Strathprints Institutional Repository

Needs Assessment of Gossamer Structures in Communications Platform End-of-Life Disposal

Malcolm Macdonald[1], Colin M^{c}Innes[2] and Charlotte (Lücking) Bewick[3]
Advanced Space Concepts Laboratory, Department of Mechanical & Aerospace Engineering, University of Strathclyde, Glasgow, G1 1XQ, Scotland

Lourens Visagie[4] and Vaios Lappas[5]
Surrey Space Centre, Faculty of Engineering and Physical Sciences, University of Surrey, Guildford, GU2 7XH, United Kingdom

and

Sven Erb[6]
ESA ESTEC, Keplerlaan 1, 2201 AZ Noordwijk, The Netherlands

The use of a gossamer structure is considered in application to end-of-life disposal of communications platforms. A wide-ranging survey of end-of-life disposal techniques and strategies is presented for comparison against a gossamer structure prior to a down-selection of viable competing techniques; solar sailing, high and low-thrust propulsion, and electrodynamic tethers. A parametric comparison of the down-selection competing techniques is presented where it was found that exploiting solar radiation pressure on the gossamer structure was of limited value. In general terms, it was found that if a spacecraft propulsion system remains functioning at the end-of-life then this will likely provide the most efficient means of re-orbiting, especially when the propulsion system is only used to lower the orbit to a point where atmospheric drag will cause the orbit to decay within the required timeframe. Atmospheric drag augmentation was found to be of most benefit for end-of-life disposal when an entirely passive means is required, allowing the device to act as a 'fail-safe', which if the spacecraft suffers a catastrophic failure would activate. The use of an atmospheric drag augmentation system is applicable to only low and medium mass spacecraft, or spacecraft that are unlikely to survive atmospheric re-entry, hence minimizing risk to human life.

[1] Associate Director, Advanced Space Concepts Laboratory, Mechanical & Aerospace Engineering, University of Strathclyde, Glasgow, G1 1XQ, Scotland. AIAA Associate Fellow.
[2] Director, Advanced Space Concepts Laboratory, Mechanical & Aerospace Engineering, University of Strathclyde, Glasgow, G1 1XQ, Scotland. AIAA member.
[3] Research Assistant, Advanced Space Concepts Laboratory, Mechanical & Aerospace Engineering, University of Strathclyde, Glasgow, G1 1XQ, Scotland. Now a Space Systems and Mission Concepts Engineer at OHB System, Bremen, Germany.
[4] Research Assistant, Surrey Space Centre, Faculty of Engineering and Physical Sciences, University of Surrey, Guildford, GU2 7XH, United Kingdom.
[5] Professor, Surrey Space Centre, University of Surrey, Guildford, GU2 7XH, United Kingdom.
[6] GNC Systems Engineer, ESA ESTEC, Keplerlaan 1, 2201 AZ Noordwijk, The Netherlands.

Nomenclature

A	=	Cross-sectional surface area, m^2
A_T	=	Cross-sectional tether area, m^2
B	=	Dipole field strength, T
C_D	=	Drag co-efficient
E	=	Specific orbital energy, J/kg
F_D	=	Drag force, N
F_{LT}	=	Low-thrust propulsion force, N
F_{srp}	=	Solar radiation force, N
F_T	=	Electrodynamic tether force, N
g	=	Standard gravity = $9.80665\ ms^{-2}$
H	=	Density scale height, km
I	=	Induced current, A
I_{sp}	=	Specific impulse, s
L	=	Lifetime duration, s
L_T	=	Tether length, m
m_{prop}	=	Propellant mass, kg
m_{sc}	=	Spacecraft mass, kg
P	=	Solar radiation pressure, $9.15\mu\ Pa$
$R_{\oplus}$	=	Mean volumetric radius of the Earth = 6371 km
R_T	=	Electrodynamic tether resistance, Ω
r	=	radius, m
v	=	velocity, $m\ s^{-1}$
η	=	geometric efficiency, a function of the solar sail steering law used
μ	=	Gravitational parameter = $3.986032\ \times 10^{14}\ m^3 s^{-2}$
ρ	=	local air density, $kg\ m^3$
φ	=	Voltage in electrodynamic tether, V

I. Introduction

The rapid development of space technology in the second-half of the twentieth-century led to the emergence of a shell of synthetic debris around the Earth. This shell of debris now poses a greater threat to spacecraft than the natural meteoroid environment within 2000 km of the Earth's surface. Resultantly, the Inter-Agency Space Debris Coordination Committee, IADC, have developed best-practice mitigation guidelines to limit the further generation of synthetic debris around the Earth. These are based on three fundamental principles; limiting debris released during normal operations; minimizing the risk of on-orbit break-ups and collisions; and, limiting the orbital lifetime of non-functioning objects in populated regions. This paper develops a needs assessment of a gossamer structure designed to address the third of these principles, end-of-life disposal.

Spacecraft end-of-life disposal by means of drag-augmentation has been widely discussed.[1–5] Drag-augmentation is typically achieved by two means, either through the deployment of a spherical envelope, for example a balloon, which has the advantage of being an omni-directional system, or by deployment of a shaped gossamer structure. Drag-augmentation concepts have also been extended to exploit enhanced solar radiation pressure on the deployed surface,[6, 7,] and the J_2 perturbation.[8] The simplicity and robustness of a spherical envelope such as a balloon for de-orbiting comes at the cost of a mass penalty when compared to a shaped gossamer structure. Considering alone the material required, the surface area of a sphere goes as $4\pi r^2$, where r is the characteristic length, e.g. radius, while the surface area of a flat disc or square scales as πr^2 or r^2, respectively.

This paper presents an assessment of the performance requirements of a gossamer structure, for example a flat rectangular surface, a shaped parafoil-like surface, or even a closed structure such as a balloon for use in disposal of communications platforms at the end-of-life. It is of note however that the simultaneous use of multiple forces is not considered within this assessment. The performance requirements are derived based on a parametric comparison against a wide-range of other disposal concepts and strategies that are also a consequence of only a single force acting on the spacecraft at any one time. Potential use-cases are developed and the applicability of a gossamer structure assessed in comparison to other disposal concepts and strategies.

II. Orbits

To enable a detailed analysis of de-orbiting concepts, strategies and/or techniques the orbital environment is sub-divided into a number of orbit categories, as summarized in Table 1. Each de-orbiting concept, strategy and/or technique will thus be rated on its applicability within each orbital regime as **I**(*napplicable*), **L**(*ow*), **M**(*edium*) or **H**(*igh*).

Table 1. Orbit categories

Category ID	Name	Altitude (km)	Removal Required
1.1	Low LEO	100 – 300	Yes
1.2	International Space Station Region	300 – 500	Yes
1.3	Medium LEO	500 – 1000	Yes
1.4	High LEO	1000 – 2000	Yes
2.1	MEO	2000 – 19000 & 24000 – 35586	No
2.2	GNSS Region	19000 – 24000	No
3	Geosynchronous Orbits	35586 – 35986	Yes
4	Supersynchronous orbits	35956 – 45000	No
5	HEO	All	Yes

III. Survey of Re-Orbiting Strategies and Technologies

A wide-range of disposal concepts and strategies will initially be surveyed to allow selection of a range of concepts for quantitate analysis and comparison to a gossamer structure. This initial survey will be based on a range of criteria introduced in the following sections.

A. Type of Method

Considering all de-orbiting concepts, strategies and/or techniques it is noted that three distinct types of method can be defined to aid the characterization, comparison and evaluation re-orbiting technology. These types are, '*Type A*', a method applied individually to a piece of debris, either as part of the initial system development or later attached to piece of debris or a non-operational spacecraft. '*Type B*', Active Debris Removal, a method remotely applied to an individual piece of debris or a non-operational spacecraft. Or '*Type C*', a method applied universally to all objects within a certain region.

B. Result of Method

Considering all de-orbiting concepts, strategies and/or techniques it is noted that two distinct solutions occur; specifically these results are either a controlled or uncontrolled re-orbit or de-orbit. When large spacecraft de-orbit significant fragments can be expected to reach the Earth's surface, as witnessed when the German Roentgen Satellite (Rosat) re-entered the atmosphere on Sunday October 23 2011 over the Bay of Bengal. The uncontrolled de-orbit of Rosat ended in a 'harmless' splashdown. However, it is of note that had Rosat re-entered as little as 10 – 15 minutes later it could have impacted the Chinese mainland in the region of the cites of Chongqing and Chengdu, with a total population of over 42 million people. Therefore, it is clear that for large spacecraft a 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 then a controlled method of re-orbiting is required to avoid unforeseen collisions and subsequent legal difficulties. Each de-orbiting concept, strategy and/or technique will be rated on its result as **C**(ontrolled) or **U**(ncontrolled). In addition, each de-orbiting concept, strategy and/or technique will be rated as Active or not, based on whether it would require active spacecraft operations to maintain the de-orbit concept.

C. Comparison Metrics

Each surveyed disposal concept will be ranked **L**(*ow*), **M**(*edium*), or **H**(*igh*) against the range of metrics defined in Table 2. It should be noted from Table 2 that not all comparison metrics should aim to be low or high, for example, a high Technology Readiness Level (TRL) and low Advancement Degree of Difficulty (AD^2) are desired.[9] In addition to this, each will be categorized for insensitivity to end-of-life orbit eccentricity and inclination.

Table 2. Comparison metrics

Metric	Low	Medium	High
Technology Readiness Level, TRL	TRL 1 – 3	TRL 4 – 6	TRL 7 – 9
Advancement Degree of Difficulty, AD^2	AD^2 1 – 3	AD^2 4 – 6	AD^2 7 – 9
Mass Efficiency	> 15% of total mass fraction	5 – 15% of total mass fraction	< 5 % of total mass fraction
Volume Efficiency	> 15% of total volume	5 – 15% of total volume	< 5 % of total volume
Sensitivity to Spacecraft Mass	< 50 kg	50 – 1000 kg	1000 kg

D. Matrix of De-Orbiting Concepts

A wide-range of de-orbiting, including re-orbiting, strategies and technologies are summarized in matrix form to allow a quantitative comparison in Table 3. Where multiple entries are given in the 'Craft Mass' column a range should be interpreted, furthermore entries in this column in bold highlight a particular efficiency. The cells are color coded to ease comparison, green is good, yellow is medium, red is poor or inapplicable (with white background). The columns for 'Result', 'Active' and 'Craft Mass' are not color coded as favorable content within these columns is contextual and cannot be generalized.

E. Initial Down-Selection of Concepts

At this initial stage, a down-selection can be performed on evidently unviable (in the 10-15 year timeframe) strategies, or strategies that only apply to regions where a gossamer structure would not be applicable. As such, all strategies and/or technologies which have a **L**(*ow*) TRL or **H**(*igh*) AD^2 are removed from further consideration at this point.

A gossamer structure can have two modes of operation. These are, 1) where atmospheric drag is used to de-orbit a satellite, or 2) where solar radiation pressure is used to either re-orbit a spacecraft at the end-of-life, for example in Geostationary Orbit, GEO, or to lower a satellite's orbit such that atmospheric drag can be used to complete the de-orbit process. It is apparent however given the high end-of-life mass (>3000 kg) of the vast majority of spacecraft in GEO that the use of a gossamer structure, of order 25 m^2, for solar sailing to re-orbit a spacecraft from GEO to the graveyard/Supersynchronous region is significantly sub-optimal when compared against an end-of-life conventional bi-propellant maneuver. Furthermore, as the magnitude of propulsive force from a gossamer structure would be small it is likely that the increase in orbit altitude per orbit would be insufficient to prevent the spacecraft entering the neighboring spacecraft slots.

Considering this it is apparent from Table 3 that the remaining, and hence down-selected concepts for further quantitate analysis are limited to the LEO region, that is up to 2000 km, where the a gossamer structure would be used to augment the atmospheric drag effects or as a solar sail to enter the region of atmospheric drag. The down-selected concepts are,

- Atmospheric Drag Augmentation
- Electrodynamic Tethers
- Mono-propellant
- Bi-Propellant
- Low-Propulsion, high specific impulse propulsion
- Solar Sailing to gain Atmospheric Drag

Note that Cold Gas propulsion was not down-selected due to its inefficiency, whilst Solid Propulsion was not down-selected due to the risk of creating further debris from propellant slag.

Table 3. De-orbiting, including re-orbiting, strategies and technologies summarized in matrix form

		Orbit Category									Technology		Efficiency		Insensitivity				Craft
Name	**Type**	**1.1**	**1.2**	**1.3**	**1.4**	**2.1**	**2.2**	**3**	**4**	**5**	**TRL**	**AD^2**	**Mass**	**Volume**	**ecc.**	**inc.**	**Result**	**Active**	**mass**
Atmospheric Drag Augmentation[1–5]	A	H	H	M	L	I	I	I	I	L	M	M	M	M	M	H	U	No	**L**M
Electrodynamic Tethers[10]	A	H	H	M	L	I	I	I	I	L	M	M	M	M	M	M	U	No	**L**M
Lorentz-augmented Deorbiting	A	H	H	M	L	I	I	I	I	L	L	M	H	H	M	M	U	Yes	L
Cold Gas	A	H	M	L	L	L	L	L	L	L	H	L	L	L	H	H	C	Yes	**L**M
Mono propellant	A	H	H	M	L	L	L	M	L	L	H	L	M	M	H	H	C	Yes	**L**M
Bi-Propellant	A	H	H	H	M	L	L	H	L	L	H	L	M	M	H	H	C	Yes	M**H**
Solid propulsion	A	H	H	M	M	L	L	M	L	M	H	L	M	M	H	H	C	Yes	LMH
Hybrid Propulsion	A	H	H	H	H	M	M	H	H	H	M	L	H	H	H	H	C	Yes	M**H**
Electrical Propulsion	A	H	H	H	H	L	L	H	M	H	H	L	H	H	M	H	U	Yes	LM
Active Solar Sailing[11]	A	I	I	L	L	L	L	M	L	L	M	M	M	M	M	H	U	Yes	LM
SRP on panels	A	I	I	I	L	L	L	M	L	L	H	L	H	H	M	H	U	Yes	LM
SRP-augmented Deorbiting	A	I	I	L	M	M	M	L	L	M	M	M	M	M	L	L	U	No	**L**M
Ground-based Laser Ablation[12–15]	B	H	H	M	M	L	I	I	I	L	L	H	H	H	M	M	U	Yes	**L**M
Space-based Laser Ablation[12–15]	B	H	H	M	L	L	L	L	L	L	L	H	H	H	M	H	U	Yes	LM
Space-based Solar Ablation[12–15]	B	H	H	M	L	L	L	L	L	L	L	H	H	H	M	H	U	Yes	LM
Multi-layered sphere	B	H	H	M	L	L	L	L	L	L	M	H	H	H	M	H	U	No	L
Foam-based ADR[16]	B	H	H	M	I	I	I	I	I	L	L	H	H	H	M	H	U	Yes	L
Ion-beam shepherd[17]	B	H	H	H	M	L	L	H	L	M	M	M	H	H	H	H	C	Yes	L**MH**
Space tug[18]	B	H	H	H	M	L	L	H	L	L	M	M	H	H	H	H	C	Yes	M**H**
Drag	C	H	M	L	I	I	I	I	I	L	H	L	n/a	n/a	M	H	U	No	**L**M
Catcher's Mitt[19]	C	H	I	M	L	L	L	I	L	L	L	H	n/a	n/a	M	H	U	Yes	L
Tungsten Dust[20]	C	L	I	M	H	L	I	I	L	L	L	H	n/a	n/a	M	H	U	No	LM

IV. Comparison of a Gossamer Structure against Down-Selected Concepts

To provide a quantitative assessment of the performance of competing de-orbit methods, a series of analytic approximations are developed that provide approximate requirements for the de-orbit device to ensure de-orbit within a fixed duration. The de-orbit timescale L is fixed at 25 years in all cases to comply with the IADC space debris best-practice mitigation guidelines. The analysis 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 stabilized 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.

A. De-Orbit Scaling Laws

1. Drag Augmentation

The decay time-scale for a drag augmented device can be estimated by considering the work done by the drag force F_D on a spacecraft of mass m_{sc} and total drag cross-sectional area A on a circular orbit of radius r. Assuming a quasi-circular orbit the spacecraft orbit speed is therefore $v = \sqrt{\mu/r}$, where μ is the gravitational parameter. The rate of change of two-body specific energy $E = -\mu/2r$ is then given by,

$$\dot{E} \cong -\frac{1}{m_{sc}} F_D v \tag{1}$$

where, the drag force F_D is defined as,

$$F_D = \frac{1}{2} C_D\, A\, \rho\, v^2 \tag{2}$$

where, C_D is the drag coefficient (assumed to be 2.1) and ρ is the local air density. In order to proceed, an analytic model of the atmospheric density is required. Using a power law fit to the 1976 standard atmosphere[21] from 150-1000 km it is found that,

$$\rho(kg\, m^{-3}) = \Lambda h(km)^{-\gamma} \tag{3}$$

where, h is height, $\Lambda = 10^7$ and $\gamma = 7.201$.

Integrating along the quasi-circular orbit decay spiral from some initial orbit radius r_0 to some final orbit radius r_1 from some duration L it can be shown that the required drag area is given by,

$$A = \frac{m_{sc}}{C_D\, L\, \Lambda \sqrt{\mu\, R_\oplus}} \frac{((r_0 - R_\oplus)^{1+\gamma} - (r_1 - R_\oplus)^{1+\gamma})}{1+\gamma}\, 1000^{-\gamma} \tag{4}$$

where $R_\oplus$ is the mean volumetric radius of the Earth (6371 km) and all distances are given in meters. For a given spacecraft mass and initial orbit, the total drag area can then be estimated for a fixed de-orbit duration, again assumed to be 25 years.

2. Solar Sailing

Solar sailing is assumed to be inapplicable for orbits below 750 km altitude, due to the dominance of atmospheric drag in this region.[22, 23] However, the orbit transfer time for a solar sail can be estimated using a similar analysis to that used in the previous section, with an appropriate sail steering law. For equatorial orbits, a simple switching law can be used that 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 with the velocity vector.

Again, the rate of change of two-body specific energy $E = -\mu/2r$ can be obtained from the work done by the solar sail generated thrust F_{srp} such that,

$$\dot{E} \cong -\frac{1}{m_{sc}} F_{srp} v \tag{5}$$

Therefore, the rate of change of orbit radius can be found from,

$$\frac{\mu}{2r^2}\frac{dr}{dt} \approx \frac{F_{srp}}{m_{sc}}\sqrt{\frac{\mu}{r}} \tag{6}$$

which can be integrated to provide the orbit radius as a function of time. The thrust F_{srp} is related to the sail area A and the solar radiation pressure P by,

$$A = \frac{F_{srp}}{2P} \tag{7}$$

assuming an ideal reflector.

For a continuous, quasi-circular low thrust spiral from some initial orbit radius r_0 to some final orbit radius r_1 in duration L it can be shown that the required thrust-to-mass ratio is given by,

$$A = \eta \frac{m_{sc}}{2\,P\,L}\left|\sqrt{\frac{\mu}{r_1}} - \sqrt{\frac{\mu}{r_0}}\right| \tag{8}$$

where, η is the geometric efficiency, a function of the sail steering law used.

Equatorial Orbits; for near-equatorial orbits, a switching law is used that requires a sail slew of 90^o 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.[23] Polar Orbits; for near-polar orbits, the sail attitude is fixed relative to the Sun and yaws 360 degrees per orbit to align the sail thrust and velocity vector's. Here the steering law efficiency is η = 2.83, which accounts for the loss of thrust due to the pitch of the sail relative to the Sun-line.[23] It can be seen that the polar orbit steering law is marginally more efficient than the equatorial steering law.

3. Low-Thrust Propulsion

Using a similar analysis to previous sections, the decay time for continuous low thrust electric propulsion can be determined. The thrust F_{LT} required to de-orbit from some initial orbit radius r_0 to some final orbit radius r_1 in some fixed duration L is found to be,

$$F = \frac{m_{sc}}{L}\left|\sqrt{\frac{\mu}{r_0}} - \sqrt{\frac{\mu}{r_1}}\right| \tag{9}$$

The required propellant mass can then be determined from the effective Δv such that,

$$m_{prop} = m_{sc}\left(1 - e^{-\Delta v/g\,I_{sp}}\right) \tag{10}$$

where I_{sp} is the specific impulse of the propulsion system, and,

$$\Delta v = \left|\sqrt{\frac{\mu}{r_0}} - \sqrt{\frac{\mu}{r_1}}\right| \tag{11}$$

4. Electrodynamic Tether

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 $\tan^{-1}\left(1/\sqrt{2}\right)$ and is kept in tension. A conducting tether of length L_T moving at speed v through a magnetic field of local strength B has an induced voltage $\varphi = L_T v B$, where B is the dipole field strength given by,

$$\mathrm{B} = B_o\left(\frac{R_\oplus}{r}\right)^3 \tag{12}$$

for field strength $B_o = 3.5\text{x}10^{-5}\ T$. The force induced in the tether is then $F_T = BIL_T$, where I is the current induced by the voltage φ. The current and voltage are related by Ohms law such that $\varphi = IR_T$ for tether resistance R_T, which in principle includes the resistance of the plasma contactors and the plasma sheath. Assuming that the tether follows a quasi-circular inward spiral, the speed v is then approximated by the local circular orbit speed $v = \sqrt{\mu/r}$. The force acting on the tether can therefore be estimated as,

$$\mathrm{F}_T = B_o^2\left(\frac{R_\oplus}{r}\right)^6 \frac{L_T^2}{R_T}\sqrt{\frac{\mu}{r}} \tag{13}$$

The tether resistance R_T can be scaled with tether length through resistivity ρ, such that $R_T = \rho\, L_T/A_T$, where the resistivity of aluminum is assumed to be $\rho \approx 2.82\text{x}10^{-8}$ Ohm-m and A_T is the cross-sectional area of the tether. The tether is assumed to have a diameter of 2 mm.

Finally, in order to model the interaction of a tether in an inclined orbit with the magnetic field a geometric efficiency factor ε is defined as,

$$\varepsilon = \frac{1}{16}\left\{\cos\left[\tan^{-1}\left(1/\sqrt{2}\right)\right]\right\}^2\{6 + 2\cos 2i + 3\cos[2(i-\delta)] + 2\cos 2\delta + 3\cos[2(i+\delta)]\} \tag{14}$$

where δ is the tilt of the Earth's dipole field relative to the equator, assumed to equal 11.5 degrees. Using Eq. (6), and integrating, the tether length L_T required to de-orbit in a duration L is given by,

$$L_T = \frac{(r_1^6 - r_o^6)}{12\,\varepsilon\,L}\frac{m_{sc}}{R_E^6 B_o^2}\frac{\rho}{A_T} \tag{15}$$

from some initial orbit radius r_0 to some final orbit radius r_1.

B. Analysis

Considering the Iridium, GlobalStar and Orbcomm communication constellations as test cases, each of the down-selected concepts can be analytically considered in-turn to quantify the relative value of each.

1. De-Orbit by Atmospheric Drag Augmentation

The analytic scaling laws can be used to determine the required drag area required, as shown in Table 4. Again, for the purposes of analysis the de-orbit time L is fixed as 25-years, and de-orbit is assumed at an altitude of 100 km.

Augmenting the area-to-mass ratio, with for example a deployable structure, is clearly a viable mechanism to decrease the de-orbit time of spacecraft in the medium LEO orbit category assuming that the deployed structure passively maintains the required increased surface area. However, the mass of applicable spacecraft is seen to significantly decease as the altitude increases towards >800 – 900 km.

Note that, assuming an Iridium spacecraft surface area of 4 and 12 square meters respectively, corresponding to spacecraft body only and spacecraft body plus maximum solar array area, it is found that an Iridium spacecraft will naturally decay in approximately 90 – 250 years. However, reducing the initial circular orbit altitude to 575 km, this natural decay timescale reduces to between 7 – 21 years. Similarly, a GlobalStar spacecraft can be assumed to, in-effect, never decay due to atmospheric drag alone, however similarly reducing its altitude to 575 – 585 km altitude reduces the decay time to atmospheric drag to approximately 21 – 24.5 years. Finally, an Orbcomm spacecraft can be assumed to decay due to atmospheric drag alone in approximately 300 years, however reducing its altitude to 600 – 605 km reduces the decay time to atmospheric drag to approximately 22 – 24 years. The effect of such limited use of propulsion will be discussed later.

Table 4. Required total drag area to de-orbit in 25 years (drag augmentation effects only)

	Iridium	**GlobalStar**	**Orbcomm**
Initial altitude (km)	781	1410	825
Bus mass (kg) EOL	526	546	100
Bus area (m^2) low {high}	4 {12}	4	1
Required drag area (m^2)	40.6	5351	12
Equivalent square side-length (m)	6.4	73.2	3.5

2. De-Orbit by Electrodynamics Tether

The analytic scaling laws for the electrodynamic tether can be used to determine the required tether length as shown in Table 5. Again, for the purposes of analysis, the de-orbit time L is initially fixed as 25-years and de-orbit is assumed at an altitude of 100 km.

The required tether lengths in Table 5 for a 25-year de-orbit are seen to be very short as direct result of the 25-year de-orbit timeframe. Most prior electrodynamic tether de-orbit analysis has sought to complete the de-orbit in a much shorter period, for example in [24] de-orbit times of 7.5 months are sought for tethers of up to 20 km.

Operationally there is no ‘strict’ need to de-orbit in less than 25-years as once deployed the electrodynamic tether should remain operationally passive, assuming sufficient on-board autonomy. Hence, reducing the de-orbit time below 25-years, and hence increasing the tether size, is not ‘strictly’ required. However, the electrodynamic tether is not an inert system, as it requires an active electrical control system to remain operational. Therefore, it is likely that for tether reliability reasons a shorter de-orbit would be preferred. Reducing the de-orbit time to 5 % of the operational life of the spacecraft, that is ¾ of a year, and 100-days assuming an operational lifetime between 5 and 15 years, the required tether lengths are also shown in Table 5.

Table 5. Required tether length to de-orbit in 25 years, assuming 2 mm tether diameter

	Iridium	**GlobalStar**	**Orbcomm**
Initial altitude (km)	781	1410	825
Bus mass (kg) EOL	526	546	100
Orbit inclination (deg)	86.4	52	52
Required length; 25 yrs (m)	23	4	0.3
Required length; ¾ yr (m)	780	125	10
Required length; 100 days (m)	2136	342	28

Given the requirements for non-passive, autonomous systems, it is reasonable to assume that short de-orbit times will be required to reduce the risk of system failure and to reduce the risk to other spacecraft. This coupled with the difficulties in deploying a long tether, especially from an inactive and potentially tumbling spacecraft make electrodynamic tethers a challenging concept.

3. De-Orbit by Low-Thrust, High-Specific Impulse Propulsion

The analytic scaling laws can be used to determine the required thrust, from a low-thrust, high-specific impulse propulsion system, as shown in Table 6. Again, for the purposes of analysis, the de-orbit time L is fixed as 25-years, de-orbit is assumed at an altitude of 100 km and the specific impulse of the propulsion system is assumed to be a conservative 3000 seconds.

The requirement to extend spacecraft operations for a further 25-years after end-of-life makes this option unattractive due to both the operations costs and the lifetime of most electric thrusters. However, it is also noted that the required thrust levels are very low in comparison to most electric propulsion systems. Reducing the de-orbit time, increases the required thrust magnitude, however it does not alter the required fuel mass (assuming a constant specific impulse). It is therefore noted that, for example, a GlobalStar spacecraft can be de-orbited, assuming continuous control, in approximately 100-days with a thrust magnitude of less than 45 mN. This notional period of extended operations can be further reduced by noting that the thrust must only be applied until the spacecraft reaches an orbit from which it will naturally decay within 25-years of the end-of-life due to atmospheric drag. It is noted that this hybrid method was how Iridium 9, which failed and was replaced by Iridium 84, was de-orbited; the on-board resistojets (low-thrust / low specific impulse: 300 mN / 350 seconds) were used to lower the spacecraft perigee in mid-late September 2000 and by early November the spacecraft was observed to be tumbling out of control. The orbit of Iridium 9 finally decayed in March 2003. Using the previous analysis that determined the maximum altitude for decay within <25-years due to atmospheric drag alone for each use-case spacecraft, the required propellant mass saving of such a hybrid scheme is shown in Table 7.

Note that an equivalent resistojet propelled de-orbit technique for an Iridium spacecraft reduces the propellant mass required from approximately 62 kg to approximately 17 kg. Such a scheme can therefore offer significant propellant mass savings, meaning that if any low-thrust propulsion system is already on-board a spacecraft in LEO it is likely a very attractive option for de-orbiting an operational spacecraft at end-of-life.

4. De-Orbit by Mono and Bi-propellant

The required propellant mass can be determined, using Eq. (10), based on the required velocity change to reduce the orbit perigee to 100 km, as shown in Table 8. The mono-propellant specific impulse is assumed to range between 150 – 225 seconds, while the bi-propellant specific impulse is assumed to range between 300 – 425 seconds.

Of the initially down-selected concepts, this is the only one that will provide a controlled atmospheric re-entry. However, if this is not required then as in the low-thrust scenario the required propellant mass can be reduced by simply moving the spacecraft to an orbit from which it will naturally decay within 25-years of the end-of-life due to atmospheric drag. The required propellant mass saving of such a hybrid scheme is shown in Table 9.

Table 6. Required thrust and propellant to de-orbit in 25 years (low-thrust only)

	Iridium	**GlobalStar**	**Orbcomm**
Initial altitude (km)	781	1410	825
Bus mass (kg) EOL	526	546	100
Required thrust (mN)	0.3	0.5	0.06
Required fuel (kg)	6.8	12.7	1.4

Table 7. Required propellant to move each spacecraft to an orbit from which it will naturally decay within 25-years of the end-of-life due to atmospheric drag (low-thrust only)

	Iridium	**GlobalStar**	**Orbcomm**
Initial altitude (km)	781	1410	825
Target altitude (km)	575	575	600
Bus mass (kg) EOL	526	546	100
Required fuel (kg)	2.0	7.8	0.4
Fuel saving (%)	71	39	71

Table 8. Required mono and bi-propellant masses to de-orbit spacecraft

	Iridium	GlobalStar	Orbcomm
Initial altitude (km)	781	1410	825
Bus mass (kg) EOL	526	546	100
De-orbit Δv (m s^{-1})	189	337	200
Mono-Propellant mass (kg)	47 – 72	90 – 140	9.5 – 15
Bi-Propellant mass (kg)	24 – 35	45 – 66	4.9 – 7.0

Table 9. Required propellant to move each spacecraft to an orbit from which it will naturally decay within 25-years of the end-of-life due to atmospheric drag (mono & bi-prop only)

	Iridium	GlobalStar	Orbcomm
Initial altitude (km)	781	1410	825
Target altitude (km)	575	575	600
Bus mass (kg) EOL	526	546	100
De-orbit Δv (m s^{-1})	110	418	119
Δv saving (%)	42	-24 %	41 %
Required mono-prop fuel (kg)	26 – 41	113 – 179	5.5 – 8.4
Required bi-prop fuel (kg)	14 – 20	58 – 83	2.9 – 4.1
Fuel saving (%)	42 – 44	- (26 – 29)	41 – 44

It is noted from Table 9 that the GlobalStar spacecraft would actually require an increased quantity of fuel in this hybrid de-orbit scenario due to the use of a circular target orbit for the maneuver. In-order to reduce the fuel requirement for the GlobalStar spacecraft an eccentric intermediate orbit would be required, with a perigee altitude of, perhaps, 300 – 400 km, reducing the required Δv to 250 – 280 ms^{-1}; a saving of 15 – 25 %. It should also be noted that an eccentric intermediate orbit might provide further propellant mass savings for the Iridium and Orbcomm platforms.

It is seen that significant propellant mass savings can once again be made through such a hybrid scheme. Meaning that if any high-thrust propulsion system is already on-board a spacecraft in LEO it is likely a very attractive option for de-orbiting an operational spacecraft at end-of-life; either directly, or by moving the spacecraft to an orbit from which it will naturally decay within 25-years of the end-of-life due to atmospheric drag.

5. Active Solar Sailing to Gain Atmospheric Drag

Assuming zero atmospheric drag when altitude is greater than 750 km, and zero solar radiation pressure when altitude is less than 750 km and noting that $L = L_{sail} + L_{drag}$, Eqs. (4) and (7) may be equated to eliminate L_{drag} as it can be assumed that the surface area of the solar sail equals the surface area of the drag device,

$$L_{sail} = \frac{\left(-\Psi\, L\, \sqrt{\mu\, R_\oplus}\sqrt{\frac{\mu}{r_0}}\eta\Lambda + \Psi\, L\, \sqrt{\mu\, R_\oplus}\sqrt{\frac{\mu}{r_i}}\eta\Lambda - \Psi\, L\, \sqrt{\mu\, R_\oplus}\sqrt{\frac{\mu}{r_0}}\gamma\eta\Lambda + \Psi\, L\, \sqrt{\mu\, R_\oplus}\sqrt{\frac{\mu}{r_i}}\gamma\eta\Lambda\right)}{20P[(r_1 - R_\oplus)^\gamma(R_\oplus - r_1) - (r_i - R_\oplus)^\gamma(R_\oplus + r_i)] - \Psi\mu\,\sqrt{R_\oplus}\,\eta\Lambda\left[\sqrt{\frac{1}{r_0}} + \sqrt{\frac{1}{r_i}} - \gamma\left(\sqrt{\frac{1}{r_0}} + \sqrt{\frac{1}{r_i}}\right)\right]} \tag{16}$$

where, $\Psi = 21 \times 10^{3\gamma}$, r_i is the radius at which the switch from solar sailing to drag occurs (equivalent to 750 km altitude) and where all radii are given in meters.

As before, for the purposes of analysis the de-orbit time L is fixed as 25-years and de-orbit is assumed at an altitude of 100 km. The required solar sail / drag surface area can thus be determined, as shown in Table 10 for de-orbit within the required total period, along with the length of time for which spacecraft operations must be extended, during the solar sailing phase, assuming passive stabilization during drag augmentation and a polar orbit.

From Table 10, it is apparent that using solar sailing to gain atmospheric drag can be immediately discarded due to the prolonged period of extended spacecraft operations required to deliver a meaningful effect. Whilst the period of extended spacecraft operations could be reduced, it would have a direct and significant impact on the required size of the gossamer structure, or the mass of the spacecraft that could be de-orbited.

Table 10. Required total sail area to de-orbit in 25 years (solar sailing and atmospheric drag effects combined)

	Iridium	**GlobalStar**	**Orbcomm**
Initial altitude (km)	781	1410	825
Bus mass (kg) EOL	526	546	100
Bus area (m^2) neglect	-	-	-
Solar Sailing Time (yrs)	2.6	17.4	5.4
Required sail area (m^2)	32.5	99.9	7.1
Equivalent square (m)	5.7	10.0	2.7

6. Re-Orbit by Low-Thrust, High-Specific Impulse Propulsion

The convention on international liability for damage caused by space objects, the Liability Convention,[25] states in Article II that a launching state shall be absolutely liable to pay compensation for damage caused by its space objects on the surface of the Earth, or to aircraft in flight. While Article III states that, the launching state shall be liable only if damage caused in space "*is due to its fault or the fault of persons for whom it is responsible*." Therefore, due to the reduced level of liability it is of interest to consider re-orbiting space objects rather than de-orbiting them with exposure to the resultant absolutely liability.

The analytic scaling laws can be used to determine the required thrust to re-orbit a spacecraft above the LEO region, as shown in Table 11. Again, for the purposes of analysis, the re-orbit time L is fixed as 25-years and re-orbit is assumed complete at an altitude of 2000 km. Once again, the required thrust level to complete the re-orbit maneuver is very low. Reducing the re-orbit time to less than 40 days for the GlobalStar platform it is found that the required thrust remains below 45 mN. It is also noted that the required fuel mass to re-orbit a GlobalStar spacecraft is significantly lower than either of the equivalent de-orbit concepts. Hence, if an electric propulsion system is already on-board a spacecraft in LEO it is likely an attractive option for re-orbiting spacecraft at end-of-life due to both the reduced, or similar, fuel mass and the reduced level of liability.

7. Re-Orbit by Mono and Bi-propellant

Using Eq. (10) the required propellant mass can be determined based on the required velocity change to increase the circular orbit altitude to 2000 km, as shown in Table 12. The mono-propellant specific impulse is assumed to range between 150 – 225 seconds, while the bi-propellant specific impulse is assumed to range between 300 – 425 seconds. Unlike the de-orbit scenario it is not possible to reduce the propellant requirement from that given in Table 12 as the spacecraft must perform two maneuver's to attain the final orbit. It is also of note that both the Iridium and Orbcomm spacecraft require additional fuel to re-orbit than to de-orbit, however as previously discussed this does not rule-out such an end of life strategy due to the increased liabilities when de-orbiting. A de-orbit of a GlobalStar spacecraft requires additional fuel when compared with two re-orbit maneuvers to the graveyard region. Finally, it is noted that the propellant requirement for a GlobalStar to reach the graveyard region is similar to that required to move to an orbit from which it will naturally decay within 25-years of the end-of-life due to atmospheric drag.

Table 11. Required thrust and propellant to re-orbit in 25 years (low-thrust only)

	Iridium	**GlobalStar**	**Orbcomm**
Initial altitude (km)	781	1410	825
Bus mass (kg) EOL	526	546	100
Required thrust (mN)	0.38	0.18	0.07
Required fuel (kg)	10.0	4.7	1.8

Table 12. Required mono and bi-propellant masses to re-orbit spacecraft

	Iridium	**GlobalStar**	**Orbcomm**
Initial altitude (km)	781	1410	825
Bus mass (kg) EOL	526	546	100
De-orbit ΔV ($m\ s^{-1}$)	564	257	541
Mono-Propellant mass (kg)	118 – 167	60 – 87	22 – 31
Bi-Propellant mass (kg)	67 – 92	33 – 46	12 – 17

8. Active Solar Sailing

The analytic scaling laws can be used to determine the required sail surface area for a 25-year re-orbit of a spacecraft to above 2000 km, as shown in Table 13 for a polar orbit. In the case of a solar sail, the requirement to extend operations by 25-years after end-of-life can only be relaxed by increasing the sail size. For example, a GlobalStar re-orbit in 12 years requires a sail of almost 12-m side length, while a re-orbit in 1.5 years (10 % of the spacecraft operational life) requires a sail side length of over 30 m. Furthermore, note that an Orbcomm re-orbit in 1.5 years requires a similarly large square sail of 18.9 m in side length.

9. Conclusions of Analysis

Of the initially down-selected concepts, the only one that does not appear to offer any added value is active solar sailing; principally due to the time required and the associated cost of extending spacecraft operations during this period. However, if the available solar radiation pressure could be exploited passively then it would overcome this cost issue.

If a spacecraft already has a propulsion system on-board, high or low-thrust, it is likely that this will offer an efficient de- or re-orbit strategy, especially when coupled with the hybridization concept of simply maneuvering the spacecraft to an orbit from which it will naturally decay within 25-years of the end-of-life due to atmospheric drag.

Some additional useful conclusions; ideally, the de-orbit technology should have an application during the operational lifetime of the spacecraft, if not, it must be of lower mass than the on-board options for de-orbiting, for example, the mass of additional fuel. De-orbiting in less than 25-years can only be justified on the basis of cost &/or risk, any period of extended spacecraft operations to allow for de-orbiting must be short; perhaps 5 – 10 % of the mission lifetime is a reasonable maximum. Rapidly de-orbiting a spacecraft that would otherwise naturally decay due to atmospheric drag within 25-years is of little commercial value within the current debris mitigation guidelines. Large mass objects, or objects that will likely survive re-entry, should however be de-orbited in a controlled manner, ensuring minimum risk to human life, or be re-orbited to a graveyard orbit.

The ‘Atmospheric Drag Augmentation’ concept and the ‘Electrodynamic Tether’ both require that the spacecraft takes some action at the end-of-life such that its interaction with the local environment is somehow altered, resulting in a new or enhanced force vector acting on the spacecraft, which in-turn acts to de-orbit the spacecraft. Both of these concepts, when in orbit, incur an increased risk of collision due to the increased surface area. However, the electrodynamic tether likely incurs an increased risk. Once deployed both concepts are mechanically inactive, however the electrodynamic tether is required to maintain an active electronic system. Furthermore, should a collision occur with another spacecraft it is likely that the electrodynamic tether will result in the creation of more debris than a similar collision with a thin-film membrane being used to augment a spacecraft’s area-to-mass ratio.

Within the medium and high LEO environment for most telecommunications constellations (or similar mass spacecraft), the use of atmospheric drag augmentation or an electrodynamic tether is of limited value when compared to the use of other technologies already on-board the spacecraft, i.e. the on-board propulsion system. However, the dual-use of such on-board technology to de-orbit the spacecraft requires that the spacecraft remains operational at the end-of-life, as indeed does an electrodynamic tether. If a spacecraft suffers a catastrophic failure, the use of on-board technology to de-orbit the spacecraft is not possible. However, an atmospheric drag augmentation system could be added to spacecraft as a ‘fail-safe’ de-orbiting technology, which if the spacecraft suffers a catastrophic failure would activate.

The principle advantage of an atmospheric drag augmentation system, such as a gossamer structure, is therefore the entirely passive operational mode and an ability to passively maximize the surface area to the mean-free flow of the atmosphere in all of the flow regimes in which it will pass through. The use of an atmospheric drag augmentation system is applicable to only low and medium mass spacecraft, or spacecraft that are unlikely to survive atmospheric re-entry, hence minimizing risk to human life.

Table 13. Required solar sail size to re-orbit in 25 years (solar sail only)

	Iridium	**GlobalStar**	**Orbcomm**
Initial altitude (km)	781	1410	825
Bus mass (kg) EOL	526	546	100
Required sail area (m^2)	116.9	55.2	21.3
Equivalent square (m)	10.8	7.4	4.6

V. Potential Use Cases

To further quantify the applicability of drag augmentation methods, a more detailed analytical model of orbit decay under atmospheric drag is applied. This enhanced model allows for the effects of atmospheric rotation, and hence initial orbit inclination, to be included in the analysis but still assumes an initially circular orbit. Using this enhanced model the approximate decay time of a circular orbit under atmospheric drag is,[26]

$$L = \frac{1}{4\pi}\left(\frac{2\beta r_c + 1}{\rho_c \beta^2 {r_c}^3}\right)\left(F^{-3/2}\right)\left(\frac{m}{AC_D}\right)\left(1 - e^{\beta\Delta}\right) \tag{17}$$

where, subscript c denotes the initial conditions of the circular orbit, Δ is the negative change in orbit radius, ρ_c is the initial atmospheric density, assuming an exponential atmospheric model from Eq. (19), β is defined in Eq. (20), and F is defined as,

$$F = \left(1 - \frac{r_p}{v_p}\omega_{atm}\cos i\right)^2 \tag{18}$$

where, ω_{atm} is the mean angular rate of rotation of the atmosphere, which is typically taken to be the same as the Earth's mean rotation rate but can vary between 0.8 and 1.3 revolutions per day. The exponential atmospheric model assumes that the atmospheric density varies as,

$$\rho = \rho_0 \exp\left(-\frac{r - r_0}{H}\right) \tag{19}$$

where, H is the density scale height and is, for a constant h,

$$\beta = \frac{1}{H} = \left(\frac{1}{h} - \frac{2}{r_0}\right) \tag{20}$$

Note that the assumption of a constant scale height requires that the density scale height is also constant. While these equations are not exact, they remain valid to an altitude of several hundred kilometers. It is found that the effect of varying inclination is negligible, less than the uncertainty due to C_D, hence an inclination of 90-degrees is assumed in future analysis.

A. 'Typical' Spacecraft

Using Eq. (17), the required ballistic co-efficient, $m_{sc}/(A\ C_D)$, to de-orbit in 25-years from a range of altitudes can be determined. Thereafter, assuming a surface area of 25 m^2, perhaps a 5 m square flat gossamer structure, the maximum mass that can be de-orbited in 25-years across the range of altitudes can be determined, assuming a passively stable attitude is maintained to maximize the surface area exposed to the atmospheric free-stream flow.

Having determined the maximum mass that can be de-orbited with a given size of gossamer structure it is noted that the 'typical' surface area of a spacecraft is,[27]

$$A = \left(\frac{\sqrt[3]{m_{sc}}}{4}\right)^2 \tag{21}$$

Thereafter the typical surface area of a spacecraft of the maximum mass that will de-orbited in 25-yrs can be determined. That is, the surface area without the gossamer structure. Note that for low-altitude orbits, the maximum mass that will de-orbit in 25-yrs will be very large, hence it can be expected that the 'typical' surface area of such a spacecraft may be large, and may indeed be larger than the assumed gossamer surface area. Having determined the typical surface area, the allowable mass of such a spacecraft can also be determined from the known ballistic co-efficient limit for a 25-year deorbit. Hence, the mass benefit of the drag augmentation device determined. However, in determining the benefit of drag augmentation device it must be noted that not only does this change the spacecraft surface area but it also changes the drag co-efficient; the drag co-efficient is however assumed constant and low, $C_D = 2$, to allow a conservative estimate to be made.

The mass or altitude benefit of a gossamer structure projecting an area of 25 m^2 into the free-stream direction, for a range of de-orbit times from an orbit inclination of 90-degrees, is shown in Fig. 1. It is seen that, for example, a 25 m^2 gossamer structure will allow a 1000 kg spacecraft to increase its operating ceiling altitude by approximately 50 km and still de-orbit within 25-years of the end-of-life. Alternatively, a spacecraft operating at 650 km altitude can increase its end-of-life mass by 736 kg (or approximately 200 %) and still de-orbit within 25-years of the end-of-life. From Fig. 1 it is seen that drag augmentation does not significantly alter the operating ceiling altitude of a spacecraft, however it does significantly alter the maximum allowable end-of-life mass.

Interpreting the data in Fig. 1, the mass benefit of drag augmentation against altitude can be further elucidated, as shown in Fig. 2, where it is seen that the peak mass benefit occurs in the altitude range 550 – 650 km and is largely independent of de-orbit time. Although Fig. 2 shows that the percentage benefit of drag augmentation increases as altitude is increased, it must be noted that this increasing altitude is a percentage of an ever-decreasing number;

hence, the ‘real’ value of drag augmentation for the ‘typical’ spacecraft is limited at altitude beyond about 800 km. From Fig. 2 it is apparent that drag augmentation is of little value for ‘typical’ spacecraft at altitudes below 550 km, as such ‘typical’ spacecraft can be expected to de-orbit naturally due to atmospheric drag below this altitude within the required timeframe without the need for drag augmentation.

Changing the surface area of the drag augmentation device allows increased masses to be de-orbited and/or allows drag augmentation to be applied at increased altitudes. It should be noted that due to the simplistic nature of the analysis, by assuming a larger drag surface area the ‘typical’ spacecraft area and mass at the same orbit also appear to increase. Hence, once again the ‘benefit’ of drag augmentation is considered rather than the actual mass or altitude variation. The direct mass comparison and effect of increasing the gossamer structure size for a 25-year de-orbit limit is shown in Fig. 3, while Fig. 4 and Fig. 5 show the benefit of drag augmentation and the effect of increasing the drag surface area for 25-year and 20-year de-orbit limits, respectively. It is noted that the lower altitude at which drag augmentation is of value to the ‘typical’ spacecraft decreases as the drag surface area is increased and that, in-general, larger sails can operate in a larger altitude range. Quantitatively, it is also noted from Fig. 5 that additional benefit is provided by drag augmentation when the required de-orbit time is reduced.

B. General Spacecraft Population

Considering the active spacecraft population in LEO, as tracked by the North American Aerospace Defense Command (NORAD) in August 2011, it is seen in Fig. 6 that two distinct peaks occur. The first is in the 800 km altitude bin and the second is in the 1400 – 1500 km altitude bins; note that each altitude bins is ±50 km. Secondary peaks in population occur either side of the 800 km altitude bin at 600 & 700 km and 900 & 1000 km altitude. A gossamer structure is of little value to the higher altitude peak in population. However, a gossamer structure, in a drag augmentation mode of operation, is of some value in the 800 km altitude bin and in the secondary peak population density below this bin. However, it must be recalled that the 800 km altitude population peak is towards the top-end of the useful regime for drag augmentation and depending on spacecraft size, may require a large projected surface area. Beyond 800 km altitude, it is assumed drag augmentation is, at best, of moderate too poor value, however this is dependent on spacecraft mass. It is reasonable to assume that the trend of launching more often to these identified altitude bins will continue in the next 10 – 15 years.

C. Communications Spacecraft

Three distinct telecommunication constellations are deployed within LEO; as discussed in previous sections these are Iridium (781 km altitude), GlobalStar (1410 km altitude) and Orbcomm (825 km altitude). As previously discussed, both the Iridium and Orbcomm spacecraft can be de-orbited in 25-years through drag augmentation, requiring mean projected surface areas of approximately 40 m^2 and 12 m^2, respectively. It is also noted that an Iridium spacecraft can be de-orbited with approximately 17 kg of fuel using its on-board resistojets. Hence, the addition of any drag augmentation device, which will incur a cost, a risk and through-life propellant usage, must offer a sufficient revenue benefit to be of value and will likely need to require, at least, 50 % less mass, that is, less than 8 kg.

D. Summary

Spacecraft of mass 1000 kg cannot de-orbit beyond the approximate altitude 650 km, 710 km, 750 km, and 810 km in less than 25-years with sail sizes of 25 m^2, 60 m^2, 100 m^2, and 225 m^2, respectively. Additionally, the benefit of drag augmentation is generally negligible for altitudes below 500 km, with the ‘typical’ spacecraft, as defined in Eq. (21) benefiting the most in the altitude range 550 – 650 km. In this altitude range, a gossamer structure can increase the allowable spacecraft mass by more than 1200 kg, with some benefit remaining up to an altitude of about 800 km, where a gossamer structure can increase the allowable spacecraft mass by more than 100 kg.

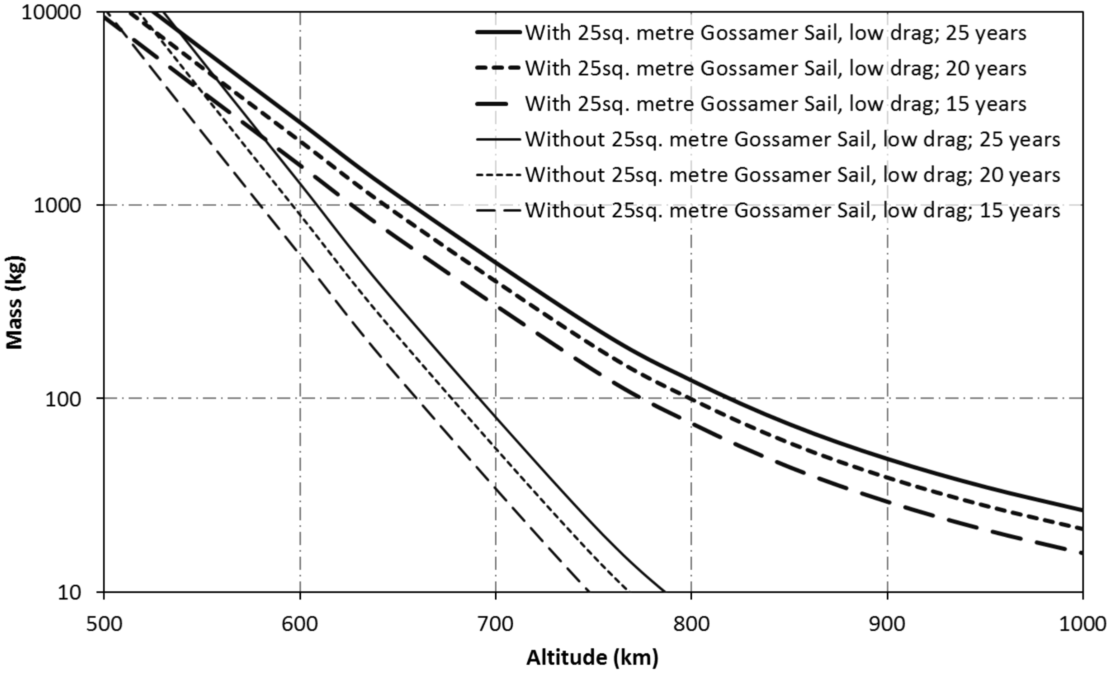

Fig. 1. Maximum de-orbit mass for a range of upper time limits from a range of orbits, with and without a gossamer structure; 90 degree inclined orbit

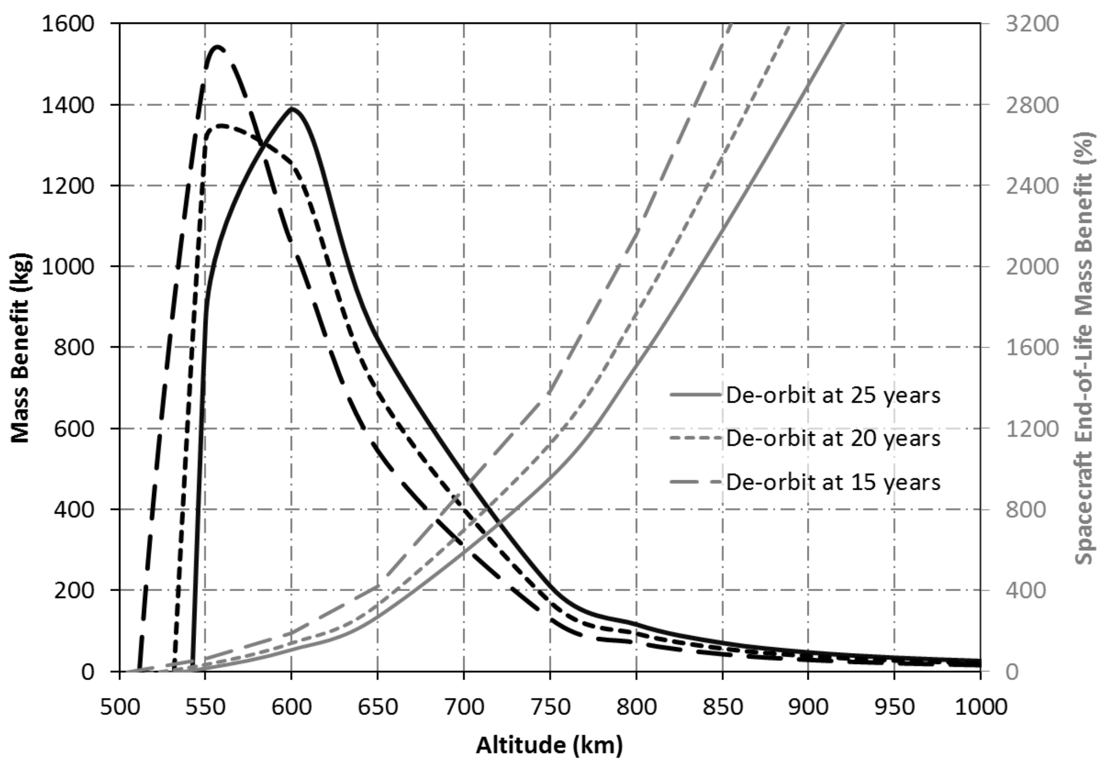

Fig. 2. Mass benefit (and percentage benefit) in the use of a gossamer structure for de-orbiting within a range of upper time limits from a range of orbits; 90 degree inclined orbit

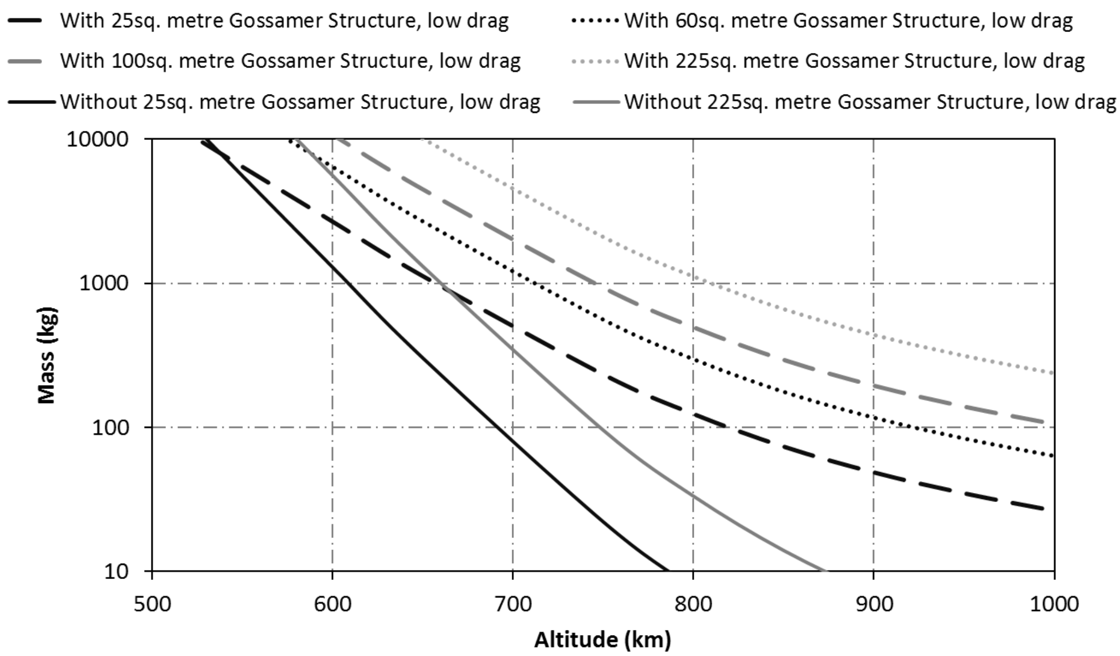

Fig. 3. Maximum de-orbit mass in 25-years for a range of gossamer structure sizes from a range of orbits, with and without a gossamer structure; 90 degree inclined orbit

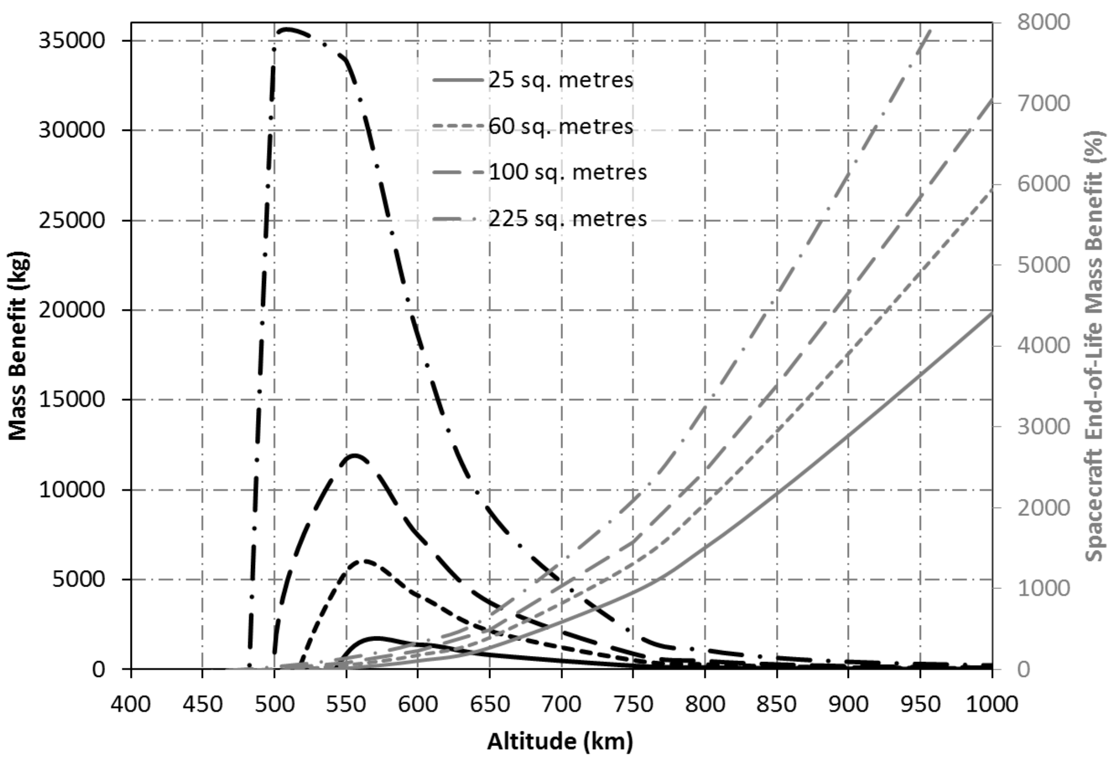

Fig. 4. Mass benefit (and percentage benefit) in the use of a range of gossamer structure sizes for de-orbit in 25-years from a range of orbits; 90 degree inclined orbit

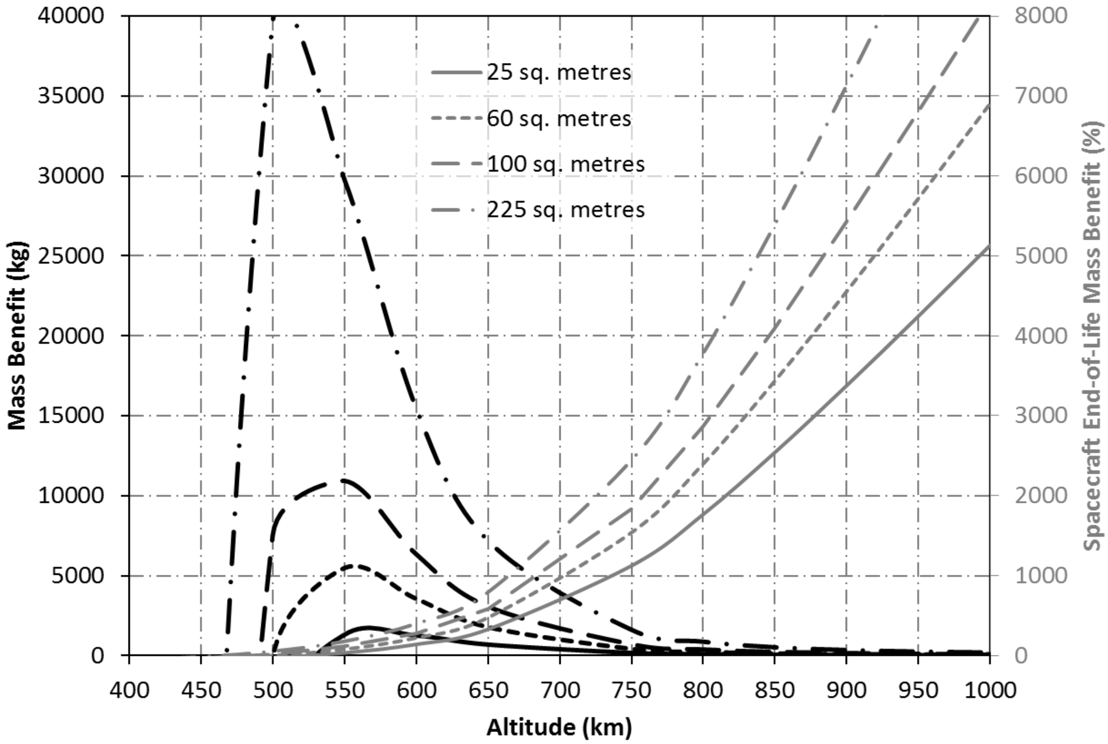

Fig. 5. Mass benefit (and percentage benefit) in the use of a range of gossamer structure sizes for de-orbit in 20-years from a range of orbits; 90 degree inclined orbit

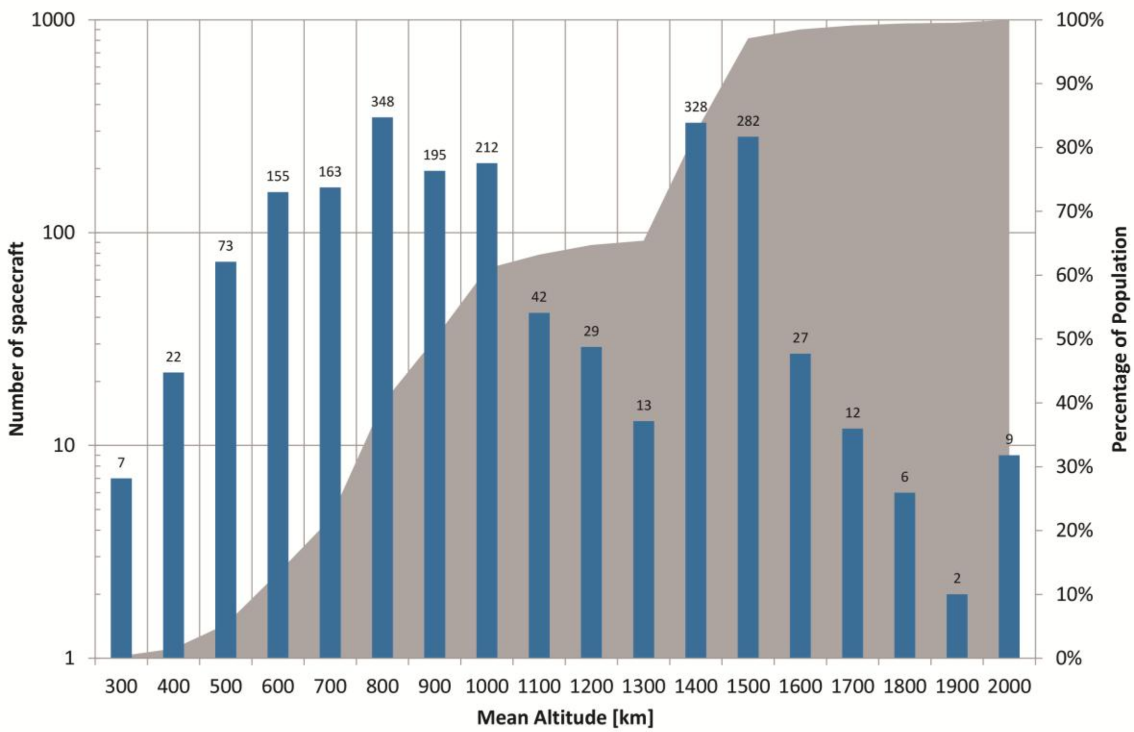

Fig. 6. Active spacecraft population as tracked by the NORAD in August 2011

VI. Assessment of Needs

Article VIII of the Outer Space Treaty states that ownership of objects launched into space is not affected by their presence in space,[28] while as previously discussed the Liability Convention defines liability for damage caused.[25] Hence, ownership and liability for damage that results from space debris remains with the spacecraft operator. If a constellation, such as Iridium, were to be wholly de-orbited by a passive means over a 25-year period, it is possible that the constellation owners would be liable for two previous generations of constellation as they de-orbit, that is over 130 spacecraft. Such a significant potential liability is likely to be unacceptable to most commercial organizations and/or insurers. However, it is of note that most spacecraft are not insured beyond the first few years of operations and hence any impact through insurance would be indirect. Due to this potential liability, it is likely that operators would want to de-orbit spacecraft in a much shorter period, probably using an on-board propulsion system; Iridium 9 was de-orbited in less than 3 years using its resistojets. However, the use of an on-board propulsion system to de-orbit a spacecraft requires that the spacecraft is operational, although perhaps only in a degraded state. If a spacecraft suffers a catastrophic failure, the use of on-board systems to de-orbit the spacecraft is not possible; however, the spacecraft operator/insurer remains liable for the risk and as such a 'fail-safe' de-orbit system may be desired. Of all the de-orbit concepts considered following the initial down-selection only drag augmentation offers the potential to be deployed from an inactive, tumbling spacecraft.

A. Passive Aerodynamic Stability

The drag augmentation device must be shown to be passively aerodynamically stable; once deployed the drag augmentation device must also be able to 'right itself', much like a shuttlecock. A significant volume of research has been conducted into such aerostabilized drag enhancement devices. For example, a hexagon-based pyramid, with six deployable booms angled back at a 25-degree sweep has been proposed for GTO to LEO transfer.[29, 30] The aerodynamic drag forces, restoring torques and oscillations that may be expected to occur during aerostabilization have also been considered for de-orbiting devices.[5, 31] In summary, the literature shows that the lower the sweep angle of the shuttlecock, that is the flatter the drag augmentation device, the greater the drag force generated, however smaller restoring forces are generated than at higher sweep angles leading to a potentially lower mean surface area to the mean-free flow of the atmosphere. A sweep angle of approximately 10 degrees is suggested as an optimal balance.[5] It is also shown in [5] that the initial angle of attack critically determines the de-orbit time as a high initial angle of attack will result in significant structural oscillations, which may not be fully damped by the time re-entry occurs, resulting in increased de-orbit times. The time of deployment can also have a significant effect on the aerodynamic stabilization characteristics of the drag augmentation device due to the atmospheric diurnal bulge, which can significantly aid structure damping.[5] Deployment between 2000 – 0600hrs, local solar time, results in faster de-orbit times, however the effect is less significant that the initial angle of attack.

Any gossamer structure must be validated aerodynamically stable and the impact of initial angle of attack and local solar time at deployment must be quantified to allow accurate estimates of de-orbit time to be generated. An alternative approach to the abovementioned validation of aerodynamic stability is to make use of a structure that offers the functional drag area regardless of the angle of attack, such as a balloon or the IDEAS system that was used on the MICROSCOPE satellite[32]. Although such a design would mitigate the need for passive aerodynamic stability it will result in a mass penalty.

B. Host Vehicle Interface & Integration

The drag augmentation device should have minimal impact on the operation of the spacecraft; however, as a debris mitigation plan is required at the Phase 0 (equivalent to pre-phase A) stage (see ISO 24113) of mission definition it is perhaps unhelpful to think of any de-orbit device as anything other than integral to the spacecraft system design.

Assuming the gossamer structure is to work up to and including the eventuality of acting as a 'fail-safe' de-orbit system, deployment of the drag-augmenting surface must be possible without a command from the spacecraft. However, an unsolicited deployment must also be assured against. Unsolicited deployment of any device can be avoided by using a conventional 'arm-and-fire' sequence of telecommands. However, this requires that the gossamer structure have sufficient on-board systems to receive, process and action telecommands while attached to a tumbling spacecraft. It also requires that these systems remain operational for longer than the operational life of the remainder of the spacecraft, and that functionality cannot be lost due to any conceivable spacecraft failure mode. As such, multiple redundant systems, perhaps including antennas, would be required. Alternatively, the deployment of the gossamer structure could be triggered by the failure of the spacecraft to halt deployment. This could, for example, be achieved by a simple countdown being reset by telecommand from the spacecraft. However, a spacecraft failure just before a reset telecommand was due to be sent could significantly reduce the time before deployment of the drag-augmenting surface and hence significantly reduce the available time to recover the spacecraft. An alternative may be for some form of countdown to be trigged by the loss of electrical power from the spacecraft, perhaps by the use of a solenoid, powered by a local secondary battery, to restrain deployment; when the local secondary battery is drained the solenoid will no longer be able to restrain deployment and the drag-augmenting surface would deploy.

The drag-augmenting surface must also be able to deploy in a verifiable and controlled manner after an extended period of time in-orbit. As such, the materials chosen must have a verified extended shelf life. For example, many rigidizable, gas-inflated, materials have only a limited shelf life of, perhaps, a few years and as such do not appear suitable for use in end-of-life de-orbit devices. Similarly, the use of stored strain energy within booms is attractive, however the thermal cycling per orbit coupled with the length of time could have the effect of relaxing the stored strain energy and altering the bending stiffness of the booms. It is notable that an analogy can be found in space-qualified air bags, which reliably deploy after, often considerable, periods of time in-orbit through the use of a chemical reaction. In addition, any thin-film must be carefully chosen to allow an extended shelf life. For example, the use of a polyimide substrate with a thin, vapor deposited Aluminum coating, incorrectly folded could lead to cold-welding of the Aluminum coating to itself. Furthermore, some polyimide substrates exhibit a stronger memory effect than others and will hence deploy less well from a folded configuration. Finally, deployment of the drag augmentation surface must be unhindered by the spacecraft peripherals, such as antenna or solar arrays.

C. Safety Aspects

The use of drag augmentation to de-orbit a spacecraft prohibits a controlled atmospheric re-entry into, for example, the southern Pacific Ocean ‘spacecraft cemetery’. As such, the use of drag augmentation for de-orbiting is limited to spacecraft that will be sufficiently destroyed during re-entry due to the thermal loads encountered; hence, minimizing the risk to Human life. However, in addition to this it is possible for the drag augmentation device to significantly alter the heating load profile during re-entry and hence could result in a spacecraft that would otherwise have been sufficiently destroyed during re-entry actually surviving to the ground. As such, it must be verified that the addition of a drag augmentation device to a spacecraft will not adversely alter the heating load profile during re-entry. However, it is likely that this should not be achieved simply by jettisoning the gossamer structure, as it may be possible for the film to survive re-entry if not attached to a much heavier object. Therefore, any supporting booms for the drag augmentation surface should be designed to buckle at a nominal atmospheric density, ensuring the heating load profile during re-entry is not adversely altered.

D. Summary of Gossamer Structure for End-of-Life Disposal Mission Needs

A Summary of the needs of a gossamer structure for end-of-life disposal of spacecraft is given in Table 14.

Table 14. Needs matrix of a gossamer structure for end-of-life disposal

Number	Need
1	Fully passive and require no ground operations support
2	Passively aerodynamically stable in an attitude that maximizes the possible surface area to the mean-free flow of the atmosphere in all flow regimes
3	Able to deploy from a tumbling spacecraft
4	Able to minimize the initial angle of attack during deployment to the mean-free flow of the atmosphere
5	Able to deploy from an inactive, non-responsive spacecraft
6	Able to deploy autonomously at a predetermined local solar time
7	Should not increase the likelihood the spacecraft impacting other spacecraft during the de-orbit phase
8	An integral part of the spacecraft systems design from Phase 0 onwards
9	Able to deploy reliably and verifiably at the end of the spacecraft’s operational life
10	Able to avoid an unsolicited deployment
11	Able to deploy without hindrance from spacecraft peripherals such as antenna or solar arrays
12	Not adversely alter the heating load profile during re-entry, such as to reduce the amount of material destroyed during re-entry
13	De-orbit an object in $\leq$ 25-years through the use of atmospheric drag that would otherwise remain in orbit through the use of atmospheric drag in this period

VII. Conclusion

A parametric comparison of a gossamer structure for end-of-life disposal against a wide-range of other disposal concepts and strategies found that the structure was best suited for atmospheric drag augmentation. Consequently, the needs assessment was limited to low-Earth orbit. It was also found that exploiting solar radiation pressure on the gossamer structure was of limited value. However, it is of note that the simultaneous use of multiple forces was not considered. In general terms, it was found that if a spacecraft propulsion system remains functioning at the end-of-life then this will likely provide the most efficient means of de- or re-orbiting, especially when the propulsion system is only used to lower the orbit to a point where atmospheric drag will cause the orbit to decay within the required timeframe.

The principle advantage of any atmospheric drag augmentation system for end-of-life disposal was found to be the entirely passive operational mode, requiring an ability to passively maximize the surface area to the mean-free flow of the atmosphere in all of the flow regimes in which it will pass through. This operational mode allows any atmospheric drag augmentation system to act as a 'fail-safe' system, which if the spacecraft suffers a catastrophic failure would activate. The use of an atmospheric drag augmentation system is applicable to only low and medium mass spacecraft, or spacecraft that are unlikely to survive atmospheric re-entry, hence minimizing risk to human life.

Acknowledgments

This work was conducted within the European Space Agency's ARTES-5.1Workplan 2010; Activity Reference 4C.021.

References

[1] Petro, A.J., "Techniques for Orbital Debris Control", *Journal of Spacecraft and Rockets*, Vol. 29, No. 2, 1992, pp. 260 – 263.

[2] Meyer, K.W., Chao, C.C., "Atmospheric Reentry Disposal for Low-Altitude Spacecraft", *Journal of Spacecraft and Rockets*, Vol. 37, No. 5, 2000, pp. 670 – 674.

[3] Campbell, S., Chao, C.C., Gick, A., Sorge, M., "Orbital Stability and Other Considerations for U.S. Government Guidelines on Post-Mission Disposal of Space Structures", In: Proceedings of the Third European Conference on Space Debris, 19 – 21 March 2001, Darmstadt, Germany. Ed.: Huguette Sawaya-Lacoste. ESA SP-473, Vol. 2, Noordwijk, Netherlands: ESA Publications Division, ISBN 92-9092-733-X, 2001, pp. 835 – 839.

[4] Harkness, P.G., "An Aerostable Drag-Sail Device for the Deorbit and Disposal of Sub-Tonne, Low Earth Orbit Spacecraft", PhD Dissertation, University of Cranfield, 2006.

[5] Roberts, P.C.E., Harkness, P.G., "Drag Sail For End-of-Life Disposal from Low Earth Orbit", *Journal of Spacecraft and Rockets*, Vol. 44, No. 6, 2007, pp. 1195 – 1203.

[6] Lappas, V., Adeli, N., Visagie, L., Fernandex, J., Theodorou, T., Steyn, W., Perren, M., "CubeSail: A Low Cost CubeSat based solar sail demonstration mission", *Advances in Space Research*, Vol. 48, No. 11, 2011, pp. 1890 – 1901.

[7] Lücking, C., Colombo, C., M^{c}Innes, C.R., "A Passive De-orbiting Strategy for High Altitude CubeSat Missions Using a Deployable Reflective Balloon", In: 8th IAA Symposium on Small Satellites, 2011-04-04 - 2011-04-08, Berlin.

[8] Lücking, C., Colombo, C., M^{c}Innes, C.R., "A Passive Satellite Deorbiting Strategy for Medium Earth Orbit Using Solar Radiation Pressure and the J2 effect", *Acta Astronautica*, Vol. 77, Aug. – Sept. 2012, pp. 197 – 206.

[9] Macdonald, M., M^{c}Innes, C.R., "Solar Sail Science Mission Applications and Advancement", *Advances in Space Research*, Vol. 48, Issue 11, pp.1702-1716, 2011.

[10] Voronka, N.R., "Technology Demonstrator of a Standardized Deorbit Module Designed for CubeSat and RocketPod Applications", In proceedings of *19th Annual Small Satellite Conference*, Logan, Utah, USA, 2005

[11] M^{c}Innes, C.R., "*Solar Sailing: Technology, Dynamics and Mission Applications*", Springer-Praxis, Chichester, 1999.

[12] Bekey, I. "Project Orion: Orbital Debris Removal Using Ground-Based Sensors and Lasers," Proceedings of *Second European Conference on Space Debris*, Darmstadt, Germany, 1997.

[13] Bondarenko, S., Lyagushin, S., and Shifrin, G. "Prospects of Using Lasers and Military Space Technology for Space Debris Removal," Proceedings of *Second European Conference on Space Debris*, Darmstadt, Germany, 1997.

[14] Rubenchik, A. M., Barty, C. P. J., Beach, R. J., Erlandson, A. C., and Caird, J. A. "High Energy Laser for Space Debris Removal" Lawrence Livermore National Laboratory, Santa Fe, NM, United States, LLNL-PROC-423323, 2010.
[15] Campbell, J. W. "Using Lasers in Space: Laser Orbital Debris Removal and Asteroid Deflection", DIANE Publishing, Occasional Paper No. 20, Center for Strategy and Technology, Maxwell Air Force Base, Alabama, 2000.
[16] Andrenucci, M., Pergola, P., Ruggiero, A., Olympio, J., and Summerer, L. "Active Removal of Space Debris: Expanding foam application for active debris removal", European Space Agency, Advanced Concepts Team, Ariadna Final Report (10-4611), 2011.
[17] C. Bombardelli, J. Peláez, "Ion Beam Shepherd for Contactless Space Debris Removal", *Journal of Guidance, Control, and Dynamics*, Vol. 34, No. 3, pp. 916–920, 2011
[18] Starke, J., Bischof, B., Foth, W.-P., Günther, H., Kinnersley, M., David, E. "ROGER a potential orbital space debris removal system", Proceedings of the *62nd International Astronautical Congress*, Cape Town, South Africa, 2011.
[19] Phipps, C. "Catcher's Mitt as an Alternative to laser Space Debris Mitigation", *AIP Conference Proceedings* Vol. 1278, No. 1, pp. 509-514, 2010.
[20] Ganguli, G., Crabtree, C., Rudakov, L., Chappie, S., "*A Concept for Elimination of Small Orbital Debris*", arXiv:1104.1401v2, 2012.
[21] NASA Technical Memorandum, "U.S. Standard Atmosphere", NASA-TM-X-74335 [NOAA-ST 76-1562], October 1976.
[22] Macdonald, M., McInnes, C.R., "Solar Sail Science Mission Applications and Advancement", *Advances in Space Research*, Vol. 48, Issue 11, pp.1702-1716, 2011.
[23] McInnes, C.R., "*Solar Sailing: Technology, Dynamics and Mission Applications*". Springer, 2004.
[24] Forward, R. L., Hoyt, R. P., Uphoff, C. W., "Terminator TetherTM: A Spacecraft Deorbit Device", *Journal of Spacecraft and Rockets*, Vol. 37, No. 2, March-April 2000.
[25] United Nations Office for Outer Space Affairs, "Outer Space Treaty", 1967.
[26] Griffin, M.D., French, J.R., "*Space Vehicle Design, Second Edition*", AIAA, Reston, VA, 2004, pp. 103 - 192
[27] Wertz, J. R., Larson, W. J., "Space Mission Analysis and Design", 3rd Edition, Microcosm/Springer, 1999.
[28] United Nations Office for Outer Space Affairs, "Convention on International Liability for Damage Caused by Space Objects", 1971
[29] Gloyer, P., "*Aerobraking to Lower Apogee in Earth Orbit with the Small Payload Orbit Transfer (SPORT) Microsatellite Vehicle*" AeroAstro Inc, SSC01-XI-8, 2001.
[30] Gloyer, P., "Aerobraking Technology for Earth Orbit Transfers" AeroAstro Inc, SSC02-VII-2, 2002.
[31] Roberts, P., "Drag Sail Dynamics for End of Life Deorbit", AAS 04-209, *14th AAS/AIAA Space Flight Mechanics Conference*, Maui, Hawaii, USA, 2004.
[32] Dupuy, C., Le Couls, O., "Gossamer Technology to Deorbit LEO Non-Propulsion Fitted Satellite", Proceedings of *The 40th Aerospace Mechanisms Symposium*, NASA/CP-2010-216272, pp. 301 – 308, 2010