Debris Mitigation by Collision Avoidance after End-of-Mission – assessment of a disposal solution for LEO satellites

Jan-Christian Meyer(1), Daniel Laugwitz(2)

(1) OHB System AG, Universitätsallee 27-29, 28359 Bremen, Germany, Email: jan-christian.meyer@ohb.de
(2) Universität Bremen, Bibliotheksstraße 1, 28359 Bremen, Email:daniel.laugwitz@ohb.de

ABSTRACT

This paper proposes an alternative end-of-life strategy for satellites and investigates the advantages and drawbacks with respect to a classical approach.

The proposed strategy keeps the satellite operational after the end-of-mission. Positive consequences of this strategy are:

- Keeping collision avoidance manoeuvre capability until re-entry
- Avoiding expensive technology developments required for complete passivation as currently demanded by space debris mitigation requirements
- Reducing the risk of creating new and very dangerous space debris clouds

The underlying assumption is that the debris population is majorly impacted by in-orbit collisions. Collisions produce large number of debris objects within a single event. In addition, naturally collisions will more likely occur in congested orbital regions. They are typically congested because they are of high interest for mission designers. Consequently any debris produced here poses a higher risk of making operational orbits unusable. Avoiding collisions is therefore the best way to reduce an increase of the space debris population. A passivated satellite cannot perform collision avoidance manoeuvres.

The work presented elaborates the risk of debris created from explosions against debris created from collisions. Cost of operations are factored in as well as impacts on the satellite system design. Furthermore, the compliance to space debris mitigation requirements is assessed and the consequences of this approach to the mission stakeholders are highlighted. The paper presents the status-quo at time of paper submission and highlights which tasks are still left to be done.

1 INTRODUCTION

Space Debris Mitigation (SDM) requirements have been introduced by national and international organisations in order to make spaceflight a sustainable business and avoid congestion of orbital regions with space debris. The question how to best limit the future debris population in terms of number and criticality has been at the basis of discussions leading to the formulation of SDM requirements. This process has been started once the population of objects in orbit became more pronounced. First recommendations were given as early as 1978 [1].

The development of the international standard ISO 24113 [4] can be considered a major achievement for the international spaceflight community. At the same time this standard represents a compromise between many options for sustainable use of space that could be agreed upon between the international partners. It has been made applicable to all European missions through the publication of [2]. Without violating the ISO 24113 requirements this paper proposes and alternative approach to end-of-life (EOL) operations. The basic assumption behind the proposed strategy is that in the future avoiding collisions will be of highest importance to limit the growth of the space debris population. This claim is justified in the following chapters leading to the development of the alternative EOL strategy before giving an outline of which work still remains to be done.

2 LEO SPACE DEBRIS ENVIRONMENT EVOLUTION

The population of space debris objects can be analysed with the MASTER2009 tool. It takes into account observation data obtained from optical and radar measurement campaigns for large enough objects. And reproduces the smaller size population through the modelling of the sources of debris creation. [5] Thus, it can be used to derive information about the debris population in the past. And since knowledge about the past has been used to extrapolate the future debris population with Monte-Carlo simulations, the MASTER2009 model is the standard European model for the development of the debris environment in the future.

The focus of this paper is the distinction between collision and explosion fragments therefore the analyses presented in the following sections are only showing those two sources. Nonetheless, the lines for total number of objects or spatial density include all other sources. Their number can be neglected for the presented assessments.

2.1 Past evolution

The evolution of debris particles created from explosions (red) and collisions (green) in the past is shown in Figure 1. The graph shows the spatial density in the LEO region between 200 km and 2000 km altitude. Particles between 1 cm and 10 m are included. These input parameters are used for all following MASTER2009 plots if not otherwise specified.

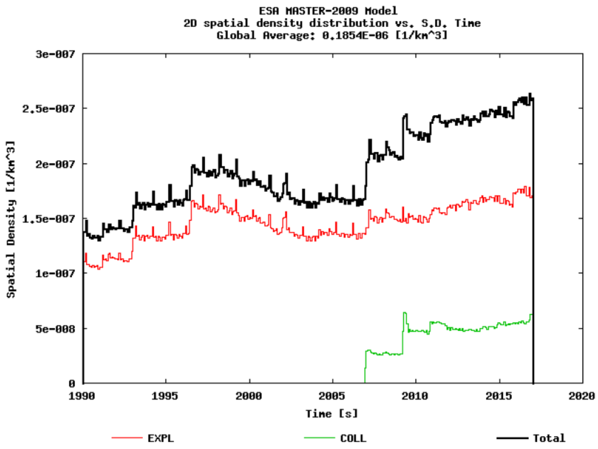

Figure 1: Past evolution of collision and explosion fragments (Source: MASTER2009)

It can be seen that there are two major collision events in the years 2007 and 2009. They correspond to the destruction of Fengyun-1C and the Cosmos-Iridium collision respectively. Today the population of collision fragments is about 82000. As can be seen from the graph these are almost entirely due to those two events.

Explosions have created notable spikes at two instances in the 1990s. Other spikes seen in the red curve quickly fade back into a more steady growth of the population. The population of explosion fragments larger than 1 cm today is about 216000.

2.2 Future evolution

The MASTER2009 future population is shown in Figure 2. The propagation is based on a business-as-usual scenario of future SDM compliance. Considering the timeframe from today until 2050 two major trends are clearly visible:

1. The number of collision fragments rises to 360000.
2. The number of explosion fragments stays globally constant with a variation related to the solar cycle.

Before 2040 the collision fragment population has reached the same order of magnitude as the explosion fragments.

The increase of collision fragments can be attributed to an increase of collisions. This is the self-accelerating effect known as Kessler syndrome.

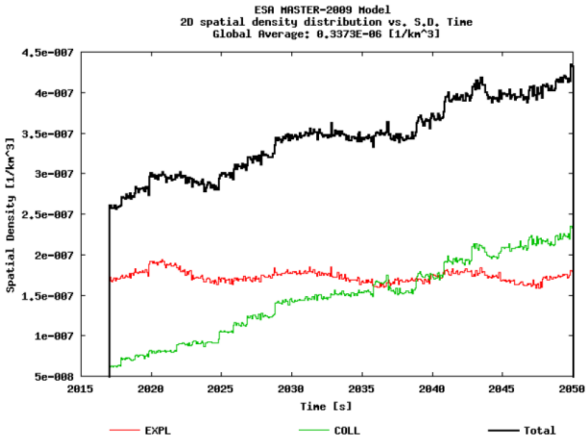

Figure 2: Future evolution of collision and explosion fragments (Source: MASTER2009)

3 CRITICALITY OF DEBRIS FRAGMENTS

A collision is more likely to occur in a more crowded orbital region. Crowded orbital regions are those that are of high interest to satellite operators for various reasons. In LEO this is particularly the case for the 800 km altitude range.

Figure 3 and Figure 4 show the distribution of collision and explosion fragments over the LEO altitudes in 2017 and 2050 respectively. The global increase highlighted above can be seen from the growth of the total curve between the two graphs. (Note the same y-axis scale). The distribution of explosion fragments remains nearly unchanged between the two points in time. There are three distinct peaks visible at altitudes of 800 km, 1500 km and 1700 km. Collision fragments – in 2017 less dominant than in 2050 – only show one peak roughly ranging from 700 km to 1000 km.

The main consequence of this concentration is that by 2050 the spatial density will have doubled in this critical orbital region while it remains at roughly constant value at higher altitudes.

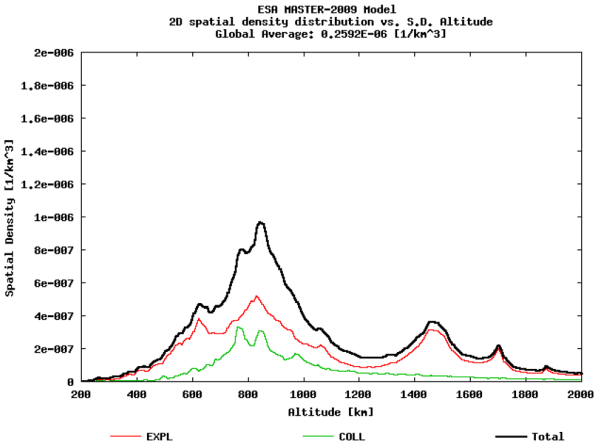

Figure 3: Space debris altitude distribution in 2017 (Source: MASTER2009)

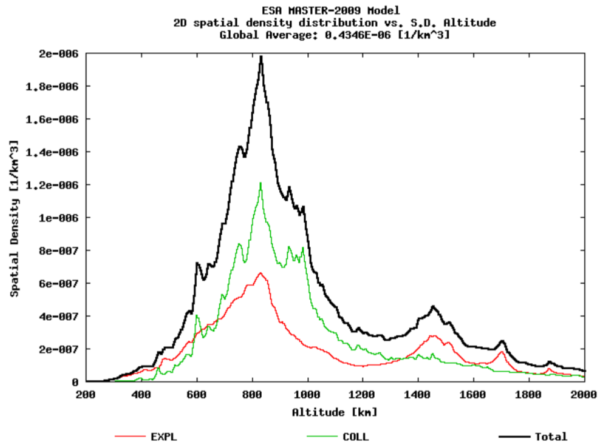

Figure 4: Space debris altitude distribution in 2050 (Source: MASTER2009)

Due to the different types of LEO missions there are huge variations in the population of space debris objects and operational satellites which are both strongly correlated.

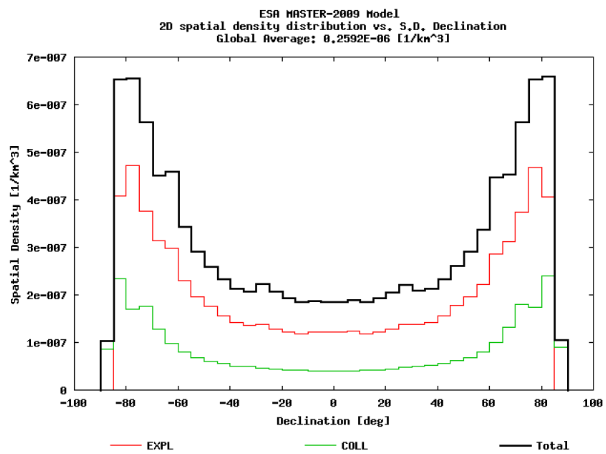

Figure 5: Space debris declination distribution in 2017 (Source: MASTER2009)

To identify the most crowded areas in the LEO regime, it is useful to first take a look at the spatial density variation for different declinations.

Figure 5 shows the distribution of objects over the declination from -90° to +90°. It illustrates that there are strong peaks in spatial density at declinations around ±90°. Taking a look at the current population of operational satellites in relation to the orbit inclination confirms the before mentioned correlation. Each operational satellite is marked by a + in Figure 6 showing its altitude / inclination combination.

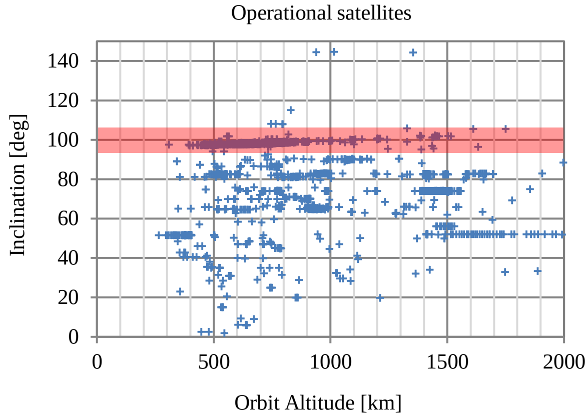

Figure 6: Operational satellites population regarding inclination and orbit altitude (Source: https://www.space-track.org)

The strongly populated regions above the polar areas can be described by the high use of sun-synchronous orbits (95°-105° inclination) within the LEO regime (nearly 30% of LEO-Satellites) which results in a greater number of collisions and thus in a higher space debris population.

The most critical orbit can now be determined by looking at the corresponding Annual Collision Probability Level which can be calculated with DRAMA-2.0.

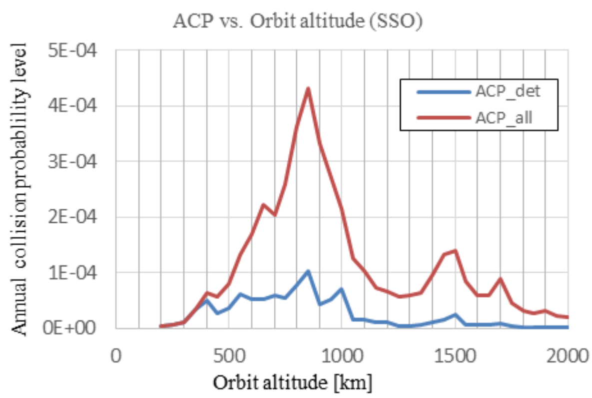

Figure 7: Annual Collision Probability Level for different orbit altitudes for SSO (Source: DRAMA-2.0)

Figure 7 shows the annual collision probability per orbital altitude for the whole population of space debris (red line) and the detectable population (blue). As it is shown in Figure 7, the highest risk of collision for an operational satellite occurs in an altitude region around 850 km with a collision probability above 0,04% per year.

In the past there have been satellite break-ups caused by battery explosions. These can mostly be traced back to the use of old NiCd battery technology. Today's satellite batteries are almost exclusively based on Li-ion technology. Although more robust, modern Li-ion technology is not safe from the risk of explosions. The question to be answered by testing is under which conditions a battery can be considered safe. According to current best knowledge the safe conditions for modern batteries are far more relaxed than those for ones with old technology. Consequently, it can be expected that the number of battery related explosions of satellites will decrease in the future. This effect is not captured in the MASTER2009 environmental model.

Also, the current trend of mega-constellations started by the New Space development is not considered in the current model. It will considerably increase the collision risk in LEO.

4 DESCRIPTION OF ALTERNATIVE EOL STRATEGY

In this chapter the new mission concept is described and it is highlighted how this can be in line with ESA SDM requirements. Finally, the parameter space governing the de-orbit strategy is introduced.

4.1 Mission and disposal concept

Based on the premise described above that being able to avoid collisions will in the future be of vital importance to the space debris environment, an alternative strategy for satellites after their end-of-mission has been defined. In essence the satellite passivation is skipped or postponed to be able to perform collision avoidance manoeuvres (CAM) also during the deorbit phase.

The concept is only applicable to satellites able to perform an uncontrolled re-entry, i.e. the ones with sufficient demisability to not violate the on-ground casualty risk requirement. For satellites that are naturally not compliant with the 25-year rule of leaving the LEO protected region at least one manoeuvre is necessary to move into a compliant orbit.

Once in a 25-year-compliant orbit the satellite will naturally de-orbit due to atmospheric drag. In this phase the satellite is operated in a low complexity survival mode to be able to perform a collision avoidance manoeuvre in case it is required. At the end of the natural decay phase the satellite will re-enter into the atmosphere and demise.

4.2 Application of SDM requirements

ESA space debris mitigation requirements demand that "during the disposal phase, a spacecraft […] shall permanently deplete or make safe all remaining on-board sources of stored energy […]." This requirement is commonly known as passivation requirement. By definition the disposal phase starts after the satellite's end of mission. [4]

In order for the alternative EOL strategy to be compliant with these regulations it is necessary to declare the "collision avoidance phase" as part of the nominal mission. This way passivation is not required to be performed before this phase is completed. The introduction of this phase into the nominal mission is shown in Figure 8.

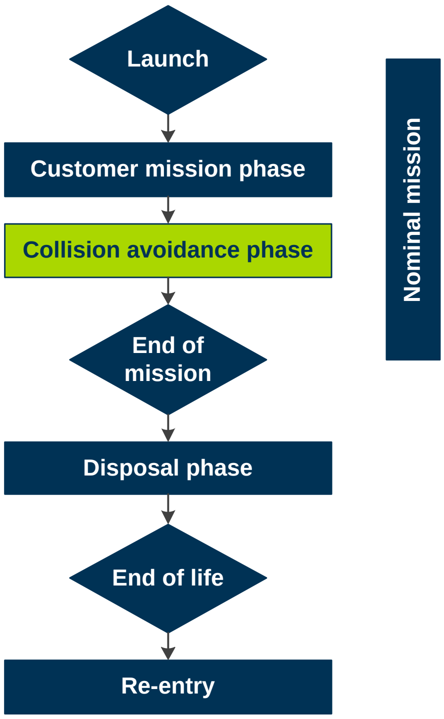

Figure 8: Mission lifecycle with alternative EOL strategy

Of course, system reliability degrades over time. The risk of accidental break-up is limited to 10^{-3} by [4]. The retardation or complete omission of passivation will challenge this requirement. Therefore, the question whether and when a passivation will need to be performed has to be answered by a detailed trade-off (cf. section 7.3).

It is challenged whether the limitation of accidental break-up (explicitly excluding external sources such as debris impacts [4]) is the best way of specifying a requirement on the risk of debris creation for the future. The collision risk is only indirectly considered through the 25-year rule. Instead, it is proposed to define a requirement on the statistical number of debris created.

This statistical value is a better measure for the risk associated with a mission. It will depend on models for break-up or collisions to identify the number of particles created in case of an event. Yet, also the probability of accidental break-up currently specified cannot be quantitatively measured and is only a statistical value.

4.3 Parameter space

In terms of identifying the most appropriate way of realizing the alternative EOL strategy there are several parameters that can be varied.

The most important parameters are:

(1) The altitude of the destination orbit (PMD-orbit) the satellite has to be transferred to
(2) The manoeuvre type for the transfer from mission orbit to PMD-orbit
(3) The propulsion type as basis for the transfer manoeuvre
(4) The Accepted Collision Probability Level (ACPL) as the tolerated risk threshold
(5) The collision avoidance manoeuvre strategy (long-term/short-term)

Variations of these parameters have more or less strong impacts on the entire space mission planning which will be specified in the following sections.

(1) The altitude of the destination orbit is the largest driver for the required Δv-budget.

(2) The transfer type mainly affects the resulting collision probability as well as a change in required Δv/propellant mass. The variation can go up to nearly 50% of additional Δv (depends on the considered PMD-Orbit) that's required for the transfer.

The regarded transfer types are:

- 1-Manoeuvre Hohmann-Transfer (circular to elliptic)
- 2-Manoeuvre Hohmann-Transfer (circular to circular)
- Low-Thrust Transfer (circular to circular)

(3) Due to the great variations in efficiency of different propulsion types this parameter primarily affects the required propellant mass. Furthermore it is to mention that all propulsion systems deliver different thrust which again impacts the applicable transfer type, the resulting transfer time and thus the collision probability.

For the considered propulsion systems check chapter 5.3.

(4) The Accepted Collision Probability Level (ACPL) describes the threshold for tolerated collision probability value and influences different aspects. Firstly it affects the frequency of collision avoidance manoeuvres because a lower collision probability threshold is violated more often than more tolerant one. A lower threshold again leads to a higher required Δv, but also to higher reduction of the remaining risk for the satellite to collide with space debris.

Table 1: Considered values for ACPL

No.	ACPL
1	1E-06
2	1E-05
3	5E-05

(5) The collision avoidance manoeuvre strategy describes how spontaneously the manoeuvres are performed, which is defined by the number of orbit revolutions before the predicted collision event. For the short-term strategy the CA-Manoeuvre is performed in the same revolution as the predicted collision. The long-term strategy provides the manoeuvre to be performed several revolutions before the collision and thus much less spontaneously. The CA-Strategy mainly influences the Δv required for the CA-Manoeuvres, where the Δv-needs increase with increasing spontaneity of the manoeuvres.

Table 2: Considered collision avoidance manoeuvre strategies

No.	CA-Manoeuvre strat.	Revolutions before pred. collision
1	short-term	0
2	long-term	5
3	long-term	10

5 METHODOLOGY

The methodology followed throughout the work and foreseen for future steps is described in this chapter.

5.1 Work flow and current status

The general idea for the strategy presented in this paper has been created in the first phase of ESA's CleanSat study performed by OHB in the year 2015. The work flow currently on-going is depicted in Figure 9.

Currently, the parameter space has been identified and the framework for assessing collision risks has been set up. Section 5.2 provides a detailed description of how the collision probability is assessed. First results are also obtained for the classical approach and some boundary-conditions alternative concepts. They are presented in chapter 6.

The assessments are done on the basis of a reference satellite mission and design. Its features are described in section 5.3.

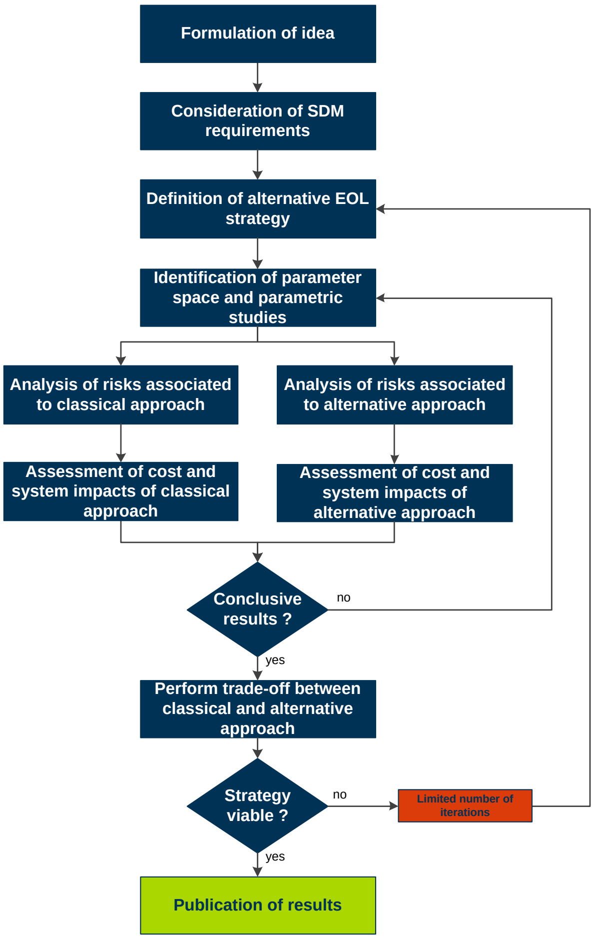

Figure 9: Methodology

5.2 Collision probability assessment

For further analysis and evaluating aspects it is important to determine how likely the satellite is going collide with a space debris object within the entire de-orbit process. This process includes the transfer from the mission orbit to the PMD-Orbit, the following decay process and ends with the re-entry.

Therefore, it is necessary to know the chronological progress of the whole trajectory which can be determined with OSCAR (DRAMA-2.0) and, for specific transfer types, with several analytical equations. Due to the fact that this method calculates several thousand values for a certain de-orbit process (one value per orbit revolution), these values have to be reduced before they can be used for collision probability assessment.

This means that the continuous trajectory (as exemplarily illustrated in Figure 10) has to be discretized with an applicable tool (e.g. Excel-macro) into a certain amount of altitude sectors with the specific Δt-value (in years) the satellite spends within each sector. For each altitude sector the Annual Collision Probability (ACP) can be calculated with ARES (DRAMA-2.0) and afterwards be multiplied with the Δt-value referring to the time spent in the sector to obtain the value for the probability of a collision within this sector. After repeating this step for each sector, these values can be summed up to obtain the collision probability for the whole de-orbit process.

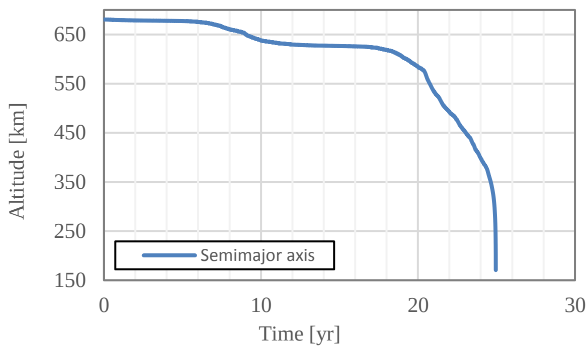

Figure 10: Semimajor axis for a natural decay process from 680 km (Source: DRAMA-2.0)

ARES delivers two different values for the ACP (ACP_w and ACP_d). The ACP_w value relates to the whole debris population including those objects that cannot be detected because of being too small for today's detection technologies. This value has to be taken into account for any calculation regarding the actual collision probabilities. The ACP_d value just includes the detectable part of the debris population and has to be considered for collision avoidance aspects.

Table 3: Altitude sectors and certain ACP_w values for a natural decay process from 680 km (25 yr duration)

h [km]	ΔT [yr]	ACP_w	CP
680	10,7077	2,14E-03	2,29E-02
634	9,1225	2,35E-03	2,14E-02
588	1,0513	1,72E-03	1,81E-03
542	1,0404	1,51E-03	1,57E-03
495	1,1499	7,75E-04	8,91E-04
449	0,8487	4,92E-04	4,17E-04
403	0,6598	8,30E-04	5,48E-04
356	0,2628	3,99E-04	1,06E-04
310	0,0821	1,69E-04	1,38E-05
264	0,0246	1,14E-04	2,18E-06
217	0,0055	3,10E-05	2,54E-07
Total from 680 km to 217 km			**4,97E-02**

Table 3 exemplarily shows the before mentioned altitude sectors with the corresponding ACP_w values and the resulting collision probability of ~5% for this natural decay process from 680 km.

5.3 Reference mission scenario

To have the same basis for all calculations regarding the classical and the alternative EOL strategy it is necessary to decide for a reference scenario first.

The reference scenario includes a satellite which is located on a specific mission orbit which marks the starting point for the PMD analysis. The goal is to consider a scenario that is as representative as possible regarding the LEO-missions that are currently in space. For that purpose the satellite databases of the Joint Space Operations Centre and Union of Concerned Scientists [6] were used to analyse what kind of satellite and mission parameters are the most representative ones for the analysis.

Based on this information the reference mission parameters were chosen as shown in Table 4.

Table 4: Reference scenario parameters

	Parameter	Value
Satellite	Dry mass [kg]	1100
	Power [W]	1700
	Dimensions satellite bus [m]	3,5 × 2 × 2
	Dimensions solar panels [m]	1,8 × 4
Orbit	Orbit	SSO
	Orbit altitude [km]	850 ($a = 7228\ km$)
	Inclination [°]	98,8
	Eccentricity	0,001 (~ circular)

This data is loosely based on the Sentinel-2 Earth observation mission in an 795 km sun-synchronous orbit with an inclination of 98,6°. Table 5 shows further parameters for Sentinel-2.

Table 5: Sentinel-2 parameters (Sources: https://directory.eoportal.org; www.wikipedia.org)

Parameter	Value
Dry mass [kg]	1120
Mass [kg]	1200
Power [W] (BOL/EOL)	2300/1730
Dimensions satellite bus [m]	3,4 × 1,8 × 2,35
Dimensions solar panels [m]	1,8 × 4

For collision probability and (de-orbit-) trajectory calculations it is necessary to determine the cross-sectional area of the satellites geometry which can be calculated with CROC (DRAMA-2.0). Here a rough model of the satellite has to be implemented which is shown in Figure 11.

CROC allows you to determine the value of the cross-sectional for different satellite movements (rotation around two axes with specified angles π and θ) and from different points of view (1, 2 and 3).

The two important geometric parameters are the maximum cross-sectional area ($A_{max} = 12{,}4\ m^2$) which has to be used for collision probability calculations and the mean cross-sectional area ($A_{mean} = 12{,}1\ m^2$) for de-orbit/decay duration purposes.

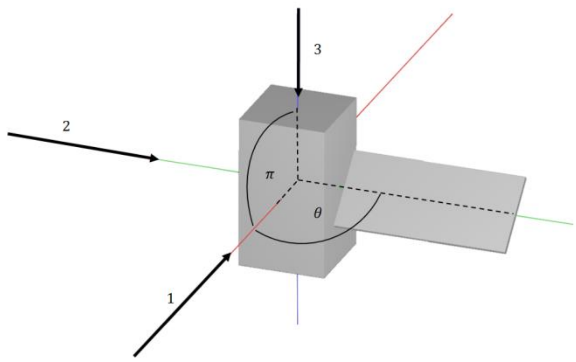

Figure 11: CROC-Model of the reference satellite

For calculations regarding transfer times and propellant masses there have to be identified different kinds of reference engines which preferably should cover a big magnitude of deliverable thrust and specific impulse.

Table 6: Reference propulsion engines (Sources: www.space-propulsion.com; www.rocket.com)

Prop. type	Reference engine	I_{sp}	F [N]	P [W]
Monopropellant	Airbus Safran 20N	222	7,9	-
Arcjet	IRS ATOS	480	0,115	750
Ion thruster	Airbus Safran RIT 10	3160	0,01	340
Ion thruster	Airbus Safran RIT 22	4750	0,175	5900

The Airbus Safran RIT 22 engine is just considered for theoretical purposes, because due to its required power budget of nearly 6 kW it would not be applicable for the regarded satellite.

The nominal mission duration was set to $T_{mission} = 5\ yr$, which has to be taken into account for reliability/qualification analysis.

6 PRELIMINARY RESULTS

To this point the preliminary results refer to collision risk and collision avoidance aspects only. Cost and reliability/qualification aspects are not considered so far (cf. chapter 7).

Table 7: Results for the classic EOL strategy regarding a 25-year de-orbit duration

Transfer	Prop. type	Δv [m/s]	$T_{transfer}$	CP_{PMD}
1-Man. HT*	Safran20N	72,46	-	5,99 %
2-Man. HT	Safran20N	88,42	~50 min	4,97 %
LT-Transfer	ATOS	88,55	9,87 d	4,98 %
	RIT 10	88,55	112,84 d	5,08 %
	RIT 22	88,55	6,47 d	4,98 %

*The 1-Manoeuvre de-orbit variant can't be split up into a transfer and a natural decay phase

As basis of the analysis regarding the alternative EOL strategy the collision probabilities, transfer times and the corresponding Δv-requirements for the transfer types mentioned in section 4.3 (2) were determined for the classic EOL strategy first. Table 7 shows the results of the analysis.

Due to the extremely low transfer time for a 2-Manoeuvre Hohmann-Transfer this option is at the current level not considered in the same level of detail as the other options.

The collision probabilities for the entire PMD were determined by adding the collision probability for the transfer phase to the collision probability for the natural decay phase (except for the 1-Man. HT).

In order to obtain a first basis for assessment and for the purpose of direct comparison all before mentioned parameters were calculated for several PMD durations regarding the considered manoeuvre types as Figure 13 shows. It illustrates the effects of choosing a specific PMD duration: A shorter duration leads to a lower collision probability but also to a higher Δv demand. Since the alternative EOL strategy proposes to operate the satellite until its re-entry, rather short PMD durations might be preferred.

The blue dotted lines connect the corresponding data points for each PMD duration.

This diagram is used to identify useful manoeuvre variations for more detailed calculations, especially regarding collision avoidance aspects. To gain a first impression which collision avoidance manoeuvre frequencies occur for certain PMD durations, these more detailed parameters were firstly calculated for three cases (25-, 10- and 1-year PMD duration). The 25- and the 1-year duration mark the maximum and the minimum cases and thus represent the range all values will be in-between.

Figure 12 shows the resulting CA-Manoeuvre frequencies for the three considered ACPL values regarding the 1-Manoeuvre Hohmann-Transfer and the LT-Transfer. In this case the LT-Transfer is calculated regarding the RIT 10 engine as it causes the longest transfer time.

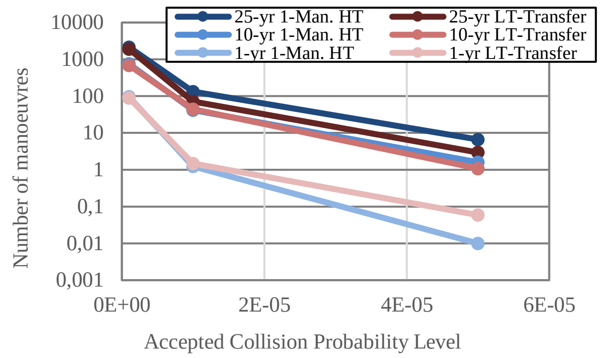

Figure 12: CA-Manoeuvre frequency for different ACPL values and PMD durations (Source: DRAMA-2.0)

For reason that the ACPL values 2 and 3 (see Table 1) result in a quite low CA-Manoeuvre frequency, the following considerations just take into account the ACPL 1 value.

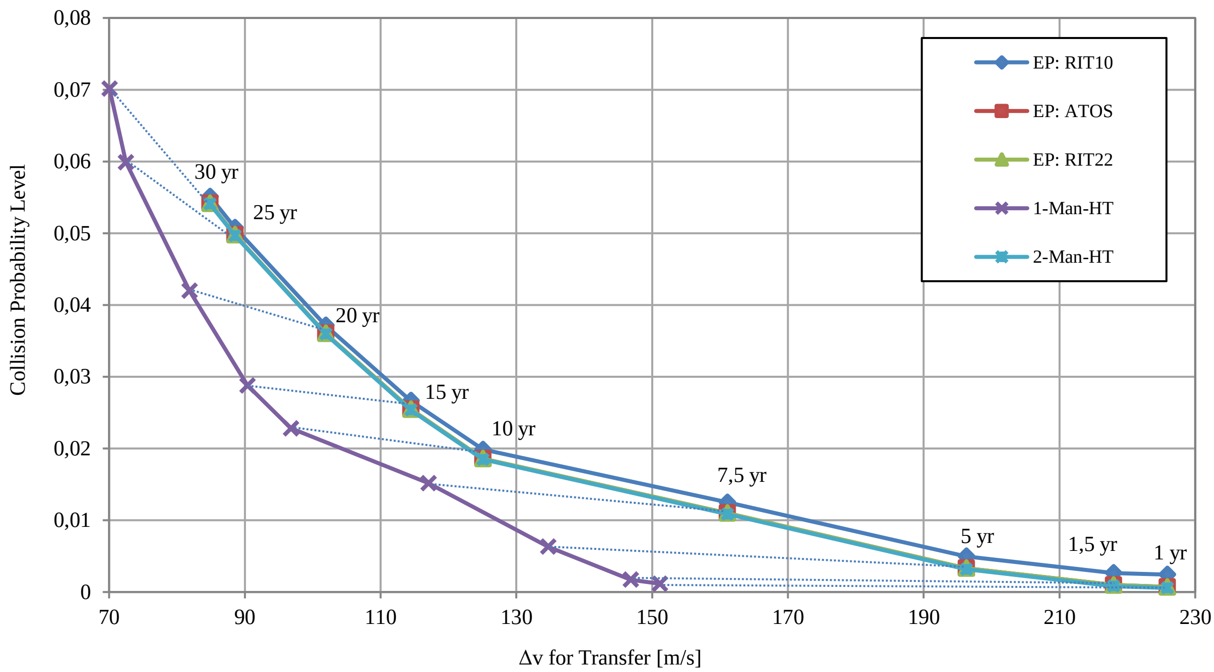

Figure 13: Collision probability and Δv-requirements for different PMD durations and manoeuvre types (Source: DRAMA-2.0)

As Figure 13 shows, the 1-Manoeuvre Hohmann-Transfer requires less Δv for a transfer than the circular-to-circular transfer types do. But at least it has to be considered that the monopropellant engine, that's required for the 1-Man. HT, has a much lower specific impulse than the electric propulsion engines. Figure 14 illustrates that this results in a much higher propellant mass.

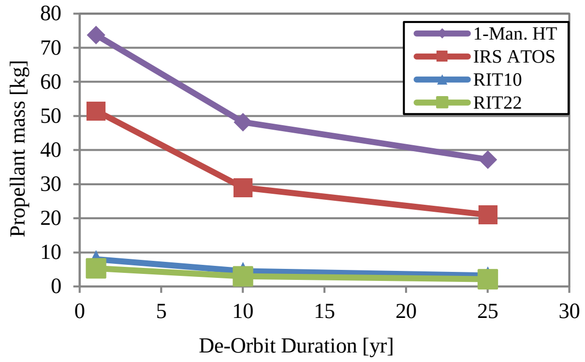

Figure 14: Propellant mass for different propulsion types

Another aspect that could be regarded for evaluation is the reduction of the collision risk and thus the remaining risk (residual risk) which is reached by performing a CAM. The residual risk refers to the detectable population and the remaining risk to the whole debris population (see DRAMA-2.0 Final report). Two parameter ratios can be used to compare the different concepts. One is the relation between the risk reduction and the collision probability (CP), the other one the relation between the risk reduction and the propellant mass. The risk reduction describes that part of the collision probability that can be reduced by performing CA-Manoeuvre which can at most be equal to the ACP_d value. The non-detectable part of the debris population, which is considered by the ACP_w value, can thus not be compensated entirely. For a more detailed definition of the risk reduction see DRAMA-2.0 Final report [3].

The first of the mentioned coefficients can be interpreted as the collision risk reduction efficiency, which describes the ratio of the compensated collision risk and the collision probability. The second coefficient may be described as the propellant cost for a given reduction of the collision probability.

Based on this, the diagram in Figure 15 compares these relations.

The data points in Figure 15 represent for each curve the following PMD durations seen from left to right: 1-, 10- and 25-year.

Figure 15 allows evaluating the advantages or disadvantages of the manoeuvre and also propulsion types regarding their efficiency in reducing risk. As it gets clear the RIT 22 engine delivers the best conditions regarding either for propellant mass or risk reduction purposes. Leaving that (theoretical) type out, the other manoeuvre types offer different advantages depending on the desired PMD duration.

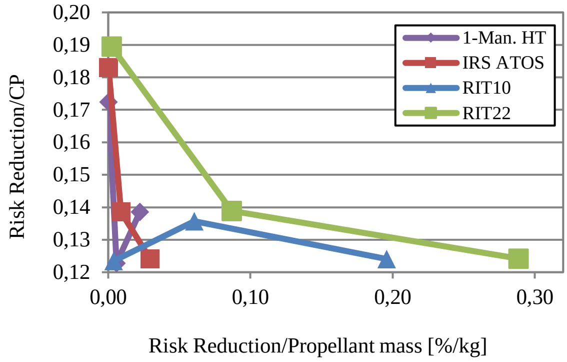

Figure 15: Comparison of the risk reduction by collision probability against risk reduction by propellant mass

For a 1-year PMD duration all options provide nearly the same value of nearly 0.00 for the ratio of risk reduction and propellant mass. For that case they can be just differentiated in terms of risk reduction efficiency, where the 1-Man. HT and the Arcjet transfer (LT-Transfer) are the most advantageous options. For 10 years of de-orbit duration it is just the RIT 10 engine which provides an advantage regarding the required propellant mass. Looking at the risk reduction efficiency there are no remarkable differences between all variants. The 25-year PMD duration provides nearly the same results, where the RIT 10 engine again delivers the best conditions in terms of required propellant mass. Looking at the relation of risk reduction and collision probability there is no manoeuver type that provides considerable advantages.

7 FUTURE WORK

The major part of the work associated with the new EOL strategy is still to be done. Preliminary results have been presented in the previous chapter. With the iterative process described in section 5.1 the concept will be further refined. In the end, the advantages and drawbacks will be compared to the classical approach to derive which mission scenarios provide overall the best solution.

7.1 Risk assessment

The alternative EOL strategy proposed and still investigated is not without risk. The following risks will be considered and carefully compared against those included in the classical disposal strategy:

- CAMs can only be performed for detectable objects. The collision risk with undetected particles is present also for the new strategy.
- With an extended mission duration the satellite's reliability will decrease. This bears

the risk of loss of control prior to passivation (if foreseen).

7.2 Cost estimation

Besides improvements in terms of debris creation and associated risks the assessment of the alternative EOL strategy will also consider cost impacts. The following cost effects will be considered:

- Extending operations will require a ground team and ground stations to be run for the prolonged time. Even though this might not be a full-time operation certain resources will be needed and accounted for.
- Retaining fuel to perform CAMs could result in extra costs on the propulsion subsystem beyond just the amount of fuel. A larger tank is the simplest example here.
- Avoiding passivation on the other hand can have positive cost impacts on the system if components such as passivation valves are not needed any more.
- If redundancy or reliability of the system have to be increased to mitigate the risks mentioned above this will have a cost impact.

7.3 Trade-off

To determine whether the new strategy globally provides advantages to satellite operators, manufacturers and space agencies a trade-off will be established. It will compare the new approach to the classical one taking into account the following criteria:

- Improvement of space debris environment (statistically)
- Satellite system impacts
- Risk
- Cost

A delicate task will be the definition of weighting criteria that for example compare a cost impact to the potential of reducing the risk of creating debris.

The outcome of the trade-off in first order is the conclusion on which strategy provides globally the best solution. In a more detailed assessment and refined analysis of the trade-off it will also enable to evaluate the parameter space. This way the results of this work can be used as an input to future missions to decide on an optimal disposal strategy.

8 CONCLUSIONS

The paper presents work in progress. The idea of an alternative EoL strategy was born from the belief that collision avoidance capability will play the most important role in the future evolution of space debris. Initial results show that the risk of creating new debris can be reduced with the proposed strategy. It is believed that this can be accomplished with means that have a smaller impact on the design of satellites than the classical strategy including a direct passivation after the end of the mission.

It is foreseen to present more detailed results in future conferences.

9 REFERENCES

1. Bonnal, C. (2016). A brief Historical Overview of Space Debris Mitigation Rules. *CleanSpace Industrial Days*, ESTEC, Noordwijk, The Netherlands, 23-27 May 2016

2. ESA/ADMIN/IPOL(2014)2. Space Debris Mitigation Policy for Agency Projects. ESA Director General's Office, 28/03/2014.

3. Gelhaus, J. (2014). DRAMA Final Report – Upgrade of ESA's Space Debris Mitigation Analysis Tool Suite. Revision 1.1. Institute of Aerospace Systems. TU Braunschweig. 12/06/2014

4. ISO 24113:2011(E). Space Systems – Space debris mitigation requirements. International Standard, Second edition, 15/05/2011

5. Wiedemann, C. (2014). Final Report – Maintenance of the ESA MASTER Model. Revision 1.2. Institute of Aerospace Systems. TU Braunschweig. 12/06/2014

6. Union of Concerned Scientists: UCS Satellite Database. www.ucsusa.org (04/10/2017)