SPACE DEBRIS REMOVAL USING MULTI-MISSION MODULAR SPACECRAFT

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ABSTRACT

The study and development of ADR missions in LEO have become an issue of topical interest to the attention of the space community since the future space flight activities could be threatened by collisional cascade events. This paper presents the analysis of an ADR mission scenario where modular remover kits are employed to de-orbit some selected debris in SSO, while a distinct space tug performs the orbital transfers and rendezvous manoeuvres, and installs the remover kits on the client debris. Electro-dynamic tether and electric propulsion are considered as de-orbiting alternatives, while chemical propulsion is employed for the space tug. The total remover mass and de-orbiting time are identified as key parameters to compare the performances of the two de-orbiting options, while an optimization of the ΔV required to move between five selected objects is performed for a preliminary design at system level of the space tug. Final controlled re-entry is also considered and performed by means of a hybrid engine.

1. INTRODUCTION

Since the beginning of the space activity in 1957 a large number of objects have been placed in orbit, without providing any measure for re-entry or re-orbiting capabilities. The result was the creation of very densely populated zones [2][13] that became a threat for the future spaceflight security. Collisional events between objects already in orbit, as happened in 2009 with Cosmos and Iridium, and accidental or intentional break-ups and explosions, as happened in 2007 with the Chinese Fengyan satellite, are symptoms of the *Kessler Syndrome*, an uncontrolled situation which could prevent any other space activity in the near future [13][14][21].

Although international agreements led to the adoption and implementation of mitigation strategies that should have limited the generation of new debris [1][3], long-term projections on the evolution of the space environment [6][17][18] suggested that the number of objects in orbit could increase rapidly in the next 20 – 30 years, even in case of drastic, unrealistic measures such as an immediate and complete halt of launches and release activities.

A shared viewpoint is that the only means to preserve the human access to space is by removing debris mass from orbit with active removal missions. The beneficial effects of ADR have already been demonstrated in [16][17][18]. A removal trend of a minimum of 5 objects per year from the most populated regions could be sufficient to stabilize the space environment for the next 200 years [6][18]. However ADR operations are severely constrained by their high predicted cost and by technical challenges that still require to be addressed [12][17].

In this framework, this paper proposes an ADR mission analysis where modular remover kits, properly sized according to the debris to be disposed, are employed for de-orbiting operations, while a servicing vehicle is responsible of the orbital transfers and rendezvous manoeuvres to carry the remover kits to the selected targets.

This concept implements a long-lasting infrastructure for ADR, since new remover kits could be launched on-demand and at low-cost in the orbital regions of interest, and the already-orbiting space tug could bring them to the new debris for removal.

In this paper, debris in sun-synchronous orbits (SSO) have been identified as priority targets since it is one of the most densely populated region, due to the intensive use for commercial interest. A preliminary design has been carried out at system level both for the space tug and the remover kits. For the former, an analysis of the orbital mechanics has been performed to determine the optimum ΔV required to rendezvous with multiple targets in a limited time. In the specific case, it has been considered to remove five objects within one year. For the latter, electro-dynamic tether and electric motors have been considered as de-orbiting alternatives and the total remover mass and orbital lifetime have been assumed as key parameters to compare their performances.

2. TARGET SELECTION

Being a potential source of many breakup fragments posing an additional collision risk, the targets for active removal should be large intact objects in crowded LEO regions characterized by a substantial orbital lifetime. They can be ranked according to $P_c \times M^{0.75}$, where P_c represents the overall catastrophic breakup probability of the object during its orbital lifetime, M is the dry mass and the exponent reproduces the trend of the cumulative number of collisional fragments according to the NASA standard breakup model [25].

The current distribution of abandoned intact spacecraft and upper stages in LEO is described in detail in [10]. Between 500 and 1100 km there are several altitude-inclination bands where efficient active debris removal might be carried out; three in particular have been recognized as the most critical [6][15] and are characterized by the following altitudes and inclinations:

1. Altitude = 1000 ± 100 km; i = 82° ± 1°;
2. Altitude = 800 ± 100 km; i = 99° ± 1°;
3. Altitude = 850 ± 100 km; i = 71° ± 1°.

Among them, the sun-synchronous orbit, the second listed, represents the most interesting region to be considered for the study and design of active removal missions. It is a very densely populated orbital regions and its high commercial and scientific interest, especially for remote sensing, make it one of the most intensively used orbit, as it can be observed from Tab. 1. In order to maintain this region safely practicable for future space activities, the removal of the existing debris objects is of the utmost importance.

Table 1: *Space Launches: Jan 1, 2011 – Feb 11, 2013*

Mission	Number	Percentage
Geosynchronous orbit	58	36.0 %
Sun-synchronous orbit	30	18.6 %
Manned space program (<500 km)	28	17.4 %
Military missions with classified or very low orbit	12	7.5 %
Navigation constellations in MEO	11	6.8 %
Miscellaneous orbits	11	6.8 %
Escape velocity launches	4	2.5 %
Globalstar (h = 921 km; i = 52°)	3	1.9 %
Gonets (h = 1490 km; i = 82.5°)	2	1.2 %
Molniya orbits	2	1.2 %

2.1. Potential targets in sun-synchronous orbit

As of 19 February 2013, there were 192 spacecraft and 74 upper stages, excluding classified objects, in sun-synchronous orbits with mean altitudes between 700 and 1100 km [23]. The mean altitude and inclination distributions, as a function of the right ascension of the ascending node at 00:00 UTC, are shown, respectively, in Figs. 1 and 2.

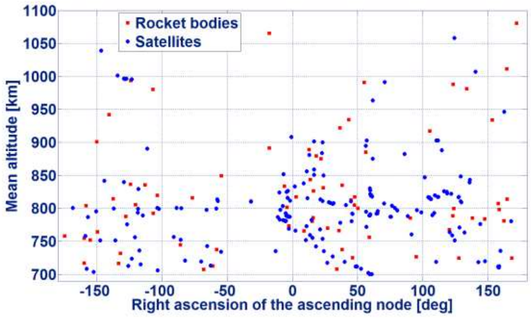

Figure 1. Distribution of unclassified satellites and rocket bodies in sun-synchronous orbits: mean altitude vs. right ascension of the ascending node

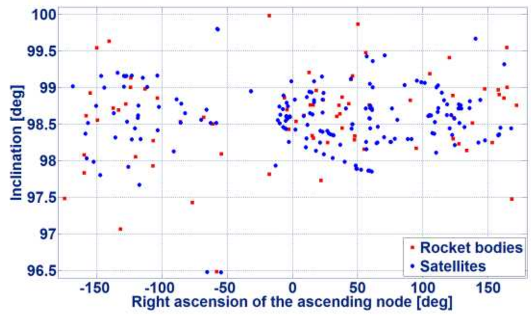

Figure 2. Distribution of unclassified satellites and rocket bodies in sun-synchronous orbits: inclination vs. right ascension of the ascending node

According to a survey carried out by the Union of Concerned Scientists [24], the operational satellites should be about 90, so the total number of potential targets to be removed is around 175. But contrary to the situation prevailing in most of the other crowded altitude-inclination bands, spacecraft and upper stages in sun-synchronous orbits are extremely heterogeneous, with the full range of models and masses represented [10]. Abandoned satellites, for instance, include both cubesats and the 8 metric tons Envisat. This means that a specific debris removal option might be used only a relatively limited number of times. However, some families of upper stages might offer attractive targets for removal, as can be seen in Tab. 2.

3. ADR CONCEPT

The active removal scenario proposed in this paper is based on the employment of a large servicing vehicle, called space tug in the following, which performs multiple orbital transfers and rendezvous manoeuvres

Table 2. Upper stages in sun-synchronous orbits

Upper Stage	No.	Dry Mass (kg)	Mean Altitude (km)	Inclination (°)
Agena D stage	3	673	784-1067	99.9-100.0
Altair stage	1	30	717	98.1
Ariane 1 H8	1	1450	786	98.8
Ariane 4 H10	5	1800	756-786	98.3-98.8
Ariane 5 EPS	1	3600	770	98.2
Burner 2 stage	17	116	707-836	97.5-99.1
CZ-2 2nd stage	4	4000	727-827	98.1-98.3
CZ-4 3rd stage	10	1000	717-918	98.2-99.4
Delta 2nd stage	6	820	708-942	96.5-99.6
Dnepr-1 3rd stage	10	2356	758-991	97.4-98.6
H-2 2nd stage	2	2700	785-1081	98.5-98.8
Molniya 3rd stage	1	879	802	98.7
PSLV 4th stage	5	920	713-835	98.4-98.9
Rokot Briz KM	1	1420	901	99.5
Scout 4th stage	2	25	754-1012	97.8-99.5
Taurus 4th stage	1	203	725	99.2
Vostok 2nd stage	2	1440	885-889	99.2-99.5
Zenit 2nd stage	2	8300	805-994	98.3-99.1

carrying several remover kits that are attached on the correspondent target to accomplish de-orbiting operations. Since the objects to be removed are extremely heterogeneous, as happens for those in SSO, the remover kits could be extremely variable. In order to develop a standardized solution to tailor proper remover kits to different debris, this paper proposes a modular remover architecture, where each remover kit is constructed by assembling a certain number of micro-satellite units, equipped with different de-orbiting devices, according to the characteristic of the object to dispose.

Several advantages could be expected from this kind of scenario. The redundancy provided by the concurrent employment of multiple modular vehicles offers a higher fault tolerance and mission reliability since the failure of one remover does not compromise the entire mission; moreover, the separation between orbital transfers and de-orbiting manoeuvres allows to optimize independently both of them. The removers modular architecture implies a high system flexibility, thanks to the possibility of tailoring each de-orbiting package according to the characteristics of the debris to deorbit, with significant mass and cost savings, and makes the concept scalable and adaptable both to large and small objects, varying the number or the size of each unit. Finally, thanks to the limited mass and size of the removers with respect to those of a single, large spacecraft, cost savings could be expected.

3.1. Remover modular configuration

The remover kits are based on the assembly of a certain number of elementary units that depends on the client debris properties (mass and altitude) and hence the de-orbiting option adopted.

Four elementary modules have been identified:

- I a *main bus*, with the ADCS subsystem, the TT&C and C&DH subsystems, the power plant, and the capture device; the main bus is the principal unit of the remover, at which the other modules necessary for the de-orbiting are hooked up;
- II an *electric thruster* unit, which includes the electric thruster itself, its own power plant (solar panels and batteries), the tank and all the devices necessary for its functioning;
- III an *electrodynamic tether* unit, with all the devices necessary for its functioning;
- IV a *hybrid rocket* unit for the atmospheric controlled-re-entry, if necessary.

Among the existing propulsive technologies, electric propulsion represents a suitable solution for active removal of massive objects thanks to the low propellant mass requirements and since it could be sufficiently mature to be used in a very near future [9]. Electro-dynamic tether appears to be a very promising propellant-less concept for active removal, but the vulnerability to the space environment of the classic cylindrical tethers, still represents one the most critical perceived obstacles for its employment [4][19][20]. The use of tape (electrodynamic) tethers that are currently being studied for deorbiting and tested [22] for their survivability to micrometeoroids impacts may remove this obstacle in the future.

Each unit was preliminary sized, assuming a maximum mass per unit of 100 kg.

The mass and the power required by the subsystems of the main bus unit were determined according to historical data and design relations reported in [26]. A preliminary mass of 50 kg was estimated, including a safety margin of 20%.

An already developed electric thruster (BUSEK BHT-200-X2B) was considered for the propulsive unit: it has a mass of about 1 kg, a specific impulse of 1350 s, 13mN of thrust and requires 200 W of power [5][8]. The power plant mass was determined according to the nominal power of 200 W, with a margin of 20%, and it included the mass of the solar panels and the batteries. A dual-junction InGaP/GaAs-on-GE solar cell technology [11] was considered to size the mass and the performances of the solar arrays and Li-ion batteries were selected as secondary power source. A 15% of mass was added for the harness [26]. Ti6Al4V, the common material used for space tank, was assumed to size the mass of the tank [26]. A margin of 25% was considered for taking into account the mass of the mounting hardware and the pressure regulator system. A total dry mass of 27 kg was obtained, with a maximum propellant mass that could be stored per unit of about 70 kg at a pressure of 70 bar.

The EDT unit was estimated to have a mass between 50 kg and 70 kg depending on the mass of the debris to deorbit: 50 kg was assumed for objects below 4000 kg, while 70 kg for the more massive objects. The unit includes the tether, tip mass and all the mechanisms necessary for the deployment and control of the system dynamics after deployment [22].

For the HEM (Hybrid Engine Module) a parametric mass evaluation was performed, being propellant and inert mass dependent on the size of the captured debris and of the chasing unit. The method here adopted for inert mass evaluation was mainly derived from [26]. The performance of HTP (High test peroxide, 87% concentration) and HTPB-based elastomer (Hydroxyl-terminated polybutadiene) at 10 bar pressure, exhausting in vacuum through a nozzle with area ratio of 100, were considered. HEM configuration was investigated at O/F = 7, about the maximum of the gravimetric specific impulse, which was about 315 s.

The masses of the elementary modules are reports in Tab. 3

Table 3. Mass of the elementary modules

Parameter	Value
Main bus mass	50 kg
EP unit dry mass	27 kg
Maximum propellant mass per unit (EP)	70 kg
EDT unit mass	50kg - 70 kg

3.2. Mission operations

The mission is considered to start with both the space tug and the removers already in the same orbit and docked together. Assuming a removal trend of 5 objects per mission, it was supposed that five remover vehicles were launched, already assembled in a single larger structure. The entire system is transferred by the space tug to each selected debris, according to an optimized sequence of manoeuvres. At the end of each transfer, a remover kit is detached from the structure to accomplish the capture and de-orbiting operations of the correspondent debris, and the space tug transfers the remnant removers to the subsequent targets.

Two scenarios are considered for the debris de-orbiting. In *Scenario 1*, the orbit of the debris is lowered through a continuous low thrust manoeuvre, while in *Scenario 2* an electro-dynamic tether is used; finally, a controlled re-entry is performed with the hybrid engine module, properly sized according to the final system mass (debris + remover kit). Once all the five removals have been accomplished, a new set of five removers is launched in orbit, together with a dedicated unit that re-supply the space tug with the propellant necessary to perform the new sequence of orbital transfers.

3.3. Mission analysis

The main objectives of the analysis were i) to determine the proper configuration of the remover kits for each selected debris (i.e. number of electric thruster units and/or EDT unit), ii) to quantify the performances of the different de-orbiting strategies and iii) to estimate the size of the space tug, according to an optimized sequence of orbital transfers between some selected debris.

The two de-orbiting scenarios previously mentioned were compared based on the total mass of the remover kits and the orbital lifetime as drive parameters.

The removal of five objects per year was assumed for sizing the mass of the space tug propulsion system. The main problem to face was the determination of the optimal sequence of five orbital transfers that minimizes the total ΔV necessary to accomplish the entire mission. Since all the targets lied in the same SSO region, the altitude and inclination variations did not affect significantly the ΔV budget; more critical were the changes in terms of RAAN. Several sets of five objects were considered and a *Matlab* code was developed to determine the best sequence of multiple orbital transfers. In this evaluation, it was first verified the possibility to take advantage of the natural precession of the line of nodes due to the Earth's oblateness to obtain a RAAN alignment within a given time limit. If the time limit is not met, then the code computes the ΔV necessary to change the RAAN between chaser and target through an impulsive manoeuvre. Considering a time interval for each rendezvous and capture phase of about 15 days, the remaining time for the orbital transfers was 290 days. Considering that four transfers are required to reach the five debris, and supposing that no impulsive manoeuvres would have been performed for RAAN changes, the time limit for the natural alignment of the RAAN was set at seventy-two days, in order to ensure a total mission time of one year.

3.4. Remover mass

The mass of the remover kits depends on the number and the type of units assembled.

In *Scenario 1*, the mass of the remover kits was the sum of the masses of the main bus, of the HEM and of the electric thruster units. The propellant mass and the number of propulsive units required for the selected debris were determined in an iterative way, starting from the base equations of propulsion (Eqs. 1-2) and assuming a maximum propellant mass per unit of 67 kg, considering a 5% of residual fuel mass [26]:

$$m_p = m_{fin}(e^{\frac{dv}{I_{sp} g_0}} - 1) \quad (1)$$

$$dv = \left| \sqrt{\frac{\mu}{a_i}} - \sqrt{\frac{\mu}{a_f}} \right| \quad (2)$$

In these equations, dv is the dv for a low thrust manoeuvre [27], I_{sp} is the specific impulse of the electric thruster, g_0 is the gravitational acceleration, a_i and a_f are the initial and final semi-major axes of the de-orbiting manoeuvre, m_{fin} is the final mass at the end of the de-orbiting phase (debris mass and remover kit dry mass).

In *Scenario 2* the mass of the remover kits was simply the sum of the mass of the main bus, of the EDT unit and of the HEM.

To size the mass of the hybrid engine module, the controlled re-entry was assumed to begin at 250 km altitude. As initial approach, the ΔV budget regarded a single firing which moved the spacecraft assembly to an elliptical transfer orbit with perigee at zero altitude. No optimization was performed on this specific aspect. Such manoeuvre requires ΔV = 75 m/s.

HEM propellant budget was obtained with Tsiolkovski equation, assuming a constant specific impulse during the firing. This is not actually true for a hybrid rocket since an oxidizer-to-fuel mass ratio shifting can occur during operations, depending on grain geometry. However, this information is beyond the level of detail for this study and proper grain design can limit such occurrence.

Inert computation considered combustion chamber vessel, injection plate, tanks for oxidizer and pressurizing gas, nozzle, the pressurizing gas itself and a 10% margin for other contributions. Structures were assumed to be made of aluminium.

The present deorbiting mission required a low ΔV budget from the chemical rocket system. That is, inert hardware mass did not represent a negligible contribution in the total budget. Actually, when deorbiting of small-medium objects is needed, the mass of inert components exceeds the one of the propellant. The leading factor is represented by the weight of such components that down scale in accordance with the quantity of propellant till a technological limit (e.g. the thickness of combustion chamber walls).

Globally the HEM inert fraction as well as the ratio between the HEM module and the target payload decrease as the mass to remove is increased. However, at this standpoint it is not yet possible to establish a cut-off weight that makes the use of this strategy convenient since other mission-specific aspects should be considered for the choice of the propulsion unit dedicated to the final firing.

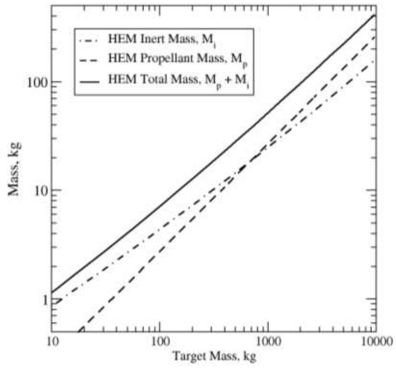

Figure 3 HEM mass for different target sizes

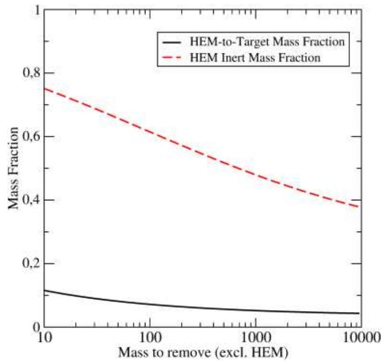

Figure 4 HEM mass fractions

3.5. De-orbiting time

In *Scenario 1* the orbital lifetime was calculated from Eq. 6, assuming constant acceleration and neglecting the propellant loss during the manoeuvre:

$$\Delta t = \frac{m_{in}\, dv}{F_{tot}} = \frac{m_{in}\, dv}{n_p F} \quad (3)$$

m_{in} is the total initial mass (debris and remover kit loaded), and F is the thrust of each electric thruster.

In S*cenario 2* the orbital lifetime was calculated according to [7], assuming a tether length of 5 km.

3.6. Space tug sizing

A chemical propulsion technology, based on liquid bipropellants N2O4/MMH, was considered for the space tug. According to [26] , a specific impulse of 300 s and an inert mass fraction of 0.17 were assumed. The total system mass was calculated according to the Eqs 4-6, including a safety margin of 10%:

$$MR = e^{\frac{dv}{g_0 I_{sp}}} \quad (4)$$

$$\frac{m_{prop}}{m_{pay}} = \frac{(MR-1)(1-f_{inert})}{(1-f_{inert}MR)} \quad (5)$$

$$f_{inert} = \frac{m_{inert}}{m_{inert}+m_{prop}} \quad (6)$$

where dv is the dv calculated for each single transfer, MR is the mass ratio, f_{inert} is the inert mass fraction, m_{pay} is the payload mass, which includes the mass of the removers that are carried in each transfer.

4. SIMULATIONS AND RESULTS

Two sets of five debris were selected as representative for the mission analysis. In the first one, a class of homogeneous objects was considered (Tab. 4): in particular, in the Ariane third stages class (Tab. 2), four Ariane 40 and the Ariane 5 were selected as possible targets. The second set of debris (Tab. 5) was composed by spent spacecraft, with mass variable between 3000 kg and 8100 kg.

Tabs. 6-7 show the results obtained regarding the removers mass and the orbital lifetimes for the selected sets of debris. It is also indicated the number of electric thruster units which should be assembled in the case of low thrust de-orbiting. In Tabs. 8-9 the difference, in percentage, between the mass and de-orbiting time obtained in the two scenarios, respect with the EDT case, are reported. The remover mass in *Scenario 1* and the de-orbiting time in both scenarios are strongly affected by the target object's mass. From the results listed in Tabs. 8-9 it emerges that for lower massive debris, as for example the Ariane 40 third stages (mass below 2000 kg), the mass of the propulsive remover kits is comparable to that of the EDT remover kit (mass differences below 10% could be neglected due to the assumption made in the remover elementary units sizing); according to the sizing procedure described in section 3.4, only one propulsive unit is necessary for the de-orbiting of these objects (see Tab.6). The de-orbiting time, instead, results higher in the *Scenario 1*, which means that the electro-dynamic force due to the tether is more effective than the electric motor thrust. For more massive objects, as the Ariane 5 third stage (Tab. 4) or the satellites in Tab. 5, the number of propulsive units required for the de-orbiting increases, and hence the propulsive thrust, since it was supposed that all the motors work simultaneously. This implies, on the one hand, an increased remover kit mass, but, on the other hand, the de-orbiting time could be significantly reduced with respect to that of the EDT scenario. These two parameters, the total remover kit mass and the de-orbiting time, play an important role during the removal mission design, driving the choice of which solution could be better in each specific case. The remover mass drives the estimation of the mission cost, especially regarding the launch costs, while the de-orbiting time represents a key parameter to consider for the mission risk assessment. One of the main threat for the EDT is represented by its high extension for few kilometres and its vulnerability to small debris, which are very numerous in the orbital regions crossed during the descending phase. Debris flux simulations could be required to determine the risk of catastrophic collisions in both scenarios, but this aspect is out of the scope of the paper and it will not be examined in more detail. Anyway, it can be concluded that the choice of which de-orbiting solution could be better than the other depends on which one allows the best compromise between the mass and the de-orbiting time, according to the costs and risk requirements stated during the removal mission design.

Table 4: Properties of the first set of debris

Name	h mean [km]	i [deg]	Ω [deg]	Mass [kg]
ARIANE 40 20443	766	98.76	35.78	1800
ARIANE 40 21610	756	98.76	58.85	1800
ARIANE 40 22830	786	98.76	15.76	1800
ARIANE 40 25261	780	98.34	15.8	1800
ARIANE 5 27387	770	98.25	27.05	3600

Table 5: Properties of the second set of debris

Name	h mean [km]	i [deg]	Ω [deg]	Mass [kg]
SPOT 5 27421	822	98.63	121.79	3000
METOP-A 29499	817	98.67	111.35	4100
METOP-B 38771	817	98.72	111.91	4424
COSMOS 2441 33272	719	98.11	110.30	6500
ENVISAT 27386	764	98.44	120.53	8100

The mass of the space tug was estimated according to equations (4), (5) and (6), considering the optimal sequence of orbital transfers determined for each set of debris; The EDT removal kits were considered for the four Ariane 40 debris in Tab. 4, while propulsive kits were assumed for the Ariane 5 and all the debris in Tab.5. The results are shown in Tabs. 10-11.

Table 6: Removers mass and de-orbiting time for the first set of debris

Name	Propulsion (Scen.1)			EDT (Scen.2)	
	Mass [kg]	n_p	T [y]	Mass [kg]	T [y]
ARIANE 40	216	1	1.40	**195**	**1.10**
ARIANE 40	216	1	1.37	**195**	**1.10**
ARIANE 40	216	1	1.45	**195**	**1.10**
ARIANE 40	216	1	1.44	**195**	**1.10**
ARIANE 5	359	2	**1.39**	**270**	2.11

Table 7: Removers mass and de-orbiting time for the second set of debris

Name	Propulsion (Scen.1)			EDT (Scen.2)	
	Mass [kg]	n_p	T [y]	Mass [kg]	T [y]
SPOT 5	327	2	**1.27**	**245**	1.91
METOP-A	401	2	**1.71**	**312**	2.57
METOP-B	422	2	**1.84**	**326**	2.75
COSMOS 2441	561	3	**1.50**	**412**	3.39
ENVISAT	675	3	**2.03**	**478**	4.52

Table 8: remover mass and orbital lifetime of the propulsive scenario compared to those of the EDT scenario, for the first set of debris

Name	Δ m (%)	Δ T (%)
ARIANE 40 R/B	10.8	36.9
ARIANE 40 R/B	10.8	37.6
ARIANE 40 R/B	10.8	35.7
ARIANE 40 R/B	10.8	30.2
ARIANE 5 R/B	33.0	**-34.4**

Table 9: remover mass and orbital lifetime of the propulsive scenario compared to those of the EDT scenario, for the second set of debris

Name	Δ m (%)	Δ T (%)
SPOT 5	33.5	**-33.4**
METOP-A	28.5	**-33.3**
METOP-B	29.4	**-32.9**
COSMOS 2441	36.1	**-55.8**
ENVISAT	41.3	**-55.2**

The ΔV budgets for the orbital transfers are significantly high, resulting in a quite massive propulsion system for the space tug, whose mass could be of the order of several tons, as it can be observed from the last column of Tabs. 10-11. This fact is due to the significant ΔV required for the impulsive manoeuvres necessary for the RAAN alignment. The total mission time is well within one year, allowing a removal trend of five objects per year as stated in section 3.3. Taking advantage of the natural alignment of the lines of the ascending node, the mass of the space tug could be reduced, but the mission time would exceed the limit of one year. Alternative orbital transfers could be considered to avoid the impulsive manoeuvres, as for example first lowering the orbit of the space tug in order to obtain a quite different nodal precession, and then raising it again when the orbital plane has rotated properly to match the next debris; also alternative propulsion systems could be implemented to lower the mass of the space tug.

Table 10: total ΔV, optimal sequence, total mission time, and space tug masses (propellant and wet) estimated for the first set of debris for the Scenario 1.

ΔV [km/s]	sequence	Time [days]	Prop. mass [kg]	Wet mass [kg]
6.87	43512	54	5269	6983

Table 11: total ΔV, optimal sequence, total mission time, and space tug masses (propellant and wet) estimated for the first set of debris for the Scenario 1.

ΔV [km/s]	sequence	Time [days]	Prop. mass [kg]	Wet mass [kg]
4.5	51243	56	2524	3345

5. CONCLUSIONS

This paper presented the analysis of an ADR mission scenario where a large space tug is employed to transfer a certain number of modular removal kits, and release them in proximity of the debris to de-orbit. Two de-orbiting scenarios were analysed: in the first one, the orbit of the debris is lowered through a continuous low thrust manoeuvre, in the second an EDT is employed. The two alternatives have been compared in terms of total remover mass and de-orbiting time. From the results obtained it emerged that for low massive objects (<2000 kg), the two options are comparable in terms of removal kit mass, but the EDT performs better as regards the de-orbiting time. For quite massive debris (>2000 kg), although the mass of propulsive removal kits increases, the de-orbiting time is significantly lower compared to that of the EDT scenario. A risk assessment and a mission costs analysis should be performed to determine which solution could perform better, according to the costs and risk requirements stated in the removal mission design. A preliminary sizing of the space tug was performed, considering a chemical bipropellant propulsion technology. An optimal sequence of multiple removals was determined for two sets of debris. Form the results obtained, it emerged that very massive space tugs are required to accomplish this kind of mission. Alternative manoeuvres and /or propulsion systems could be considered to reduce the mass needed for the entire mission.

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