ACTIVE REMOVAL OF LARGE DEBRIS: ELECTRIC PROPULSION CAPABILITIES

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ABSTRACT

The risk for current operational spacecraft or future market induced by large space debris, dead satellites or rocket bodies, in Low Earth Orbit has been identified several years ago. Many potential solutions and architectures are traded with a main objective of reducing cost per debris. Based on cost consideration, specially driven by launch cost, solutions constructed on multi debris capture capacities seem to be much affordable The recent technologic evolutions in electric propulsion and solar power generation can be used to combine high potential vehicles for debris removal.

The present paper reports the first results of a study funded by CNES that addresses full electric solutions for large debris removal. Some analysis are currently in progress as the study will end in August. It compares the efficiency of in-orbit Active Removal of typical debris using electric propulsion

The electric engine performances used in this analysis are demonstrated through a 2012/2013 PPS 5000 on-ground tests campaign. The traded missions are based on a launch in LEO, the possible vehicle architectures with capture means or contact less, the selection of deorbiting or reorbiting strategy.

For contact less strategy, the ion-beam shepherd effect towards the debris problematic will be addressed. Vehicle architecture and performance of the overall system will be stated, showing the adequacy and the limits of each solution.

I. INTRODUCTION

Since the beginning of the spatial era, the number of debris in terrestrial orbit has been dramatically increasing. The distribution of the debris around the Earth is not uniform and Low Earth Orbits have been highlighted as a peculiar critical zone. This status is threatening the future utilization of those orbits for commercial and scientific missions. According to a sensitivity study of the effectiveness of active debris removal in LEO by M. Liou ([RD3 & RD4]), this phenomenon is for a large part generated by large debris located in altitude between 400 and 1000 km.

Among this population, a dedicated analysis has been performed in 2011 by Thales Alenia Space though CNES OTV (for Orbital Transfer Vehicle) contract ([RD2]), to focuse on the most important family who requires small impulsive deltav to go from one orbital debris to another one. The Figure I-1 highlights the family at high inclination from 95 to 100 deg which contains various debris among satellite in end of life or large launcher upper stage.

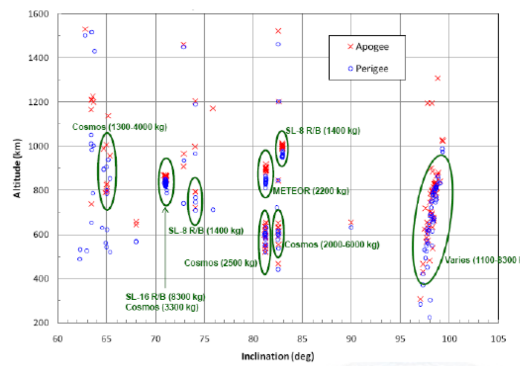

Figure I-1: family populations of debris >10cm in LEO

For our analysis, two typical debris are considered among the cible and represented in Figure I-2 with a "medium" mass of 3000 kg.

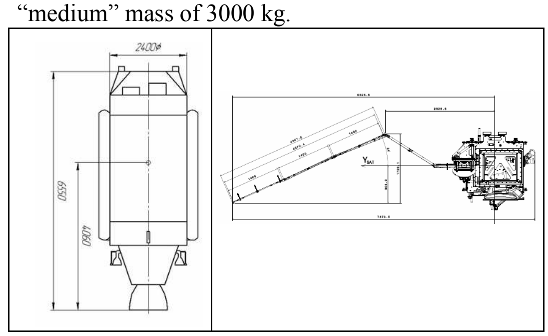

Figure I-2: Typical debris considered in the current study

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The conclusions of the OTV study are briefly summarized hereafter:

- The deorbiting strategies based on the installation of kits onto the debris generally lead to lower OTV mass/debris removed, therefore lower cost/debris,
- The AR5 class multiple launch of medium size OTV's appears to be the most efficient approach,
- The low thrust based deorbiting mission architectures are the most cost efficient solutions to place large debris on a 25 years re entry orbit.

This conclusion leads CNES to propose a dedicated analysis focused on a medium size OTV propulsed by low thrust with possible implementation of additional solid propulsion kits and able to remove more than one unique debris.

Considering low thrust propulsion, two strategies are possible. First one, the satellite is propulsed by the electric propulsion engine, captures the debris and changes the orbit of the complete stack. Second one, the satellite changes the debris' orbit using the ion beam shepherd concept.

Which strategy would allow to treat the larger number of debris, i.e. minimize the cost per debris removed? Are those strategies more efficient for de-orbiting or re-orbiting? How can we consider the impact of LOS constraint (25years orbit or direct re entry orbit)? What are the key technical points for each strategy ? This is all what the study reported in this paper is about: to compare mission concepts using electric propulsion applied to the deorbiting of a large debris population in LEO.

II. ANALYSIS PRINCIPLE

The approach followed to analyse and compare different removal concepts is shown in Figure II-1. The first step consists in establishing reference trajectories using two typical debris (refer to Figure I-1) and PPS 5000 demonstrated performances. The trajectory starts from the reference injection orbit and reaches either safe high orbit or low orbit for 25 years natural re-entry.

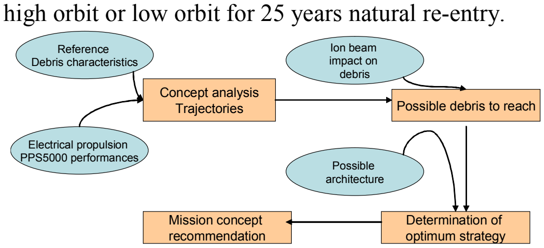

Figure II-1: study logic.

With the computed ΔV and the manoeuvres duration, a preliminary mass budget will be defined to evaluate the loaded on-board propellant mass.

For the contact less strategy, ion beam system simulations impact on debris allows determining the thrust and torque applied to the debris. The possible architectures are then traded and some general conclusions can be drawn. Each of these steps are further detailed in the following sections.

III. ELECTRIC PROPULSIVE DATA

Recent evolution of electric propulsive performances and solar array high power lead to consider high thrust electric propulsion. Through the HiPER program, SNECMA has enlarged his family from PPS 1350 (which has an in-flight validation and demonstration) with the recent PPS 5kW and PPS 20kW engines. The test campaign hold beginning of 2012 allows showing very high performance using different propellant such as Xenon, Krypton and Argon. During this campaign, tension is adapted to reach a large panel of propulsive range as presented in Figure III-1. A second test campaign on an improved PPS5000EM is foreseen mid 2013 with increased performances expected.

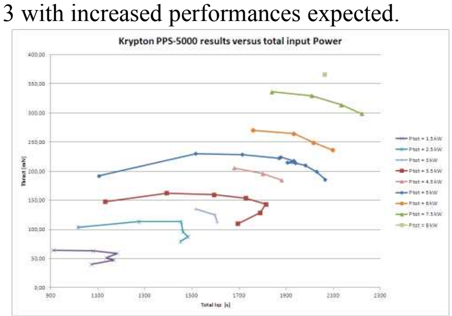

Figure III-1: Isp versus engine thrust demonstrated on Kr test campaign.

The amount of propellant intended for the chasing vehicle is around 2 tons. Considering current price per kilogram, the gap between Kr and Xe propellant is around 1M€ per mission. Furthermore, the use of electric propulsion by commercial satelite and therefore of xenon, will likely increase in next year's with incertitude on world supply capacity. Even if ionisation cost is a bit more expensive (potential of 1st ionisation 16eV for Krypton, compared to 12ev for Xenon), Krypton tests show comparable performance between the two propellants. Subsequently Kr has been chosen in baseline in this study.

In order to perform exhaustive system simulations of thrust impact on debris, it has been necessary to elaborate an effective coefficient of thrust obtained on the exposed surface. This data is computed from the divergence angle measurements on the test bench during Kr test campaign:

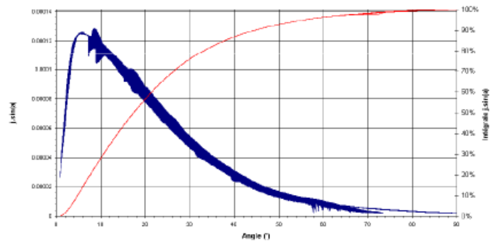

Figure III-2: Distribution of the divergence flow measured during PPS 5000 test campaign.

IV. CATEGORISATION OF MISSION STRATEGY

IV.I principles

For both capture and contact less strategies, the beginning of the mission is identical. The chasing vehicle will approach the targeted debris. After a dedicated observation phase, the chaser will determine the attitude and angular rates evolutions of the debris. Then the follow-on sequence will differ for the two possible strategies.

1/ For grapping strategy, the chaser will determine the best way to approach the debris and to capture it. Many TAS in-house activities have been performed to define the best capture strategy according to the debris motion (refer to RD [2]). The debris is then captured with a robotic arm and stacked on the upper deck of the chaser to minimize perturbation and ease the control of the complete stack for de-orbiting or re-orbiting manoeuvres.

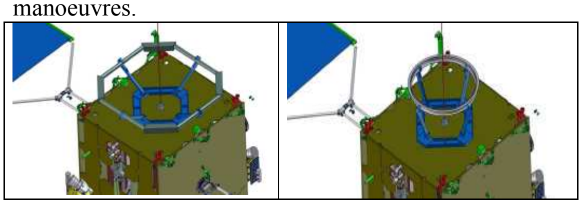

Figure IV-1: debris tong support system

2/ For the IBS (Ion Beam Shepherd) strategy, the approach is considered to stay at a safe distance of a debris in motion and the flow of a PPS 5000 on the top of the chaser will push the debris, without any contact between the two bodies. The chaser emits a quasi-neutral plasma beam directed towards the debris. The impingement of the ion beam on the debris surface produces a force F_3 and pushes forward the debris. The impulse toward the debris (F_2) is to be compensated by an opposite thrust (F_1). In accordance with the acceleration supported by the debris, the thrust (either F_1 or F_2) will be adapted during the entire mission to keep a safe distance wrt the debris, as shown on following picture:

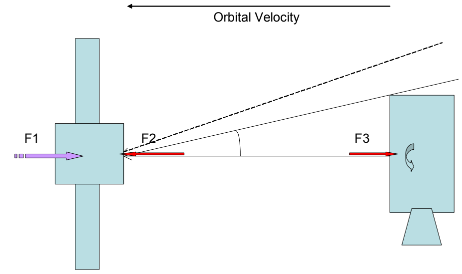

Figure IV-2: ion beam impact on debris

For both strategies, after de-orbiting mission of a first debris, the chaser satellite will used its electric main propulsion to go back to the closest debris in the selected area. Using electric propulsion offers high Isp, considering time for transfer is not a challenge for the mission. The different strategies are represented in Figure IV-2:

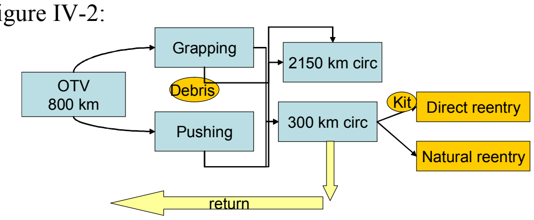

Figure IV-3: combination mission strategy

The deltaV cost is mainly driven by the changes on orbital altitude, and the changes on the ascending node. As follows, we consider that the ascending node will be modified using the natural drift induced by J2 effect, and only the semi-major axis will be changed at each desorbitation of a debris and following reorbitation of the spacecraft.

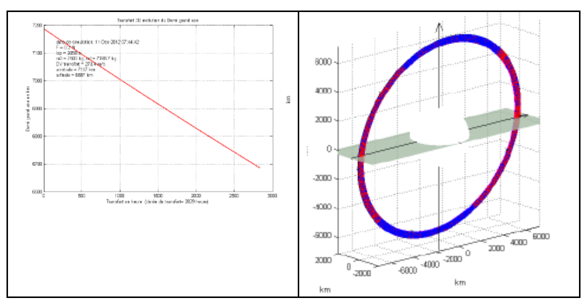

Figure IV-4: : Trajectory description for one debris de-orbitation

The optimal trajectories are computed using in-home low-thrust optimization software, T3D ([RD8]), which is based on the application of the Pontyagrin principle. One of the main constraints that are taken into account is the eclipse time. In order not to oversize the battery, the thrusters are switch off during eclipses. This strategy induces more operations and coaching from ground, combined with autonomous and reliable on-board software. The duration of the eclipses depends on the altitude and the local time of the orbital plan. Typical eclipse duration can be stated to 1000 seconds. One of the main characteristics of the optimal trajectory is the thrust duration. A maximum qualified lifetime of 15000 hours is considered to define the engines number to install on the chaser.

IV.II Respect of LOS

It is underlined that the removal process will have to follow the Space Law rules i.e. as far as LEO orbits are concerned: to place the debris on a targeted direct re entry orbit for the debris shown to constitute a human casualty risk or to put the debris on an orbit which will guarantee a re entry (non targeted) of the debris in less than 25 years.

For electric propulsion, the thrust is very low and cannot guarantee an in atmosphere re-entry slope sufficient to compute an impact area for controlled direct re-entry.

To guarantee direct strategy, a desorbitation kit has to be installed on the debris. The propulsion kit is evaluated as simple and cheapest as possible and will be based on solid propellant.

The re-entry of the chaser itself needs to be controlled. Trade-off activities will define in next future if heated hydrazine is sufficient to insure the direct de-orbitation or if a dedicated solid propulsion kit is necessary.

IV.III System simulation for ion beam impact

To study the IBS concept, it is necessary to model the plasma beam in order to estimate the plasma plume effects on the impinged surface. The tool used to perform this kind of simulation and to predict the plasma effect is ISP (Interaction Spacecraft Propulsion) developed by the Moscow Aviation Institute (MAI) in collaboration with TAS from the early 2000 (RD[5]).

The main interaction plasma plume/surfaces effects are: forces and torques, thermal flux, surfaces degradation due to sputtering and redeposition. The impact of the high energetic ions of Xenon on the spacecraft surfaces can lead to surface material erosion and the sputtered particles can contaminate other surfaces like the dissipative surfaces and the optical devices.

Simplified geometrical debris shapes are considered (radius 2,4 m / height 6,6 m) and distance (8m from the cylinder axis to the exit of the thrusters) is chosen to guarantee a safe debris motion.

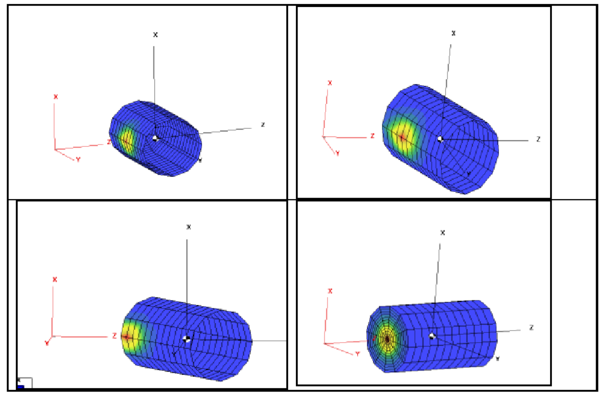

Figure IV-5: Plasma plume impingement on a cylinder – pressure angle =0° / 20°/ 45°/ 90°

The resulting force in Z direction varies with the pressure angle and provides a nominal value of 60% at 0° angle : $F_3 = F_2 * 60\%$

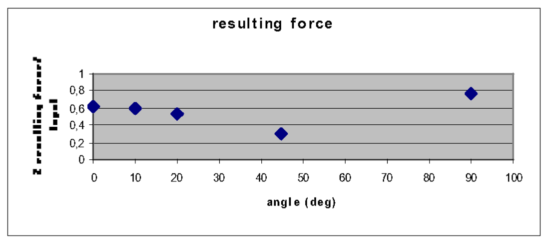

Figure IV-6: resulting force on the cylinder°

Same process is done with a “typical” satellite model including a body (2400mx2000mx6000m) and a solar array (2000m x8000m) . The safe distance between the thrusters exit and the S/C axis is supposed to be 9m.

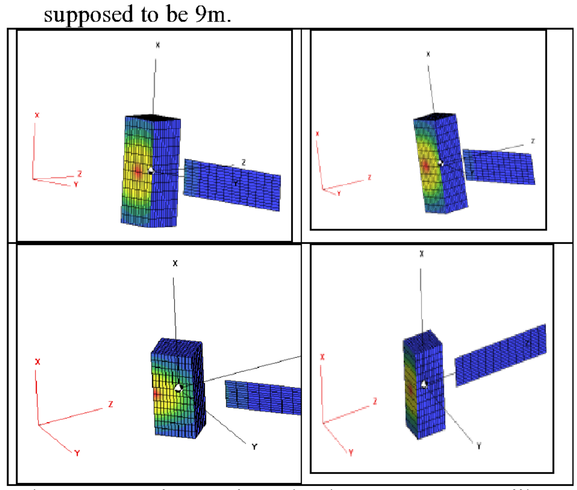

Figure IV-7: Plasma plume impingement on a satellite – pressure angle =0° / 20°/ 45°/ 90°

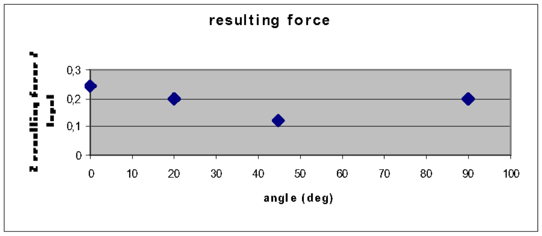

Figure IV-8: resulting force on the satellite

The pressure on the solar array panel induces also a non-neglictible torque (10^{-2} Nm) around X-axis.
To improve IBS efficacity, a clear understanding of satellite motion is needed to authorize the chaser to come closer to the debris.
Furthermore, the impact of high energy ions beams induces surfaces degradation through the sputter yield which corresponds to the number of molecules of the material sputtered by one incident ion. For each material, it depends on the energy of the incident ion and on its incidence angle.
Another phenomenon is linked to the erosion: the redeposition of the sputtered molecules, which can lead to a pollution of the surfaces. Depending on the re-emission law (diffuse or specular), the sputtered particles can reach sensitive surfaces such as radiative surface for thermal control or optical surfaces.

A preliminary calculation on the satellite test case is done to evaluate the thickness of the pollution resulting from the erosion. The considered materials are kapton for the satellite body and cover glass for the solar array.
The deposition rate on a “wall” behind the thrusters is ~5 .10-9 m/day = 5 nm/day. With the equivalent IBS propulsive duration around one year for this mission, the impact on optical surfaces sensors for example could be critical and needs to be evaluated and consolidated.

V. TRADED MISSION ARCHITECTURES

V.I Assumptions

The mission architectures that have been analyzed are all practicable combinations of the options defined in Figure IV-2 logic. The mass budgets have been established from credible spacecraft design assumptions, from Thales Alenia Space existing spacecrafts and help to define the debris number possible to treat.

The assumed launcher is a shared Ariane 5 launch in LEO orbit with 4500 kg OTV separated mass in 800 km altitude circular orbit at the aimed inclination.
The PPS 5000 at the nominal functional point will deliver 200mN and 2050s Isp. To decrease the thrust, the input tension will be modified.
Deorbiting kit is dimensioned at 130 kg for orbital change deltaV need and few additional equipments, derived from the ATK solid propellant engines catalogue.

A rendezvous phase between the chaser and the debris is considered, managed by hydrazine propulsion. This statement is still in process to clearly state the necessity to add an additional propulsion sub-system on-board. Attitude control by Xenon or Krypton cold gas could be considered depending on the debris motion velocity and capacity of the satellite control.

V.II Simulation results for grapping strategy

The OTV vehicle concept with a launched mass of 4500 kg will have to consider a capture mechanism with stereo camera system, a laser range finder, a robotic arm, and dedicated avionics for arm control and pose estimation. The Tracking, Telemetry and Control sub-system will be based on a standard S-Band with hemispheric antennas for omni-directionnal RF coverage. During the Rendez-vous phase, a high TM rate is necessary in X-band through several ground stations or TDRS system to insure a quasi permanent coverage during critical phases.
The Guidance, Navigation and Control sub-system is responsible to provide pointing and absolute/relative positioning observables and control for each phase, but also in case of contingencies. The chaser satellite is supposed to be a 3-axis stabilized with standard AOCS equipments: coarse three-axis gyro, 4 reaction wheels, magneto-torquers, star tracker, coarse sun sensor and GPS. A sub-system trade-off is in process to select sensors and actuators.

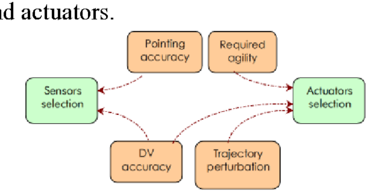

Figure V-1: AOCS sub-system trade-off
The precision during last phase rendez-vous is managed through precise observation sensors.
The propulsion sub-system consists of low-thrusts engine PPS installed on Thrusters’ Orientation Mechanism, a Power Processing Unit to convert satellite power to engine needs, a Filter Unit for plasma oscillations filtering, tanks loaded with “super critic” Krypton at 20°C regulated temperature, a Flow Regulator, and a Pre-card for electronic command of the regulator. Only one PPS 5000 will be “on” at one time.

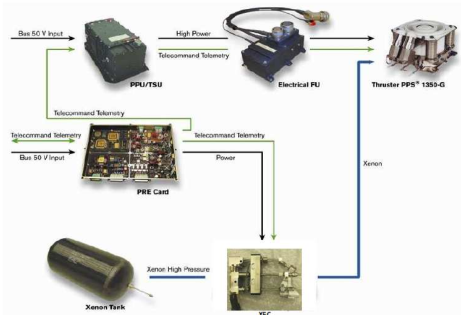

Figure V-2: Propulsion sub-system synoptic

The Power System architecture is based on a centralized design with a 100 Volts regulated bus and composed of solar arrays, SA drive mechanism, battery, pyro devices, harness and Power Conditioning and Distribution Unit ensuring battery and solar array management and regulating power voltage. With the standard system margins and PCDU losses, the spacecraft power demand is around 7700 Watts which fits into the current TAS Solarbus panels.

The preliminary mass budget could allow taking on-board up to 7 de-orbitation kits to ensure 7 debris removals. This capacity would depend on the occurrence of hydrazine and its capacity of providing sufficient V for OTV re-entry. Of course, if natural safe re-entry can be demonstrated for some satellite (i.e. no risk to leave the debris at 300 km altitude providing a natural re-entry in less than 25 years), the number of possible debris removal can increase up to 10. The cost of implementing the solution with kit would be much more expensive than the cost to place the same debris on 25 years re entry orbits using low thrust propulsion. This highlights the importance of the analyses of the human casualty risk caused by the large debris re entry.

The other strategy to reach a non-critical orbit at 2150 km altitude requires higher delta-v, but does not request to use solid propulsion de-orbitation kit. The preliminary estimation with comparable hypothesis lead to ensure 6 debris removals. Even if one debris less can be eliminated, one can point out in term of risk and maturity, this solution is better and does not present any technical issue at this time.

V.III Simulation results for IBS strategy

The modification of the propulsion sub-system impacts the global vehicle architecture. Same propellant tanks will feed one PPS 5000 for main satellite propulsion and one other PPS 5000 for IBS.

The thrust of both PPS are adjusted to guarantee a safe distance between bodies:

Using force equation equality for the debris (1),

$$\frac{dv}{dt}m_{deb} = K\frac{dm_2}{dt}v_{e2} \quad (1)$$

(where v_e is the gaz ejection velocity) and necessity to keep a constant distance between the debris and the chaser:

$$\frac{dm_{sat}}{dt} = -\frac{F_1}{v_{e1}} - \frac{dm_2}{dt} \quad (2)$$

The variation of the satellite mass is then function of engine parameters:

$$\frac{dm_{sat}}{dt} = -\frac{F_1}{g_0}\left[\frac{1}{I_{sp1}} + \frac{m_{deb}}{I_{sp2}\left(Km_{sat} + m_{deb}\right)}\right] \quad (3)$$

With this formula, and the system simulation, one can find the evolution of the thrust facing the debris all along the trajectory.

The strategy to reach a low orbit at 300 km is only considered for natural re-entry. The direct re-entry cannot be reach easily. One idea would consist in installing a solid propulsion kit by a harpoon need. Nevertheless, the accuracy prediction wrt centre of gravity location could be not compatible with this strategy. The credibility to guarantee a safe direct re-entry without any contact does not seem realistic at the current step of the study. With a 25-years natural re-entry, around 9 upper stage debris can be treated. If the debris motion and center of gravity localisation can be properly estimated through observation phase, the swept volume can be safely decreased. This allows the chaser to approach closer to the debris and induces one additional debris treated.

The other strategy to reach a non-critical orbit at 2150 km altitude requires higher delta-v. The preliminary estimation with comparable hypothesis lead to ensure 5 debris removals. Those evaluations need to be consolidated in term of propulsion sub-system mass and accomodation, in term of AOCS capacities, and associated system mass budgets.

V.IV Technological step

The grapping strategy will face :

- capacities of robotic arm to catch a non-controlled tumbling debris and to stick it on the chaser
- AOCS capabilities to control the stack.
- Kit installation and de-orbitation manoeuvres for direct rentry if any.

The robotic arm on-ground demonstration and in orbit demonstration (space station) allows to consider performances as reachable. A complete on-ground simulation loop would help to validate the complete process.

The ion beam shepherd analysis performed points out possible critical points:

- the direct re-entry cannot be reach easily.
- the impact of IBS on global debris motion
- the IBS efficiency

- the trajectory closed loop to keep constant distance between bodies
- the impact of high energy ion beam induces surface degradation and erosion. Redeposition of sputtered molecules cans lead to a pollution of the OTV surface, facing sensitive surfaces such as optical surface or thermal control.

An in-flight small demonstration on cubesat order or Piggyback satellites would help to caraterize and analyze those phenomena.

VI. SOME CONCLUSIONS

The analyses performed have shown the interest of using electric propulsion for deorbiting of a set of large debris in LEO :

- Using an enhanced RAAN drift strategy to transfer to each debris position in case of multi debris removal missions.
- The deorbiting strategies based on the installation with a robotic arm of solid propulsion kits onto the debris allows to consider direct re-entry,
- This direct re-entry could not be affordable with IBS
- IBS strategy is affordable to bring a debris to a 25 years re entry orbit
- Reaching high orbit far from the critical zone can be a good first step for IBS strategy

Some comments on the frame of the analyses shall also be made, in particular:

Operations costs and processes could be significant for such long duration missions.

A detailed analysis of the on-board regulation capacities and software control is needed if battery is not sized to cover eclipse phase.

Improvements of electric propulsion capacities will increase in the next future the advantage of those solutions.

A coupled dynamics calculation of the debris motion would need to be adjusted with in-flight demonstration, even on a small size approach such as cubesat.

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