Low-thrust Missions for Expanding Foam Space Debris Removal

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Abstract: Active space debris removal missions are a necessary way to control and reduce debris growth, in order to permit future sustainable space activities. The method here proposed can be classified as a momentum exchange method, and in particular it can be thought as a drag augmentation system. The core idea is to develop a system able to apply an expanded foam ball around a target debris enlarging its area-to-mass ratio to increase the natural atmospheric drag and thus significantly decelerating and deorbiting the debris. The drag augmentation system here proposed does not require any docking system; an uncontrolled re-entry takes place ending with a burn up of the object and the foam in the atmosphere within a given time frame. The method requires an efficient way to change orbits between two debris. The present paper analyses such a system in combination with an Electric Propulsion system. Mission time, propellant and foam mass requirements are assessed and a preliminary platform sizing is presented.

Nomenclature

A	=	debris cross-sectional area
A_D	=	debris cross-sectional area
a	=	semi-major axis
$\bar{a}$	=	mean acceleration
$\vec{a}_{drag}$	=	atmospheric drag perturbation
C_d	=	drag coefficient
Δt	=	maneuver time
E	=	eccentric anomaly
e	=	eccentricity
i	=	inclination
m	=	debris mass
m_0	=	spacecraft mass at the beginning of thruster firing
m_p	=	propellant mass
$\bar{m}_{SC}$	=	average spacecraft mass
μ	=	Earth gravitational parameter

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ρ	=	atmospheric density
T	=	thrust
V	=	orbital velocity
V_D	=	debris volume
$\vec{V}_r$	=	relative velocity
ω	=	argument of perigee
Ω	=	right ascension of the ascending node

I. Introduction

THE increase of space debris is considered as one of the main threats for future sustainability of space activities and space access. The risk of debris collisions and the potential cascading effects could, due to their number and broad distribution, prohibit future human and robotic space missions.

Between 1957 and 2008, indeed, approximately 4600 launches have placed some 6000 satellites into orbit. Among these, about 400 were launched beyond Earth into interplanetary trajectories, but of the remaining ones only about 800 are nowadays still operational. This means that roughly 80% of space objects are uncontrolled debris. To these, launcher upper stages have to be added in order to have a rough idea of the large debris population. Furthermore, considering also smaller debris caused by explosions, fragmentations, collisions, accidental discharge and similar events, the whole debris population comprises millions of objects[1]. Space debris are not uniformly distributed, but are concentrated in the most used and thus currently most useful orbits, in particular in the Low Earth Orbit (LEO) and in the Geostationary Earth Orbit (GEO).

In the recent years, space debris mitigation guidelines have been adopted at international level, though while necessary, indications are that even if fully implemented, these might not be sufficient to solve this problem in the long term. Active debris removal missions might, indeed, be necessary to clean up certain target space regions where the debris threat is more hazardous both for potential commercial or human mission and for the risk of further debris collisions[2]. Thus, a number of active debris removal concepts has been described, such as: electromagnetic methods (i.e. electrodynamic tethers[3], magnetic sail), momentum exchange methods[4] (i.e. solar sail, drag augmentation device[5]), remote methods (i.e. lasers[6]), capture methods (i.e. nets[7-8]) and modification of material properties or change of material state.

In this paper, an innovative debris removal system based on expanding foams and its associated electric propulsion system is proposed, analyzed and described. As it intrinsically relies on atmospheric drag, its application is limited to the debris population in the LEO region and, considering the high total impulse typical of multiple target missions, electric propulsion has been chosen as main propulsion system for the platform in charge of delivering the foam. More in detail, the method is based on the ejection of polymeric foams for the nucleation of a debris-tailored foam ball around the target objects, see Fig. 1. The resulting increase of the area-to-mass ratio augments the natural drag effect and thus leads to a considerable reduction of deorbiting time with respect to the natural one. Due to its innovative nature, this method allows overcoming the usual difficulties of typical active deorbiting systems (i.e. docking), though it adds as key technological development step the reliable foam expansion in vacuum and its attachment to the debris.

This paper presents, in Sec. II, the core idea of the method together with the main advantages and some drawbacks. Section III describes the preliminary mission analysis of the proposed active space debris removal method assessed by means of analytical approximations to estimate the velocity increment required to acquire target orbits. The main maneuvers taken into account are changes of the semi-major axis, inclination and right anomaly of ascending node. Foam balls deorbiting time is assessed in a medium atmospheric density scenario by means of an averaging the atmospheric drag effect during the re-entry, a simplification justified for such first order analysis due to typical deorbiting mission durations.

The spacecraft in charge of carrying and spraying the foam is sized in Sec. IV. It is supposed to be equipped with electric thrusters to be able to efficiently change orbits and thus serve multiple targets. For the platform design, both chemical and electric propulsion schemes are considered and compared. The gain offered by the electric propulsion in such a high total impulse missions with limited time constraints is highlighted and, accordingly, the preliminary sizing of the candidate spacecraft is carried out. The whole design is addressed at subsystem level considering the performance of a medium class launcher in Sun-synchronous orbits.

Finally, in Sec. V the performance of the foam-based method is evaluated in terms of the amount of foam and propellant required by the dedicated platform to accomplish its task. As test case, a representative list of potential space debris is considered, where foam balls to be nucleated around debris are sized considering both a suitable deorbiting time, compliant with the IADC guidelines[9], and the resulting increased impact probability.

II. Description of the method

The proposed active space debris removal concept belongs to the field of the momentum exchange methods, and in particular it can be thought as a drag augmentation system. The proposed system intends to define a reliable and easy way to perform such a drag augmentation. The core idea of the method is to develop a platform able to realize a foam ball around a target debris that enlarges its area-to-mass ratio such that the atmospheric drag can exert a significant influence to decelerate the debris. In this way, debris that would have required hundreds of years to deorbit, will re-entry in a prescribed time.

The above-described method can be thought as composed of different phases. These are, actually, the mission scenario phases:

- *Launch*: The platform in charge of targeting and deorbiting the debris has to be launched into an initial orbit. It is reasonable to assume that the most suitable choice for the launch orbit of the spacecraft is the one of the first target debris, thus the first maneuver the platform has to perform is the first debris rendezvous.
- *Foaming process*: In this phase the target debris has been reached and the actual foaming process takes place. During this stage the foam has to be ejected from the platform and reach the target debris, to stick to the debris surface, to grow in volume, and to eventually cover at least part of the targeted debris.
- *Debris Deorbiting*: The debris is now either partly or entirely encompassed by the foam and the natural deorbiting of the system (composed by debris and foam) begins.
- *Targeting of next debris*: The platform, operating its (electric) thrusters, performs a set of orbital maneuvers aimed at the interception of the next target debris, i.e. debris rendezvous.
- *Platform self-disposal*: Once the platform has completed its mission, the thrusters can be finally used to lower the orbit perigee and deorbit the spacecraft within the 25 *years* limit, as recommended by the IADC guidelines[9].

These phases are schematically represented in Fig. 1, where it is shown, in particular, the target debris interception (left), the foaming process (centre) and the debris deorbiting while the platform starts targeting the next debris (right).

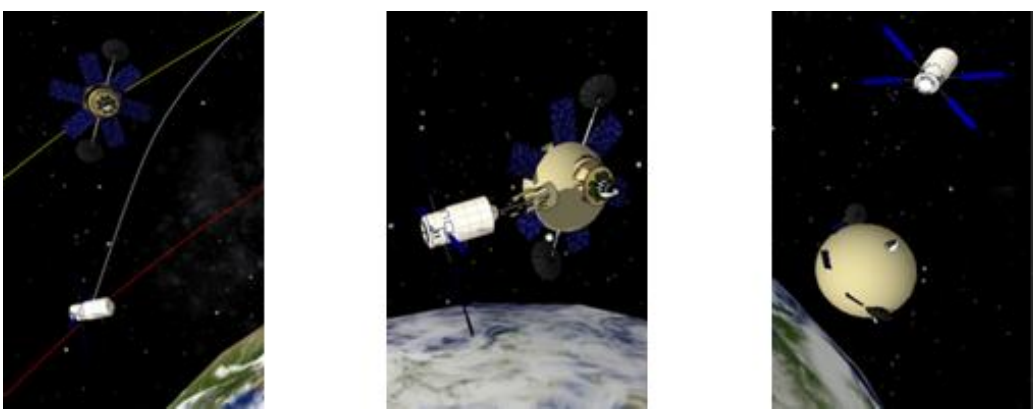

Figure 1. Representation of the proposed method: target debris interception (left), foaming process (centre), debris deorbiting (right).

The reliability of a foam-based strategy is related to its simplicity, e.g. the absence of control during the deorbiting time and the absence of any potential impact damage. The resulting object can be thought as a ball that (theoretically) offers always the same cross section; as a conservative assumption, it can be thought as containing in itself the target debris, which thus is assumed as no longer contributing to the drag and the foam can be nucleated also at a distance from the deorbiting platform. Furthermore, the most remarkable advantages of the proposed method are:

* A docking mechanism is not required, thus all the technological concerns related and all the potential hazards deriving from the docking with non-cooperative debris do not apply.
* The resulting foam structure does not require any control during the re-entry. Since the foam will ideally expand isotropically in space (vacuum and microgravity conditions)[10-11], resulting in a spherical form presenting always the same cross section, an uncontrolled re-entry (no thruster and a limited ground segment) can take place.
* The resulting foam structure around the debris is sturdier than any tether, sail or net-based structure.
* The momentum exchange is given only by the drag force decelerating the debris until it re-enters and burns up in the atmosphere.
* There are no potential hazards related to ground based systems, e.g. lasers passing through the atmosphere (although other kind of hazards might exist).

The main drawbacks of this method, instead, are related to the possibly difficult foam nucleation, incomplete attaching or not complete expansion in vacuum conditions. These phenomena could limit the foam-based method performance. Furthermore this method has nowadays a low TRL, especially concerning the foam behavior in vacuum and microgravity conditions. Accordingly, a foam able to expand in vacuum condition represents the key technology to be developed for the applicability of the proposed method.

Foams characteristics are extremely various and wide-ranging due to the different nature of the various substances that may form their gaseous part as well as their liquid or solid part. Two characteristics can be used to obtain a rough classification of foams: the structure of its cells and the material they originate from. There exist *closed-cell* foams, in which an insulating gas is retained within the cells, and *open-cell* foams in which the gas is free to pass within the foam cell lattice[12]. Several materials can compose the firm part of foams: glass, ceramic, metals, polymers.

A preliminary analysis[13] led to focus on polymeric foams, that turn out to offer the best combination of characteristics for the considered application. Polyurethane foams, in particular, have been chosen due to their reasonably high expansion factor (typically), flexibility and their relatively simple production. These foams can expand hundreds of times their initial volume on the ground and can reach densities of few kg/m^3. It has been assumed in the test case presented in Sec. V that the next generation foam, *ad-hoc* developed for such application would reach, at the end of the expansion phase, density values of 1 kg/m^3 [13].

The analysis of hazards and risks of the removal method proposed identified ground handling, launch, foam ejection and the debris deorbiting as the most critical phases. The latter two are the ones deserving particular attention as a potential fault could lead to a wide range of undesirable consequences both for the mission and for spacecraft operations. In case the foam ejected from the spacecraft does not reach or does not stick to the target debris, very likely it would start orbiting. However, it has been proven[13] that, by way of example, a small ball of foam of 0.1 kg would deorbit from 900 km within 4 months and from 600 km in less than two weeks, thus the threat it might represent is considered marginal. During the deorbiting phase, instead, the foam ball can hit or be hit by other debris. However, given the peculiar property of porous solids as foams and aerogels, most potential impacts would end with the impacting (small) debris encompassed by the foam, potentially without generation of new debris[13]. Potential benefits deriving by this effect have been assessed considering well-known debris population generated by satellite impacts[13]. For instance, the population generated by the *Cosmos-Iridium* impact would be reduced by 40% by an orbiting foam ball of 10 m radius thanks to this domino effect.

III. Mission Analysis Approach

Before starting the analysis of the proposed foam-based method, it is mandatory to provide methodologies to compute the behavior of the deorbiting object. In particular, in this section, some models required to assess the deorbiting time and the impact probability are provided. Furthermore, also the numerical methodology by which the deorbiting time is computed through the work is given.

The specific atmospheric model is of fundamental importance to estimate this deorbiting time, the specific one here considered has been chosen as it offers the chance to model different density regimes, from low to high. This model is also sketched in this section.

Moreover, considering only the atmospheric drag, it is clear that the larger the foamed debris, the better this method works. This does not hold anymore if also the impact probability is considered. The NASA90 impact probability model is here chosen and applied below (among several options detailed in Ref. 13]), as it can be easily implemented without relying on specific libraries.

A. Estimation of deorbiting time and impact probability

Considering a direct method, as the one described in Ref. 14, the integration of small acceleration values for the very long time needed by an object to reach high density atmospheric layers burning out, could be very expensive in terms of computation time. An alternative solution is the implementation of the Encke's method[14] based on the assumption that the integrated variables are small and so it is expected to be the integration error. This method needs, however, a very careful rectification to avoid numerical instabilities or large loss of precision[15], thus nowadays it tends to be avoided. An additional valid alternative to model the perturbation effects is represented by the Gauss form of the Lagrange Planetary Equations[16]]. These equations model the time evolution of the classical orbital parameters (semi-major axis a, eccentricity e, inclination i, right ascension of the ascending node Ω, argument of pericenter ω and mean anomaly M) under the influence of a non-conservative perturbation. This perturbation changes along the orbit due to the different contributions of the various terms composing this acceleration as Sun and Moon third body acceleration, atmospheric drag, Earth non-spherical gravitational field, solar radiation pressure and thruster acceleration[17]. As the typical orbital altitude of the target debris considered is between 500 km and 1000 km, it is possible to assume that the main perturbation affecting these objects is the atmospheric drag, which can be expressed as[17]]:

$$\vec{a}_{drag} = -\frac{1}{2}\rho C_d \frac{A}{m} V^2 \frac{\vec{V}_r}{\left|\vec{V}_r\right|} \tag{1}$$

where ρ is the atmospheric density value, C_d a dimensionless number reflecting the object configuration sensitivity to drag force, A and m respectively the object cross-sectional area and mass and V its relative velocity vector with respect to Earth atmosphere.

The two orbital elements more affected by this force are the semi-major axis and eccentricity, thus the focus in the estimation of deorbiting time is given to these elements. Typically both tend to decrease, but the initial value of eccentricity for debris in the LEO region is already close to zero. By means of some several substitutions for the relative velocity and the eccentric anomaly rate of change, it is now possible to obtain an expression for the instantaneous semi-major axis change due to the atmospheric drag with respect to eccentric anomaly (E):

$$\frac{da}{dE} = -\rho \frac{C_d A a^2 \left(1+e\cos\left(E\right)\right)^{3/2}}{m\sqrt{\left(1-e\cos\left(E\right)\right)}} \tag{2}$$

With a similar procedure, it is possible to obtain the instantaneous rate of change of eccentricity with respect to the eccentric anomaly[18].

$$\frac{de}{dE} = -\rho \frac{C_d A a}{m} \frac{\sqrt{1+e\cos\left(E\right)}}{\sqrt{1-e\cos\left(E\right)}}\left(1-e^2\right) \tag{3}$$

The simultaneous integration of Eqs. (2) and (3) has to be carried out numerically. This has been done through an adaptive Lobatto quadrature of the atmospheric force over the eccentric anomaly[19]. This quadrature is based on a four-points Gauss-Lobatto formula, i.e. a quadrature rule approximating a definite integral, by means of a weighted sum of function values at specified points within the domain of integration.

A specific model for the atmospheric density variation with the orbital altitude is then required for these integrations. Since our preliminary analysis does not assume a specific mission scenario, neither in terms of beginning of the mission nor in terms of mission duration, it is reasonable to assume a simple static model for the atmospheric density. One of the simplest static models is the Harris-Priester model[17,20], which relies on a number of tables listing reference density values obtained from observational data within a complete solar cycle.

It is important to note that for this method, as for any other method based on drag augmentation, it is more realistic to assume that an active debris removal mission would take advantage of a medium or even a high atmospheric density period. Consequently, some of the target debris should complete their deorbiting phase within a high atmospheric density period, while others would require more than few years to complete their deorbiting phase. Under this assumption, the deorbiting time provided by the medium atmospheric density model used in the present paper represents a realistic, likely conservative, scenario.

Together with the computation of the deorbiting time, the assessment of impact probability for orbiting objects due to the proposed method and for the foam balls themselves is a mandatory task for the estimation of hazards and risks related to the method and of a suitable debris-tailored foam ball size. We only focus in the following on possible space debris impacts, neglecting those due to micrometeoroids. This assumption is justified below 2000 km, where the orbital debris environment represents the major threat for space flight compared with the meteoroid environment[21].

NASA90 model[22] provides a simple and very fast debris flux calculation for orbital altitudes below 1000 km, but it does not take into account the existence of a large number of particles on eccentric orbits. While this model has been the first more or less detailed description of the debris environment, it is no longer considered up to date[21]. In spite of that, it remains one of the most valid options for preliminary analyses and impact probability estimations and is also used in the present paper. The NASA90 model computes the debris flux versus the impactor diameter by means of an analytical formulation. The flux is defined as the cumulative number of impacts on a spacecraft in circular orbit per square meter and year on the surface of an object randomly rotating around its center of mass[22]. This value is obtained as function of the minimum impactor diameter, the considered orbit altitude and inclination, the mission epoch and the solar radio flux.

One of the main advantages offered by the proposed method is the possibility to tailor the foam ball on the target debris orbit and characteristics. The applied foam ball sizing procedure is based on the assessment of debris deorbiting time and its impact probability for different values of the drag exposed area. The two quantities show opposite trends for growing values of the foam ball radius. The deorbiting time is computed by means of the approach described above in this section and the impact probability is obtained assessing the impact flux value through the NASA90 model for a minimum impactor diameter of 10 *cm*. The minimum of the curve given by these two contributions provides the sought foam ball radius for a given debris.

B. Low-thrust maneuvers

After the short description of the adopted approach, this section focuses on the methodology used to assess the performance of the method. In particular, we describe a mission scenario centered on a platform equipped with an electric thruster that moves from a debris to the next one after each the foaming process. In order to implement the preliminary low thrust mission analysis given the large number of potential target objects, some assumptions have been made:

- At the beginning of each mission, the platform is released by the launcher on the exact orbit of the first target, thus eliminating the need for additional orbital maneuvers. This means that, for the first targeted debris, the only task to be performed is the debris foaming.
- The orbits of debris are considered unchanged for the whole time needed to target each debris, i.e. no drag or Earth oblateness perturbation (J2 effect) is considered during the low thrust transfers neither for the debris nor for the platform.
- Platform and debris orbits are always accounted as circular and, (see plot in Fig. 5), the argument of perigee change maneuver has been neglected.
- No orbit phasing is considered to adjust the platform true anomaly with respect to the one of the target. This assumption seems reasonable, since a more sophisticated thrusting strategy would definitely allow avoiding further maneuvers.
- The time needed to encompass the debris with foam is neglected with respect to the time needed for orbital maneuvers.
- The mass of the spacecraft is considered constant during each orbital manoeuver and its value is estimated as the average between the mass at the beginning and at the end of the thruster operations.
- The platform trajectory is obtained by the semi-major axis and inclination change maneuver followed by a RAAN change.

- The natural change of the orbital RAAN, due to J2 effect, is neglected both for the platform and for the debris. This assumption results in a conservative mission profile, since it is always reasonable to assume a thrusting strategy capable of exploiting this perturbation.
- The total maneuver cost is obtained as the maximum velocity increment between the one necessary to perform the combined semi-major axis and inclination change and the one for the RAAN change maneuver. This assumption still provides reasonable results considering the previous considerations about natural RAAN drift.

Considering the assumptions listed before, it is possible to describe the mission profile as a sequence of predefined maneuver, each one assessed by means of analytical approximations to actively target each debris. The selection of the debris sequence is based on the comparison of the maneuver cost for all debris of the list.

In particular, the ΔV required for the combined semi-major axis and inclination change is assessed by means of the analytic Edelbaum approximation[23]:

$$\Delta V = \sqrt{V_0^2 - 2V_1V_0\cos\left(\frac{\pi}{2}\Delta i\right) + V_1^2} \tag{4}$$

where V_0 and V_1 represent, respectively, the orbital velocity on the initial and final orbit and Δi is the desired inclination change angle. It considers a constant acceleration to compute the low thrust transfer velocity increment between two circular inclined orbits by linearizing the Lagrange Planetary Equations around a nominal circular orbit[23].

The other maneuver that is modeled is the RAAN change. It can be analytically approximated by[24]:

$$\Delta V = \frac{\pi}{2}\sqrt{\frac{\mu}{a}}\left|\Delta\Omega\right|\sin(i)\ , \tag{5}$$

where $\Delta\Omega$ is the desired change in RAAN. This maneuver is performed using out-of-plane thrusting with burn arcs centered on the apices (i.e., the maximum and the minimum latitude points) under the assumption of almost circular orbits[24].

Once these values have been assessed, the mass of propellant needed to perform each of these maneuvers can be computed by means of the Tsiolkovsky Equation[18]. It is, moreover, possible to assess the time needed to perform each manoeuver, under the assumption of constant acceleration, by means of:

$$\Delta t = \frac{\Delta V}{\bar{a}} = \frac{\Delta V}{T}\bar{m}_{SC} = \frac{\Delta V}{T}\left(m_0 - \frac{m_p}{2}\right) \tag{6}$$

where $\bar{m}_{SC}$ is the average spacecraft mass, m_0 the value of the spacecraft mass at the beginning of thruster firing and $\bar{a}$ the resulting average acceleration on the spacecraft.

The mass of foam required to deorbit each targeted debris can be computed considering the foam density (1 kg/m^3 assumed) and the volume of the ejected foam. This value is obtained as the difference between the volume of the debris after and before the foaming process. Since the shape of target debris is unspecified, both the foamed debris and the specific debris are assumed as spherical objects. Under this assumption, the original debris volume, V_D can be estimated by its cross sectional area as:

$$V_D = \frac{4}{3}\pi\left(\frac{A_D}{\pi}\right)^{3/2}, \tag{7}$$

where A_D represents the average exposed area of the target debris. The final volume of the foam ball, instead, is computed by taking into account the foam ball radius for a specific debris[13] given by the procedure sketched at the end of Sec. III.A.

IV. Deorbiting Platform Sizing

The aim of this section is to assess a preliminary sizing of the deorbiting platform and its propulsion system since the results of the mission analysis depend on its mass and performance.

From an analysis on available European launchers vehicles, a medium size launcher (Soyuz) turns out to be the most suitable for our purpose. Bearing in mind that most debris lie in the SSO region at approximately 800-900 km, the reference launch mass has been defined assuming the Soyuz launcher performance into a SSO of this altitude (4600 kg) [25].

As already stated, considering the high total impulse of the investigated active space debris removal mission, an electric propulsion system, in particular, the SNECMA Moteurs PPS-5000 Hall effect thruster[26], is considered as main propulsion system to maximize the payload mass fraction. This thruster operates at a nominal power of 5000 W, it is able to provide around 3000 s of specific impulse with an efficiency larger than 50% [26]. This means that (conservatively) a single PPS-5000 provides a thrust of 200 mN. Moreover, as larger solar power generation systems are feasible for the considered orbits, two identical thrusters are chosen to be operated simultaneously. This doubles the required power and available thrust, while still 3000 s are considered for the specific impulse. It is worth mentioning that the same thrusters can also be used for the fine attitude control during the foam ejection process and for counterbalancing the thrust due to the foam ejection.

The preliminary design of the deorbiting platform has been carried out by means of a top-down approach based on statistical data on the mass of each subsystem, see Fig. 2.

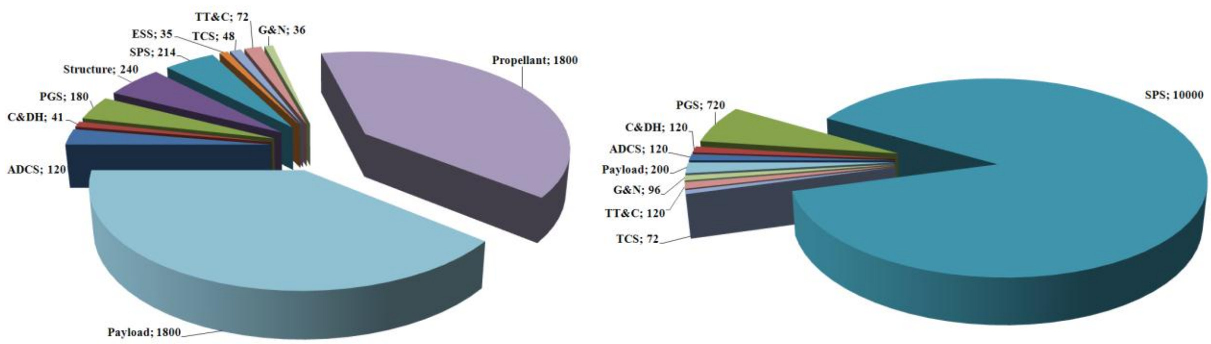

Figure 2. Deorbiting platform preliminary mass (left) and power (right) breakdown.

Assuming 20% of contingency for each subsystem, the total dry mass of the spacecraft is 1000 kg, thus about 3600 kg can be allocated for the propellant mass, the foam with its tank and the foam ejection device. In a rough (and conservative) estimation, this mass value is equally divided (1.8 tons each one) between the payload, composed of the foam itself and by the foam ejection device, and the propellant.

This represents a first guess assumption and we consider this just as reference value. Indeed the foam and the propellant mass fractions depend on the specific mission strategy and, in general, their value depends on the size and the spatial distribution of the targeted debris population.

The platform is equipped with about 40 m^2 of independent, deployable, sun-tracking solar panels capable of providing up to 12000 W. This power level is required to operate the thrusters and the other subsystems during the foam ejection phase, which is the most power-demanding phase of the mission.

This preliminary platform sizing is completed considering a comparison, in terms of subsystems masses and propellant mass fraction, between an electrical and a chemical propulsion option.

Considering, from the preliminary mass breakdown, that thermal, telecommunication, on-board data handling and attitude control subsystem do not depend on the propulsion scheme, the comparison is made by sizing the propellant mass fraction and the power generation system mass for the two cases. As the exploitation of a low thrust propulsion scheme increases the mission ΔV, a meaningful comparison needs to consider the same performance of the two systems in terms of orbital maneuver. A representative value for semi-major axis (200 km) and inclination (3 deg) changes are here assumed for such a comparison. Considering the velocity increment consistent with these changes in the two cases (425 m/s for the electric case and 362 m/s for the chemical case) the Tsiolkovsky equation can be used to estimate the propellant mass consumption considering for the chemical case a realistic specific impulse value of 250 s. The analysis so far described is represented in the two plots of Fig. 3.

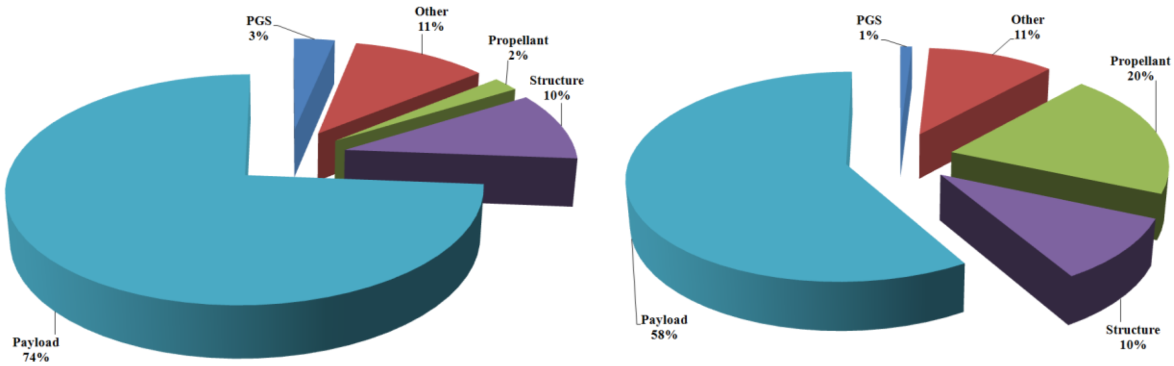

Figure 3. Comparison between electric (left) and chemical (right) propulsion system configurations.

As shown in Fig. 3, the electric configuration allows 600 kg of additional payload mass as the propellant is almost 10% of the one of the chemical configuration (from more than 900 to less than 100 kg) and the power generation system mass increases only by 125 kg (from 25 to 150 kg). It is worth stressing that the above analysis is quite conservative as only a single transfer between two target debris has been considered. Taking into account the whole mission, or even just several transfers, the performance of the electric scenario would be even better.

For the sake of completeness also the preliminary design of the foam ejection device is addressed. This device has to be able to approach the debris and cover its surface in such a way that the resulting foamed object resembles a sort of foam ball, thus the foaming coverage should be as regular and distributed as possible. Since the kind of foam identified is originated by the mixing of two components, and in order to simplify the device (limiting the number of possible failure modes) and reduce its mass, a static mechanism, namely a mixing chamber and/or a mixing nozzle, is assumed as the best option to produce and eject the foam. Thus, a foam ejection nozzle mounted on a robotic arm (see Fig. 4) represents the candidate device proposed for our method.

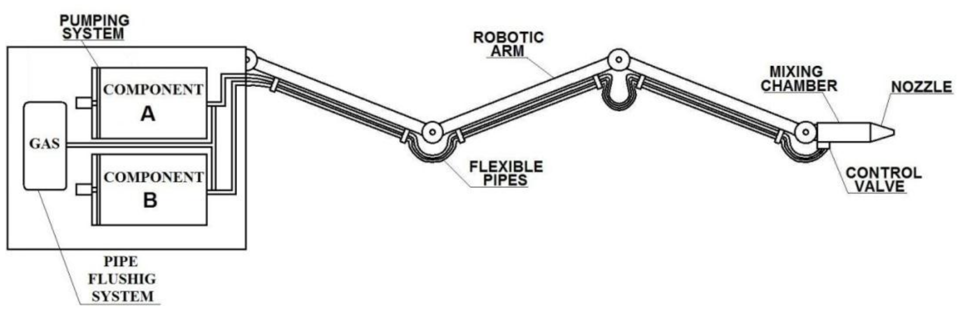

Figure 4. Schematic design of the foam ejection nozzle on the extendible robotic arm.

V. Test Cases

Considering the mission analysis approach described in Sec. III, a list of space debris is required to assess the performance of the method in terms of number of debris targeted per year and deorbited mass. Databases with catalogued space debris have all some access limitations, especially concerning the nature and thus size and shape of tracked objects. In order to obtain a broad description of the real space debris environment, the list here considered is based on a list of the currently tracked objects. The Union of Concerned Scientists (UCS)[27] makes available a list of hundreds of tracked objects (downloaded on November 2010). These objects are mainly active spacecraft, but some filters can be applied in order to have a list of possible candidate debris. Considering the object launch date and its expected lifetime, a filter here applied is to consider an object as debris if it is still in the UCS list after the end of its lifetime (no mission extension assumed). Moreover, debris mass and orbital altitude have been arbitrarily limited to 2000 kg and 1000 km. In this way the UCS list returns 237 objects, which are for the purpose of this assessment assumed as possible candidates for to be deorbited debris objects in LEO.

Instead of providing a full list, Fig. 5 shows mass, altitude, inclination and eccentricity of the considered objects. It is worth stressing that these objects are not, at least today, space debris and the list can change according to future update of the UCS database.

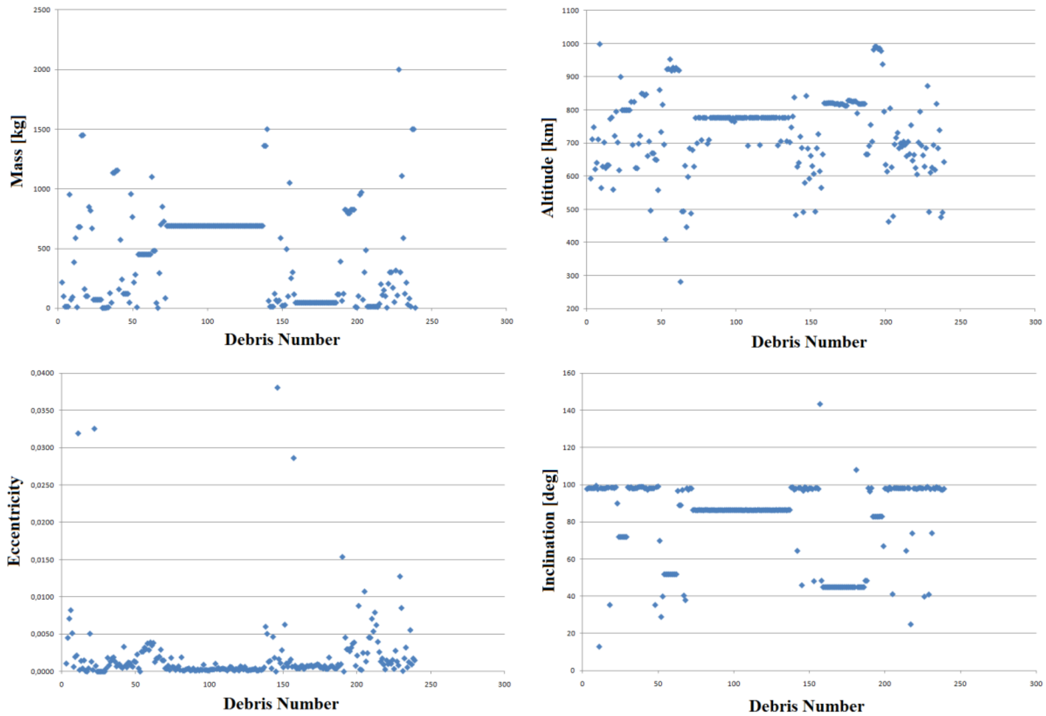

Figure 5. Physical and orbital characteristics of possible future debris according to the UCS list.

In Fig. 5, the long horizontal plateau at approximately 670 kg (at an altitude of 777 km) is due to the *Iridium* satellite series. The smaller one, around 45 kg (at approximately 820 km), corresponds to the 28 satellites of the *ORBCOMM-FM* series and the last one, very small, around 450 kg (at 920 km) is due to the *Globstar FM* satellites (9 in this list). The eccentricity plot shows that these orbits are almost circular, as more than the 80% of debris has an eccentricity below 0.003. Finally, the inclination plot shows that the orbits are mainly crowded into highly inclined orbits, around 80 and 100 deg, i.e. the Sun-synchronous region. It is worth stressing that the presence of few debris out from this range causes a very large mission ΔV in order to change inclination and reach just these few debris. This means that considering dedicated missions, for instance operating at different inclination ranges, would significantly reduce the time required moving from a given debris to the next and accordingly the propellant mass required.

The initial value of the platform inclination has been obtained by means of a coarse grid on this value and considering the most suitable value as the one resulting in the largest number of debris deorbited per year and the smaller number of missions required. Figure 6 shows the results of the low-thrust mission analysis applied to the filtered UCS list.

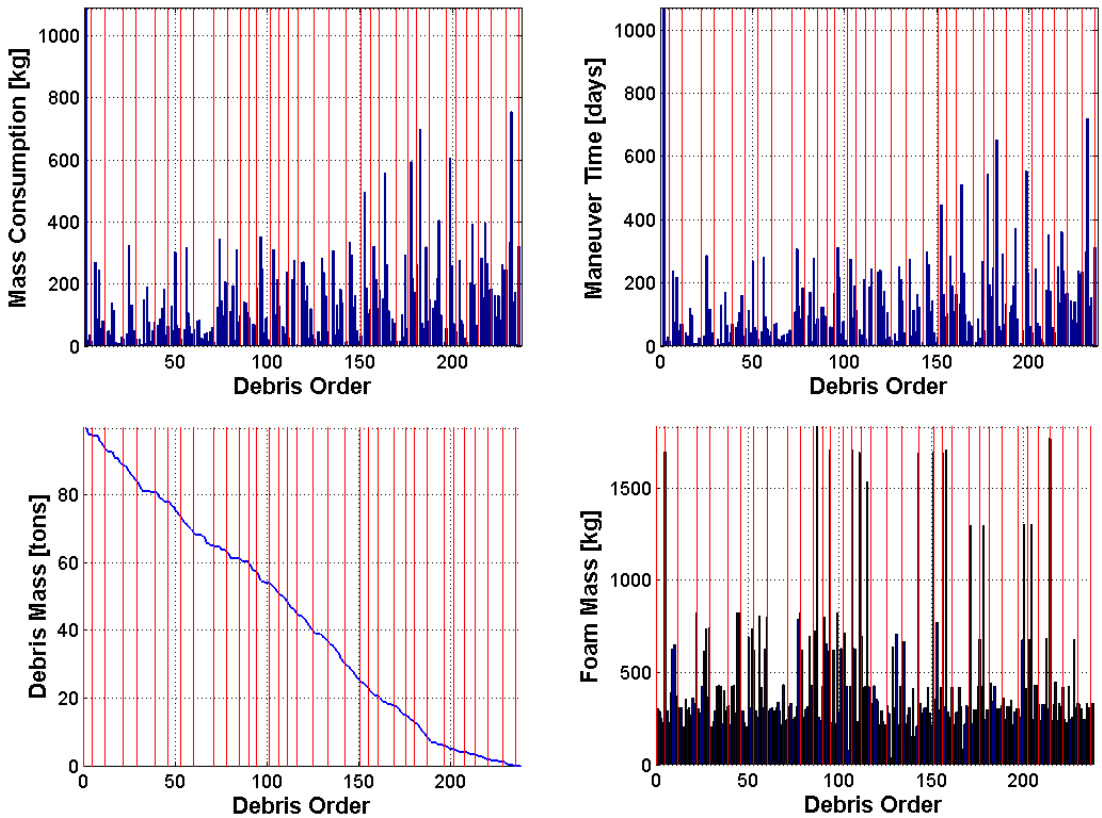

Figure 6. Mission analysis result for the UCS list in the medium density scenario. From the upper left corner: platform mass consumption, manoeuver time, foam mass and total mass of debris to be still deorbited trends with respect to the debris index number.

In Fig. 6 the vertical red lines represent the end of a single mission. A mission is intended to be over if the sum of propellant and foam used has reached the total amount allocated for the two quantities, i.e. 3.6 tons. Since the beginning of each mission there are more debris to target, the propellant mass cost is typically lower than the foam mass cost. For this reason some missions, especially the first ones, present higher values of foam mass consumption and small values of propellant mass consumption. On the contrary, last missions present higher values of the propellant mass consumption and require much more time to be completed due to the small number of debris yet to target. In particular, the UCS list requires 34 missions in order to reach all the 237 debris with a mission duration that ranges from 1 to 4.4 years. Assuming to perform one mission after the other, 3.28 debris can be targeted each year on average with a corresponding value of the debris mass per year of 1.38 tons/year, so each mission is sufficient to target 2.93 tons of debris. As other possible alternative, 34 parallel missions can be carried out at the same time. In this case, the number of foamed debris per year becomes 115 and all missions would be completed within 4.4 years. Given the wide range of semi-major axis and inclination values of this list, each mission on average requires a velocity increment on the order of 12 km/s. Most of the time is required to move from a debris to the next one and changing the considered thruster (or the power level) would reduce this time increasing the mission performance. In general, the electric thruster, the foam density, the initial platform orbit and its mass, the atmospheric model and the debris list considered are the fundamental factors affecting the mission performance[13].

VI. Conclusion

This study summarizes the first analysis of an innovative space debris removal method focusing on the low-thrust mission analysis of the deorbiting platform. The expanding-encompassing-foam concept here proposed aims to augment the drag acceleration acting on a given debris such that an uncontrolled re-entry can take place with the burn up of the object in the atmosphere within a given time frame.

The proposed mission and the method performance depend on the target debris list, the space region they cover and their number. The deorbiting platform in charge of targeting and encompassing with foam each target debris is a

4.6 tons spacecraft to be launched in LEO / SSO by means of a Soyuz-class launcher. The electric propulsion system allows a significant propellant mass saving and a reasonable increment in the transfer times from a debris to the next one. This scenario requires a large total impulse, thus the electric propulsion options can easily overcome chemical solutions. No particular technological issues arose in the power and mass budgets assessment.

In general, up to several tons debris can be deorbited and with the mission analysis carried out it is shown that more than 1 ton of debris per year can be deorbited with such foam-based method. This corresponds to 3 debris deorbited per mission with a maximum mission duration of the order of 4.4 years. In this way it is shown to be possible to reduce debris lifetimes by 80% in average atmospheric density conditions.

The mission analysis outcome can be significantly improved both by exploiting an optimization approach for the design of minimum propellant mass maneuver from a debris to the next one and including the main perturbations typical of the LEO region. Nonetheless, a more detailed mission analysis on a real subset of targets with an exhaustive investigation of the close-approach and foam ejection phases would represent a major refinement of the present work.

It is worth stressing that foam density affects the method performance and the range of target debris mass. For this reason, the development of foam with characteristics suitable for the proposed method is a fundamental step to be performed in the near future. The foam should have a high expansion factor (i.e. the volume ratio before and after expansion), stick to debris surfaces and resist to atmospheric drag. Moreover, the design of a static mixer capable of mixing and ejecting a two-component foam in the space harsh environment represents a required additional technical development.

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