IAC-17-A6.5.10

Taxonomy and Analysis of Issues Facing Post Mission Disposal Concept

Emma Kerr[a], Malcolm Macdonald[b], Philipp Voigt[c]

[a] *Mechanical and Aerospace Engineering, University of Strathclyde, 75 Montrose Street, Glasgow, G1 1XJ,* emma.kerr.100@strath.ac.uk
[b] *Mechanical and Aerospace Engineering, University of Strathclyde, 75 Montrose Street, Glasgow, G1 1XJ*
[c] *Future Programs, Airbus Defence & Space, 88039 Friedrichshafen, Germany*

Abstract

In order to ensure a sustainable space environment for future generations a strategy for all spacefarers must be developed in order to mitigate the growth of the space debris population. To this end, this preliminary analysis is the first step towards the development of a cost-efficient but highly reliable PMD (Post Mission Disposal) module. This PMD module will be attached to the spacecraft on ground and will ensure the removal of the spacecraft at the end of the nominal operational lifetime or act as a removal back-up in the case of loss of control of the spacecraft. The PMD module will be scalable and flexible, enabling the PMD of any future spacecraft in an Earth orbit. Ultimately, the gap between the 90% PMD success rate required by ISO 24113:2011(E) and the current success rate of 50%-60% can be closed. A survey of de- and re-orbit techniques and concepts was carried out and a taxonomy of approximately 40 concepts, including 12 which do not appear in the literature, is presented. A qualitative analysis was carried out on the concepts identified in the taxonomy, and a comparison matrix was built including 12 different comparison metrics. The 5 most promising concepts for the PMD module were down-selected from this matrix. These concepts were: drag augmentation, solar sailing, electrodynamic tether, low thrust propulsion and high thrust propulsion. A further 3 additional concepts were also defined by considering combinations of the down-selected concepts. A quantitative analysis of the down-selected concepts was performed using a purpose built analytical analysis tool. This tool was designed to rapidly predict re-entry epochs of space objects, given specific mission parameters. The analytical nature of this tool allowed for a Monte Carlo analysis, resulting in trade-off analyses within and between the different concepts for various mission parameters. The output of the quantitative analysis provided preliminary mission parameters, systems sizing and trade-off data on each of the down-selected concepts and combination concepts. From this analysis it was concluded that each system had its advantages, and challenges, so recommendations were made on how each system could be used to its maximum potential and which systems were more effective than others in specific situations. The most prominent of these results were the need for the PMD to de-tumble the spacecraft prior to deployment of the removal system, and the fact that none of the down-selected concepts were recommended for use in long term missions.

Keywords: Post Mission Disposal; De-orbit; Space Debris

1. Introduction

This paper outlines the mission analysis undertaken in support of the TeSeR (Technology for Self-Removal) project. The TeSeR project is a European Commission funded project (Grant Agreement 687295) to provide a standardised PMD (post mission disposal) module for spacecraft. This project intends to standardise not only the disposal device but the attachment interface to the spacecraft. Further details can be found at the project website.[1] The mission analysis was undertaken to outline possible technologies for use in the PMD module, and requirements of such technologies. It should be noted that while the findings and recommendations presented herein were derived specifically for the TeSeR project, many are of such a general nature that they could be applied to other missions considering post mission disposal.

2. Pre-determined user, mission and system requirements

Prior to beginning any analysis the following requirements were determined by gaining an insight into the users requirements and examining pertinent industry standards, and best practice guidelines.

- The PMD module shall remove S/C (spacecraft) from LEO (low Earth orbit) after its end of operation and as a goal the PMD module should remove S/C from any Earth orbit after its end of operation.
- The PMD module shall ensure that the S/C is removed out of the LEO or GEO (geosynchronous) protected region within 25 years after its end of operation.
- In case of a re-entry into Earth atmosphere the casualty risk on ground shall be $<10^{-4}$.
- The removal shall cause no new space debris on purpose.
- The PMD module shall not increase the probability of collision with other space objects.
- The PMD module shall be able to remove different S/C (but only one S/C per PMD module), and as a

goal the PMD module shall be able to remove any S/C (but only one S/C per PMD module).

- The PMD module shall be able to remove a S/C in a reliable way after end of operation with a probability of success of at least 90 %.
- The PMD module shall be able to remove a S/C after end of operation in a cost efficient way.

3. Taxonomy and Survey of De- and Re-Orbiting Strategies and Technologies

Prior to beginning an analysis of the possible concepts for the PMD, a taxonomy was developed in order to capture all possible de/re-orbiting strategies. This is shown in Fig. 1. To build the taxonomy the question of what is the overarching aim of this project must first be posed. The answer being: to make space safer to operate in. This project aims to do this by designing a device or series of devices to de- or re-orbit spacecraft post-mission. However, the fundamental principle of the problem is actually to reduce the probability of collisions occurring, therefore this taxonomy is built beginning there. The taxonomy then branches into all the different methods possible to achieve this goal.

This taxonomy was then used to produce a list of concepts by mapping concepts to the taxonomy, as shown in Fig. 2. Some of these are already present in the literature but other, more unconventional concepts can also be generated such as the idea of heating the entire atmosphere in order in increase the atmospheric drag experienced by spacecraft.

It can be seen from Fig. 1 that there are only 4 fundamental methods to alter the probability of collision and of those altering the orbital energy has the most potential applications. Fig. 2 also highlights that this strategy has produced the most concepts within the literature. Fig. 2 was then used to determine where there were gaps, such as utilising the weak force in some manner. These gaps could be further explored to determine if there are any other concepts possible.

4. Qualitative Analysis

To enable a detailed analysis all de- and re-orbiting strategies and technologies are summarised in Table 6 (see end) to allow qualitative comparison. The 'In Lit' column defines whether a concept appears in the current literature. The cells are colour coded to ease comparison, green is good, yellow is medium, red is poor and black is inapplicable. The columns for 'Control', 'Passive' and 'S/C Mass' are not colour coded as favourable content within these columns is contextual and cannot be generalised.

The orbital environment is sub-divided into a number of orbit categories for this analysis, as summarised in Table 1. The final column pertains to the removal of space objects under ISO 24113.[2]

Table 1. Orbit Regimes

ID	Name	Altitude (km)	Removal Required
1.1	Low LEO	100 – 350	Yes
1.2	ISS Region	350 – 450	Yes
1.3	Medium LEO	450 – 1000	Yes
1.4	High LEO	1000 – 2000	Yes
2.1	MEO	2000 – 19000 & 24000 – 35586	No
2.2	GNSS Region	19000 – 24000	No

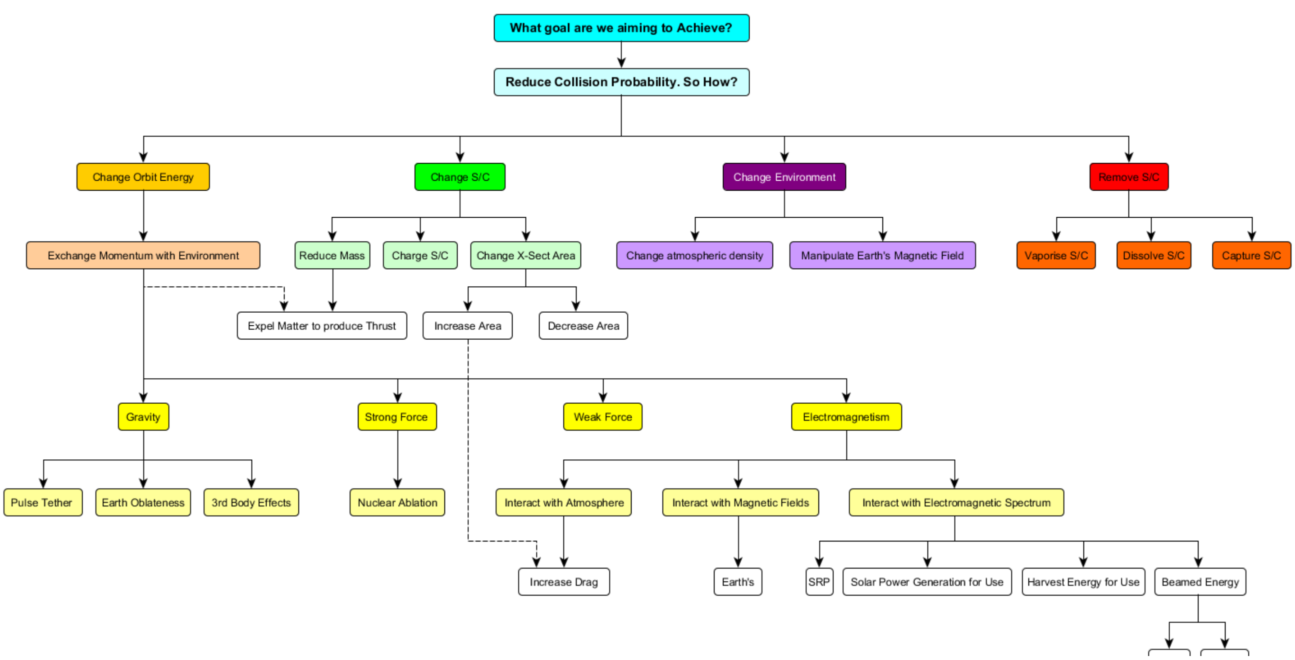

Fig. 1. Taxonomy of De/Re-Orbiting Strategies.

3	Geosynchronous	35586 – 35986	Yes
4	Supersynchronous	35956 – 45000	No

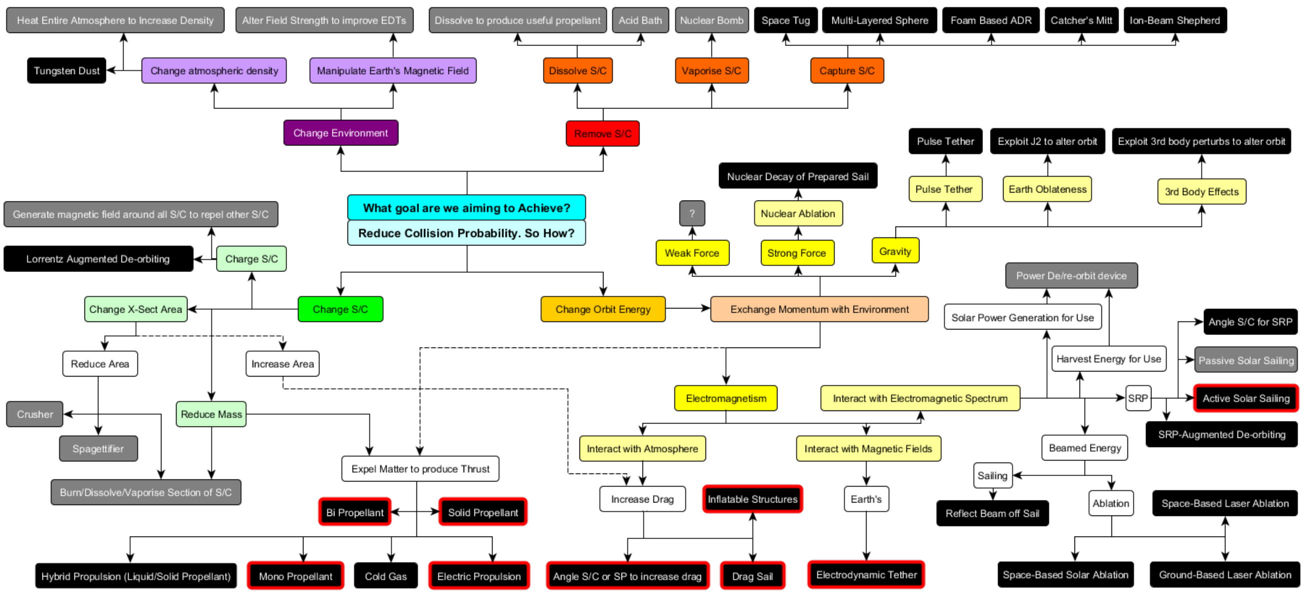

Fig. 2. Taxonomy of De/Re-Orbiting Strategies including concepts. Note concepts in grey are concepts not seen before in the literature, concepts in black represent those previously studied and concepts outlined in red represent those initially down-selected.

method of re-orbiting is required to avoid unforeseen collisions and subsequent legal issues. Each de-orbiting concept, strategy and/or technique will be rated on its result as, Controlled, Semi-Controlled and Uncontrolled (C, S or U respectively). In addition, each de-orbiting concept, strategy and/or technique will be rated as passive or not, based on whether it would require active spacecraft operations to achieve the desired de- or re-orbit.

Each de-orbiting concept, strategy and/or technique will be rated on its applicability within each orbital regime as, Inapplicable, Low, Medium and High (I, L, M and H respectively).

Considering all de-orbiting concepts, strategies and/or techniques it is noted that three distinct types of method can be defined to aid the characterisation, comparison and evaluation of re-orbiting technology.[3] These types are: Type A (method applied to an individual piece of debris, typically incorporated in the initial system development but could also be attached to piece of debris or a non-operational spacecraft), Type B (active debris removal: method remotely applied to an individual piece of debris or a non-operational spacecraft) and Type C (method applied universally to all objects within a certain region).

It is also noted that three distinct solutions occur; specifically the result of the technology is a controlled, semi-controlled or uncontrolled re-orbit or de-orbit. When large spacecraft de-orbit significant fragments can be expected to reach the Earth's surface, therefore, it is clear that for large spacecraft a controlled, or at least semi-controlled, de-orbit is highly desirable to ensure minimum risk to human life, typically by ensuring a splashdown in the southern Pacific Ocean 'spacecraft cemetery'. Meanwhile, for small spacecraft, where complete destruction is ensured due to atmospheric heating, an uncontrolled de-orbit is acceptable. It is noted that when a spacecraft is being re-orbited a controlled

The concept of technology readiness levels (TRL) are used widely in aerospace to assess and define the maturity of a technical concept, capability or product. For the comparison of de-orbiting concepts, strategies and/or techniques TRL bands will be applied in-place of detailed TRL analysis, these bands will be, Low (TRL 1-4), Medium (TRL 3-7) and High (TRL 6-9).

The Advancement Degree of Difficulty system provides nine levels of risk, associated with the advancement of a technology from one TRL to the next. For the comparison of de- and re-orbiting concepts, AD2 bands will be applied in-place of detailed AD2 analysis, these bands will be, Low (AD2 1-4), Medium (AD2 3-7) and High (AD2 6-9).

De- and re-orbiting concepts, strategies and/or techniques will be categorised for mass efficiency as, Low (>15% of total end-of-life mass fraction), Medium (5–15% of total end-of-life mass fraction) and High (<5% of total end-of-life mass fraction).

De- and re-orbiting concepts, strategies and/or techniques will be categorised for volume efficiency as, Low (>15% of total end-of-life volume), Medium (5–

15% of total end-of-life volume) and High (<5% of total end-of-life volume).

De- and re-orbiting concepts, strategies and/or techniques will be categorised for insensitivity to end-of-life orbit eccentricity and inclination, simply as Low, Medium and High insensitivity.

De- and re-orbiting concepts, strategies and/or techniques will be categorised for suitability to three different spacecraft end-of-life mass ranges. These are, Low (<50kg), Medium (50–1000kg) and High (>1000kg).

5. Initial Down-Selection of Concepts

At this initial stage, a down-selection can be performed on evidently unviable strategies based on the user, mission and system requirements detailed in section 2.

In order to address these requirements, all strategies not consisting of a technology to be placed on a spacecraft to de-orbit or re-orbit it are removed. This eliminates all Type B and C concepts. Furthermore, all concepts with a low TRL or high AD2 are removed at this point as they are unlikely to result in the success rate required. Finally, cold gas is removed due to its low mass and volume efficiency, and the remaining solar radiation pressure concepts, excluding active solar sailing, are removed due to their high sensitivity to orbit eccentricity and inclination.

Therefore the concepts initially down-selected are: Atmospheric Drag Augmentation; Electrodynamic Tethers; Propulsion, High Thrust (Mono-Propellant, Bi Propellant, Solid Rocket) and Low Thrust (Electric Propulsion); Active Solar Sailing.

These concepts will be taken forward for further study to determine the limitations of each technology. Combinations of these technologies will also be considered.

6. Quantitative Analysis of Down-Selected Concepts

In order to provide a quantitative comparative assessment of the performance of the down-selected concepts a series of analytical solutions have been developed to provide the approximate sizing for the various concepts. In order to complete this analysis certain parameters in the analysis had to be explicitly defined. The first of these was the timescale of removal. Therefore the time for removal was set at 1 year for fast removal and 25 years for slow removal in all cases.

The analysis completed herein assumes a quasi-circular low thrust spiral affected by air drag, solar radiation pressure or low-thrust electric propulsion. For air drag de-orbit it is assumed that the drag device is stabilised and is always normal to the velocity vector. For the solar sail, two steering laws are required, either for near-equatorial or near-polar orbits.

In order to compare all the different concepts a common parameter must be studied, in this case the comparison is drawn using the payload mass each system can de- or re-orbit from a range of altitudes. This payload mass is calculated assuming the de-orbit technology is a standalone system so is essentially a post mission disposal module attached to a spacecraft of the calculated payload mass. Three different scenarios are considered: de-orbit from LEO, re-orbit from LEO to 2000km and re-orbit from GEO to 40000km. De-orbit from GEO could be studied however it is considered to be inefficient compared to re-orbit. De- and re-orbit from all regions discussed in Section 4 could be considered using this analysis. However removal from LEO and GEO regions are of paramount concern as these are the protected regions as discussed in ISO24113 so removal is required.[2]

6.1 Analytical Models

6.1.1 Drag Augmentation

Whether Active or Passive it is assumed that the maximum area required to de-orbit within the required time would be the same for any drag augmentation device. The active system may complete a few more or less revolutions at the very end of the deorbit period; however, it is assumed that this would not significantly affect the area required. Therefore, the active and passive systems have the same sizing procedure. It is assumed that drag devices are only used in the low-mid LEO region below 1000km, as above this, the area of device required to deorbit even small spacecraft becomes prohibitively large. Given the nature of atmospheric drag it is assumed that drag augmentation always results in de-orbit therefore re-orbit concepts including drag are not considered.

The time required to de-orbit can be calculated many ways, both numerically and analytically. Here an analytical solution developed in house is utilised. This solution includes analytical models of atmospheric density and solar activity. However, for the purposes of this study it is assumed that solar activity is at a constant moderate level, in order to keep the fidelity of the drag model equivalent to the models for solar sailing and propulsion. [4–8]

Assuming a quasi-circular orbit the de-orbit lifetime can be calculated as

$$Lifetime = \frac{H\,T_0\,M}{2\pi\,\rho_0\,a_0^2\,F\,A\,C_D}\left[1 - e^{\frac{a_0}{H}\left\{\left(\frac{T_f}{T_0}\right)^{\frac{2}{3}} - 1\right\}}\right] \quad (1)$$

where T_0 and T_f are the initial and final orbit periods, a_0 is the initial orbit semi-major axis, ρ_0 is the local total atmospheric density at the initial altitude, H_0 is the scale height at the initial altitude, F is a parameter used to take atmospheric rotation into account, C_D is the drag coefficient (assumed to be 2.2), A is the area of the spacecraft projected in the flight direction and M is the mass of the spacecraft. By rearranging this equation the area-mass ratio required to de-orbit within a given time can be calculated. Then by assuming the device used will be a drag sail of a specific sail configuration the payload mass that that sail could deorbit in a given time can be calculated.

6.1.2 *Solar Sailing*

The boundary between the dominance of atmospheric drag and solar radiation pressure has been studied and it has been shown that the changeover altitude varies depending on the solar intensity, specifically the solar flux output. For the purposes of this study, it is assumed that solar sailing would not be used below 700 km altitude. Therefore, only re-orbit concepts are considered using solar sailing. Appropriate steering laws must be applied; the two extremes, equatorial and polar orbits, are used for comparison. For equatorial orbits, a simple switching law can be used which requires a sail slew of 90 degrees twice per orbit. For polar orbits, the sail attitude can be fixed relative to the Sun, but the sail must yaw 360 degrees per orbit to align the sail thrust vector opposite to the velocity vector. For near-equatorial orbits, a switching law is used which requires a sail slew of 90º twice per orbit. It can be shown that the steering law efficiency is $\eta=\pi$ which accounts for the loss of thrust due to the switching law. For near-polar orbits, the sail attitude is fixed relative to the Sun and yaws 360 degrees per orbit to align the sail thrust vector opposite to the velocity vector. Here, the steering law efficiency is $\eta=2.83$ which accounts for the loss of thrust due to the pitch of the sail relative to the Sun-line.

The removal time for a solar sail can be estimated by considering the work done by the force on a spacecraft of mass, M, and total sail area, A, on a circular orbit of radius, R. For a continuous, quasi-circular low thrust spiral from some initial orbit radius, R_0, to some final orbit radius, R_f, the removal time can be calculated as

$$Lifetime = \eta \frac{M}{2PA}\left|\sqrt{\frac{\mu}{R_f}} - \sqrt{\frac{\mu}{R_0}}\right| \tag{2}$$

where A is the area of the sail, M is the mass of the spacecraft, P is the solar radiation pressure, and μ is the gravitational parameter of the central body.

As with drag augmentation by rearranging this equation the area-mass ratio required to de- or re-orbit with a given lifetime can be calculated. Then by assuming a specific sail configuration the payload mass that a given sail could de- or re-orbit in a given lifetime can be calculated.

6.1.3 *Electrodynamic Tether*

Given an EDTs dependence on the Earth's magnetic field, the efficiency of such a system falls as altitude increases. It is therefore assumed that the EDT is used only in LEO for both de- and re-orbit. In order to account for the variation in tether attitude, it is assumed that the tether is aligned off the local vertical at a mean angle of approximately 35° and is kept in tension. For a conducting tether of length L_T the time required to de- or re-orbit is given by

$$Lifetime = \frac{\left(R_f^6 - R_0^6\right)}{12\,\varepsilon\,L_T}\,\frac{M}{R_E^6\,B_0^2}\,\frac{\rho}{A_T} \tag{3}$$

where R_0 is the initial orbit radius, R_f is the final orbit radius, ε is the geometric efficiency factor, M is the mass of the spacecraft, and R_E is the radius of the Earth and A_T is the cross-sectional area of the tether. The tether is assumed to have a diameter of 2 mm, the assumed field strength is B_o=3.5E-5T and the resistivity of aluminium is assumed to be $\rho\approx$2.8E-8 Ohm-m. Rearranging this equation the payload mass that a given tether could de- or re-orbit in a given lifetime can be calculated. It should be noted that this equation will give a very approximate result.

6.1.4 *Low Thrust Propulsion*

These concepts are considered valid for both de- and re-orbit and are suitable for use in any region.

Again using a similar analysis to drag and solar sail sections, the decay time for continuous low thrust electric propulsion can be determined. The time required to de- or re-orbit from some initial orbit radius R_0 to some final orbit radius R_f with a given thrust F can be calculated as

$$Lifetime = \frac{M}{F}\left|\sqrt{\frac{\mu}{R_0}} - \sqrt{\frac{\mu}{R_f}}\right| \tag{4}$$

where μ is the gravitational parameter of the central body. By rearranging this equation the total mass that can be removed can be calculated and the required propellant mass can be determined. Finally, the mass of the propulsion system can then be estimated using the method developed by Ceriotti, Heiligers and McInnes, allowing a final estimation of the payload mass a given system can de- or re-orbit in a given timeframe. [9]

6.1.5 *High Thrust Propulsion*

As with low thrust propulsion these concepts are considered valid for both de- and re-orbit and are suitable for use in any region. However unlike the low thrust manoeuvre which follows a spiral trajectory, the high thrust manoeuvre is based on the Hohmann transfer method.

In the case of high thrust propulsion the lifetimes required are considerably shorter and the mass of propellant required is not linked to the lifetime but instead to the total burn time of the propulsion system. Therefore the analysis is conducted in a slightly different manner. The dry mass of the spacecraft is then calculated as

$$m_{dry} = \frac{F\,t}{g\,I_{sp}}\left(e^{\frac{\Delta V}{g\,I_{sp}}} - 1\right) \tag{5}$$

where F is the thrust, g is acceleration due to gravity of the Earth, I_{sp} is the specific impulse, t is the total burn time, and ΔV is the required change in velocity. Then using an estimation of the mass of the propulsion system the payload mass can be estimated.

7. Analysis

7.1 Drag Augmentation

Many types of drag augmentation devices exist such as sails, balloons etc., however in order to calculate the payload masses required for system comparisons it is assumed that a simple square drag sail is deployed. The payload mass that can be de-orbited by various sizes of sails within the given 1 year and 25 year timeframes can be compared against the payload mass that can be de-orbited without the addition of the sail. These payload masses are the masses that could be carried by a standalone de-orbit system, so are the total mass de-orbited minus the sail assembly mass. This sail assembly mass is calculated by assuming a Mylar sail of 10µm thickness, then multiplying by a factor to account for the mass of the deployment mechanism, boom masses etc. The assembly loading used is approximately 32g/m^2. This sail assembly loading, as with others later in the document, is a best case scenario and represents an aggressive design point. Therefore a second assembly loading of 150g/m^2, based on current CubeSat flight models, is also studied. Four sail sizes are studied; these are sails areas of 25m^2, 100 m^2, 400m^2 and 1200m^2. If the masses that each sail can de-orbit are plotted against initial altitude it could be seen that including a drag sail significantly increases the available payload mass. Alternatively, for a given payload mass the maximum initial altitude the satellite could launch to while still adhering to the prescribed de-orbit period is significantly increased and the available payload mass is also increased for the 25 year de-orbit period compared to the 1 year period. These conclusions can also be drawn from the information in Table 2. Note, in Table 2, for the sail concepts a range is given based on the best and worst case sail assembly loading.

7.2 Solar Sailing

Again assuming that a square sail is deployed the mass that can be re-orbited by various sizes of sails within the given 1 year and 25 year timeframes can be compared. In this case the sail assembly loading used is approximately 19g/m^2, calculated in the same manner as the drag sail assembly loading but assuming a 5µm thick Mylar sail with a 0.5µm coating of Aluminium. As with the drag sail this is a best case scenario so a second assembly loading of 150g/m^2 is also studied. The same four sail sizes are used as in the drag augmentation analysis. When studying the results in Table 2, or graphically the payload mass that can be re-orbited from LEO to 2000km, or from GEO to 40000km in the 1 and 25 year periods, it can be seen that increasing the size of the solar sail included significantly increases the available payload mass. Alternatively for a given payload mass the minimum initial altitude the satellite could launch to while still adhering to the prescribed de-orbit period is significantly decreased. As expected, the available payload mass is also increased for the 25 year de-orbit period compared to the 1 year period. However the 1 year period is already sufficient time to re-orbit a 1 tonne spacecraft from any altitude to 2000km using a 25m^2 sail, for the GEO region again a 25m^2 sail is sufficient to re-orbit a 1 tonne spacecraft to 40000km. Therefore it is recommended that Solar Sailing be used only for short re-orbit periods.

7.3 Electrodynamic Tether

Assuming a simple wire tether with circular cross-section is deployed the mass that can be de- or re-orbited by various tether lengths within the given 1 year and 25 year timeframes can be compared. As with the Drag and Solar Sails an approximation is used to calculate the mass of the system. It is assumed here that an Aluminium tether of 2mm diameter is deployed. The mass of the tether is then multiplied to include the mass of the deployment system. It should be noted that this approximation is very loose as currently tether technology is still developing rapidly, and is only being used on small spacecraft so this approximation should be reconsidered as more data becomes available. Tether lengths of 1m, 100m, 1km and 10km are considered in order to cover a full range of possible spacecraft from CubeSats all the way up to large communications satellites. When studying the payload masses that can be de- or re-orbited from LEO in the 1 and 25 year periods, graphically or from the information given in Table 2, it can be seen that increasing the tether length increases the available mass. In this case depending on the tether length and payload mass the 25 year period may be appropriate.

7.4 Low Thrust Propulsion

Three different systems of varying specific impulse and thrust are considered. The first system is based on the Astrium RIT-10 with I_{sp}=3200s and F=0.015N, the second is based on the Astrium RIT-XT with I_{sp}=4500s and F=0.15N and the third is based on the more advanced futuristic JPL Nexis with I_{sp}=8500s and F=0.5N. Low thrust propulsion could be used in any capacity, to de- or re-orbit from either of the protected regions. If the payload mass was plotted against initial altitude, it could be seen that the second system is sufficient to complete either the de-orbit or re-orbit to 2000km of a 1 tonne spacecraft within the 1 year period and the first system is sufficient to complete the re-orbit to 40000km of a 1 tonne spacecraft within the 1 year period. It is therefore recommended that low thrust propulsion be used for short duration de-orbit periods. These conclusions can also be drawn from Table 2.

7.5 High Thrust Propulsion

Again three different systems are considered this time of varying propellant types. The first system is a mono-propellant system with I_{sp}=230s and F=5N, the second is a bi-propellant system with I_{sp}=300s and F=10N and the third is a solid propellant system with I_{sp}=300s and F=25kN. Since high-thrust propulsion systems do not thrust continuously it is assumed that these systems would only be used for short duration de- and re-orbit periods. In order to calculate the required propellant the maximum operational (burn) time of each system are assumed as 500000s, 500000s and 100s respectively. Again, if the available payload mass is plotted against initial altitude, or by examining the information given in Table 2, it can be seen that the mono-propellant and solid propellant systems give very comparable answers while the bi-propellant system can de- or re- orbit significantly more mass.

7.6 Possible Combination concepts

Two different approaches can be taken in combining technologies: technologies could be used sequentially or they could be used concurrently. Due to the involved nature of concurrent concepts, only sequential concepts are considered herein.

Table 2. Parametrically derived payload masses (kg) systems can remove (blank entries indicate system inapplicable)

	1 year			25 years		
	de-orbit from 1000km	re-orbit 1000km-2000km	re-orbit 30000km-40000km	de-orbit from 1000km	re-orbit 1000km-2000km	re-orbit 30000km-40000km
drag without sail	0			0		
25m^2 drag sail	0-0.4			27-30		
100m^2 drag sail	0-1.8			110-121		
400m^2 drag sail	0-7			438-485		
1200m^2 drag sail	0-21			1315-1456		
25m^2 solar sail		5602-5606	6707-6710		140150-140153	167764-167767
100m^2 solar sail		22410-22423	26828-26841		560599-560612	603950-603961
400m^2 solar sail		89638-89691	107311-107364		2242395-2242448	2684221-2684273
1200m^2 solar sail		268915-269072	321934-322091		6727186-6727343	8052662-8052818
1m EDT	6	3		138	65	
100m EDT	551	260		13794	6529	
1km EDT	5509	2604		137938	65293	
10km EDT	55094	26036		1379384	652932	
LowT 0.015N 3200s	920	1040	1246	23472	26492	31632
LowT 0.15N 4500s	9160	10272	12329	235820	263622	315032
LowT 0.5N 8500s	29877	33214	40069	789809	873240	1044614
HighT MonoProp	4525	4986	6089	4525	4986	6089
HighT BiProp	9291	10216	12424	9291	10216	12424
HighT Solid Prop	4646	5108	6212	4646	5108	6212
25m^2 combi sail	50-53			1351-1353		
100m^2 combi sail	202-213			5402-5414		
400m^2 combi sail	807-853			21608-21654		
1200m^2 combi sail	2420-2558			64825-64962		
6m LowT -> Drag	682			1108		
1y LowT -> Drag				2237		
2y LowT -> Drag				4215		
5y LowT-> Drag				9949		
HighT -> Drag	6968			11563		

7.6.1 Sequential Combination Concepts

The addition of a second system increases the complexity of a concept and increases the risk of failure so given the efficiency of the individual systems the first combination solutions worth consideration are those that do not require additional mass. Another option worth consideration is the addition of a small high thrust propulsion system to be used only to control re-entry in the uncontrolled systems on large spacecraft. The addition of this small propulsion system is assumed to be of negligible mass when compared to the payload masses available. Therefore these concepts are not studied further herein. The only sequential combination concepts not requiring additional mass therefore are the combination of solar and drag sailing as the same sail could be used for both, and the use of a propulsion system to lower the altitude to a point where the spacecraft will naturally decay due to drag in the prescribed time period. These concepts are discussed further in the following sections.

7.6.2 Solar/Drag Sailing

Assuming that a square sail is deployed for solar sailing down to 700km (where atmospheric drag becomes dominant) then drag-augmentation to re-entry the mass that can be de-orbited by various sizes of sails in 1 year or 25 years can be compared. In this case a 10µm Mylar sail with 0.5µm coating of Aluminium is assumed giving a sail loading of 35g/m^2. As with the drag and solar sails a second, more conservative, sail assembly loading of 150g/m^2 is also studied. The same four sail sizes are used as in the drag augmentation and solar sailing sections. If the payload mass was plotted against the initial altitude, it could be seen that, because the solar sails are significantly more efficient than the drag sails, the payload mass the sails can de-orbit is determined by the payload mass that the drag sails can de-orbit from 700km. The combination sail also allows de-orbit from above 1000km where the drag sails are ineffective, extending the use case for drag sails. Comparing the results given in Table 2 it can be seen that the combination sails perform better than the original drag sails.

7.6.3 Low Thrust Propulsion -> Natural Decay Due To Drag

As low thrust propulsion requires active control during operation its applicability to long duration missions is limited due to the risk of failure. However, if the low thrust system was used for a shorter duration, for example 1 year, to lower altitude before the spacecraft is allowed to decay naturally due to drag the risk of failure is reduced, thus increasing the applicability of the technology. For the case of the 1 year de-orbit period it is assumed that propulsion is used for the first 6 months only, while for the 25 year period four options are considered, these are that propulsion is used for 6 months, 1 year, 2 years or 5 years. If the payload mass was plotted against the initial altitude, it could be seen that the use of the propulsion system in combination with drag would increase the payload mass or allowable initial altitude slightly. However, this benefit is only gained in the range of 400-600km therefore the increase in applicability is marginal. Though, it could also be seen that while using propulsion in combination with drag is not as efficient as propulsion alone there is a marked increase in the payload mass and initial altitude allowable between the 1 year pure propulsion and 1 year propulsion with 24 year drag. The use of propulsion for 1 year followed by the 24 year drag period would have the same risk of failure as the 1 year pure propulsion so the increase in applicability is clear, however it should be noted that as with all concepts the longer the spacecraft is allowed to remain in space the more likely it is to collide with other spacecraft.

7.6.4 High Thrust Propulsion -> Natural Decay Due To Drag

As high thrust propulsion is assumed herein to be used for a single impulse manoeuvre its applicability to long duration missions is limited. However, if the high thrust system was used simply to lower altitude before the spacecraft is allowed to de-orbit due to drag the applicability of the technology is increased. Again, if the payload mass was plotted against the initial altitude it could be seen that the use of the propulsion system in combination with drag would increase the payload mass or allowable initial altitude for either de-orbit periods.

7.7 Comparison of Concepts

The previous sections focussed the comparison of various sizes of systems within a specific group, this section deals with the comparison of the various different concepts. As comparing all systems would create a series of complex graphs, one system has been chosen for each concept. Figures 3 and 4 show the payload mass that one system from each concept can de-orbit within 1 year and 25 years respectively.

It can be seen that for the 1 year period the EDT and high thrust propulsion systems seem to be the most promising while for the 25 year period low thrust propulsion surpasses high thrust but the EDT still appears most promising.

The combination sail still does not compare to the EDT and propulsion methods for the 1 year de-orbit, however the high thrust propulsion and drag concept is seen to be more efficient. For the 25 year de-orbit the 100m^2-1200m^2 combination sails perform better than the high thrust propulsion but the EDT and low thrust propulsion methods still prove most promising. However, again it can be seen that the high thrust propulsion and drag concept is more promising than the high thrust propulsion alone.

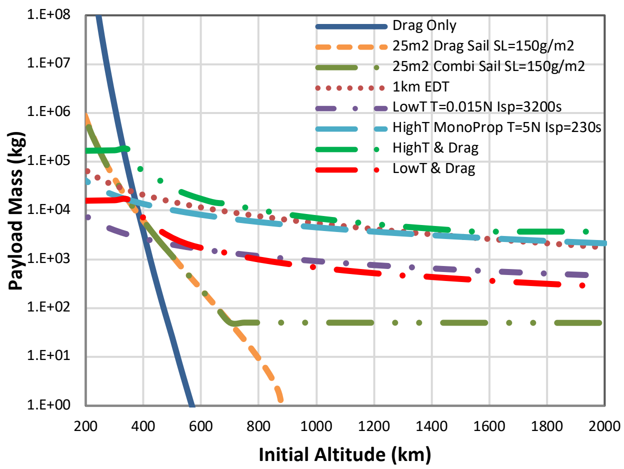

Fig. 3. Comparison of concepts for 1 year de-orbit

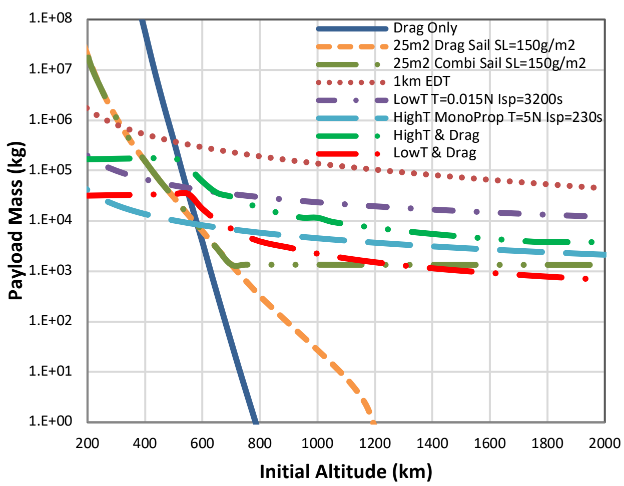

Fig. 4. Comparison of concepts for 25 year de-orbit

The combination sail still does not compare to the EDT and propulsion methods for the 1 year de-orbit, however the high thrust propulsion and drag concept is seen to be more efficient. For the 25 year de-orbit the 100m^2-1200m^2 combination sails perform better than the high thrust propulsion but the EDT and low thrust propulsion methods still prove most promising. However, again it can be seen that the high thrust propulsion and drag concept is more promising than the high thrust propulsion alone.

Figures 5 and 6 show the payload masses that one system from each concept can re-orbit to 2000km within 1 year and 25 years respectively. It can be seen that for the 1 year period the solar sail and high thrust propulsion seem to be the most promising while for the 25 year period the EDT and low thrust propulsion surpass high thrust but the solar sail still appears most promising.

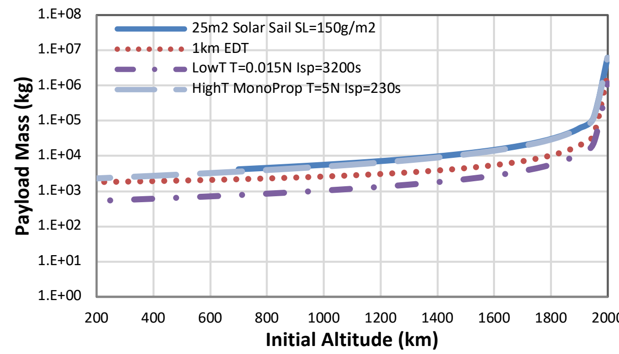

Fig. 5. Comparison of concepts for 1 year re-orbit to 2000km

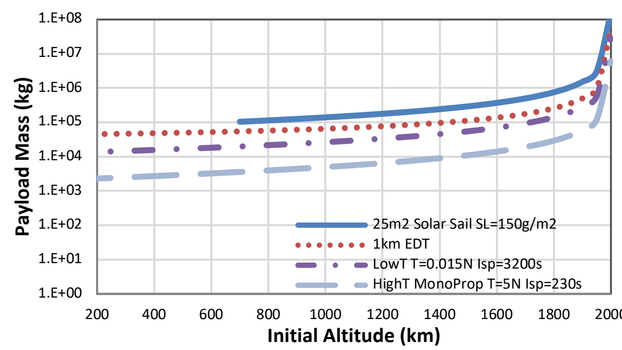

Fig. 6. Comparison of concepts for 25 year re-orbit to 2000km

Figures 7 and 8 show the payload masses that one system from each concept can re-orbit to 40000km from GEO within 1 year and 25 years respectively. It can be seen that for the 1 year period the solar sail and high thrust propulsion seem to be the most promising while for the 25 year period low thrust propulsion surpasses high thrust but the solar sail still appears most promising.

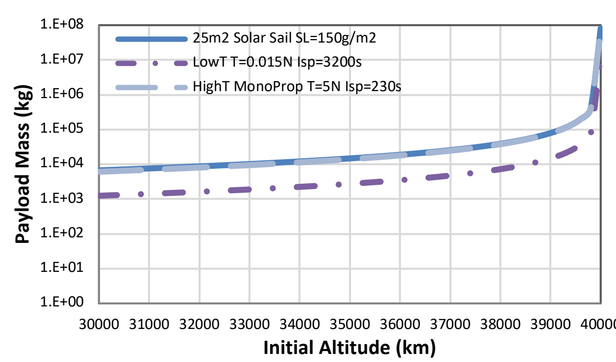

Fig. 7. Comparison of concepts for 1 year re-orbit to 40000km

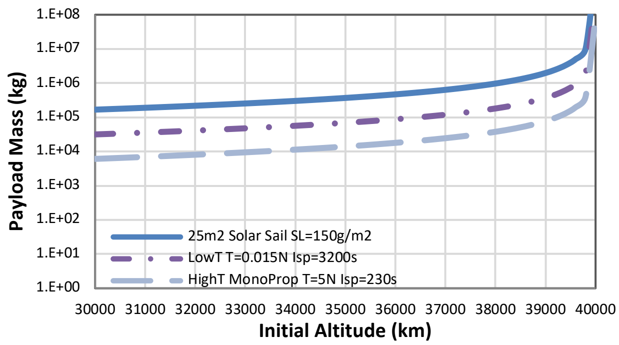

Fig. 8. Comparison of concepts for 25 year re-orbit to 40000km

Figures 3-8 give an interesting overview of how the various systems compare, however due to the exclusion of the various differently sized systems the overview is incomplete. Table 2 shows the payload mass that all of the systems considered can remove for a given orbit in a given timeframe for a more direct comparison.

The applicability of the concepts can be summarised briefly for comparisons sake by combining the parametric results and initial qualitative analysis, as can be seen in Table 3-Table 5. In these tables each concept is denoted by its initial; D(rag), S(olar Sailing), E(lectrodynamic Tether), L(ow Thrust Propulsion), H(igh Thrust Propulsion), C(ombination Sail), LD(Low Thrust Propulsion and Drag) and HD(High Thrust Propulsion and Drag). Each concept is then colour coded for applicability, green being good, yellow moderate and red bad. It can be seen that none of the down-selected concepts have been recommended for the 25 year re-orbit manoeuvres, as each system considered is active, therefore the failure risk increases over the long duration.

Table 3. Concept Applicability Comparison for de-orbit from LEO

	1 year								25 years							
	D	S	E	L	H	C	LD	HD	D	S	E	L	H	C	LD	HD
<1kg	H	/	H	L	M	H	L	M	H	/	NR	NR	NR	H	L	M
1-10kg	H	/	H	L	M	H	L	M	H	/	NR	NR	NR	H	L	M
10-100kg	M	/	H	M	M	H	M	M	H	/	NR	NR	NR	H	M	M
100-500kg	L	/	H	M	H	H	M	H	M	/	NR	NR	NR	H	M	H
500-1000kg	L	/	H	H	H	H	H	H	M	/	NR	NR	NR	H	H	H
1000-2000kg	L	/	M	H	H	H	H	H	M	/	NR	NR	NR	H	H	H
>2000kg	L	/	M	H	H	M	H	H	L	/	NR	NR	NR	H	H	H

Table 4. Concept Applicability Comparison for re-orbit from LEO

	1 year								25 years							
	D	S	E	L	H	C	LD	HD	D	S	E	L	H	C	LD	HD
<1kg	/	M	H	L	M	/	/	/	/	NR	NR	NR	NR	/	/	/
1-10kg	/	M	H	L	M	/	/	/	/	NR	NR	NR	NR	/	/	/
10-100kg	/	M	H	M	M	/	/	/	/	NR	NR	NR	NR	/	/	/
100-500kg	/	M	H	M	H	/	/	/	/	NR	NR	NR	NR	/	/	/
500-1000kg	/	H	H	H	H	/	/	/	/	NR	NR	NR	NR	/	/	/
1000-2000kg	/	H	H	H	H	/	/	/	/	NR	NR	NR	NR	/	/	/
>2000kg	/	H	M	H	H	/	/	/	/	NR	NR	NR	NR	/	/	/

Table 5. Concept Applicability Comparison for re-orbit from GEO

	1 year								25 years							
	D	S	E	L	H	C	LD	HD	D	S	E	L	H	C	LD	HD
<1kg	/	M	/	L	M	/	/	/	/	NR	/	NR	NR	/	/	/
1-10kg	/	M	/	L	M	/	/	/	/	NR	/	NR	NR	/	/	/
10-100kg	/	M	/	M	M	/	/	/	/	NR	/	NR	NR	/	/	/
100-500kg	/	M	/	M	H	/	/	/	/	NR	/	NR	NR	/	/	/
500-1000kg	/	H	/	H	H	/	/	/	/	NR	/	NR	NR	/	/	/
1000-2000kg	/	H	/	H	H	/	/	/	/	NR	/	NR	NR	/	/	/
>2000kg	/	H	/	H	H	/	/	/	/	NR	/	NR	NR	/	/	/

8. Findings and Recommendations

The analysis provided here captures each system's maximum potential, so while solar sailing and electrodynamic tethers appear the most promising that is not necessarily to say that they are the best systems. Each system has strengths and weaknesses; for example high thrust propulsion is the most promising concept in terms of producing a controlled re-entry. Take for example a nanosat of mass 10kg, with a requirement to de-orbit within 25 years. Adding a large propulsion system would be wasteful as it would add unnecessary additional mass. The system is also active requiring operations throughout the lifetime adding complexity and cost. It is therefore recommended that systems should be chosen based on the spacecraft to be removed, and the initial proposed orbit as well as by balancing which system appears most promising with which is most cost efficient and least likely to fail. The analysis performed herein will allow a user to rule out some concepts however, for example if the spacecraft is of mass 1000kg and is to de-orbit within a year the analysis concludes that a drag sail will not be sufficient therefore other concepts must be considered. Sections 8.1-8.8 give a summary of each concept studied.

8.1 Drag Sail

Drag Sails should only be used to de-orbit rather than re-orbit.

For a 1 year de-orbit the benefit of including a drag sail begins at approximately 300km, beyond this altitude including a drag sail can either increase the allowable payload mass to be launched to a specific initial altitude or increase the allowable initial altitude for a specific payload mass to be launched to. As the initial altitude increases the mass benefit continuously increases until it reaches a peak at the altitude where no mass could be de-orbited without a sail (approximately 750km) then steadily decreases. Conversely the benefit in the initial allowable altitude can be seen to increase as payload mass decreases. This means that larger spacecraft see less benefit to the allowable initial altitude by including a drag sail than small spacecraft do.

For a 25 year de-orbit the benefits of including a drag sail follow the same trends as for the 1year de-orbit period. However for the 25 year period the benefits begin at approximately 550km and the peak mass benefit occurs at approximately 1000km.

A drag augmentation device could be deployed from a tumbling spacecraft, however typically the system would then have to stabilise its attitude therefore when employing drag if the system cannot passively stabilise itself an attitude control system should be included.

It may also be possible to reduce the spin rate sufficiently, when using a drag sail, by deploying the booms thus increasing the moment of inertia. It should be noted however that this method would only work if the spacecraft is tumbling relatively slowly.

Drag augmentation generally provides an uncontrolled re-entry so a re-entry control system, most likely a small propulsion system, should be included in order to control re-entry of large spacecraft in order to adhere to the requirement to minimise the casualty risk as large spacecraft are unlikely to burn up completely upon re-entry.

8.2 Solar Sail

Solar Sails should only be used for re-orbit rather than de-orbit, unless used in combination with a second de-orbiting technology.

A solar sail cannot be deployed from a tumbling spacecraft, therefore when using a solar sail an attitude control system should be included with the sail in case the spacecraft is tumbling to de-tumble before the sail can be deployed. As with the drag sail, it may be possible for the boom deployment to increase the moment of inertia enough in order to deploy the sail safely.

8.3 Electrodynamic Tether

Electrodynamic tethers are dependent on the Earth's magnetic field so they should only be used in the LEO region for either de- or re-orbit.

An EDT does not have the same efficiency for de- and re- orbit, the altitude of equal efficiency for de- and re-orbit calculated in this parametric analysis is approximately 1325km. Therefore below 1000km de-orbit is recommended while above 1500km re-orbit is recommended.

An EDT cannot be deployed from a tumbling spacecraft, therefore when using an EDT an attitude control system must be included in the system in case the spacecraft is tumbling to de-tumble before the EDT can be deployed.

Typically an EDT provides an uncontrolled re-entry so when using an EDT a re-entry control system, most likely a small propulsion system, must be included in order to control re-entry and adhere to the requirement to minimise the casualty risk.

It may, however, be possible to control re-entry by jettisoning the tether prior to re-entry thus creating a momentum exchange which could be exploited to affect a controlled or at least semi-controlled re-entry.

8.4 Low Thrust Propulsion

Low thrust propulsion can be used in any region.

Low thrust propulsion is just as efficient for de-orbit and re-orbit, therefore in the LEO region if the spacecraft is above 1000km re-orbit is recommended while below 1000km de-orbit is recommended if the low thrust propulsion system has sufficient thrust to effect a semi-controlled re-entry. However if the system cannot effect at least a semi-controlled re-entry it is recommended that either a second system be included to control re-entry or re-orbit should be completed.

A low thrust propulsion system with sufficient attitude awareness could be used to de-tumble therefore a low thrust system could be used on tumbling spacecraft.

While low thrust propulsion appears to be one of the more promising systems in the frame of the TeSeR project it has some challenges. A low thrust propulsion system requires a large power system which can limit its use to large spacecraft. Also the possibility of a failure occurring increases as system complexity increases and low thrust propulsion is potentially the most complex system of the down-selected concepts.

8.5 High Thrust Propulsion

High thrust propulsion can be used in any region.

High thrust propulsion is just as efficient for de-orbit and re-orbit, therefore in the LEO region if the spacecraft is above 1000km re-orbit is recommended while below 1000km de-orbit is recommended if the high thrust propulsion system has sufficient control to effect at least a semi-controlled re-entry. However if the system cannot effect at least a semi-controlled re-entry it is recommended that either a second system be included to control re-entry or re-orbit should be completed.

A high thrust propulsion system with sufficient attitude awareness could be used to de-tumble therefore a low thrust system could be used on tumbling spacecraft.

Arguably high thrust propulsion systems, specifically solid propellant systems offer the most promise in the frame of the TeSeR project as they have a long shelf life so are less likely to fail even when used on spacecraft with long mission and long de-orbit lifetimes. Bi-propellant and Mono-Propellant systems are more complex and so again have an increased risk of failure. High Thrust systems also offer the most promise in terms of controlled re-entry.

8.6 Combination Sail

Due to the drag component combination sails should only be used to de-orbit rather than re-orbit.

For any de-orbit period the benefit of a combination sail over a drag sail begins in at approximately 700km allowing the spacecraft to be launched to higher initial altitudes.

A sail cannot be deployed from a tumbling spacecraft, therefore when using a combination sail an attitude control system should be included with the sail in case the spacecraft is tumbling to de-tumble before the sail can be deployed.

As with the drag and solar sails, a combination sail would typically provide an uncontrolled re-entry so a re-entry control system, most likely a small propulsion system, should be included in order to control re-entry and adhere to the requirement to minimise the casualty risk.

8.7 Combination Low Thrust Propulsion and Drag

Due to the drag component this concept should only be used to de-orbit rather than re-orbit.

The inclusion of the natural decay due to drag extends the use of low thrust propulsion to long duration de-orbit manoeuvres as well as the short durations low thrust alone could be used for.

For a 1 year de-orbit the benefit of including a period of natural decay due to drag is in the altitude range of approximately 400-700km. Beyond 700km the low thrust propulsion alone is more efficient and below 400km, drag alone is more efficient.

For the 25 year period the benefit of including a period of natural decay due to drag begins at approximately 550km. The benefit here it should be noted is a reduction in mass for an increased probability of success or in other words a decreased probability of failure.

8.8 Combination High Thrust Propulsion and Drag

Due to the drag component this concept should only be used to de-orbit rather than re-orbit.

As with the low thrust propulsion-drag concept, the inclusion of the natural decay due to drag extends the use of high thrust propulsion to long duration de-orbit manoeuvres as well as the short durations high thrust alone could be used for.

For a 1 year de-orbit the benefit of including a period of natural decay due to drag begins at approximately 350km. Above this altitude by including a period of drag there is an increase in the initial allowable altitude or the payload mass available.

For the 25 year period the benefit of including a period of natural decay due to drag begins at approximately 550km. Above this altitude by including a period of drag there is an increase in the initial allowable altitude or the payload mass available.

Acknowledgements
The work presented herein was funded by the European Commission in the H2020 framework (grant agreement number 687295). We would also like to thank the other partners involved in the project; Airbus DS GMBH, Weber-Steinhaus & Smith, Beazley Furlonge Limited, Universitaet der Bundeswehr Muenchen, D-Orbit SRL, Aalborg Universitet, University of Surrey, Gomspace APS, PHS Space Limited, and Hyperschall Techologie Göttigen GmbH.

References

1 “TeSeR – Technology for Self-Removal” Available: http://teserproject.eu/.
2 International Organization for Standardization,

ISO 24113:2011 Space systems - Space debris mitigation requirements, 2011.

3 Macdonald, M., McInnes, C., Bewick, C., Visagie, L., Lappas, V., and Erb, S., "Concept-of-Operations Disposal Analysis of Spacecraft by Gossamer Structure," *Journal of Spacecraft and Rockets*, vol. 52, Mar. 2015, pp. 517–525.

4 Kerr, E., and Macdonald, M., "Improving the Accuracy of Orbit Lifetime Analysis using Enhanced General Perturbation Methods," *26th AAS/AIAA Space Flight Mechanics Meeting*, Napa, CA, USA: 2016, pp. 1–14.

5 Kerr, E., and Macdonald, M., "General Perturbations Method for Orbit Lifetime Analysis Incorporating Non-Spherically-Symmetrical Atmospheres," *AIAA/AAS Astrodynamics Specialist Conference*, Reston, Virginia: American Institute of Aeronautics and Astronautics, 2016.

6 Kerr, E., and Macdonald, M., "Improving the Accuracy of General Perturbations Methods for Spacecraft Lifetime Analysis," *66th International Astronautical Congress*, Jerusalem, Isreal: 2015, pp. 1–6.

7 Kerr, E., and Macdonald, M., "A General Perturbations Method For Spacecraft Lifetime Analysis," *25th AAS/AIAA Space Flight Mechanics Meeting*, Wiliamsburg, VA, USA: 2015, pp. 1–15.

8 Kerr, E., and Macdonald, M., "Incorporating Solar Activity into General Perturbations Analysis of Atmospheric Friction," *In peer review for: Journal of Guidance, Control, and Dynamics*.

9 Ceriotti, M., Heiligers, J., and McInnes, C. R., "Trajectory and Spacecraft Design for a Pole-Sitter Mission," *Journal of Spacecraft and Rockets*, vol. 51, Jan. 2014, pp. 311–326.

Table 6. Summary of de- and re-orbiting concepts, strategies and technologies

Name	Type	In Lit	TRL	AD2	Mass Eff.	Vol. Eff.	Orbit Category 1.1	1.2	1.3	1.4	2.1	2.2	3	4	5	Insensitivity Ecc.	Inc.	Control	Passive	S/C Mass
Passive Drag Augmentation	A	Yes	H	L	M	M	H	H	M	L	I	I	I	I	L	M	H	U	Yes	LM
Active Drag Augmentation	A	Yes	M	L	M	M	H	H	M	L	I	I	I	I	L	M	H	S/C	No	LM
Electrodynamic Tethers	A	Yes	H	M	M	M	H	H	M	L	I	I	I	I	L	M	M	U	No	LM
Lorentz-Augmented De-orbiting	A	Yes	L	M	H	H	H	H	M	L	I	I	I	I	L	M	M	U	No	L
Cold Gas	A	Yes	H	L	L	L	H	M	L	L	L	L	L	L	L	H	H	C	No	LM
Mono Propellant	A	Yes	H	L	M	M	H	H	M	L	M	M	H	L	M	H	H	C	No	LM
Bi Propellant	A	Yes	H	L	M	M	H	H	H	M	M	M	H	L	H	H	H	C	No	MH
Solid Propellant	A	Yes	H	L	M	M	H	H	M	M	M	M	H	L	H	H	H	C	Yes/No	LMH
Hybrid Propulsion	A	Yes	M	L	H	H	H	H	H	H	M	M	H	L	H	H	H	C	No	MH
Electric Propulsion	A	Yes	H	L	H	H	H	H	H	H	M	M	H	L	H	M	H	C	No	LM
Active Solar Sailing	A	Yes	M	M	M	M	I	I	L	L	L	L	M	L	L	M	H	C	No	LM
Passive Solar Sailing	A	No	L	H	M	M	I	I	L	L	L	L	M	L	L	M	H	C	Yes	LM
SRP on Panels	A	Yes	H	L	H	H	I	I	I	L	L	L	M	L	L	L	L	U/S	No	LM
SRP-Augmented De-orbiting	A	Yes	M	M	M	M	I	I	L	M	M	L	L	L	M	L	L	U	No	LM
Nuclear Sail	A	Yes	L	H	L	L	L	L	L	L	L	L	L	L	L	H	H	U	No	LM
Vaporise	A	No	L	L	H	H	H	H	H	H	H	H	H	H	H	H	H	n/a	No	LMH
Pulse Tether	A	Yes	L	M	M	M	H	H	M	L	I	I	I	I	L	M	M	U	No	LM
Ground-based Laser Ablation	B	Yes	L	H	H	H	H	H	M	M	L	I	I	I	L	M	M	U	No	LM
Space-based Laser Ablation	B	Yes	L	H	H	H	H	H	M	L	L	L	L	L	L	M	H	U	No	LM
Space-based Solar Ablation	B	Yes	L	H	H	H	H	H	M	L	L	L	L	L	L	M	H	U	No	LM
Multi-layered sphere	C	Yes	M	H	H	H	H	H	M	L	L	L	L	L	L	M	H	U	Yes	L
Foam-Based ADR	A/B	Yes	L	H	H	H	H	H	M	I	I	I	I	I	L	M	H	U	No	L
Ion-Beam Shepherd	A/B	Yes	M	M	H	H	H	H	H	M	L	L	H	L	L	H	H	C	No	LMH
Space Tug	A/B	Yes	M	M	H	H	H	H	H	M	L	L	H	L	L	H	H	C	No	MH
Crusher	A	No	L	M	H	H	H	H	M	M	L	L	H	H	L	H	H	U	No	LM
Spagettifier	A	No	L	H	H	H	L	L	L	L	L	L	H	H	L	H	H	U	No	LM
Beam Sailing	B	Yes	L	H	H	H	H	H	M	L	I	I	I	I	L	H	H	C	Yes	LM
Acid Bath	A	No	L	H	H	H	H	H	H	H	H	H	H	H	H	H	H	n/a	Yes	LM
Drag	C	Yes	H	L	n/a	n/a	H	H	M	L	I	I	I	I	L	M	H	U	Yes	LM
Catcher's Mitt	C	Yes	L	H	n/a	n/a	H	H	M	L	L	L	I	I	L	M	H	U	No	L
Tungsten Dust	C	Yes	L	H	n/a	n/a	L	M	M	H	L	I	I	I	L	M	H	U	Yes	LM
Atmospheric Heating	C	No	L	H	n/a	n/a	L	L	M	H	I	I	I	I	L	M	H	U	Yes	LM
Magnetic Field Manipulation	C	No	L	H	n/a	n/a	H	H	M	L	I	I	I	I	L	M	M	U	Yes	LM