Guidelines for Active Removal of Non-Functional Targets Designed to Assist Rendezvous and Capture

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Abstract— **With the increasing awareness of the space debris issues, the option to remove the satellites from their orbits in case of failure or at the end of the operational life is being considered since the design phase. The removal will be likely exploited in autonomous way by means of space-robots, and to include specific devices onboard spacecraft could greatly ease these complex robotic operations. The selection of suitable devices is obviously related to the proposed servicing techniques. The paper discusses the pros and cons of different possible solutions, offering a view of the so-called SDRS (Situational Awareness, Active Debris Removal and On-Orbit Servicing) technologies.**

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1. INTRODUCTION

Access to space is a key technology for a large number of scientific and industrial activities, from mobile communications to television broadcasting, from remote sensing and astronomy to Earth observation and weather forecast. In the last decades the demand for active satellites in low Earth orbits has produced a population of dead bodies, or debris, mainly made of last stages of launchers, dysfunctional satellites, spacecraft at the end of life, and smaller fragments. This population will keep growing because of ongoing space activities, on-orbit explosions, and accidental collisions.

Already in 1978 Kessler [1] developed a model which considered the major source and sink terms for the growth of the satellite population in earth orbit. By applying techniques formerly developed for studying the evolution of the asteroid belt, Kessler concluded that collisional breakup of satellites will become a new source for additional satellite debris in the near future, and that once collisional breakup begins, the debris flux in certain regions near the Earth may quickly exceed the natural meteoroid flux. Over a longer time period the debris flux will increase exponentially with time, even though a zero input rate may be maintained. These famous conclusions have become known as the "Kessler Syndrome".

In this context, it is important to notice that upcoming large space programs could deeply affect the space population. New economy companies like Google, Facebook and SpaceX will play a primary role on the commercial use of LEO orbits. Actually X-Space (Google/Fidelity Investment) planned to build up 4000 satellites @ 1100Km LEO to be launched by Space X launchers to operate by 2020. More recently the firm OneWeb (Virgin Galactic/Qualcomm) has planned to build up about 1000 satellites @ 1200 Km LEO to be launched by Virgin Galactic Launcher One S/C.

The above scenario shows that before the end of this decade a new generation of satellites should be designed to cope with the Kessler syndrome and its relevant domino effect. Recent studies have demonstrated [2] that commonly adopted mitigation measures, such as limiting post-mission orbital lifetimes of satellites to less than 25 years, will slow down the population growth, but it will be insufficient to stabilize the environment. Even if spacecraft are designed to achieve an EOL (End Of Life) compliance with these Space Debris Mitigation (SDM) requirements, a failure of the spacecraft, or other unforeseen events, may lead to the satellite being non-operational in the protected regions. Therefore such a satellite may then be required to be removed by another spacecraft through active debris removal.

For LEO satellites the removal is usually associated with the concept of de-orbiting, i.e. lowering the debris attitude until it is destroyed by the atmospheric drag, while for higher altitudes (such as GEO satellites) the purpose is to bring the body to a graveyard orbit that is of no operational interest and do not intersect remarkable orbits. In both cases, the task is far from trivial, since the removal strategy is effective if quite large target bodies are taken into account. Moreover, targets can be characterized by a non-negligible tumbling motion and by uncertain structural conditions.

The three main tasks that a chaser satellite must accomplish in order to perform a successful de-orbiting of the target are:

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1) Determination of the relative position and attitude of the target with respect to the chaser

2) De-tumbling of the target

3) Contact (capture) or contactless de-orbiting

This paper shall focus not on the chaser spacecraft, but on the target satellite itself. In fact, it is possible to design technological devices for future spacecraft and for those currently under development, that could contribute to ease the task of the chaser. These devices are sometimes defined as SDRS (Situational Awareness, Active Debris Removal and On-Orbit Servicing) technologies.

2. Navigation Issues and Problems

RF navigation

Different solutions to the problem of weak received signal in radio frequency (RF) based navigation could be arranged. A passive measure taken to improve the power of the return signal is the placing of a corner-cube reflector on the target [3]. An application of the passive corner-cube reflector would define a reference point on the target, which can be distinguished from all other spurious reflections by the power of the signal. Such a solution would have a low impact on the integration of the satellite, but its effectiveness would be much lower than active methods.

Another method used to increase significantly the power density of the return signal consists of placing a transponder on the target; this amplifies the received signal and re-transmits it toward the original source using a different frequency. The transponder de-couples transmitted and received signals, and no switching between transmit and receive mode is needed. The time delay of the signal on its way through the transponder must of course be known and must be constant over time. A transponder on the target would improve the discrimination of the direct signal from spurious reflections and refractions by the target structure, but would require additional active equipment and power on the target side. Of course, the issue of powering a transponder on board a non-functional satellite arises and must be faced. A possible solution is to operate the transponder in such a way that a low-consuming stand-by mode is interrupted only when an incoming wake-up signal is received. In this way a battery pack would only be necessary for the short time of the actual debris removal. On the other hand, this means that a miniaturized communication system should be included in the SDRS device. Technologies for inexpensive (in terms of power and mass budgets) transponders are currently investigated, as proofed by a recent European Space Agency activity [4].

LIDAR

LIDAR (light detection and ranging) is an active sensing device with its own source of laser illumination. Compared with passive optical and active radar/microwave instruments, LIDAR systems provide substantially more accurate and precise measurement without reliance on natural light sources and with a long detecting distance. As a result, LIDAR technology for space applications is actively investigated [5], [6]. However, in absence of a proper reflector on the target, the received signal could be affected by noise and clutter; in particular the relative attitude measurement would be seriously affected. For the measurement of range and direction (Line-of-Sight) only a single retro-reflector is needed, even though to increase the returning signal power, it may be useful for very long distances to increase the number of reflectors on the target. At least three retro-reflectors will be necessary for the measurement of relative attitude [3]. After having identified them, either by a search scan in the total FOV (Field-of-View) or by previous tracking of the complete pattern at longer ranges, the sensor will track them individually. By knowing the coordinates of the reflectors on the target and measuring the distance to each of them, the sensor can establish the angles of the coordinate frame established by them on the target. The accuracy of the relative attitude measurement is given by the distance between the reflectors and the center of the pattern and by the range resolution of the laser range finder sensor.

LIDAR technology can be divided in scanning and non scanning devices The latter basically recover the global scene following each laser pulse, and are especially effective in Simultaneous Location and Mapping (SLAM) applications. The first ones, also known as structured or modulated-light devices, offer high output rates (and possibly more convenient power requirement management) at the cost of the modulator element complexity (see the photonic mixer devices [7] as an example of these sensors).

Visual Navigation

Visual-based navigation can be one of the most appealing choices in the case of uncooperative rendezvous, since they can be considered a low cost, mainly passive and accurate sensor for pose determination at close distances. The main issue is that these qualities can be effectively exploited only if the target is correctly identified by processing the acquired image.

The problem of difficult identification of general shapes, as well as the association of primitive shapes (points, lines, circles) of the target to a known model of the target itself, can be solved by including in the satellite design of easily detectable optical features. They can be passive (reflectors) or active (LEDs).

Concerning the passive reflectors, it must be noted that the target may have a relative attitude angle with respect to the camera axis on the chaser, and the projection of the pattern plane on the plane normal to the camera axis maybe shortened, therefore the measurement of the coordinates of two reflectors is not sufficient. A minimum of three reflectors is indeed requested to define a plane on the target. However, in many applications a configuration with four reflectors in a plane is chosen to eliminate the ambiguity of solution [3]. A problem that affects the simple reflector approach is that the chaser computes the relative pose according to the received (reflected) light. The operational range of a camera sensor will be limited by the signal-to-noise ratio of the received signal. In order to increase the power density at the camera optics, the area of the reflectors has to be increased, leading to the design of multi-reflector spots in case of long range targets, : in fact, they require less mass and volume than a single reflector with larger diameter, and allows for a greater flexibility in accommodation.

Another visual feature commonly used in space application can be found as an example in the Automated Transfer Vehicle operations. In this application it is possible to determine the relative 3D position and attitude of the two platforms thanks to the use of a specific visual target, already known. The target consists of two main parts on different planes, a larger circle and a smaller circle with an impressed cross mark. It has been shown [8] that such a target is sufficient to determine the two circles centers and radii, and also the orientation of the two arms of the cross sketched on the smaller circle (see Figure 1). The Hough transform can be the principle for relatively simple algorithms, able to quickly recognize basic features as lines and circles with limited requirements for the on-board computer. Of course in this case the optical feature, even though passive, could have a larger impact on the satellite (or launcher) integration and operations, due to its larger dimensions and to the constraint for its accommodation in specific areas of the external surface.

Concerning the active visual features, it is possible to refer to the recent PRISMA mission [9]. In that case, two different operational modes of the camera have been used: a Far Range and a Close Range camera. In the Far Range the camera is used as a star-tracker, and the target is seen as a bright spot (no attitude information is required). If pose determination is requested, as of course is the case when reaching close distances (typically 20-30m), more detailed information of the target must be acquired. In the case of the PRISMA mission, the visual navigation does not operate by relying on natural illumination. In fact, five LEDs on each face of the target, arranged in specific patterns, are used to generate feature points in the image. To ensure a sufficient contrast ratio to the natural illumination background, the VBS cameras and LEDs are operated in synchronous pulse mode.

With respect to other active methods, the power consumption of a LEDs' system is very low. Of course, it cannot be active relying on its own batteries from the launch to the final removal. Yet, it is possible to design a protocol that checks for the satellite status at a given rate. When the query returns a failure message of the satellite, then the LEDs can be turned on, waiting for the chaser to perform the removal.

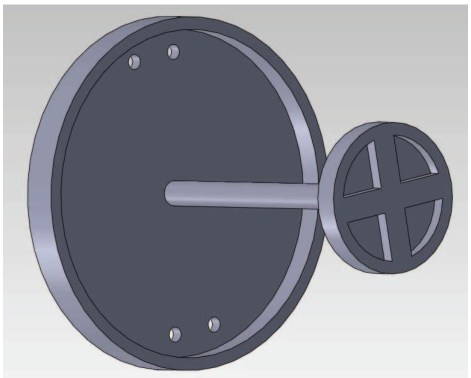

Figure 1 An example of a visual feature, allowing for relative pose autonomous determination. Use of a similar kind of marker has been analyzed in [8]

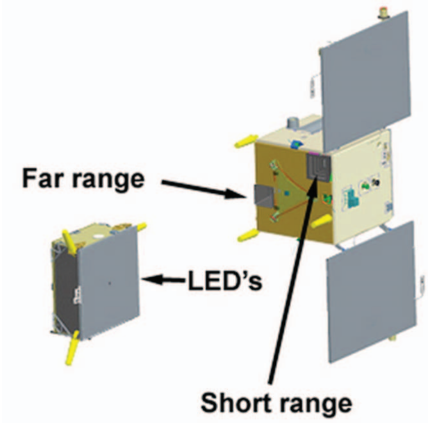

Figure 2 Far range camera and short range camera on the chaser of the PRISMA mission, with visual navigation assisted by the presence of pulsed LEDs on the target (adapted from [10])

GNSS

A solution for navigating in LEO –and still usable in higher orbits [11] – is represented by Global Navigation Satellite System (GNSS) technology. By providing the kinematic state of the target and making such a knowledge available to the chaser can be of course really helpful in the approach and rendezvous. Furthermore, as also the chaser, as it is likely the case, is equipped with a GNSS receiver, it is possible to exploit a differential navigation with far better accuracy. Even if limited to data concerning only position and velocity, as attitude determination would require a far more complex configuration, the information is accurate enough up to the docking phase, where the need for sub-centimeter accuracy would arise. Different technical solutions can be envisaged in order to limit the requirements for the equipment to be accommodated onboard the target:

1. A GNSS translator, i.e. a piece of hardware including a receiving antenna, a frequency translator and a re-irradiator, lacking any computation capability [12]. Such a solution has been used at the beginning of the GPS era in military applications (missiles) and is currently a bit out-of-date due to advances in miniaturization and in GNSS mass production which made the navigation processor (i.e. the computation part of the receiver) far less bulky and expensive.

2. A GNSS transponder, i.e. a complete GNSS receiver that broadcasts its computed navigation solution. Nowadays this kind of equipment is extremely widespread in terrestrial applications, and attained levels of miniaturization that make it certainly suitable for the specific, proposed use [13]. However, the preliminary analysis of this class of products highlighted two issues: the first and obvious one is the need for a check of the capabilities for in-space use, including the possibility to inexpensively correct the firmware without altitude/speed limits, while the second is related to the consumption, in the order of tens of milliwatts, that make this products unsuitable for a continuous operations. As a result, GPS transponders should be used only when the capture attempt approaches, meaning that a way to activate them needs to be considered.

3. A GNSS "bullet", i.e. a GNSS transponder which is fired against the target from the approaching chaser: such a solution, already implemented to track terrestrial vehicles [14], could obviously solve the previous issue. Adaptation to space would require above all the certainty that neither additional debris would be produced at the impact, nor the attitude motion of the target would be affected in a way that makes capture unfeasible.

Estimation

Notice that, whatever sensor, or suite of sensors, should be selected for the relative position and attitude of the target, a powerful estimation function will be needed in the navigation system. In fact, extremely variable accuracies in the measurements (reflective surface variations for RF and LIDAR techniques, changes in light conditions for imaging devices, multipath from GNSS) will need careful processing of the sensors' output. Such a data processing needs an educated inclusion of the foreseen relative behavior between chaser and target, based on knowledge of orbital and attitude dynamics and on chaser's maneuvering. While different approaches are available in the long range, approach phase, the extended Kalman filter (EKF) is likely to be the selected choice for the close-range, final phase of the rendezvous [15]. Accurate kinematic state is for sure needed for an successful guidance of the docking phase, through design of effective maneuvering while limiting collision risks. In many cases the current state estimate has also to be fed-back to imaging systems (i.e. to the sensors) to speed up the scenario processing reducing the relevant required computation effort.

3. ATTITUDE STABILIZATION

An important distinction for attitude control concepts is between passive and active attitude control. In the case of attitude stabilization of a non-functional satellite, the residual power and computing capability are limited to the SDRS device that is on board. As a consequence, the passive attitude stabilization solution is by far more attractive; the hardware required is less complicated and relatively inexpensive.

Natural physical properties of the satellite and its environment are used to control the attitude [16]. In particular, a well-known approach consists in exploiting the Gravity Gradient (GG) torques, which act on the spacecraft for the fact that it is in a gravitational field, and its moment of inertia are not equal (i.e. it does not have a spherical symmetry). Depending on the relations among the three moments of inertia, the satellite belongs to stable or unstable region for attitude dynamics, as shown in Figure 3.

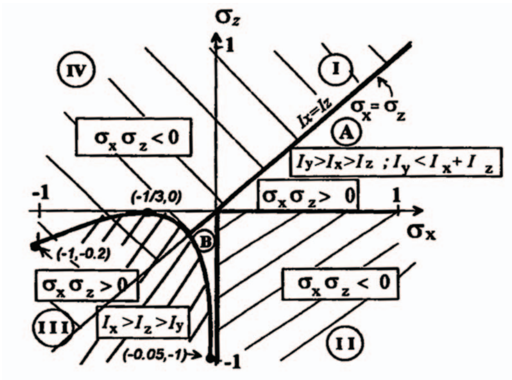

Figure 3 Stability regions for GG-stabilized satellites (from [16])

In order to move the system in one of the stable regions, the moments of inertia of the satellite can be modified thanks to the deployment of a boom. Booms suitable for stabilizing a platform beyond its operational lifetime (i.e. not part of the standard attitude control system) have been developed in the past years, for example by DLR [17]. Specifically, Ultra-light carbon-fiber reinforced plastic struts (CFRP booms) have been manufactured, deployment techniques have been developed and deployment tests at zero gravity have been also performed: these booms are foreseen to be inflated and rigidized in space. As reported in [17], 14-meter long booms were built during the demonstration phase, whereas the design was worked out according to future mission applications for a length of 28 meters. The specific mass of the booms amounts to 100 grams per meter only.

Following the lightweight philosophy of the boom design, the internal energy of the booms, which is stored during the packaging procedure, can also be used for the deployment. The self-deployment test depicted in Figure 4 - completely driven by elastic deformation energy - takes only 10 seconds.

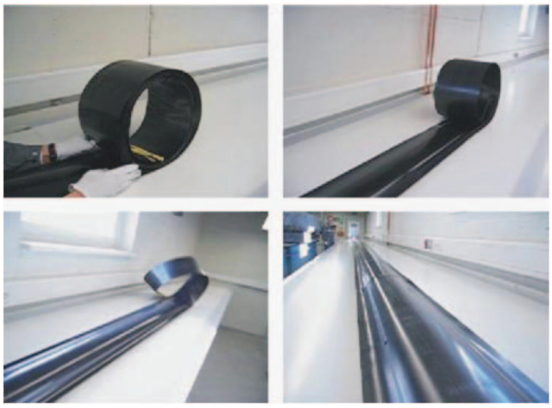

Figure 4 The internal deformation energy of a stowed boom package enables the self-deployment of the 14m long boom (from [17])

The gravity gradient stabilizes the spacecraft in the sense that there remains an amplitude-bounded harmonic angular motion about an average bias value. In order to obtain an effective de-tumbling, the harmonic oscillatory angular motion must be damped and reduced to a minimum. The existence of external disturbances initiates the oscillatory motion and, without effective damping devices, the time to appreciably decrease the oscillatory motion might be very long .

Several kinds of passive dampers have been investigated, usually for the purpose of attitude control of low cost small satellites. A short list is reported in the following [16]:

- Point-mass damper: the principle of this very simple damper is to use a mass-spring/dashpot system accommodated inside the satellite. The energy dissipated in the device helps to damp the oscillatory motion of the satellite.

- Dampers combined with the boom: the extended boom that increases the moments of inertia in order to achieve desired GG stabilization can be mounted on a large external spring. On the tip of the spring, which has a helical form, a fluid damper can be mounted to increase the internal energy dissipation inside the satellite so that a damping effect is produced.

- Magnetic hysteresis rod damper: if a rod of magnetically permeable material is located inside the satellite, then the angular motion of the satellite with respect to the earth's magnetic field will induce magnetic hysteresis losses. These devices have been used on many satellites but, unfortunately, their effectiveness is limited to rather low orbit due to the decreasing strength of the earth's magnetic field.

- Damping by boom articulation: many GG designs are based on a suitable mechanism at the hinges that join the augmenting booms to the satellite's body. This mechanism usually has two degrees of freedom and so provides damping about two axes; this enables damping of both the pitch and the roll angles.

- Wheel damper: a wheel, immersed in a container holding a viscous fluid, can be effective in damping the angular motion of a satellite.

Focusing on the SDRS technologies, the preferred solutions will be the ones that have the least impact on the satellite both in terms of performance of the nominal operations, and in terms of integration. In this sense, a magnetic hysteresis rod damper have a large influence both on the satellite attitude and on the on board electronics all along the satellite lifetime. A point-mass damper does not affect the onboard electronics, but it affects the attitude maneuvers even when the satellite is functional.

Fluid dampers could instead represent a proper solution, provided that the fluid motion is prevented before the satellite becomes non-functional. The fluid can be initially maintained fixed inside a ring. When the fluid motion is ''unlocked'' an exchange of the angular momentum between the rotating spacecraft and the viscous fluid will provide a dissipating effect which in turn will reduce the angular velocity of the spacecraft. Of course the damping effects are governed by the fluid ring parameters, in particular by their global geometrical characteristics (i.e. the inertia moments) and by fluid properties (i.e. the dynamic viscosity). On account of that, it will be of interest to find the "best" combination of these parameters which are able to reduce the libration motion inside assigned threshold in minimum time [18].

Viscous damping of slosh waves in low-g is somewhat greater than it is in high-g because of the greater wetted area and presumably because of the smaller fraction of the liquid mass that participates in the sloshing [19]. This can also be regarded as the amount of mechanical energy lost by viscous friction (transferred into heat) over one cycle of sloshing in relation to the total mechanical energy [20].

Without dampers, the attitude of a rigid satellite under gravity-gradient torque in a circular orbit remains stable in the Lagrange and Debra-Delp regions. However, in the presence of energy dissipation, the Debra-Delp region disappears and the attitude of the satellite remains infinitesimally as well as asymptotically stable in the Lagrange region only [21].

As underlined in [22], the settling times involved can be quite long (on the order of hundreds of orbital periods). This happens because the dissipation torque is not proportional to the angular rate of the satellite, but to the relative rate between the spacecraft and the fluid, which is significantly less than the former.

The simulations performed in [23] show that, if an optimal pyramidal configuration (see Figure 5) is designed for the fluid ring dampers, a faster damping can be achieved. In the case of a preceding satellite in a circular orbit with a radius of 7000 km, the satellite attitude is asymptotically stabilized in the roll, pitch, and yaw directions; however, even after 50 orbits, the satellite attitude angles are reduced only by approximately 50% of the initial value in the roll direction and about 70% in the pitch and the yaw directions. Even though it could be concluded that the damping effect is not fast enough for a satellite that must be operative, in the case of the stabilization of a non-functional satellite, this is certainly not an issue.

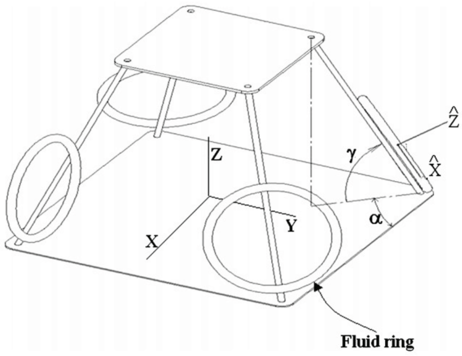

Figure 5 Pyramid configuration of fluid rings (from [23])

As preliminary design of the proposed SDRS device for attitude stabilization, it can be therefore proposed a deployable boom, together with a small three axis ring fluid, as the one tested for nano satellites in [22], and reported in Figure 6. The SDRS device should be able to determine when the satellite has become (or is going to become) non-functional. At that point the boom will be deployed, and the sets that prevent the fluid motion during the satellite operative life will be removed. The satellite will reorient according to the gravity gradient torques stable equilibria, and the external disturbance and residual oscillations will be slowly damped out.

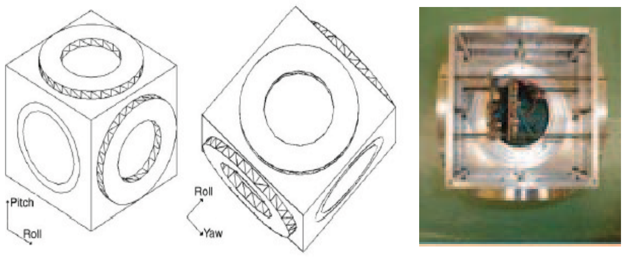

Figure 6 A three axis fluid ring (from [22]).

4. Capture and De-Orbiting

Robotic Manipulators

The capture of a space object can be quite difficult not only for the need to coordinate the position and the attitude of the chaser and the target, but also due to the grasping action itself. In fact the structural strength of the selected contact region of the target (likely orbiting since a long time, and possibly damaged) could be not sufficient. Of course the problem greatly depends on the peculiarities of the case to be investigated.

As an example it is possible to analyze a spent Ariane 4 stage (H10) for a possible capture, followed by a possible removal by means of a propulsive kit to be applied to the stage [24]. A first issue deals with the identification of the meaningful zones for grasping by means of the end effector(s). In particular, the areas that could be of greater interest are:

1) Nozzle and neighboring parts;

2) Top of the stage where the fairing and payload adapters are located.

The center of mass of the spent stage is located close to the nozzle and therefore, for minimizing the torques exerted on the H10, it would be convenient to grasp it near the nozzle. The option (zone 1) presents however some issues due to the geometric and structural characteristics of the area, especially because of the limited thickness of the structure of a nozzle (3-4 mm of 7075 aluminum alloy) that could hardly withstand the grasping operations. A grabbing in the zone 2 seems indeed a convenient solution (see Figure 7).

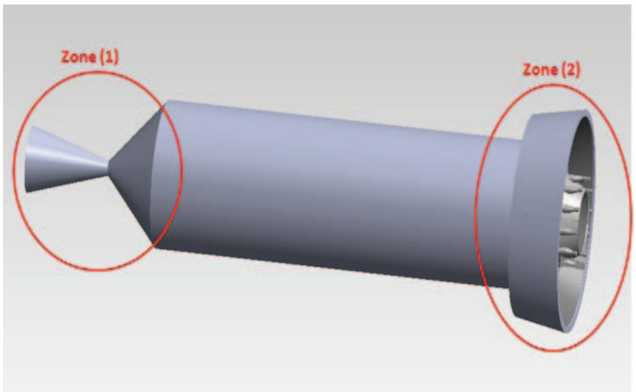

Figure 7 The two possible grasping sites of H10: near the nozzle (zone 1) and near the payload adapter (zone 2).

Depending on the mission scenario, more contact points could be required. As an example, a first contact point could be devoted to the grasping, while another region should be selected to apply the propulsive kit and withstand the relevant forces. The choice of the zone (2) remains valid also in this scenario, in which the stage must be grabbed by the end effector and docked by the propulsive kit at the same time. In this case it is proposed to select the fairing adapter interface for the end effector grasping, and the payload adapter interface for the propulsive kit docking. In such a way there is no mechanical interference between the two mechanisms. These two areas are highlighted in Figure 8. For the launcher stage case, as a conclusion, there seems to be enough reliable zones for the grasping, without the inclusion of purposely designed mechanisms.

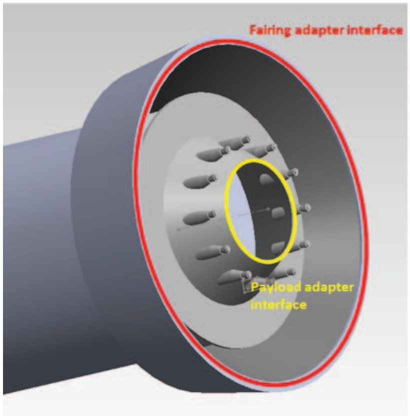

Figure 8 H10 grasping zones: Fairing adapter interface for the Mothership's manipulator end effector, and payload adapter interface for the propulsive kit's docking mechanism.

The case of satellites can be more complicated. In fact, by analyzing the qualitative design of a typical GEO-class platform, shown in Figure 9, it is possible to see that most of the lateral surfaces of the bus are occupied by solar arrays, antennas, and devices for telecommunication and telemetry. These structures are not designed to be grasped, and could not resist to the mechanical load applied when the chaser ignites its motor for performing the de-orbiting.

The proposed solution is to include additional structural elements which can act as handles for their grasping in the unlucky event they become non-functional. Of course they should be connected to the most resistant parts of the bus, and should not interfere with the other external devices.

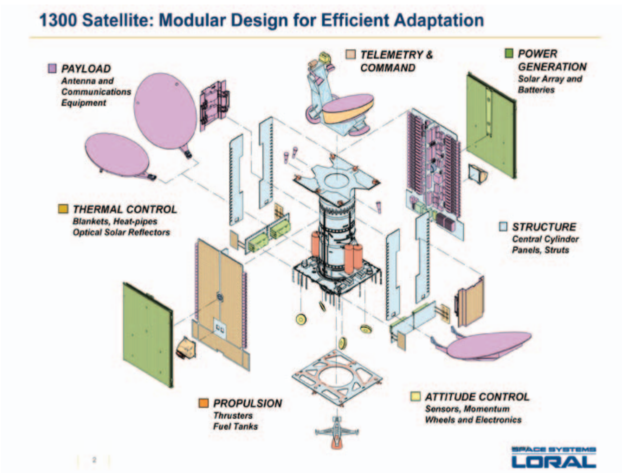

Figure 9 Example of the structural design of a satellite (http://sslmda.com/images/aboutssl/1300modular.jpg)

In fact, a typical mission [25] requires a large number of electrical components, which are substantial producers of heat and need to be mounted on external walls to promote efficient heat radiation to space. In addition, such units need to be grouped into systems creating the need for large areas. A quite usual solution is to employ a box shape using large aluminum alloy honeycomb panels. Load from the panels is carried to the central tube through four shear panels (see cyan colored part in Figure 10) and adjacent box closure panels.

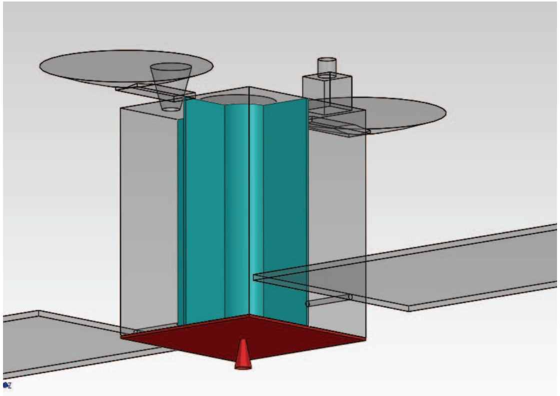

Figure 10 Qualitative design of a GEO test case. The bottom panel should be the most appropriate for including a SDRS for assisting the grasping.

The most favorable side of the bus for the including the handles seems to be the bottom surface (the red one in Figure 10), which typically hosts only the apogee motor. The apogee engine is fitted into the interior of the central tube, at the scope of transmitting the large axial loads to the most structurally resistant part of the satellite.

For the same reason, the central cylinder is also the structure destined to withstand the loads at launch. In fact, as shown in Figure 11, the launcher payload adapter is usually connected to this panel; therefore also when designing a handle for this panel, it must be considered the presence of the interface with the launcher adapter.

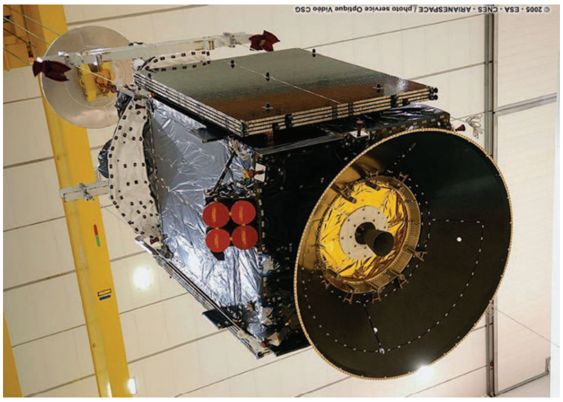

Figure 11 An example of a satellite during its integration on the launcher (source: http://www.arianespace.com).

The competition between launch vehicle suppliers encourages the use of standard interfaces giving the customer the option of using alternative launch vehicles with the same interface design [25]. As an example, Figure

12 shows a sectional view of a manacle clamp used to attach the spacecraft. A number of accurately machined clamp blocks are placed to form a segmented ring over correspondingly accurately machined wedge shaped lips on both the launch vehicle and the spacecraft side of the interface. The clamp band is then tightened over these blocks by two pyrotechnic bolts.

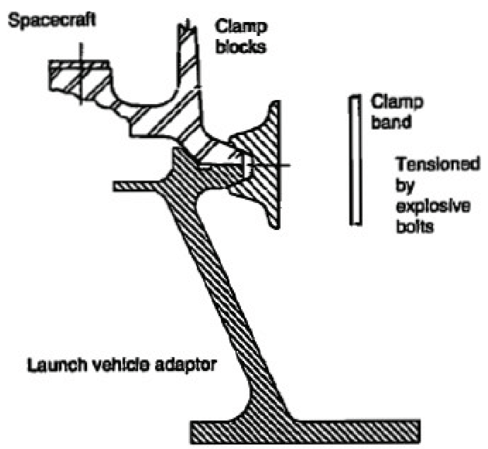

Figure 12 Spacecraft attachment clamp band (from [25])

The design of the handle as a SDRS device must take the presence of this interface into account s a convenient option. Different solutions can be developed.

In one of the possible design options, the handle can be considered as part of the launcher interface itself. This structural element is obviously course very robust, but it could not have the shape for a reliable grasp by means of a robotic arm. Therefore its shape must be modified, by increasing its dimensions, so that it is possible to include a circular handle, not different from the one designed for the DLR DEOS mission. This configuration has the advantage that the clamp would be performed in a very safe area from the structural point of view, since it is directly connected to the central cylinder (see Figure 13 – left frame). M oreover, if the capture should end up with the application of a propulsive kit, the thrust applied for the de-orbiting maneuver would have an action line very close to a principal axis of inertia of the satellite: this load would therefore follow the same load paths happening during the launch. The nozzle is quite far from the modified interface and should not suffer from its presence.

There are also some drawbacks too. In fact, the overall dimensions of the spacecraft integrated on the launcher are increased (see Figure 13- right frame), and this could be a problem for very large satellites, that could exceed the maximum available volume of the fairing.

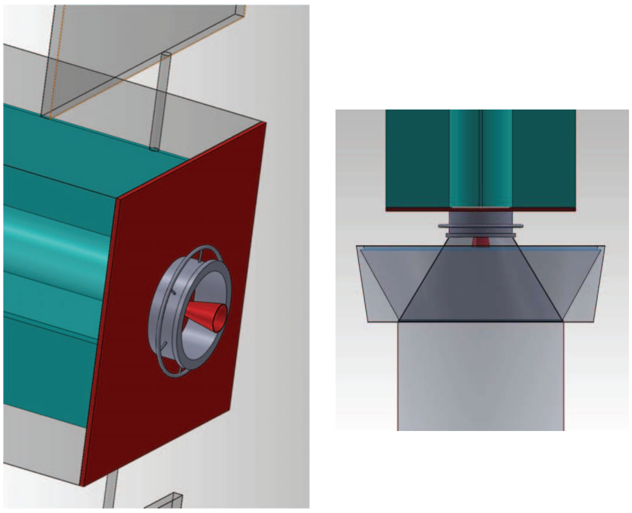

Figure 13 A possible solution for including a robust handling, by modifying the launcher adapter.

Another drawback is that the modification of such an important part from a structural point of view, could lead to a total re-design of the satellite geometry. Considering the structural design process in Figure 14, the requirement for the design of this modified interface generates constrains that should be included at the same level as the launcher constraint block. Therefore, such a solution for satellites that have been already under development could present some problem.

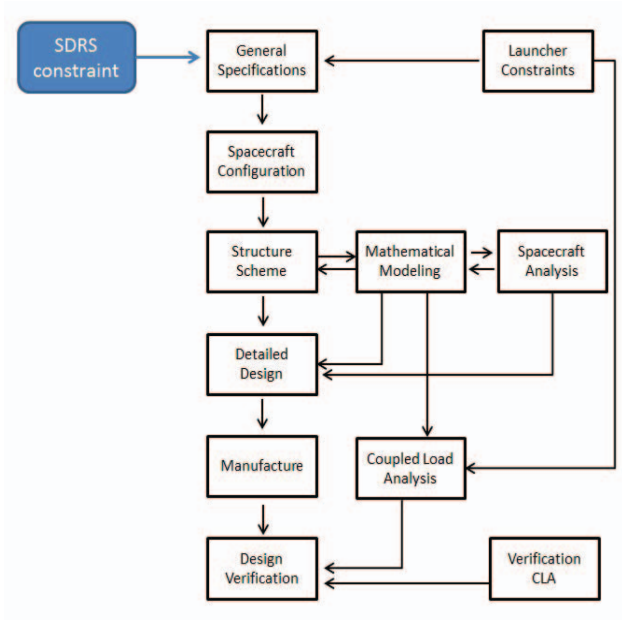

Figure 14 Structural design process (modified from the original in [25] with the inclusion of the SDRS device)

A different and better solution where the launcher adapter is not modified, but an additional structure is included will be described in the following.

In this case the interface to the structure is connected to the central cylinder, just like the nozzle. Its dimensions are such so large so that it does not interfere with the nozzle operations, indeed it should be small enough to fit inside the launcher adapter. A visual representation of the proposed solution is reported in Figure 15 – left frame. In this configuration the nozzle is inside a sort of cage, which offers a number of possible robust grasping points for the manipulator of the chaser. As in Figure 15 – right frame, in this case, the integration on the launcher is not altered, and therefore the design of the satellite should not be modified as a result of the inclusion of the handle.

In fact, even though the details of the mechanical connections of the handle with the spacecraft structures would require of course a careful analysis, this device represent a modification of a detail of the structural system, and not one of its main parts. The impact on the satellite is therefore lighter than the one proposed as first solution, as it is possible to understand supposing that the requirements for the handles apply to a later block of the design process (see Figure 16).

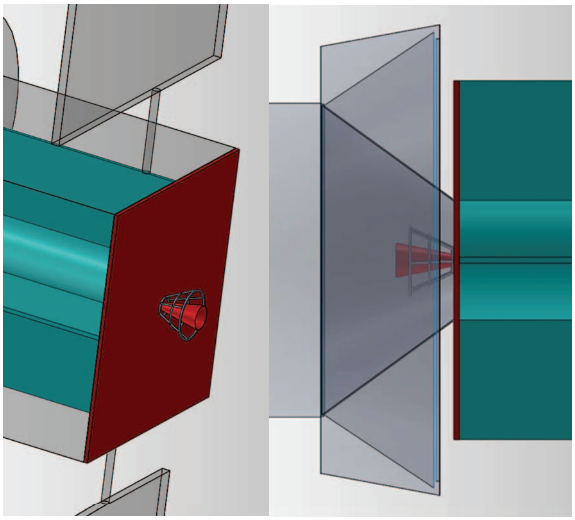

Figure 15 A second possible solution for including a robust handling, by adding a structure that is external to the nozzle and internal to the launcher adapter.

Overall, capturing and grasping operations have to be considered as extremely complex. The command of one (or more) robotic arm(s) needs to take into account the constraints given by the allowed torques and agility and to strictly consider the collision risk [26]. To gain the requested accuracy it is likely that a set of cameras are required to capture the scene from different locations and help in navigating the arm(s) to the selected rasping point(s). Control of the robotic arms, to be autonomously carried out in real time onboard – and therefore with limited computation resources - would be a challenging task, to be carefully studied [27]. This is an important reason to consider also removal techniques which do not involve a chaser. These techniques can be exploited by already equipped platforms, therefore perfectly matching the SDRS-easing concept. Two examples (tethered systems and deployable sails) will be depicted in the following.

Tethered systems

Space electrodynamic tethers offer the opportunity for in-space “propellantless” propulsion and power generation around planets (like the Earth) with a magnetic field and an ionosphere. In general, moving a conductor across a magnetic field generates an electromotive force (EMF) to drive current through the conductor if a means to “close the circuit” is available. The main problem is that tethers are particularly vulnerable to small artificial and natural debris impacts, because – at the very high relative velocities characterizing the collisions – even a particle smaller than one half of the tether diameter may cut a single strand wire.

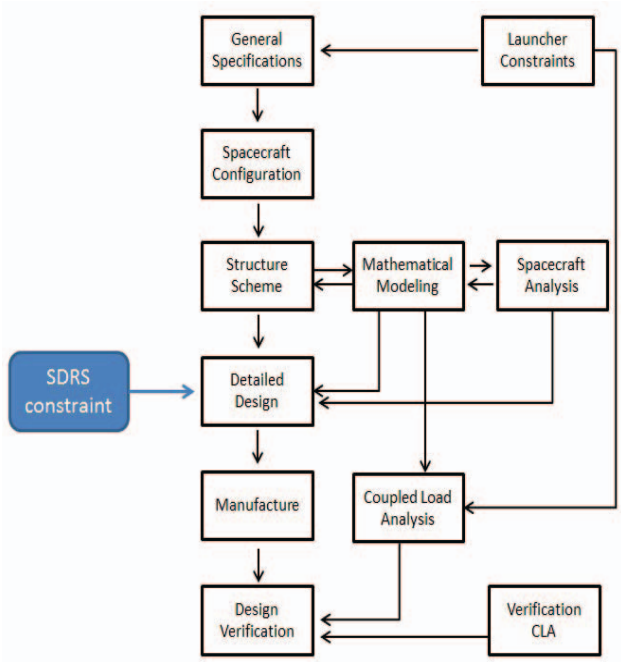

Figure 16 Structural design process (modified from the original in [13] with the inclusion of the SDRS device)

A single hit by a very small particle may therefore produce a critical failure of the tether, while reducing its lifetime to times much shorter than the mission duration. Of course, although single line tethers lifetimes can be improved by increasing the tether diameter, this incurs a prohibitive mass penalty as well as additional operational problems. Therefore, strands with ultra-high strength characteristics, together with more creative tether designs, like multi-strand structures, should be realized to reduce the tether vulnerability to space debris impacts [28].

The critical size of a particle able to sever the tether is affected by the tether material as well as by the tether design, and it can only be determined by the hypervelocity impact test results. However, many new resistant materials and combinations of these have not been thoroughly characterized yet for the effects of hypervelocity impacts by either meteoroids or orbital debris particles. On the other hand, only a few laboratory experiments have been carried out using aluminum tethers of normal design, proving that an aluminum, single strand, tether may be cut by a particle 1/3 of its diameter, while one of woven aluminum could be severed by particles 1/2 of its diameter.

An innovative configuration to reduce the system vulnerability to space debris impacts was proposed. The new design envisaged a two bare metallic strands, 0.7 mm in diameter each, forming N loops tied together in N+1 equidistant knots along the tether (Figure 17) . Simulations and experiments show that this innovative configuration increases the survival probability of an electrodynamic tether (EDT) system during typical de-orbiting missions.

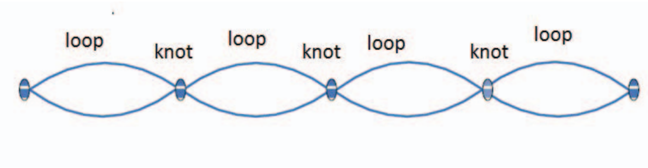

Figure 17 An example of tether Double Strand Solution (from [28])

Another approach, implemented by the JAXA, is the tape EDT. The tape tether is designed to be more survivable in the orbital debris environment than a normal string tether since its projected area in the direction of the thickness decreases and the wide width direction assures resistance against the severance. A spaceflight validation of bare electrodynamic tape tether technology has been conducted on 31 August 2010, when 132.6m of tape tether have been successfully deployed over 120 seconds in a ballistic flight [29].

Once the non-functional satellite attitude has been stabilized, the electrodynamic tether can be deployed. Two main strategies can be investigated. In the first one, the tether is a SDRS device on board the debris. According to this solution, a large number of technologies for removal should be designed on the satellite, both for attitude stabilization (booms and fluid ring dampers) and for deployment of the tether.

According to a different scenario, the EDT can be attached to the satellite by a chaser spacecraft. In this case, only a structurally robust region on the satellite's external panels should be included in the design, so that the chaser can safely attach the EDT device, without the risk of damaging the satellite structures, thus creating new debris.

Sails

In the future, satellites might carry a packaged ultra-lightweight 'gossamer sail' to open as they head towards retirement. The increased aerodynamic drag would pull the craft out of orbit to burn up in the high atmosphere, reducing the risk of catastrophic collisions and creating a sustainable space environment for future generations [30].

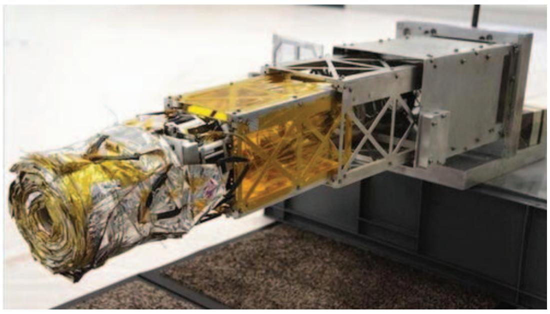

Figure 18 Partially deployed gossamer sail during testing (from [30])

Many agencies are developing deployment systems for large solar sails. The first in-orbit demonstration has been performed by the Japanese Space Agency (JAXA) that launched the solar power sail demonstrator "Interplanetary Kite-craft Accelerated by Radiation Of the Sun: IKAROS", on May 21th, 2010. IKAROS demonstrates a solar power sail technology, the first in the world, and aim to expand the solar power sail that diameter is 20 meter class [31]. This device was not aimed to de-orbit purposes, in fact the scope was to supply propulsion for deep space mission. Moreover, the dimension was so big that it could be hardly considered as a SDRS device for de-orbiting LEO satellites, see Fig.18.

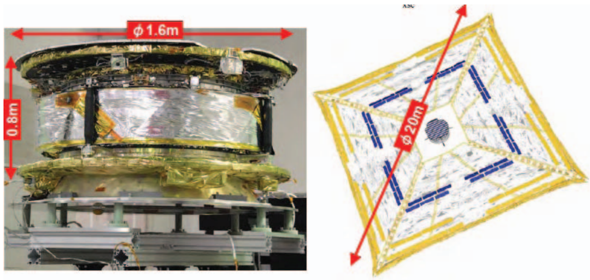

Figure 19 Stowed deployment mechanism (left) and deployed sail (right) for the IKAROS project (from [31])

In the early 2000s, NASA made substantial progress in the development of solar sail propulsion systems for use in robotic science and exploration of the solar system [32]. Two different 20-m solar sail systems were produced. The sail systems consist of a central structure with four deployable booms that support each sail. These sail designs are robust enough for deployment in a one-atmosphere, one-gravity environment and are scalable to much larger solar sails – perhaps as large as 150 m on a side. As a technology demonstrator, the mission "NanoSailD" has been performed. The sailcraft consists of a sail subsystem stowed in a 3-U CubeSat. The demonstration flight happened in 2011 and fulfilled the two primary technical objectives: (1) to successfully stow and deploy the sail and (2) to demonstrate deorbit functionality. The solar sail subsystem occupies the lower 2/3 volume of the spacecraft. Sail closeout panels provide protection for the sail and booms during the launch phase of the mission. These panels have spring-loaded hinges that will be released on orbit, under the command of the spacecraft bus. Figure 19 shows the fully deployed NanoSail-D in a ground test.

The limited dimensions of this kind of devices make them a candidate choice for de-orbiting non-functional spacecraft in LEO. Of course an accurate prediction of the re-entry area in this case is quite difficult, but this could not be an issue for small LEO satellite, that would completely burn into the atmosphere. The increase in the de-orbiting efficiency would be maximized if a stable attitude could be maintained, keeping the maximum cross area orthogonal to the velocity direction. Furthermore, the approach would work even if a tumbling motion would be present.

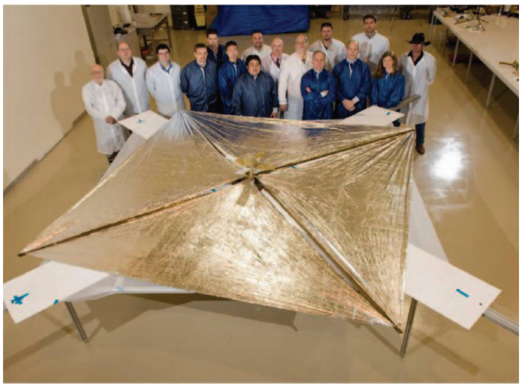

Figure 20 The NanoSail-D development team is shown with the flight system following a successful ground, non-vacuum deployment test (from [32]).

The deployment of the sail requires that no obstacles are in the close distance from the stowed device. Therefore, a solution is to perform a double deployment: first a boom is deployed, and then the stowed sail, on top of the boom, is unfurled. Of course the possibility for the SDRS device to detect the time when the satellite has become non-functional is crucial. A wrong decision in this sense would lead to catastrophic consequences on the operational activities of the satellite.

A possible configuration for a midi/mini-small LEO satellite that have become non-functional is depicted in Figure 20. Depending on the test case selected, the appropriate side of the bus should be selected, taking into account both the internal volume and the antennas, payloads, and other devices that could occupy the external surface of the bus. In this proposal two qualitative scenarios are shown. With respect to the attitude stabilization problem analyzed in the previous pages, in this scenario the boom does not have the purpose to modify the moments of inertia of the system, in order to exploit the gravity gradient torque. Instead, the scope is only to impart a safety distance between the sail and the satellite. This configuration, however, could also have a stabilizing effect on the target, that should be investigated.

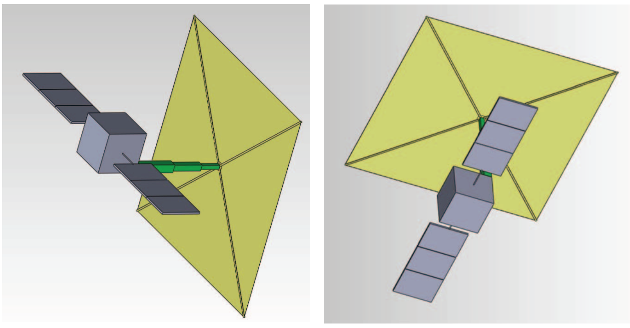

Figure 21 Two possible configurations for the drag augmentation sail SDRS device for the case of a mini LEO satellite.

5. FINAL REMARKS

The paper presented and discussed possible solution aimed to facilitate the capture and removal of spacecraft once their operational lifetime ends. A specific attention has been devoted to the case of the spent final stages of the launchers, that are obviously some of most threatening debris.

Rendezvous and capture by means of a chaser is currently considered the baseline solution, as the only one capable to manage large objects already in orbit. While consistent knowledge on proximity operations has been and is continuously built, the case for uncooperative targets is challenging. Difficulties in the determination of the relative kinematic state, in the approach (taking into account collision risk) and in the grasping of poorly prepared surfaces need to be carefully considered. These issues promote the effort to actively deal with the end-of-life removal process since the design phase for the new satellites. A number of technologies capable to help during the recovery have been therefore reminded. Their requirements in terms of power and accommodation onboard (as parts not instrumental to the platform's main goal) seem a significant constraint.

Alternative, emerging removal solutions that do not require the intervention of a chaser have been also depicted. While they can save from complex rendezvous and docking operations, they are likely to present even more serious constraints to the platform. Tests already in progress in several labs and first in-flight experiences will indicate if these solutions can become a widespread addition to spacecraft in order to fix the end-of-life issue.

It is especially important to remark that so-called SDRS (Situational Awareness, Active Debris Removal and On-Orbit Servicing) technologies are definitely playing an increasing role in space engineering, so that a survey – even preliminary - of possible solutions can be valuable for both mission analysis and platform design. To this aim, different suitable techniques have been introduced, together with their background. Study history and suitable relevant references have been reported.

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