. Illustration of a drag sail.

As reported in [51], some sails have been already successfully deployed in space and therefore have a quite high Technology Readiness Level (TRL). However, these tests did not demonstrate the deployment after a long on orbit storage and the capability to survive the environment for many years after the deployment, which is why this kind of technology is still evolving.

Within CleanSat, HPS is currently developing and testing a 25 m^2 drag sail. A drag sail of such size covers already a pretty wide range of satellite applications, includes satellites with a mass below 750 kg and flying at an altitude below 650 km. Deorbiting at higher altitudes could be achieved if the satellite mass would be further reduced or the drag sail design area was increased. Sails usually have a modular design, so that they can be scaled up or down to be adapted for different mission scenarios. Lastly, when sizing the drag sail, it is important to take into account that after deployment the spacecraft would usually not have a stable attitude, which would maximise the sail performances, but instead would be tumbling during the decay. This usually leads to an over-sizing of the sail with respect to the optimal scenario. However, it has been pointed out in [40] that it should be possible for this system to be self-stabilising by having the correct distance between centre of pressure and centre of mass. The satellite may not stabilise in the higher orbits, but could become stable at lower altitudes. This could be a promising solution to not be forced to oversize the sail.

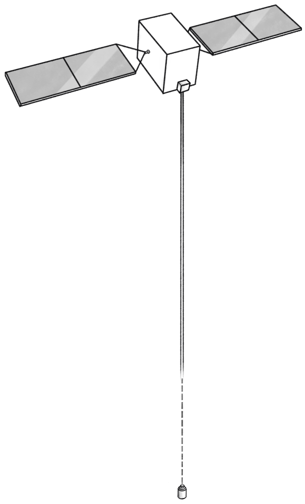

Figure 6. Illustration of a breaking tether.

Inflatable devices are a more compact alternative with respect to drag sails. They have a lower TRL, but their main advantage is that they do not require the usage of mechanisms for the deployment. Indeed, the booms or support structures are inflated by a pressured tank and hold the skin that increases the drag area, as shown for example in [18]. The main risk linked to their usage is the one of possible leaks.

Tethers Electrostatic and electrodynamic tethers are a low-weight and low-cost solutions to accelerate the decay of space objects. An impression of how such tethers could look like is shown in Fig. 6.

The first concept that was developed is commonly known as electrodynamic tether. This technology requires a current to flow in the tether, which exploits Lorentz drag to slow down the satellite. [39] introduced later on the concept of an electrostatic tether, which makes use of a thin charged tether to tap momentum from the plasma ram flow by Coulomb drag. Since in this case no current is required to flow in the tether, this concept needs only a conducting tether and a voltage source to maintain the potential difference between the spacecraft body and the tether. Consequently, this technology entails an order of magnitude lower mass and power consumption than electrodynamic tethers. Moreover, electrostatic tethers are known to be more effective in the solar wind, where electrodynamic tethers are useless. The main advantage of electrodynamic tethers with respect to the alternative is that, according to [50], in LEO Lorentz drag is more than two orders of magnitude larger than Coulomb drag . Moreover, electrostatic tethers are thinner than electrodynamic and therefore more prone to be cut by debris.

Electrostatic tethers were further developed under the CleanSat framework with [31]. In this case, the chosen baseline was a spring deployed gravity-stabilised tether. In the context of this study, the TRL of a plasma brake module increased from 1 to 2-3. They also proved that deorbiting a spacecraft up to 800 kg from 850 km or up to 200 kg from 1200 km within 25 years would be possible by using a 2 kg one-module device. Using two modules would allow to deorbit larger spacecraft, but also to increase reliability and performances.

In general, the main advantage of tethers is that they are a very scalable technology. Their length is proportional to their performances and it is mainly limited by the tensile strength, which depends on the chosen material.

The main disadvantages of tethers are that they can be extremely long (up to around 3 km) and therefore could increase the risk of collisions in LEO, especially if used in multiple missions at the same time. Moreover, the main threat is that the tether deployment fails and that the device ends up tangling around the spacecraft itself, becoming ineffective. Moreover, if the spacecraft is tumbling, it would not be possible to deploy the tether at all, because it would be impossible to stabilise it and it would end up entangling around the spacecraft.

There have already been some space missions that successfully deployed a tether. Another interesting point that it is often source of misunderstandings, is that usually tethers do not pose a threat for operative spacecraft in the case of a collision, because their thickness is very small (around 50 μm) and the impacting area is also relatively small. This feature distinguishes tethers from other passive devices, such as sails, which have rigid booms and a much larger impacting area.

3.1.4. Design for Demise

Design for Demise is the intentional design of space system hardware to maximise its ablation and minimise the overall casualty risk, during an uncontrolled atmospheric re-entry. Of course, these changes in the design shall not affect negatively the spacecraft overall performances (*e.g.* mass, stability).

D4D can be achieved in many different ways. For example, the heat required to ablate the components can be minimised by decreasing their mass or by manufacturing them with materials that have different properties, *e.g.* lower melting temperature. The heat transfer can be also optimised, for example by including orifices in the structures through which the hot flux can reach the internal equipment. Of course, despite the changes imposed by D4D, the spacecraft still has to be capable to reliably carry out its functions and successfully complete the mission for which it has been developed. However, understanding and applying D4D is not an easy task, for two main reasons. The first is that some of the re-entry processes, such as fragmentation, have not been fully understood yet. The second is that it is extremely challenging to simulate and mimic the re-entry conditions on ground. In particular, heat flux changes throughout the re-entry cannot be reproduced and there is a size limit for on-ground tests, due to the available facilities. Furthermore, it is complex to test high heat fluxes and mechanical stresses at the same time. Lastly, in reality the fragments spin and rotate during the re-entry, while they are stationary during on-ground testing. The replication of the varying re-entry attitude is currently under investigation.

Considering all the challenges related to performing demise verification and casualty risk assessment, it has been recognised that there is a need for commonly accepted guidelines for demisability assessment and verification. This need has been partially addressed by [24].

To start developing demisable technologies, the most critical pieces of equipment needed to be identified. This was done by [11], [58] and [20]. First, a critical element was defined as an element that is large or heavy, that is usually shielded by other pieces of the spacecraft, that is common to many platforms or that has an high proportion of critical materials, which are characterized by a high heat of demise or very high melting temperature. The aforementioned studies allowed to identify some system level criticalities and some equipment level critical items, such as tanks, reaction wheels (RW), magnetorquers (MTQ), driving mechanisms, balance masses, batteries and optical instruments. For some of these, possible demisability solutions were identified and eventually pursued by other studies, which will be detailed in the following paragraphs.

System level At system level, there are various techniques that would allow to reduce the overall casualty risk of a re-entry. First, it is straightforward that a prompt opening of the outer satellite structure during the re-entry helps to reduce the casualty risk on ground, because it exposes the internal equipment to the heat flux earlier on. To tackle this, various solutions were investigated, in particular SMA dismantle mechanisms by [6] and [12] and demisable joints by [17] and [59].

OHB also investigated and compared various solutions that would allow to obtain an early break-up. An interesting open point on early break-up is that at current break-up altitudes the Multi Layer Insulation (MLI) that covers the spacecraft is usually removed by the mechanical forces and by the interaction with atomic oxygen in the atmosphere during the mission lifetime and orbital decay phase. This means that the MLI could still be in place at higher altitudes, because of the lower mechanical stresses and lower amount of oxygen present there. This remaining layer could prevent the heating of demisable joints or SMA actuators, and in turn jeopardize the early break-up.

Further investigations could be conducted about the influence of harness and propulsion or heat pipes on the fragmentation.

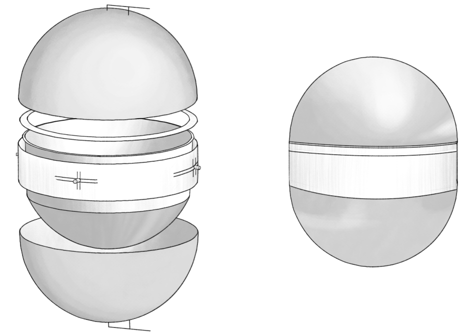

Figure 7. Illustration of a monopropellant tank.

A different D4D system level technology is the usage of exothermic reactions, which increase the amount of available energy. This was investigated specifically on RW by [21] and [19], who identified the need of further studies and concepts that would allow to properly contain the energy released to support melting process. Moreover, selecting the amount and placement of thermite to support demisability during re-entry is a difficult optimisation problem that could not be tackled during past studies.

Tanks Tanks were identified as a critical element of the re-entry phase because they are usually made of Titanium, which has a very high melting temperature, or include a composite overwrap that is hardly demisable. On top of this, tanks are the only confirmed re-entry debris. To increase their demise, various activities were performed within the CleanSat framework, by first distinguishing the tanks in some categories. The design of tanks used in space mission can greatly vary depending on the propulsion system. Thus the first distinction is between high pressure tanks, used for electric propulsion and usually made of a Composite Overwrapped Pressure Vessel (COPV), and liquid propellant tanks, traditionally metallic and used for mono and bi-propellant propulsion systems (see Fig. 7).

The former option has been recently gaining a lot of market interest, because space industry is currently switching to Electric Propulsion (EP) and composite tanks are the only ones that can hold gas efficiently. Up to now, the most used gas for EP was Xenon at 200 bars, which is becoming very expensive due to its rarity and due to the increasing demand. Therefore, a switch to Krypton could happen in the next years, because it is a cheaper alternative. This would have a negative impact on the tank demisability because Krypton requires a higher operational pressure (300 bars) and therefore thicker tanks. The demisability of the COPV tanks was investigated in the past by [2]. The main conclusion was that even if the Aluminium liner were to melt inside the COPV, it is not clear how and if it could escape the composite over-

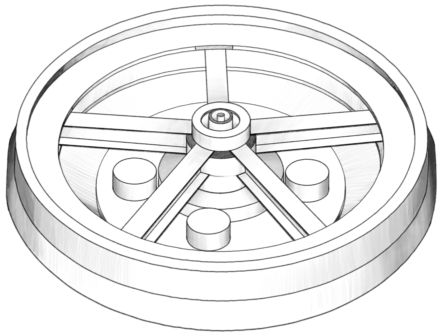

Figure 8. Illustration of a reaction wheel.

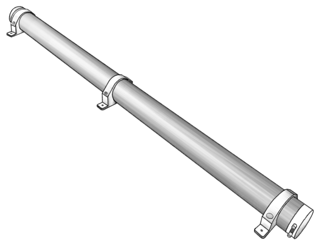

Figure 9. Illustration of a magnetorquer.

wrap. Indeed, the thermal characteristics of the composite layer are not yet well known and this makes it difficult to model it. Moreover, it is challenging, if not impossible, to extrapolate the results related to one type of composite to apply them to a different one. Currently, there are on-going studies, for example by Peak Technology, that focus on enhancing the demise by using different manufacturing processes for the COPV tanks.

Instead, the prevailing solution for liquid propellant tanks, which are usually kept at a pressure around 25 bars, is to enhance their demisability with a change of material. In particular, [5] and [44], proposed to substitute the commonly used Titanium alloy (TiAl6V4), which has high melting temperature and heat capacity, with Aluminium alloys, such as Al-Cu, Al-Mg and Al-Li. The main advantage of the first one was its maturity. The Al-Mg alloy would be compatible with green propellant but its manufacturability still had to be demonstrated. Al-Li had good structural properties and it would have allowed for thinner tanks, but even in this case its manufacturability was not well known. Alternatively, [3], which was focusing on green propellants, proposed a thermoplastic liner with a carbon composite overwrap. However, this solution would have had the same issues pointed out for COPV tanks, and potentially leak problems because of the lack of a metallic liner. In the past, only the tank shell was modelled and analysed, while nowadays a more comprehensive assessment is performed, including interfaces and Propellant Management Devices (PMD), that should be demisable as well. Currently, various volume ranges have been identified and distinguished. Large tanks, with a volume range between 170 and 220 L, are currently being investigated by MT Aerospace, while demisable alternatives for small ones, with a volume range between 40 and 52 L, are being researched by the Polish Institute of Aviation.

Reaction Wheels Reaction wheels are used to control the attitude of a satellite without the use of thrusters (see Fig. 8). Analysis showed that medium and heavy RWs can survive the re-entry. Moreover, satellites typically employ four RWs at a time, which means that a demisable alternative could greatly benefit their casualty risk. However, RWs are very complex mechanisms, which means that every change in their design takes a long time to be qualified. Among the proposed solutions in the context of [13], the most promising ones were to change the materials of the most critical sub-components, such as the flywheel or parts of the ball bearing unit that should not affect the RW heritage and require a full life test that would be very long. Another option would be to try to upgrade the electronics to obtain a down-sized design with increased speed. The former will be investigated in the future, while the latter would be more complex to develop, because reducing the size of the flywheel and increasing the speed of the RW with respect to current used speeds (typically up to 6000rpm) would require a life test, which is very time consuming and can be quite expensive.

Another proposed method was the usage of exothermic reactions, which were tested on RW within [21] and [19].

Magnetorquers Magnetorquers, or magnetic torquers, are a satellite system for attitude control, detumbling, and stabilization (see Fig. 9). MTQs were identified as critical elements because their more internal elements are exposed to the heat flux only later on during the re-entry phase. That is why medium and heavy cores were expected to always survive the re-entry. When setting up the re-entry conditions for the MTQs investigated within [43], it was taken into account that they are usually attached to a panel of the spacecraft, which can shield them from the flux. Therefore, while other components had to demise given a release altitude of 78 km, the MTQs got a stricter release altitude requirement: 65 km. Among the proposed solutions to enhance the demisability of MTQs, the most promising ones aimed at exposing the core as early as possible during the flight. This was suggested since the core material itself could not be changed, as it is strictly linked to the functionality of the MTQ. Therefore, the material of the feet was changed to support an earlier separation and the housing material was changed to guarantee an earlier exposure of the core. Another option that

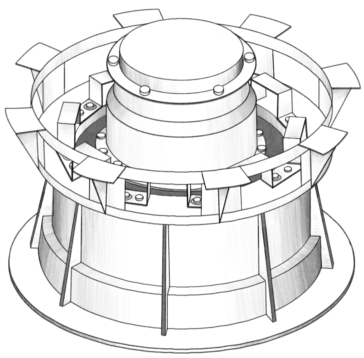

Figure 10. Illustration of a solar array driving mechanism.

was investigated was to split the core in juxtaposed cylinders.

Driving Mechanisms Large mechanisms are often made out of steel and titanium because of load and stiffness requirements, and are therefore hard to demise. In the past, driving mechanisms and in particular Solar Arrays Driving Mechanism (SADM), which can be visualised in Fig. 10, were identified as critical because they contained such critical materials.

Two main kinds of SADM can be distinguished: the ones that allow for continuous rotation and the ones that do not allow it. Both can rotate for $360°$, but one can keep on rotating whereas the other needs to rotate back during eclipse. The difference is that continuous rotation requires a slipring, while the other option uses a twist capsule or cable wrap. The former may be harder to demise, due to the slipring materials and due to its higher complexity and mass. However, the missions that include the continuous rotation SADM are usually heavy and could therefore not be able to perform uncontrolled re-entry even if a demisable SADM alternative was available. Lastly, many Radar Earth observation missions exploit a dawn-dusk orbit and do not even require a SADM. In any case, a more demisable SADM may be useful or needed for a restricted number of applications.

The SADM has one external face, but the rest is inside the spacecraft and often attached at the end. Despite having a great amount of harness holding it, it could potentially be ripped out when the solar array detaches from the spacecraft.

A demisability assessment of SADM is currently being carried out by KDA.

Balance Masses In early studies, heavy balance masses were found to be prone to survive re-entry. A solution proposed by [34] was to develop layered balance masses. This concept would have been combined with a passive release system of the layers. According to their simulations, the balance masses would have always completely demised if this was put into practice.

However, it must be remembered that balance masses are different for every mission and can normally be easily adapted. Therefore, for the time being, it was deemed to be inconvenient to investigate further this demisability solution, since it would have not been generic and applicable to all cases.

Batteries The assessment of the battery demise initially suggested that they were a critical item, but a refined analysis inclusive of fragmentation to cells resulted in complete demise. [16] suggests that the key aspect of the demisability is the failure of the Glass Fiber Reinforced Plastic (GFRP) which contains the cell packs in the batteries. Where this is predicted to fail, there is no obvious physical reason why the fragmentation to cells will not occur. To guarantee this, the break-up process should be studied in more detail.

Optical instruments One of the particularities of payload, and in particular optical instruments, is that they are not generic. Instead, they are adapted to the mission. It is therefore difficult to propose a unique demisable solution that could be used in every mission. Moreover, they have many design constraints that need to be fulfilled and that are hard to be transcended while applying D4D. Notably, some materials cannot be easily substituted, such as ceramics, which are needed because of their thermal stability, and the glasses and mirrors that are used for the lenses. A secondary challenge that has been identified when trying to apply D4D techniques to optical instruments within [46], is that the most undemisable components, such as ceramics and glass, are not well characterized. Indeed, the ceramic breakage is hard to predict, while glass can have a viscous behaviour that is difficult to model. Among the investigated solutions, the most promising turned out to be containment, because it would allow to keep together all the pieces known to be undemisable, reducing the number of fragments and in turn the overall casualty area. Other options that were taken into account but that were deemed to be less promising were design for fragmentation, *i.e.* divide lenses into multiple smaller components that shall easily separate during re-entry, and the usage of pyrotechnic devices, such as pyrobolts.

3.2. Controlled Re-entry

This kind of re-entry targets a specific uninhabited and inherently safe zone (*e.g.* South Pacific Ocean Uninhabited Area), to minimise the casualty risk. To perform controlled re-entry a high thrust and high ΔV manoeuvre is required at the EOL to perform the so-called final burn.

Sometimes, the last burn is anticipated by a perigee lowering manoeuvre.

The main system impact of this re-entry strategy is that a high thrust capability is needed, which in turn means that chemical propulsion must be used and that a high mass of propellant is required. In the worst case scenario the higher mass can lead to the choice of a larger launcher, which in turn has a dramatic impact on the overall mission cost. Moreover, at the EOL, blow-down systems often used in LEO for mono-propellant thrusters usually have a lower propulsive efficiency, which needs to be carefully checked and possibly enhanced with a repressurisation system, to be able to perform the required final burn. The thrust is also constrained by the maximum load that the appendages can bear without breaking. It is also important to notice that performing a controlled reentry entails the usage of an overall system that is more complex than the one required by uncontrolled re-entry, and is therefore inherently riskier. Lastly, it is a common practice to lower the orbit as much as possible before the last burn to reduce the effort required by the last manoeuvre. This can pose a challenge in terms of controllability when the satellite is close to the orbit pericentre and may require specific Attitude and Orbit Control Systems (AOCS) modes. It may also affect the thermal design or even the communication system, since both are typically designed for higher orbits.

3.2.1. Low thrust manoeuvres

EOL manoeuvres can be performed to lower the altitude of a spacecraft. This is usually done to decrease its time in orbit after EOL, and it is normally done by the spacecraft Reaction Control Thrusters using the propellant system on-board. Considering that this is usually a monopropellant thruster, low thrust and higher specific impulse engines can be considered to obtain a mass optimised system to perform the Controlled Re-entry.

Electric propulsion [53] explored the possibility of using ion thrusters to perform these manoeuvres. As a result of this study, a thruster baseline for two case scenario (controlled re-entry and uncontrolled re-entry) has been been envisaged. The same EP sub-system architecture for both thrusters was selected. This was deemed to be an interesting thruster and similar ones have been chosen for present and future constellations. Indeed, with EP it is possible to raise and lower satellite orbits with a minimised mass impact, which makes it possible for operators to send more satellites within one launcher to a lower orbit. EP is also the solution used in telecommunication constellations. In general, this BB was not pushed further in the framework of the CleanSat initiative because it was not specifically related to the EOL.

Moreover, a very interesting comparison that was carried out in the framework of this activity was the comparison of Krypton and Xenon as propellant of choice for electric propulsion. According to [23], the main effect of Krypton on the thruster performance with respect to Xenon, is a reduction of the thruster efficiency and an increase of the specific impulse values. Moreover, Krypton can be more than ten times cheaper than Xenon, as reported by [32]. As a main drawback, Krypton has a lower density than Xenon, which means it requires to be stored at higher pressures, and therefore needs more voluminous and heavy tanks. Depending on the driving requirements being thrust level, specific impulse, cost or overall mass, one may be chosen over the other.

Arcjets This technology was investigated by [54] and [38]. It creates an electrical arc that is used to heat up a flow of propellant, such as ammonia or hydrazine. Therefore, arcjets can be used to generate an hybrid propulsion system, in combination, for example, with hydrazine chemical monopropellant engines. Arcjets provide a lower thrust level, but have a higher specific impulse, than their chemical counterpart. This implies they could have been successfully used for all operations that did not require high thrust, such as perigee rise and lowering, using the same tanks as the monopropellant systems. Despite this, they were always perceived as a separate system. They are a potentially cheaper alternative with respect to Electric Propulsion and exploit a simpler Power Processing Unit. However, they have a high power consumption and they are more sensitive to pressure with respect to hydrazine chemical thrusters. Moreover, at the time of the first analysis of this technology, it was noticed that additional developments would have been required to qualify such thrusters for a very high number of ignitions and long operation times. Given these disadvantages, and the fact that EP quickly became the standard for small satellites and it now has a great heritage, arcjets were not investigated further.

An important realization that was achieved in the framework of the aforementioned CleanSat activities was that green propellants, such as LMP-103S, are not suited to an arcjet system. More analyses and tests should have been performed for future applications, with inevitable lengthening of development time and increase of cost.

3.2.2. High thrust manoeuvres

There are three main thruster options available to perform the final burn of a controlled re-entry: monopropellant, bipropellant and Solid Rocket Motors (SRM).

Monopropellant Thrusters These thrusters are the standard ones used for LEO Earth Observation missions and therefore constitute the easiest and currently cheapest option. However, they are not normally pressure regulated, which means that they may require repressurisation to be able to perform the last burn, depending on the final thrust-to-mass ratio. On top of this, they

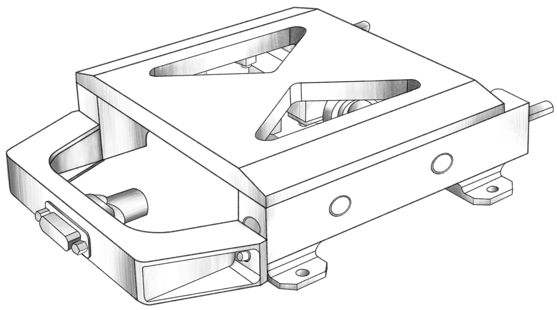

Figure 11. Illustration of a mechanical pressure regulator.

are the ones providing the lowest propellant efficiency with respect to the alternatives.

Two different kinds of propellant are usually distinguished in the context of these thrusters. One is hydrazine, the traditionally most used propellant, while the other is the family of green propellants (*e.g.* LMP, hydrogen peroxide), which are non-toxic alternatives.

The investigations performed within the CleanSat framework on this topic were various. Concerning hydrazine-based systems, [9] focused on a re-design of the thruster to increase the available specific impulse and lower the cost with respect to a reference thruster. Indeed, it was assessed that the performance could be increased with a higher expansion ratio of the nozzle and that the cost could be reduced by approximately one third with respect to the reference. [8] focused on achieving the same but with a green propellant based thruster. In particular, they proved that a deorbit thruster with LMP-103s as propellant can be realized with higher performance than a classical hydrazine engine. However, due to the need of high temperature stable materials (procurement and manufacturing), the cost would be higher than an hydrazine thruster. Possibly, these conclusions would have been different if hydrogen peroxide as an alternative green fuel had been used in the design. Indeed, LMP requires the heating of the catalyst bed, while hydrogen peroxide and hydrazine do not. However, this issue is much more impactful on AOCS manoeuvres, rather than on deorbit ones. Thus, it could possibly be overcome in this second case. The LMP's specific impulse and density are better than the ones for hydrogen peroxide, while hydrogen peroxide has a higher density but worse specific impulse with respect to hydrazine. Hydrogen peroxide was the propellant chosen by [55], by whom two baseline concepts were analysed: one low thrust and one high thrust motor. The selected concentration of hydrogen peroxide for both the baselines was 98% by weight.

Pressure regulation As previously stated, sometimes controlled re-entry may demand an active pressure regulation in order to provide the correct engine inlet conditions for boost, ensuring to meet the window for the manoeuvre properly. That is why one of CleanSat BBs, [7], focused on the development of a repressurization module which would exploit a high pressure gas tank and solenoid valves. They proved that the active pressure control of the propellant tank under defined conditions would be feasible with the proposed technology. It was also considered that the developed design could be applicable for GEO missions, given that the high pressure tank volume would have to be increased due to bigger downstream ullage volumes. Other ways to achieve pressure regulations are through mechanical (shown in Fig. 11) or electronic re-pressurisation. Mechanical repressurization is already used in GEO, but it is used at the Beginning Of Life (BOL). Its usage is not recommended at EOL because it can present aging issues, and would therefore need a re-qualification. Moreover, there are no mechanical regulator suppliers in Europe. Regarding an electronic pressure regulator, the fact that the required solenoid valves are not produced yet in Europe poses a constraint on its development.

In any case, as long as the pressure and the thrust level at the EOL are high enough to perform the last manoeuvre (when the perigee has to be decreased significantly in a single manoeuvre) there is no need for re-pressurization, but this often requires a significant over-sizing of the propulsion system. Nowadays, due to the fact that Vega launcher performances are up to 2 tons in LEO, this is the most common technique.

Bipropellant Thrusters Bipropellant thrusters are complex and heavy systems that require at least three tanks to function, which makes them costly and more prone to failure. However, they guarantee considerable advantages. First of all, the pressure of the propellant system can be regulated, which means that the thrust can be kept constant throughout the whole spacecraft lifetime. Bipropellant thrusters usually exploit mechanical pressure regulators at BOL, to raise their orbit to reach GEO. Yet, they would still benefit from the qualification of mechanical pressurization systems for the EOL or from electronic pressure regulators with less aging issues. Secondly, they provide the highest propellant efficiency with respect to the other high thrust alternatives, which means that they can allow important savings in terms of fuel mass. Considering the drawbacks of this propulsion system, its utilisation is usually taken into account only for large spacecraft (*i.e.* with a mass greater than two tons), for which the usage of monopropellant may lead to the selection of a larger launcher.

Solid Rocket Motors Fig. 12 shows how an SRM would appear. SRMs are modular systems that can be added to the main propulsion system used for nominal operations and that ensure the required deorbit thrust and

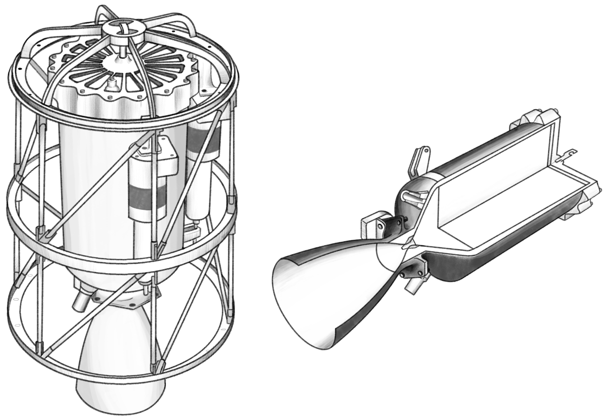

Figure 12. Illustration of a solid rocket motor.

ΔV. On top of this, they have a higher specific impulse than monopropellant thrusters. Another advantage is that, unlike the liquid propellant alternatives, they generate no sloshing and therefore they have a stable contribution to the position of the Centre of Mass (CoM). This can be an important asset to missions where an accurate knowledge of the CoM is a necessity.

Within CleanSat, SRMs were analysed by both [42] and [45]. The main constraint on SRMs during these studies was the maximum acceleration allowed of 0.02-0.04g, to guarantee the integrity of appendages and prevent the creation of new debris. This greatly bounds the SRM thrust level, leading to regimes of combustion where the SRM are less efficient. Moreover, this increases the burning times needed to achieve the same total impulse, which in turns leads to the need for higher insulation mass. To overcome this issue, three solutions are usually proposed. One was to develop high specific impulse and low thrust solid rocket motors, to allow for lower thrust levels without impacting too much on the propulsive performance of the motors. The second one was to use clusters of SRMs to relax the requirements applied to each SRM, by using smaller motors instead of a long-time burning single SRM. A cluster configuration also has the advantage that accommodating several small motors can be easier than a single large one. Moreover, if only one optimised motor was to be used for every spacecraft, every different size would need to be qualified, increasing the non-recurring cost in total. Considering that a SRM might be used in a cluster configuration and that if only one is used, depending on the size of the satellite, different lengths of the motor are needed, the interface would need to be as universal and scalable as possible. The last proposed way to cope with the requirement on the maximum thrust level was to enhance the robustness of the current satellites' weak points, such as deployable hinges and supporting structures, to relax the constraint. This would lead to unavoidable spacecraft dry mass increase, however the mass saving on the SRM may justify and overcome such a penalty, leading to an overall system-level simplification and mass reduction.

It was also envisaged a possible usage of SRMs for small satellites. Indeed, these do not usually have appendages and their compact shape would allow for a much higher thrust-to-mass ratio. This would reduce the burning time and would make the SRMs lighter and smaller.

When designing a SRM, the selection of an appropriate Thrust Vector Control (TVC) plays a very important role, because it is required to be able to perform controlled reentry. Nevertheless, it comes with higher costs and thrust losses. The TVC is needed to guarantee the alignment of the thrust with the Centre of Mass, to avoid uncontrolled spinning of the spacecraft. TVC can be achieved for example with a movable nozzle, vanes or a gimbal.

Within [22], the possibility of a fully autonomous SRM was investigated. In the framework of this actvity, it was assumed that the electronics would be divided into modular boxes with standard dimensions and mechanical interfaces, connected by a belt structure. In the autonomous deorbiting system, two additional tiles would be included with respect to the non-autonomous version: a fully independent attitude and dynamics control system and the telemetry, tracking and command unit, which allows the decommissioning system to directly communicate to ground without relying on the platform.

4. LESSON LEARNT ABOUT INTEGRATION OF NEW TECHNOLOGIES IN FUTURE MISSIONS

The general aim of CleanSat is to achieve an evolution of the LEO platforms including technologies that would guarantee compliance with the SDM requirements. Thus, the integration of newly developed SDM technologies in as many missions as possible is essential. This initiative has been going on for almost a decade and its co-engineering approach has been evolving and refining in time. This means that it is now possible to draw high-level conclusions about the development and integration of new space technologies.

Regarding the technology development, it was recognised that it can be difficult to predict the evolution of the market. Indeed, technologies developed in the present will enter the market in no less than five years. This issue is worsened by delays in the development, which can lead to changes in the target missions. This is why **scalability** and **flexibility** are a key asset for technologies under development and are often requested by the LSIs. However, this has drawbacks in terms of cost, because different processes or tools may be needed, and this leads to some resistance from suppliers. Moreover, it implies that the mass cannot be fully optimised.

For them to be integrated in new missions, it was assessed that technologies should be taken into account in the earlier phases of the project, *i.e.* before Phase B2. An interesting conclusion that was identified about EOL mission architecture choices, is that institutions such as agencies tend to be more conservative than the private sector, and therefore, in case of doubt, always baseline controlled re-entry to avoid later costs. This means that Design for Demise has been pushed less by institutional missions and users so far, but seems to entail an higher potential for the commercial market, which appears to be ready for higher risks during early design phases.

5. CONCLUSIONS

It is now common knowledge that the fast growth of space debris is jeopardizing the usage of the most important Earth orbits. Space debris mitigation has been identified as a vital step to invert the trend. For this reason, international guidelines and regulations on Space Debris Mitigation are being enforced worldwide. The requirements that were put in place and adopted by many nations and agencies generated an urgent need and a high demand for technology solutions. They were the source of great innovation, because many unforeseen challenges needed to be overcome, but also generated more and more competition. A predefined and commonly agreed-upon approach to verify SDM requirements has been identified as one of the most urgent needs. This is necessary to achieve a levelled play-field and fair competition among industries, in particular when it comes to the evaluation of the casualty risk and on the usage of Design for Demise, that can make the difference between Controlled and Uncontrolled Re-entry, with major cost and risk impacts.

In the frame of this paper, various technologies related to SDM that were investigated or developed under ESA's CleanSat initiative were described, together with the lessons learnt from each of them and the decisional processes that led to certain design choices rather than others.

Lastly, it was recognised that it is paramount to establish and mature standardised procedures in this area and to push them at technology and project level from early phases.

ACKNOWLEDGEMENTS

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All the illustrations contained in this paper have been drawn by Sacha Berna.

LIST OF ACRONYMS AND ABBREVIATION

Table 1. List of acronyms and abbreviations.

Acronym	Meaning
AOCS	Attitude & Orbit Control System
ATOX	Atomic Oxygen
BB	Building Block
BOL	Beginning-Of-Life
CoM	Centre of Mass
COPV	Composite Overwrapped Pressure Vessel
D4D	Design for Demise
ESA	European Space Agency
EOL	End-Of-Life
EP	Electric Propulsion
GEO	Geostationary Earth Orbit
GFRP	Glass Fiber Reinforced Plastic
LEO	Low Earth Orbit
LSI	Large System Integrator
MLI	Multi-Layer Insulation
MPPT	Maximum Power Point Tracking
MTQ	Magnetorquer
PCDU	Power Conditioning & Distribution Unit
RW	Reaction Wheel
SA	Solar Array
SADM	Solar Array Driving Mechanism
SDM	Space Debris Mitigation
SMA	Shape Memory Alloy
SRM	Solid Rocket Motor
S3R	Series Switching Shunt Regulation
TRL	Technology Readiness Level
TVC	Thrust Vector Control

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