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Self-cremating Satellite

A solution to reduce lifetime of space debris

Abstract:

Deorbiting sail devices have already been developed to reduce space debris especially in low earth orbit. However, they are still limited to nano or micro satellites. In this paper, limitation and challenges of current deorbiting devices have been studied. A new concept, the so-called self-cremating satellite is proposed. This concept can be implemented on future satellites as well as upper rocket stages in low earth orbit. This deorbiting device uses existing membranes of the satellite to increase drag forces through a dedicated mechanism. It also exposes the satellite structure to harsh space environment, which degrades it eventually. The issue of uncontrolled rotation of the satellite after end of life (EOL) is also addressed. The self-cremating satellite promotes sustainable use of the orbital environment as well as the earth's atmosphere by reducing the rate of generation of space debris.

Esha	Student or Professional

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List of abbreviation

ACS: Attitude Control System
AO: Atomic Oxygen
EOL: End of life or post-mission orbital life of satellite
ISS: International Space Station
LEO: Low earth orbit (< 2000 km from earth)
LLDP: Linear Low-density Polyethylene
MLI: Multi-Layer Insulation
PMMA: Polymethyl methacrylate
PMP: Polymethylpenent
S/C: Satellite
SCS: Self-cremating Satellite
SRP: Solar Radiation Pressure
UN: United Nations

Self-cremating Satellite

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1 Introduction

Accumulation of space junk is increasing exponentially due to the so-called Kessler effect, which refers to non-terminating process of collision between the fragmented space junks [1]. Collision creates debris that leads to more collision. In addition to this, the production of satellites increased exponentially during the last decade [2] [3]. The Current rate of production is 20 satellites/ week [4]. Therefore, the issue of growing space debris is of great interest for the sustainable development of space technology. The UN implements and regulates space debris mitigation guidelines. The international standard ISO-24113[5] specifies each satellite launched in LEO should re-enter the earth's atmosphere before 25 years of its end of life (EOL). Despite these guidelines, 90% of satellites weighing 500-1000 kg and 60 % weighing 100-500 kg would fail the 25 years EOL regulation based on an EOL estimation of satellites launched between 2015-2020 [6]. There are several reasons behind this non-compliance. The higher the orbit of the satellite is, the longer it will take to re-enter [7]. Larger satellites are generally launched to higher altitudes. The natural orbital decay period of an object decreases with the increase in its area to mass ratio [8]. However, the development of lightweight structures for heaver and larger satellites is still a challenge. Hence, there is a requirement of using effective deorbiting devices for heavy satellites.

Several deorbiting devices have been developed to accelerate the natural orbital decay rate by exploiting perturbation forces, which exist in the orbit, such as solar radiation pressure (SRP), earth's magnetic forces, aerodynamic drag etc. These devices increase the effect of these perturbation forces to reduce its orbital energy. Reduction of the orbital energy forces the satellite to a lower orbit. One of the deorbiting devices, which will be studied in this paper, is sail device. Over the last decade, several sail devices have been developed utilizing a deployable large surface [9-19]. This increases the surface area and therefore increases the area to mass ratio.

2 State of the art

There are two types of sail devices. When the sail device uses solar radiation pressure created by momentum-transfer of photons on the large reflecting surface to reduce orbital energy, it is called a solar sail. It is generally used for higher altitudes. At lower orbits (<500 km), the atmospheric drag is the dominating force as the density of the air increases with decrease in altitude. Drag forces created by air molecules on the large surface area working against the velocity vector of the satellite reduce its kinetic energy. Hence, this device is called a drag sail. A sail device consists of a large thin membrane, a stowing device to keep it folded until it is needed and a deploying device. During the mission period, the membrane is stowed within the satellite. After end of life (EOL), this membrane is deployed by a deploying mechanism. The area of the sail membrane depends upon the mass of the satellite as well as the altitude of the orbit and the targeted deorbiting time. Some sail devices and its capabilities have been studied and summarized in Table 1. In this table, some deorbiting satellites are compared with respect to its overall mass, its maximum possible projected area without the sail membrane, its orbiting altitude, the sail area and estimated deorbiting time. Nano-Sail D2 uses a sail membrane of 10 m^2, which is 333 times bigger than the area of the satellite [16]. It takes less than one year to re-enter. However, Asteroid Finder takes more than one year to re-enter with a sail membrane of 25 m^2, which is 43 times larger than the projected area of the satellite [13]. Similarly, Gossamer-2 takes 4 weeks to re-enter with a sail membrane, which is 1333 times bigger than the projected area of the satellite [20].

Table 1 Analysis of current sail devices with comparison of sail area and deorbiting time and mass of the satellite

Satellite	Mass of satellite (kg)	Projected area of satellite (m^2)	Orbit (km)	Sail area (m^2)	Deorbiting time	Sail device	Reference
2U cubeSat	2.66	0.02 (2U)		3	-	AEOLDOS	[9]
Nano-Sail D2	4	0.03 (3U)	640	10	<One year	Nano-Sail	[16]
TechDemoSat-1	150	0.63	686	>6.6	<25 years	Icarus DOS	[19]
Asteroid Finder	140	0.572	850	25	>One year		[13] [10]
Gossamer-2	57	0.3	500	400	4 weeks	Sail with ACS	[20]

According to present sail technology, the sail membranes are 2 to 4 magnitude larger than the projected area of the satellite. Therefore, they are limited to mini-satellites (100-500 kg). Satellites weighing >500 kg, are generally launched to higher orbit and would need even larger sail areas. The larger the sail area, the higher is the risk of collision of the sail membrane with space debris. Hence, the probability of successful deorbiting without generating further debris reduces. Moreover, folding such a big membrane and stretching it after EOL requires a complicated mechanism. Furthermore, it occupies a specific mass and volume of the payload, which reduces economic effectiveness. Therefore, rather than creating the possibility to increase the sail area, the efficiency of current sail devices could be improved by optimizing the design of sail membranes and controlling its orientation (attitude control) in the orbit.

2.1 Attitude control of sail device

For effective reduction of the orbital energy throughout the deorbiting period, the sail must be oriented against sunlight in the case of a solar sail, or against the velocity vector in case of a drag sail. However, due to several perturbation forces caused by solar activity, gravity gradient and atmospheric density distribution etc. a satellite can tumble. Hence, an attitude control system (ACS) is required to stabilize the orientation of the satellite against a particular direction. An active attitude stabilizing system (ACS) controls and changes the orientation of the satellite to maximize the deorbiting rate [21] [22]. However, in passive stabilizing systems, the orientation of the satellite is kept constant or stabilized in a fixed direction throughout the deorbiting period. This can be done by optimizing the sail design and the initial deploying attitude. [22]

2.2 Design of sail device

A flat drag sail device cannot control tumbling of the satellite in the absence of an ACS. Hence, the probability of the sail membrane orientated against the velocity vector is 1/6 in case of a cuboidal satellite. A DLR mini satellite “Asteroid Finder” takes 390 days with a sail membrane of 25 m^2 to re-enter the earth’s atmosphere (Table 1). However, the same satellite, in a simulation takes 21 days assuming that the sail offers a constant area throughout the period. A graph of cross-area to orbiting time, shows that most of the time, the sail membrane was not orientated against the air-velocity vector [13]. Similarly, NanoSail-D2 mission took 240 days to de-orbit whereas the estimation was 70-120 days [15-16]. Due to uncontrolled tumbling of the satellite, the effective drag area during deorbiting period is much lower than nominal deployed sail area. This effective area can be maximized by changing the 2D sail device to a 3D sail device. This will increase the probability of the sail membrane being orientated against the velocity vector or solar radiation. Some example of 3D sail devices are discussed in the following sections:

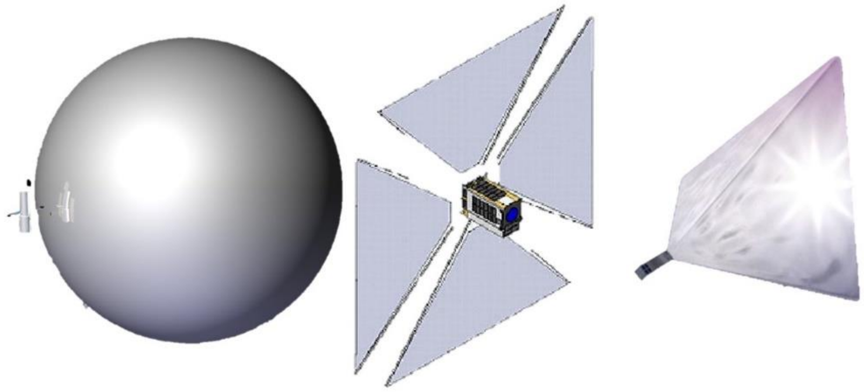

Figure 1 some 3D sail devices. left: Gossamer (GOLD) balloon[23], center: shuttlecock shape [24] and right: Quasi-rhombic pyramidal device [25]

2.2.1 Inflating balloon sail device

Gossamer orbit lowering device (GOLD) presents a spherical sail device in the form of a balloon as shown in Figure 1(left). It inflates a balloon using a large ultra-light thin inflatable envelop under a controlled deployment, inflation and pressure system. This system can be used for micro to medium satellites[1]. It utilizes torque generated by solar pressure, atmospheric drag as well as the gravity gradient all together to reduce the orbital energy. Due to its spherical shape, there is no need to control its orientation against a particular direction to de-orbit. However, inflating a balloon in space where the outside pressure is a few pico- bars is extremely challenging. There is also a risk of deflation due to striking of the balloon with space debris or meteoroids as well [26]. Kapton film and LLDP have been proposed as the balloon material [23], but a detail investigation of the feasibility has not been carried out yet.

A conical balloon shape was also proposed which uses similar technique of inflating the balloon into a conical shape but by means of composite springs. The satellite will be at the apex of this cone, which will create the desired equilibrium condition against perturbation forces. However, this system is quite heavy compared to the limited payload [27-28].

To author's knowledge, a super-tough material that can sustain extremely high strains associated with inflation of a football-field-sized balloon, maintain pressure and sustain collisions with space debris is still under development and not state of the art today. The area of such a balloon for heavier satellite is of the size two to four orders of magnitude of the satellite area.

2.2.2 Shuttlecock sail device

A 3D sail device which has been mentioned in [29] and shown in Figure 1 (center), utilizes aerodynamic torque to stabilize the rotation of the satellite. The sail device is similar as badminton shuttlecock, which has feathers like membranes with the satellite at the cock region of the shuttlecock. Hence, the center of the drag pressure is behind the center of the mass, which controls pitch-yaw rotation. Apart from this, an active magnetic control system is also inserted for roll stabilization. However, this sail system is complicated to deploy. It has both passive and active ACS and limited to orbit below 500 km.

2.2.3 Quasi-rhombic pyramidal sail device

The University of Glasgow proposed a 3D sail device for nano-satellites, which deploys a series of membranes such that it forms a quasi-rhombic pyramid (QRP) as shown in Figure 1(right). Due to the shape, the apex of the pyramid is always pointed towards the sun. Any perturbation in orientation of

[1] For the classification of small satellite see Appendix

pyramid is damped with the help of fluid dampers. Hence, the QRP stays in an oscillatory equilibrium condition like a pendulum. This system is complex and requires an optimized initial attitude for deployment in order set the equilibrium position towards the sun. [21] [30-32].

All these devices increase the area to mass ratio of the satellite by spreading a membrane in order to increase the projected area. However, the ratio can also be increased by decreasing mass. Moreover, these devices use external membranes and complex stowing and deploying mechanisms. This increases the parasitic weight. This parasitic weight increases with increase in sail area for heavier satellites. This weight can be decreased by eliminating external membranes and its stowing mechanism. In this paper, a new design of a deorbiting device (self-cremating satellite) will be proposed where the main focus is to eliminate the need of external membrane and reduce the mass of the satellite during deorbiting time. Another focus is to reduce the effect of tumbling (rotation) of the satellite on drag area.

3 Self-cremating satellite (SCS)

A self-cremating satellite (SCS) uses existing multi-layer insulation (MLI) instead of additional membranes. This can be done by peeling off the MLI from the satellite after EOL. MLI is used for thermal insulation purposes in almost all satellites except some pico or nano-satellites. It covers the outer-surface of satellite in order to protect it from the outside environment. It is made of multiple layers of very thin, reflective metal coated plastic films, which are separated by low conducting or insulator spacer material. For LEO satellites, generally 15 to 20 layers are used [33]. These layers are extremely thin comprising about 5-12 mm thickness overall [33-36]. However, after end of life, the function of the MLI is undesirable as it increases the life of the satellite by protecting it from the space environment. A special mechanism will be installed between the satellite structure wall and the MLI. This mechanism will detach the MLI. Hence, the structural wall of the satellite come into direct contact to atomic oxygen in combination with ultraviolet radiation, which will oxidize it at a faster rate as compared to a conventional satellite degradation process. MLI will be detached and stretched out as a flap or peeled off like banana as shown conceptually in Figure 2.

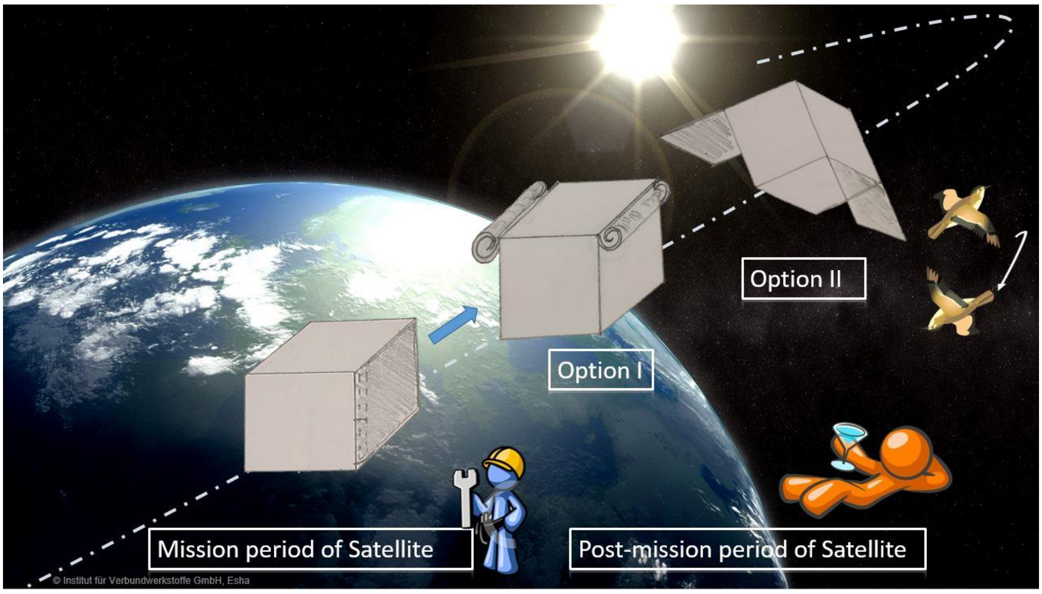

Figure 2 Conceptual sketch of self-cremating satellite (SCS)

3.1 Mechanism

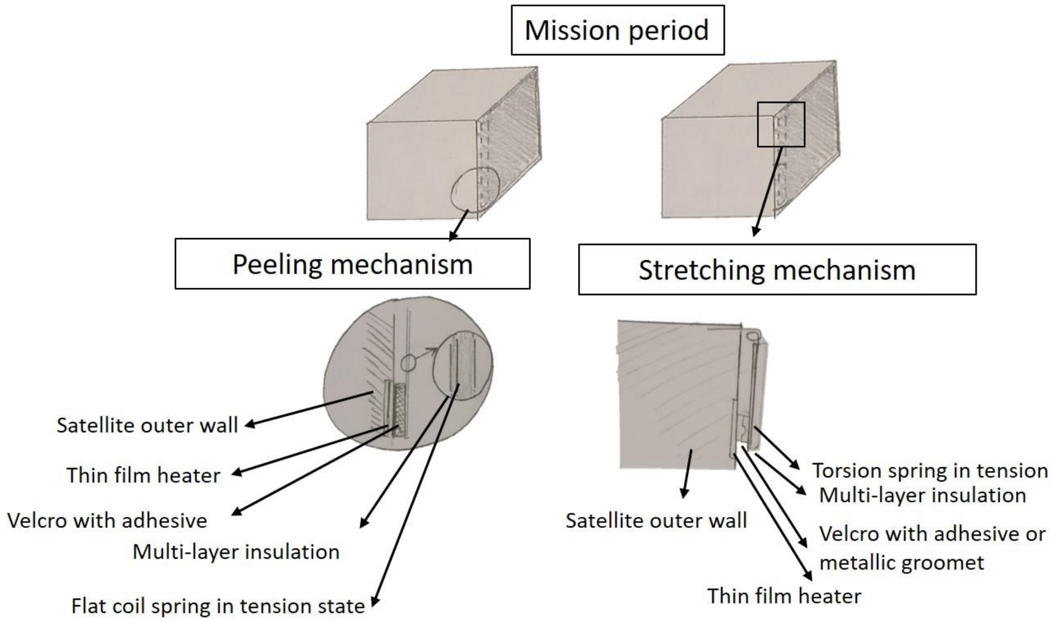

Figure 3 Proposed mechanism for peeling of MLI (left) and stretching of MLI (right)

As shown in Figure 3, a peeling/stretching mechanism will be attached between the MLI and the satellite outer structural wall. This mechanism will remain inactive throughout the mission period. After the mission period, this mechanism will be activated through an automatic system triggered by an on-board computer. As shown in Figure 3, there are different ways to remove the MLI, one of is the peeling of the MLI using a peeling mechanism and another is stretching the MLI as flap using a stretching mechanism. Removing the MLI is an analogous to getting sun-burnt on earth which damages the skin. Similarly, if the MLI will be removed from the satellite, the material of structural wall will start degrading. Therefore, both parameter of area to mass ratio will be tackled after EOL.

In the peeling mechanism (left side of Figure 3), there is a thin film heater, a flat coil spring inserted inside MLI in stressed condition similar to a slap ruler bracelet for kids, and adhesive or attaching agent of the MLI. Whereas, in the stretching mechanism (right side of Figure 3), there is a torsional spring inside the layers of MLI in stressed condition instead of the coil spring. The thin film heater is a heating element printed on a thin polyimide film or any other material qualified for space application. Such heaters are widely used in computer hardware, micro-electronics, pressure vessel technologies, medical industries, military and aeronautics etc. [37]. They can be printed in almost any shape and size as needed. They are comprised of three thin layers of size < 2 µm [38]. The outer layers are electrically insulating whereas the core layer is electrically resistive. This core layer is connected to an electrical power source. It will be attached to the satellite structure beneath the MLI through a proper space application qualified adhesive.

After EOL or after the satellite loses its functionality due to disconnection from ground or collision with space debris, these thin film heaters will automatically be activated, for example by connection to the solar cells or the power source of the satellite. The heat produced must be designed such that it is capable to melt the adhesive of the attaching agent during sun face time. The torsional spring attached to the MLI in case of the stretching mechanism or the coil spring in case of peeling mechanism will return to its neutral position as shown in Figure 2 (option I for peeling mechanism and option II for stretching mechanism).

3.2 Functionalities of self-cremating satellite (SCS)

The concept of a self-cremating (SCS) has three major functionalities in regards to deorbiting. First, it increases the ballistic coefficient by increasing the drag area and second, it accelerates material degradation by exposing the unprotected satellite to the space environment. The structural walls of the satellite come into direct contact to atomic oxygen in combination with omni-directional ultraviolet radiation and solar radiation. This will lead to oxidation of the satellite material in the form of CO_2, H_2O and metal oxides. However, the material degradation is very slow. For example: after 4 years of space exposure in ISS, a sheet of PEEK polymer lost 0.1077 g and 0.0252 cm thickness. [39]

High thermal cycling will generate fatigue in the material. Hence, the structural walls will not only eroded but also embrittled. Material embrittlement and its effect were studied in MISSE 5 polymer mission [40]. 37 polymer samples were exposed to omni-directional charged particle radiation, thermal cycling and low doses of atomic oxygen and solar radiation for 13 months. 18 samples were severely damaged showing strain-induced surface cracks and erosion even though they were exposed to low doses of solar radiation and atomic oxygen under a 127 µm thick layer of Kapton. Polymers like PMMA and PMP became so brittle that they fractured on-orbit into pieces. However, these samples are 50.8 µm thin. This shows that the material degradation is very slow. Hence, it ensures that unwanted disintegration of the satellite during deorbiting period is very unlikely to happen. However, at the time of re-entry, an eroded and embrittled satellite will disintegrate faster than a conventional satellites. Hence, it increases the probability of disintegration during re-entry.

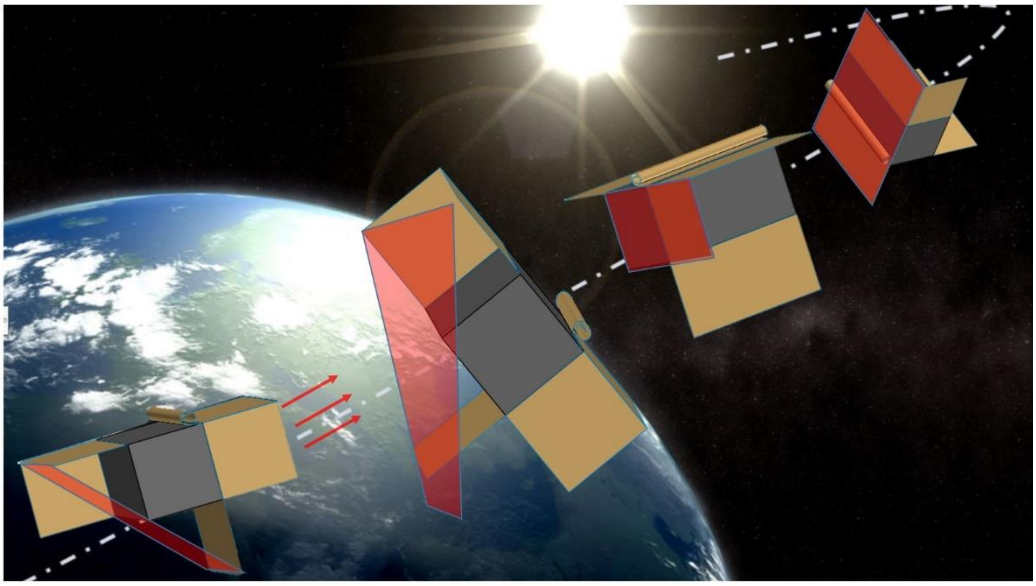

Figure 4 An example of tumbling of self-cremating satellite. One projected membrane will always be orientated normal to velocity vector or solar radiation irrespective of rotation. Red plane is the project drag area at each rotation

Another functionality of the SCS is the efficient use of the MLI as sail/drag membrane. Spreading out the MLI in SCS will be optimized and planned in such a way that it will spread the sail membrane in all possible directions. (example; Figure 4) Therefore, one sail membrane will always be orientated against solar radiation or air-velocity vector. Hence, tumbling of the satellite will not affect the drag area in case of drag sail or SRP incident area in case of solar sail as shown in Figure 4. The projected area is always bigger than the satellite in each orientation. Hence, there is no need of an attitude stabilization system in a self-cremating satellite.

4 Application and benefit of Self-cremating satellite

A self-cremating satellite can be a satellite or a rocket upper stage, which is initially covered by multi-layer insulation during mission period. After EOL, the same insulating layer can be used to increase the area to mass ratio. Hence, this concept can be used only to those systems, which use MLI. It is especially beneficial for bigger satellites as they have a larger surface area of MLI.

Some of the benefits of using this concept for future satellites or rocket bodies are:

1. This idea deals with a technique through which satellite can lose mass and increase area after their operational time. This will increase the rate of orbital decay throughout the EOL.
2. The effect of mass loss is more evident on bigger satellites. Bigger satellites have more surface area hence they will receive more AO flux [39].
3. Multi-layer insulation, which is wrapped around the satellite during mission, will be used as sail membrane after EOL. Hence, there is no need to increase the weight budget for external membrane and its associated stowing mechanism.
4. It does not use any propulsion system or any weight or cost expensive mechanism.
5. Tumbling of the satellite does not affect its projected area. Therefore, it uses the area of sail membrane more effectively as compared to conventional sail devices.
6. Material degradation due to harsh space environment is a slow process [39]. Rate of degradation depends upon the property of the material and the area of exposure. However it embrittle the structure. Hence, it leads to safe disintegration during re-entry.

5 Discussion

Sail devices for satellite systems have been in development for more than a decade. However, they are still limited to nano or micro satellites due to the requirement of a relatively large sail membrane. Through the concept of a self-cremating satellite, the area of sail membrane can be used efficiently. Instead of spreading one big membrane, it spreads several small membranes in every direction. This reduces the effect of tumbling on drag area. Hence, it can be used for heavier satellites without increasing the sail area dramatically. Moreover, mass loss due to exposure to a harsh environment will be more effective in the case of mini to large satellites as the surface area will be larger. It will not only degrade the structure but also increase the probability of safe disintegration during re-entry and thereby reducing casualty risk. Bigger satellites are more prone to increasing space debris generation due to cascading collision as described by the Kessler effect.

Hence, it is important to design a sail device for satellites heavier than 100 kg. With the concept of self-cremating satellite, this can be addressed without using huge external membranes or increasing parasitic weight. The mechanism described in this paper is currently a concept. In further studies, aerodynamic analysis of the projected area of the self-cremating satellite is required to optimize the effective drag area. A patent application has already been submitted. This project aims to decrease the accumulation rate of space junk and its influence on environmental footprints.

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7 Appendix

Classification of cube satellite on the basis of weight

Type of satellite [41]	Mass (kg)
Large satellite	>1000
Medium satellite	500-1000
Small satellite	<500
Mini satellite	100-500
Micro satellite	10-100
Nano-satellite	1-10
Pico-satellite	0.1-1