Evaluation of Different Concepts for Active Debris Removal

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Abstract

The highest densities of orbital debris can be found at orbital altitudes near 900 kilometres. Within this orbit region, weather satellites, as well as Earth observation satellites like Envisat, perform their missions. While orbital debris already poses a threat to these satellites mainly in the sun-synchronous orbits, the risk of collisions will further increase in the future. Due to the high relative velocities of the objects of up to 15 km/s, collisions would result in highly energetic crashes, leading to a complete fragmentation of the participating objects. Such a collision will most likely be a so-called catastrophic collision. The resulting fragments might themselves collide with other objects causing a so-called feedback collision.

Today, catastrophic collisions are not a major problem, occurring approximately once in every five to ten years. However, the ongoing increase of space debris objects will increase the future collision risk. In order to avoid catastrophic collisions to become the leading source of space debris in the future, mainly the large objects, which apparently trigger collisional cascading, have to be removed from their orbit after their mission. As many satellites and rocket bodies are not able to perform a deorbiting maneuver by themselves, active deorbiting has to be performed for those objects.

In this paper different concepts already proposed for active removal of objects are further investigated. The first part will contain an introduction to the properties of the different techniques. This will be done with respect to satellites and rocket bodies which are likely to be primary objects for an actual mission. These objects are filtered by defining priority criteria, which can, for example, be derived by examining object flux (using ESA's MASTER-2009 model) and object mass. In the second part the relevant mission parameters, mission duration and system mass, will be determined for the different techniques.

Keywords: Active Removal, Deorbit, Space Debris

1. INTRODUCTION

Recent studies have shown that the current object population especially in low Earth orbits (LEO) has reached a critical state in view of the projected population increase in the near future [1, 2, 3]. Even without any future launch activities the object population will increase due to mutual collisions between the debris objects generating new objects which are likely to collide themselves and produce even more debris. This effect, known as collisional cascading, was first analysed by Donald Kessler in 1978 [4], and is thus often referred to as the "Kessler-Syndrome".

In order to stabilise the LEO environment, mitigation guidelines have been proposed by the IADC [5]. However, the mitigation measures will only slow down the increase of the population by

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accidental collisions [3]. In addition to mitigation measures the active removal of non-operating spacecraft, as well as rocket bodies, will be mandatory to stabilise the critical regions at altitudes of 600-1000 km. Therefore, an annual rate of about five actively removed objects has been estimated in recent analyses [1].

The active debris removal (ADR) presupposes a dedicated spacecraft, which is launched from Earth, and performs rendezvous and docking maneuvers to finally deorbit a previously determined target. Within this study three different concepts to remove targets from their orbits are evaluated. For this purpose a software tool has been designed, which estimates the key parameters *total system mass* and *total mission time* for chemical (CP) and electrical (EP) propulsion systems, as well as electrodynamical tether systems (EDT). Different scenarios shall be evaluated, also including a multi-target mission approach, where multiple similar targets are removed within a single mission. The evaluation focuses on deorbiting those objects, which pose the greatest threat to their environment in order to achieve the highest benefit-to-cost ratio as stated in [3].

2. PRIORITY TARGETS

The first step in ADR is the identification of the prioritised target objects. Those objects have the highest risk of being involved in a catastrophic collision, which leads to a total fragmentation of the respective collision partners. The resulting debris cloud will further increase the risk of future collisions within the orbital region the collision occurred in. The onset of the collisional cascading will most likely be within the orbital region with the highest spatial density, which, in general, correlates with the collision risk. For the MASTER-2009 model, Fig.1 shows that the most densely populated regions are between 600 and 1000 km for high declinations. This includes on the one hand the sun-synchronous orbits which are used by many operational spacecraft e.g. Earth observation, weather and military reconnaissance satel-

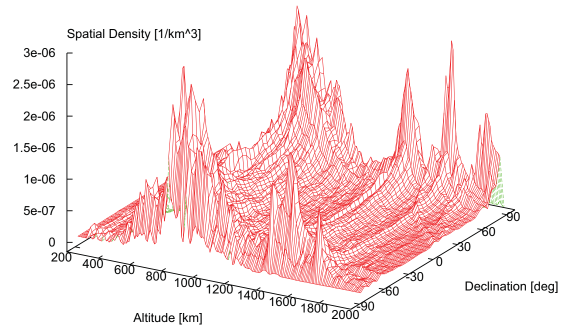

Figure 1: Spatial density vs. altitude and declination for LEO objects > 1 cm (MASTER-2009)

lites. These satellites are often located within a narrow inclination band of approximately 98 degrees. On the other hand, there are mainly Russian military and Earth observation satellites in the same altitude band, but with inclinations about 82 degrees. This further increases the probability of catastrophic collisions as head-on collisions occur with relative velocities of up to 15 km/s, resulting in extremely high kinetic energy impacts.

2.1. Defining Priority Criteria

In order to achieve a maximum benefit-to-cost ratio for each individual ADR mission, potential targets within the critical orbit region have to be ranked by defining an appropriate priority criterion R. In [1, 2] this criterion was defined as

$$R_i = P_c(i) \cdot m_i, \quad (1)$$

where P_c is the collision probability and m the mass of the i^{th} object. While it is clear that with increasing collision probability the ranking value should also increase, the mass indicates, how many new debris objects will result from a catastrophic collision when, as in [2], the NASA Breakup-Model is applied. Thus, an increasing mass implies an increasing risk to other objects.

An alternative method to derive a priority ranking is based on the flux, which can be processed for each object using the ESA MASTER-2009 (Meteoroid and Space Debris Terrestrial Environment Reference) model. MASTER considers all relevant space debris sources and meteoroids down to one micron for historical as well as future populations. In this study only objects larger than one centimeter in size are considered for flux calculations, including cataloged objects, launch- and mission-related objects, explosion and collision fragments, solid rocket motor slag and sodium-potassium droplets. This is due to the fact that only objects greater than about 1 cm possess enough kinetic energy to cause a total fragmentation event. The ranking criterion shall be defined as

$$R_i = F(i) \cdot m_i^{0.75}, \quad (2)$$

where F is the flux and m the mass of the i^{th} object. The mass of each object is raised to the power of 0.75 according to the NASA Breakup-Model. Only objects with a mass greater than 100 kg are considered for the determination of the ranking, neglecting those objects which spend only a small fraction of time in the critical region, e.g. objects in high-eccentricity orbits. Furthermore, those objects, which are operational and are able to perform an appropriate re- or deorbit maneuver at their end-of-life according to the IADC guidelines, are not considered in this analysis. In Fig.2 the first 500 objects of the priority ranking derived by Eq. 2 are shown. The result is similar to the one derived in [1], where Eq. 1 was used, thus giving further confirmation for the validity of the process.

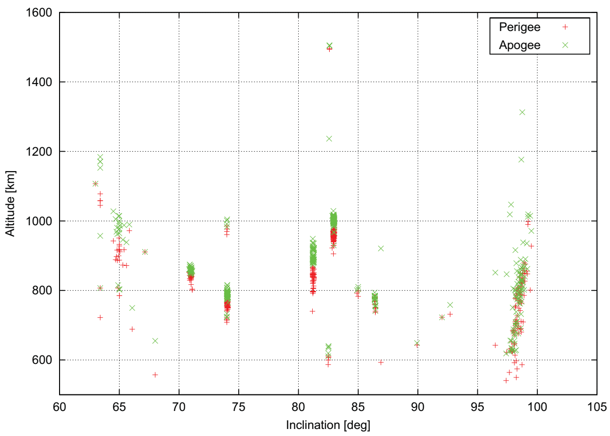

Figure 2: Top 500 objects in LEO

2.2. Target Objects

The priority ranking shows, that there are a lot of geometrically similar rocket bodies among the top listed objects. Within the Top-30 objects, the second stage of the Zenit-2 launcher is clearly dominating with 21 entries. Objectively speaking, the Zenit-2 stages would be ideal objects when starting with ADR, as the design of a dedicated spacecraft would only require minor adaptions for each mission, if any. It could also be possible to actively remove multiple rocket bodies of the Zenit-2 type within one single mission, assuming one could use the same de-spin and docking mechanisms for all objects. However, there might be political and/or legal constraints when deorbiting objects from other countries or organizations. In order to avoid problems of this type, national agencies or even companies could focus on removing their own objects, for example defunct satellites within constellations.

In this study two different rocket bodies have been chosen to analyse the ADR characteristics. Firstly, the Zenit-2 second stage as the object which would most likely be the primary one for the first missions, and secondly, the Ariane-4 upper stage of type H10, which is also repeatedly listed within the Top-500 objects. The H10 upper stage would be an example, if ESA decided on actively removing objects, but political or legal aspects would still be prevailing.

3. ADR TECHNIQUES

Three different concepts are analysed in this study to actively remove the objects, which have been identified in the previous section. Other concepts, as the solar sail, may be subject of further studies. A software tool to simulate the ADR of user-specified objects was developed. It allows for a declaration of the residual lifetime of the objects, which could, for example, be 25 years according to the IADC guidelines. Such a scenario would allow for removal of multiple objects when using electrical propulsion or electrodynamical tethers, as otherwise (in case of the direct deorbit) the ADR spacecraft (ADRS) would have to spiral down into the denser layers of the Earth's atmosphere, and not being able to spiral up for the next target.

The algorithm for all techniques is similar and consists of the following phases:

1. Rendezvous & docking (RVD, including despin of potentially tumbling objects)
2. Deorbit maneuver or burn-phase
3. Undocking and going for the next target (if required), including the estimation of the time required for the alignment of the ADRS' and the target's right ascension of ascending node (RAAN phasing)
4. Re-Orbit maneuver or burn-phase to get to next target's orbit (if required)

The third phase may take a considerable amount of time, as the orbital planes of subsequent objects typically do not coincide. While the inclination of the considered objects is within a narrow band, the problems arise from the difference in the line of nodes (RAAN). The ADRS would have to perform costly out-of-plane maneuvers to rotate the orbital plane, or it could wait within the disposal orbit (e.g. 25-year lifetime orbit) until the lines of nodes match. The latter behaviour is investigated within this study.

The actual line-up of the target objects in a multi-target mission is an optimization problem. The first approach would be to deorbit the target requiring the highest amount of fuel, thus minimizing the total fuel mass. However, this could be not the best solution, as the required mission time to deorbit all targets may exceed accepted levels due to eventually increased RAAN phasing times between subsequent targets. Different scenarios are investigated in section 4.

3.1. Chemical Propulsion

The chemical propulsion (CP) system is advantageous due to its high reliability and technical maturity, as it is widely used to re- or deorbit satellites at their end-of-life. The system is modeled, assuming a specific impulse of 450 s for a liquid hydrogen/oxygen propellant, and an engine dry mass of 100 kg. The RVD operations are assumed to take up four orbits, requiring an additional ΔV of 30 m/s for each target. The targets are assumed to be deorbited by a single maneuver each, decreasing the perigee altitude according to a specified residual lifetime of 25 years (delayed deorbit).

The software tool also allows for the analysis of direct deorbit maneuvers for the CP system, even for a multi-target mission. In this scenario, the ADRS would perform the un-docking and the subsequent reboost maneuver directly after the deorbit maneuver has taken place, thereby raising its perigee to an altitude corresponding to the next target. However, in this study only a delayed deorbit scenario is looked at, where the ADRS performs the reboost maneuver during its next perigee pass, first increasing the apogee to the subsequent target's altitude. A further maneuver is performed to raise the perigee as soon as the ADRS reaches its apogee.

3.2. Electrical Propulsion

The active removal of objects by means of electrical propulsion (EP) is interesting due to the fact that the required propellant mass is considerably lower when compared to chemical propulsion. However, the increased power demands imply a mass penalty for additional solar arrays or power converters, a point which also has to be considered. The ADRS may use electrothermal, electromagnetic, or ion thrusters. This study focuses on ion thrusters, as they are mainly used in practical applications today. An example is the RIT-10, which has already been successfully operated in space during the EURECA and Artemis mission, respectively. [1]

The main advantage of electrical propulsion systems is the high specific impulse, which is as-

[1] `http://cs.astrium.eads.net/sp/spacecraft-propulsion/ion-propulsion/index.html`, July 2011

sumed to be 3300 s in this study, according to the RIT-10 thruster. The mass of the thruster is 2 kg. Due to the low thrust (15 mN) a direct deorbit is only possible for single-target mission, as the ADRS would have to spiral down into the denser layers of the atmosphere. Thus, this system may only provide delayed deorbit capabilities if multiple targets have to be removed. The duration of RVD operations is assumed to about six orbits.

3.3. Electrodynamical Tether

The major disadvantage of techniques using propellant, is the increasing total system mass for multiple targets, which may severely limit the allowed number of targets for one mission. By directly converting electrical energy into a retarding force, electrodynamical tethers do not require any propellant. The tether is a moving electrical conductor within the Earth's magnetic field, thus inducing a Lorentz force according to

$$\vec{F}_L = \left(J \cdot \vec{l}\right) \times \vec{B}, \tag{3}$$

where J is the electric current, $\vec{l}$ is the tether length, $\vec{B}$ the Earth's magnetic field and $\vec{F}_L$ is the Lorentz force. The maximum force is applied when the satellite is moving perpendicular to the magnetic field, that is in the geomagnetic equatorial plane. The Earth's magnetic field is modeled as a single magnetic dipole with its axis tilted by 11.5 degrees away from the Earth's rotation axis, according to [6].

The tether is assumed to be rigid, based on the radial alignment of the tether caused by the gravity gradient, which also provides a tensioning force. However, dynamical problems, which may for example arise during tether deployment, are neglected. Due to its length of several kilometers, the tether has a high cross-sectional area, which has a negative impact on collisional flux. Therefore, the tether has to be designed to provide a sufficiently high survival probability to achieve the mission goals. Several designs to increase the reliability and impact tolerance have been proposed, like the Terminator Tether™ [7], or a double line multi-loop tether [8]. Due to the high inclinations of the priority targets, the acting Lorentz force may be strongly reduced. However, it was shown in [6], that even objects at sun-synchronous orbits may be deorbited within an acceptable time frame.

It is further assumed in the model that the electrodynamical tether (EDT) system mass, including the tether as well as the deployment mechanism, power supply, and further support structures, makes up 1 % of the maximum target object mass, as estimated in [9]. The tether length is assumed to be 7.5 km providing an electric current of 0.3 A. Similar to the electrical propulsion technique, the ADRS spirals down until it reaches the desired graveyard orbit of the target. After undocking, the direction of the electric current is changed, thus inverting the Lorentz force and causing the ADRS to increase its altitude, heading for the next target.

4. ANALYSIS OF ADR SCENARIOS

For each deorbit technique defined in the previous section, several scenarios are considered. For each scenario the key parameters *system mass* and *total mission time* are estimated.

1. ADR of one priority target
2. ADR of multiple targets sorted by mass, starting with the heaviest object
3. Multiple targets with modified sorting criteria

In the first scenario one single target out of the top ranked objects is removed from its orbit to a 25-year graveyard orbit. The active removal of multiple objects within one single mission is considered in the second scenario. The object with the highest mass is removed first, as it most likely accounts for the highest amount of required fuel and deorbit time respectively. Additionally, a major advantage in case of an eventual mission abort would be in having already removed one or several objects with the highest mass from orbit. The third scenario is similar to the second one, with the difference being in the sorting of the target objects. As already said, the process of the orbital plane alignment (RAAN phasing), with respect to the subsequent target, is time-consuming and may exceed some well-defined limit when total mission time is considered. In this case, the objects may be deorbited in an alternative sequence in order to save time.

4.1. Simulation Assumptions

In the following, the three techniques, which are chemical propulsion (CP), electrical propulsion (EP) and electrodynamical tether (EDT), shall be evaluated based on the active removal of priority ranked Zenit-2 and Ariane H10 upper stages for the different scenarios defined in the previous section. The following assumptions are made:

- The dry mass of the ADRS is 800 kg.
- Deorbit Time is measured from apo- to perigee in case of CP and from initial to final orbit for EP and EDT.
- Final orbit of target corresponds to a 25-year residual lifetime orbit.
- Launch and early operations phase is omitted.
- ADR mission starting in May 2011, considering monthly predicted solar flux values [2] for residual lifetime estimations.

4.2. Scenario 1

Table 1 shows the results for single-target missions. For the Ariane H10 upper stage, using a CP system, the target is deorbited to a graveyard orbit within a few hours. As should be expected, EP and EDT both result in a significantly increased deorbit time of 4 and 2.5 months, respectively. However, the additional mass is only 6.6 kg for EP and 17.6 kg for EDT compared to 123.4 kg for CP when deorbiting the Ariane H10 upper stage. Another disadvantage of the CP may be the fact that the target satellite's apogee still remains within the critical region. Thus, the target will cross this region during its apogee passes for a few years.

		CP	EP	EDT
Ariane H10	m [kg]	923.4	806.6	817.6
(m = 1764 kg)	t [d]	0.3	115.6	75.7
Zenit-2 R/B	m [kg]	1070.0	847.0	890.0
(m = 9000 kg)	t [d]	0.3	1123.2	418.7

Table 1: *Scenario 1* - Delayed Deorbit (25-year graveyard orbit) for single-target mission

The deorbiting of the Zenit-2 rocket body is much more demanding in terms of additional mass and required deorbit time, mainly due to the target mass, which is approximately five times higher than for Ariane H10. For the CP system, 170 kg of fuel are required, while the deorbit time remains fairly the same. The major difference for the Zenit-2 R/B can be seen in the increased deorbit times for the EP and EDT system with 3 and 1.1 years, respectively. This is not only due to the increased target mass, as the Zenit-2 R/B in this example also has a higher initial semi-major axis and a higher ballistic parameter, resulting in a significantly increased altitude difference between initial and graveyard orbit. The design life of the RIT-10 ion thruster is specified with approximately 20,000 hours (≈833 days of continuous thrust). Thus, in this example, it would not be possible to deorbit the Zenit-2 R/B, by using the EP system. The use of a more powerful engine may be an option. For example, the RIT-XT[3], which has only been tested in several configurations so far, would reduce the total mission time to 113 days (about 10 % of the RIT-22 value) or 2712 hours of operation. However, any new engine would first have to demonstrate its reliability considering such a long thrusting phase. This study focuses on the RIT-10 engine, as it is already flight-proven.

4.3. Scenario 2

In this scenario multiple targets shall be removed from their orbits within one mission. The

[2] `http://celestrak.com/SpaceData/`

[3] `http://cs.astrium.eads.net/sp/spacecraft-propulsion/ion-propulsion/ion-thruster-rit-xt.html`, July 2011

major problem of such a scenario is to maneuver after having removed one target and heading for the next one. The orbital planes of subsequent targets are generally not aligned, so that costly out-of-plane maneuvers have to be performed in order to induce a secular precession of the orbital plane. Alternatively, the ADRS may wait in its initial orbit (identical to the graveyard orbit of the previously deorbited target), until the orbital planes match due to the different precession rates of the ADRS and the target's orbit planes caused by the Earth's oblateness. The latter option is simulated for all three techniques within this scenario.

The resulting mission time for the active removal of multiple objects is shown in Fig.3 for the top Zenit-2 rocket bodies. While a mission

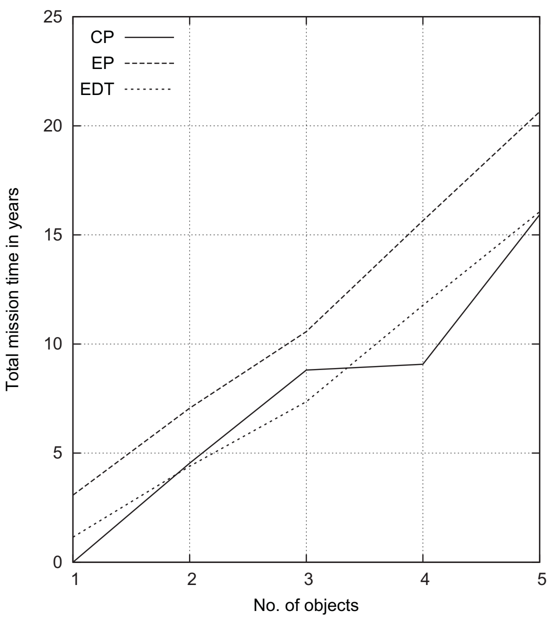

Figure 3: Total mission time for multiple-target ADR

deorbiting one single object clearly favours the CP system with respect to mission time, a multi-target mission strongly depends on the RAAN separation of subsequent targets. As a CP system is characterized by a negligible amount of deorbit time when compared to the other techniques, the total mission time mainly comprises of the phasing time for orbital plane matching. It can be seen, that an EDT system may conduct a multi-target mission within a shorter period of time in this specific line-up of the target objects, if two or three objects are removed from orbit. However, if the line-up is chosen with respect to the RAAN of subsequent targets, the total mission time for a CP system may be significantly reduced, as can be seen between the third and fourth target. In general, the EP and EDT system both require less time for RAAN phasing, as the difference in the secular precession of the RAAN is higher for a circular graveyard orbit when compared to an elliptical graveyard orbit, which still has its apogee at the initial altitude of the target satellite. For the simulation, which is shown in Fig.3, the difference in the secular rates of the RAAN is approximately $0.30°/\text{day}$ for the EDT and EP system, while it is only $0.15°/\text{day}$ for the elliptical graveyard orbit of the CP system. This means that a CP system needs about six days to compensate one degree in RAAN separation, while the EP and EDT system need about three days. The situation is even worse for sun-synchronous orbits with high inclinations. For the Ariane H10 upper stages, each in an orbit with an inclination about 98°, the differential precession of the line of nodes is about half the value as for the Zenit-2 rocket bodies ($i \approx 71°$), thus doubling the required phasing time. It would seem logical to line-up the targets with respect to the separation in the RAAN, as it is done in scenario 3. This, however, could result in not removing the top ranked objects first, thus reducing the effectiveness of such a mission. Therefore, a trade-off has to be performed, in order to reach the mission goals and also doing this within an acceptable time frame.

As already said, the ADRS does not have to solely rely on the natural nodal regression. With the help of out-of-plane maneuvers, this process may be accelerated to shorten the time-consuming phasing time, especially for a CP system. For an EP system the contributions of each mission phase to the total mission duration are shown in Fig.4. The major contribution to the total mission time is

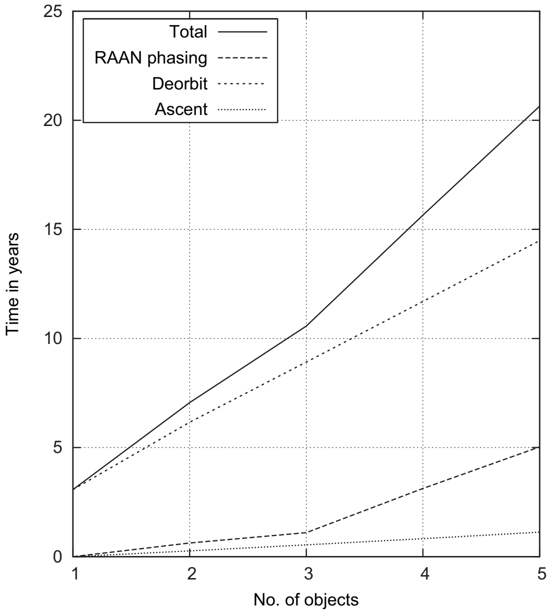

Figure 4: Breakdown of single contributions to total mission time for an EP system

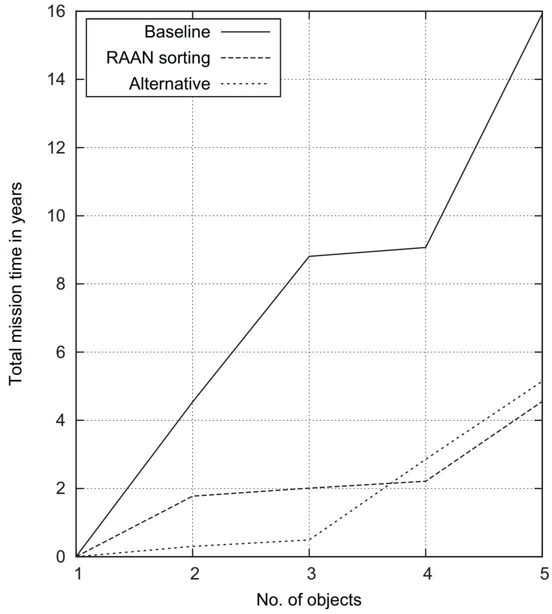

Figure 5: Impact on total mission time by different sorting for CP system

due to the descent or deorbit phase. Each Zenit-2 rocket body has a mass greater than 8,000 kg, which leads to about three years of required deorbit time for each target, compared to only about three months for the subsequent ascent phase. Deorbit maneuvers lasting for over three years, of course, can not be managed by the engine used within this analysis. However, more powerful engines or multiple engines of the same type would be able to reduce the mission time, thus enhancing the EP system performance.

4.4. Scenario 3

This scenario shall show, how different sorting of the priority objects may affect mission characteristics. The baseline scenario, which is used for comparison, is equal to the second scenario analysed in the previous section. In Fig.5, the impact on the total mission time for the CP system is shown, if the Zenit-2 targets are deorbited in a different order. The “RAAN sorting” scenario sorts the target objects according to their RAAN at mission start. In this scenario, the total mission time is reduced to 4.5 years, while there were about 16 years in the baseline scenario. A third option, which is shown as “Alternative” in Fig.5 shows, that basically the fourth and fifth target have an unfavorable RAAN, while the first three targets may be deorbited within six months. The last two targets may be replaced by two other targets which meet the RAAN criteria, in order to further reduce the total mission time. However, this would again result in not removing the priority targets first. An option could be to consecutively group four to five objects with appropriate RAAN values for multi-target missions and conduct these missions simultaneously. The total system mass is barely affected, with a difference of 3 kg between “Alternative“ and “Baseline“ scenario. This is due to the fact that all target objects are in similar orbits, with 3 km being the maximum difference in the semi-major axis for two objects.

For the EP system, Fig.6 shows, that a RAAN-based sorting is appropriate only for the CP system. For the EP system, the mission will be prolonged by five years, if the targets are sorted by RAAN. It can also be seen, that the ”Alternative” scenario with a different line-up does not shorten

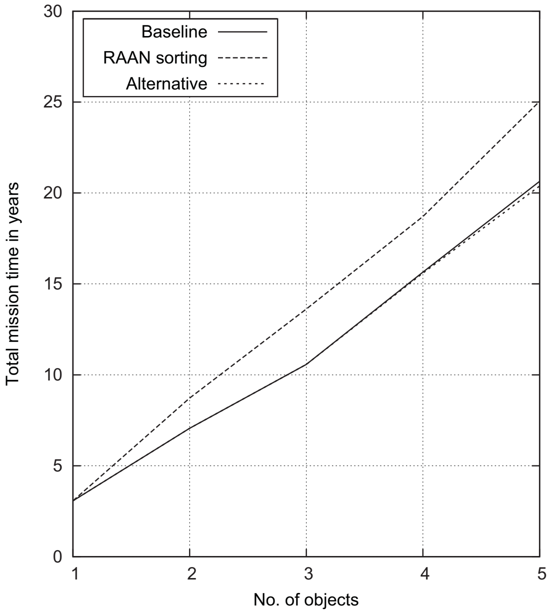

Figure 6: Impact on total mission time by different sorting for EP system

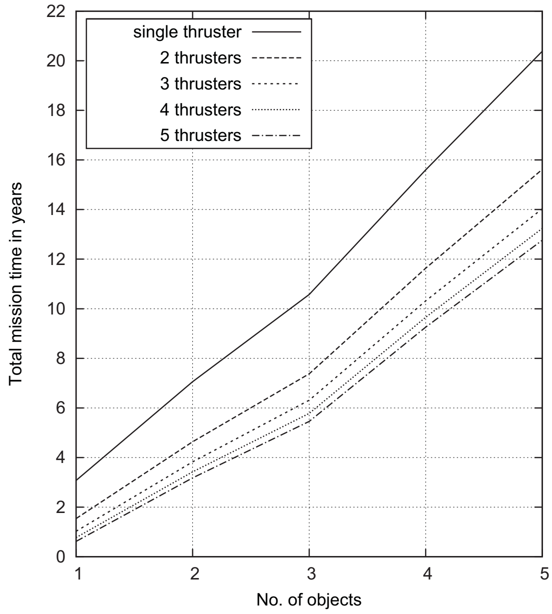

Figure 7: Variation of the number of ion thrusters in the EP system.

the mission significantly. Since the ascent and descent (deorbit) phase account for about 80 % of the total mission time, the optimization of the target object line-up with respect to the RAAN does not have the desired effect. In order to achieve better results for the total mission time of the EP system, one option would be to investigate the properties of other ion thrusters. Another option is to use multiple thrusters of the same type. This is shown in Fig.7 for up to five thrusters. It can be seen, that the required time for a five-target-mission can be reduced by more than four years, if a second thruster is used. Adding more thrusters will lead to significantly decreasing required time for the ascent and descent phases, but simultaneously increasing the RAAN phasing time. Therefore, simply adding further engines will only solve the problem of reducing ascent and descent phases, while the problem of aligning orbital planes still has to be addressed.

The results for the EDT system are shown in Fig.8. Similar to the EP system, sorting the objects by RAAN has no positive effect. Due to the required deorbit time of 13-14 months per object, the secular change in the RAAN is about 80° for each object[4]. The properties of the secular RAAN change have been considered in determining the alternatives 1 and 2 in Fig.8. The total mission time for deorbiting five objects is reduced by 2-3 years. In Fig.8, it can also be seen, that especially the last two objects require the highest amount of RAAN phasing time, as ascent and descent phases are similar for each target object with respect to required time. As for the CP system, an optimization may be to replace those targets by more appropriate ones. However, in order to significantly reduce the total mission time, other tether configurations have to be investigated, as ascent and descent phase in this scenario make up about 80 % of the total mission time, similar to the EP system.

5. CONCLUSION

The active removal of space debris is proposed by several authors in order to stabilise the future

[4] $\approx 2°$per day

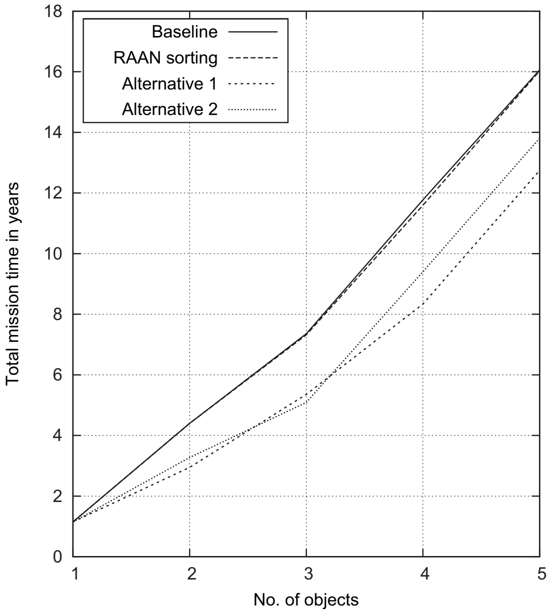

Figure 8: Impact on total mission time by different sorting for EDT system

environment in LEO. In this study three different techniques have been investigated, trying to quantify critical mission parameters as total mission time and system mass. For this purpose, a software tool has been developed, which allows ADR analysis of this type. Two different, general concepts for an ADR mission have been investigated: single-target missions, where only one target object is removed from orbit, and multi-target missions, deorbiting multiple objects within one mission. However, the first step of ADR is to identify those objects, which pose the highest threat to the current LEO environment, which was the first part of the study. The results show, that the definition of an alternative ranking criteria based on object flux and mass leads to similar results, with respect to the top ranked objects, when compared to previous studies ([1]). The majority of the top-ranked objects is found in densely populated orbit regions in LEO, particularly in sun-synchronous altitudes.

Three scenarios have been looked at. In the first scenario, a single-target mission was conducted with a CP, EP and EDT system, respectively. For the CP system, an ADR mission may be performed within a short time-frame, compared to more than one year for deorbiting single objects with the EP and EDT systems. However, the CP system has the disadvantage of significantly higher mass needs. The second and third scenario both introduce the idea of deorbiting multiple objects within one mission. It has been shown that the greatest problem is to align the orbital planes of the ADR spacecraft and the subsequent target, after having deorbited one target in another orbital plane. Only the effect of the natural secular precession of the line of nodes due to the Earth's oblateness has been looked at in this study. The total mission time can be considerably reduced by lining up the target objects with respect to the mutual orientation of subsequent target's orbital planes. However, there is no uniform distribution in RAAN for the top-ranked objects, so that there will not be many possible missions, where the targets are ideally positioned. Moreover, the effectiveness of ADR missions will be highly reduced, if the top objects are not removed within the first missions [3]. Possible solutions to further reduce the required mission time, e.g. out-of-plane maneuvers, shall be the subject of further studies.

This study is only one step towards possible ADR missions. Many problems still have to be solved, apart from the difficulties related to orbital mechanics. Besides the development of guidelines and regulations to overcome legal issues, also cost analyses have to be performed. However, the active removal of space debris will be inevitable in order to preserve the LEO environment for present and future satellites.

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