EARLY PHASE RE-ENTRY ANALYSIS OF ESA EARTH OBSERVATION MISSIONS TO ADVANCE SATELLITE DESIGN CHOICES

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ABSTRACT

At OHB System, a large effort is invested to design satellites which comply with the new Clean Space Policy and Space Debris Mitigation Guidelines. This paper provides the outcome of the re-entry analysis performed during the early phases of satellite missions at OHB, within the Space Systems Studies department. In specific, the re-entry analysis of different Earth observation satellites are presented, which have been the drivers of design decisions between uncontrolled and controlled deorbit strategies. In view of the system implications, an uncontrolled re-entry strategy is often the preferred option. In specific, the disposal strategy is driver for the selection of the propulsion system and launcher. Thus, an overview of the factors with greatest impact on the demisability of the satellite components is here presented, together with the different Design for Demise strategies proposed to achieve compliance with the 1/10000 casualty risk threshold. General conclusions between the different satellite systems are drawn, differentiating between radar and optical payloads, and between chemical and electric propulsion systems. Missions that cannot achieve the casualty threshold need to plan a controlled re-entry targeting the debris impact over unpopulated areas to minimize the casualty risk for human population.

1 INTRODUCTION

In the early phases of Earth Observation (EO) missions, the disposal strategy is highly interconnected with system level trade-offs for selection of the propulsion system and launcher. VEGA-C is the baseline launcher for the recent ESA EO missions, which is for large satellite systems a design driver. A higher dry mass can be achieved by performing an electric raising manoeuvre from a low injection orbit. However, a chemical propulsion system is required for performing a controlled re-entry. An uncontrolled re-entry is only possible by demonstrating compliance with the 1/10000 threshold. In the impossibility of compliance, a hybrid propulsion system can be considered for satellites demanding a higher mass. Only then, if launcher performance are not sufficient, a different baseline launcher is considered. Chapter 3 presents the flow-diagram followed at OHB for the selection of the disposal strategy, propulsion system and launcher in recent ESA EO missions.

The roadmap for the selection of the disposal strategy is presented in Chapter 2. The DRAMA tool from ESA is the baseline software at OHB for re-entry and casualty risk analysis. In Chapter 4, the methodology of the computation is summarized.

The break-up altitude is a paramount design parameter for the demisability of the components. This is retrieved with dedicated Monte Carlo simulations, considering the default thermal criterion as the trigger for the spacecraft fragmentation.

Guidelines from ESA are followed for the modelling of the spacecraft, presented in Chapter 5. However, there is quite an uncertainty for the modelling of components such as CFRP panels and electronic units. The default CFRP model from DRAMA seems not representative for structural panels, and thus material properties from reference are considered for comparison. Casualty risk analysis results from different EO missions analysed at OHB are presented in Chapter 6. Surviving components from platform and payload are highlighted, differentiating between radar and optical missions, and chemical and electric propulsion systems. Given the large impact the demise of CFRP panels have on the casualty risk budget, a research on dedicated CFRP test campaigns is carried out to understand their demisability at spacecraft re-entry.

Design for demise solutions (D4D) can be applied to reduce the casualty risk. For missions at the edge of compliance, considering containment solutions such as tethers joining bipods, or demisable reaction wheels has led to compliance with casualty risk threshold to perform an uncontrolled re-entry.

Finally, in the impossibility for performing an uncontrolled re-entry, a controlled re-entry shall be considered. To minimize the delta-V cost for performing the final burn, the highest perigee altitude that allows the debris footprint area to fit within the target area (i.e.: South Pacific Ocean Uninhabited Area) is assessed in Chapter 8. A perigee altitude around 70km results as the threshold value for the missions analysed.

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2 ROADMAP TO SELECTION OF DISPOSAL STRATEGY

At the end of mission, satellites shall be placed out of the LEO clearance region within 25 years, in compliance with ISO 24113 and ESA Space Debris Mitigation (SDM) requirements defined by the ESA policy ESA/ADMIN/IPOL(2014)2 [1].

The end of life assessment aims to identify the most suitable de-orbiting solution for the mission. In order to avoid the need to rely on de-orbiting services (i.e. space tugs), the satellite will have to de-orbit by its own means. Different de-orbiting solutions exist, mainly the satellite can be de-orbited with a controlled or an uncontrolled re-entry. With the update in 2019 of the ISO 24113, the disposal to a graveyard orbit above the LEO region is no longer permitted. The selection driver between a controlled or uncontrolled disposal strategy is the resulting casualty risk, which shall comply with the 1/10000 threshold. The trade on the disposal scenario is driver for the mission, as strongly impacts the required propellant to be stored on-board, as well as the required propulsion system, and consequently, the launcher selection.

The following diagram shows the common process followed for disposal manoeuvre selection of LEO satellites. The atmospheric re-entry is achievable through either propulsive means or passively through atmospheric drag. The satellite decay period and the compliance to casualty risk are the deciding factors.

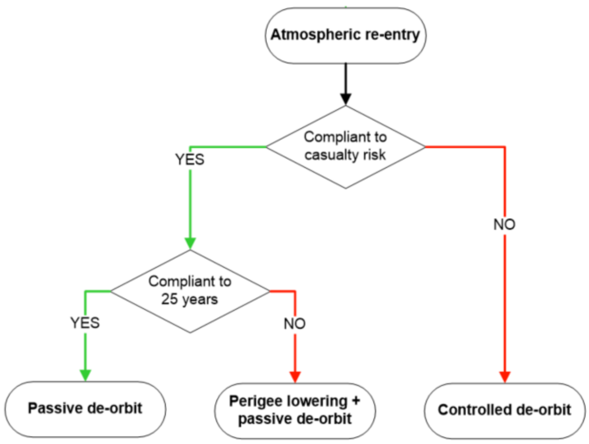

Figure 1. Process for selection of orbital disposal strategy for LEO spacecraft

The DRAMA-SARA tool from ESA is used to calculate the satellite components which may survive and to compute the casualty risk for both the controlled and uncontrolled scenarios. The casualty risk is highly impacted by the re-entry year. It increases with the world population. The survivability of the components is significantly related to the break-up altitude of the main parent. A 78km is commonly approved by the Agency, however, following the ESA Space Debris Mitigation Guidelines [1], the prediction of the break-up altitudes can be based on valid physical considerations, similitudes or probabilistic assessments. Dedicated analysis are performed for this purpose (Chapter 6).

Mainly large and high melting temperature components survive re-entry. Design for demise solutions (D4D) can be applied to reduce the casualty risk. There exist different solutions, with the aim of reducing the casualty area (e.g.: containment tethers), accelerating the break-up altitude (e.g.: new structural joining technologies) or replacing components with other of higher demisability materials.

In case of compliance with the 1/10000 threshold, the orbit decay is analysed to assure the satellite re-enters within 25 years. The satellite is manoeuvred down to a perigee altitude from which, after 25 years, it will re-enter into the atmosphere.

On the other hand, in the impossibility to comply with casualty risk requirement by means of an uncontrolled disposal strategy, a controlled re-entry scenario shall be planned to target the debris impact over unpopulated areas (i.e. SPOUA: South Pacific Ocean Uninhabited Area). A controlled re-entry scenario requires a larger Delta-V demand and therefore higher propellant mass. Moreover, the final boost will require a high thrust not achievable by means of electric propulsion.

In a controlled re-entry scenario, a Monte Carlo campaign is conducted to simulate the uncertainties of the final boost and estimate debris footprint area. In Chapter 8, targeting a perigee altitude around 60km is shown to guarantee the footprint size to fit within the SPOUA.

3 SYSTEM IMPACTS OF DISPOSAL STRATEGY SELECTION

In the early phases of Earth Observation satellite missions, the disposal strategy is highly interconnected with system trade-offs for selection of the propulsion system and launcher.

As required by the Agency in most cases of the recent EO missions, VEGA-C shall be the baseline launcher. Examples are the Copernicus expansion missions like CO2M, LSTM and PICE satellites, Earth Explorer missions as SKIM, Harmony, Forum, Hydroterra and the Next Generation Sentinel satellites. For large satellite systems, this might be a challenging requirement given the performance of the launcher.

Figure 2 illustrates an example of the flow diagram for selection of launcher, disposal strategy and propulsion system in the recent ESA missions at OHB. As can be seen, the selected propulsion concept is dependent on launcher performance and compliance to casualty risk. The starting point is a mission concept with VEGA C as

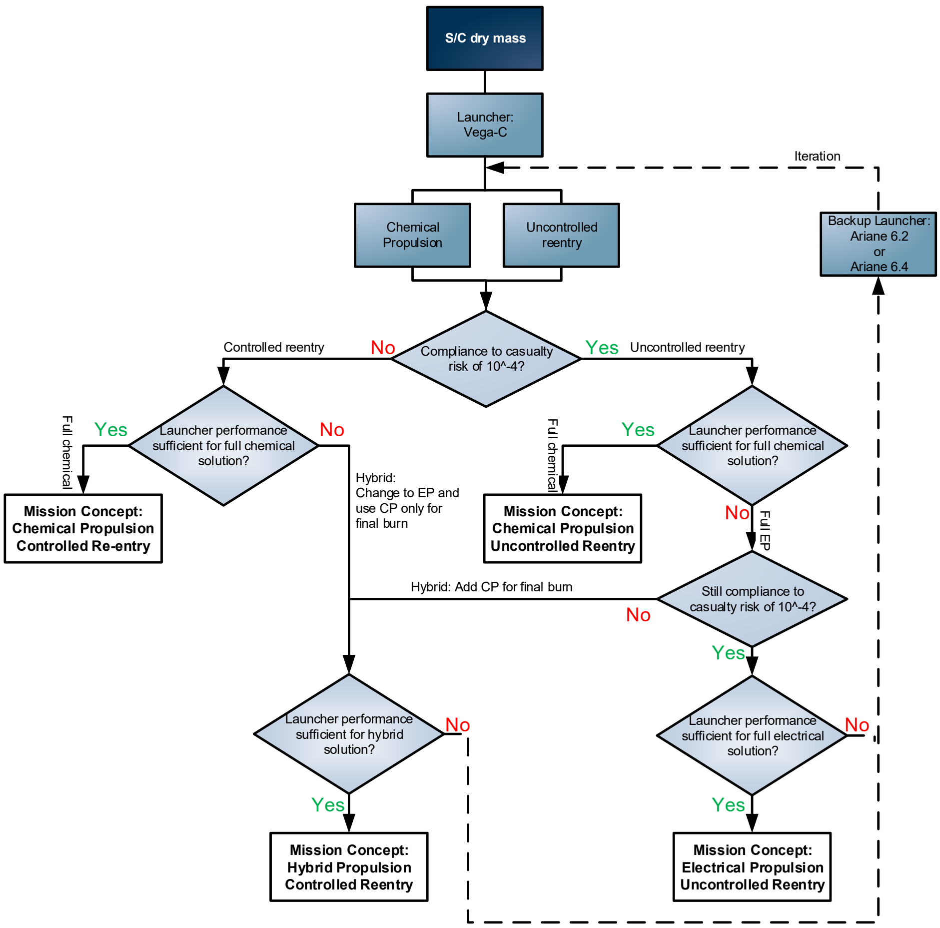

Figure 2. Impact of Disposal Strategy on Launcher and Propulsion System Selection (Example from recent ESA mission at OHB)

baseline launcher, and a chemical system with uncontrolled re-entry to simplify the satellite system. However, an uncontrolled manoeuvre is only possible by demonstrating the casualty risk at impact is less than 1/10000. With the required propellant mass for performing a controlled manoeuvre, the remaining satellite dry mass with a VEGA-C launch might be compromised. For instance, an estimation of the top-down mass breakdown for a representative SSO at 800km, with VEGA-C performance of 2080kg [2], is provided in Table 1. This shows the remaining satellite dry mass is ~1790 kg.

For large satellite systems requesting higher launch mass, changing the baseline launcher is left as the very last option. Instead, a full-electric propulsion system is first analysed. With a launch to a low SSO orbit at 500km altitude, VEGA-C performance is of 2.4 tones [2]. With a full-EP system, the spacecraft will then perform an electric orbit raising up to the nominal orbit. Following with the previous example of the representative SSO orbit at 800km, Table 1 shows the remaining satellite dry mass can be increased up to 2240kg. In the impossibility of compliance with the casualty risk threshold, a hybrid propulsion system for performing the final burn is considered. Finally, missions requiring larger satellite dry masses, a launcher offering higher performance shall be considered.

Table 1. Top down mass breakdown for best and worst case dry mass scenarios for a SSO at 800km

Top-down mass breakdown	Mass [kg] Chemical Controlled	Mass [kg] EP Uncontrolled
Launcher performance	2080 (@800km)	2380 (@500km)
LVA	95	95
Spacecraft wet mass	1985	2285
Propellant mass	197*	46**
Spacecraft dry mass	1788	2239

(*) Launcher dispersions + 60km targeted perigee, 220s Isp

(**) Raising and lowering manoeuvre to 800km, 1600s Isp

4 CASUALTY RISK ANALYSIS METHODOLOGY AT OHB

The DRAMA tool from ESA is used to calculate the components of satellite models which may survive the re-entry and to compute the casualty risk for an uncontrolled and a controlled scenario. The SARA module of DRAMA executes the re-entry survival and risk analyses in two steps using the following tools: SESAM (to assess re-entry survival) and SERAM (to assess re-entry risk).

The updated version of DRAMA allows defining relationships between the spacecraft components. A main parent has to be defined, then the rest of components can be included in or connected to it. The main parent is defined as Platform Module, which has an equivalent box shape to the external shape of the spacecraft structure. All units contained in the spacecraft bus will be included and the external units will be connected to it, as the payload housing (with connected and included payload children), antennas and the launcher adapter.

Components are modelled with simplified shapes (boxes, spheres, cylinders and cones), each composed of a single material characterised by density, melting point, specific heat capacity, heat of fusion and emissivity. Default properties from DRAMA database are used when available.

Surviving fragments are identified and added to the casualty area budget. The kinetic energy threshold criterion of 15-Joules (accepted as the minimum level for potential injury to an unprotected person) is applied to filter surviving components in the casualty risk budget.

The survivability of the components is significantly related to the break-up altitude of the main parent. The higher the break-up altitude, the earlier the exposure of the children to the flow and the longer the time these have to reach a higher temperature. In previous versions of DRAMA, the break-up altitude was fixed at 78 km. In this updated version of DRAMA (version 3.0.3) the break-up altitude can be triggered based on the integrated time histories of the aerothermodynamics of the fragment model along the propagated trajectory [3]. The thermal criterion is the default trigger for the spacecraft fragmentation. Following the ESA Space Debris Mitigation Guidelines [1], the prediction of the break-up altitudes can be based on valid physical considerations, similitudes or probabilistic assessments.

The re-entry analysis at OHB is performed in two phases. Based on the default criterion of DRAMA (total demise of the parent primitive), a first analysis is performed to estimate the worst-case break-up altitude of the main parent. The survivability and casualty risk analysis is then performed in a second simulation forcing the break-up of the main parent at the pre-determined break-up altitude. The worst-case (i.e.: latest break-up) corresponds to the mean-3σ altitude resulting from a Monte Carlo campaign on a simplified model of the spacecraft, with the uncertainties impacting the spacecraft trajectory.

The uncertainties to perform an uncontrolled re-entry are associated to the last boost (thrusting acceleration, direction and duration) and the spacecraft trajectory (spacecraft mass, drag coefficient and atmospheric density). For the performed analysis in the early phases, preliminary quantities are defined based on the dispersions considered for the ESA ATV controlled re-entries [1], see Table 2.

Table 2. Uncertainties for controlled re-entry

Parameter	**Uncertainties**
Atmospheric density	±20% (Uniform)
Drag coefficient	±25% (Uniform)
Spacecraft mass	±20 kg (Uniform)
Thrust level	±10% (Uniform)
Thrusting time	±5 s (Uniform)
Thrusting direction	±1° (Uniform)

In order to define the dispersions to conduct the Monte Carlo campaign in DRAMA, the identified uncertainties of the last boost are combined to be defined as the ones available in the Monte Carlo entries DRAMA list (See Figure 3).

The effect of the last boost on the orbital elements can be derived with the analytical equations that describe the rate of change of the parameter subject to a perturbation acceleration. Assuming an in-plane manoeuvre, the affected parameters are the semi-major axis (a), the eccentricity (e) and the argument of perigee (ω). The analytical equations are given in Table 3, where r and v are the magnitude of the position and velocity vector, μ is the Earth gravitational parameter and θ the true anomaly.

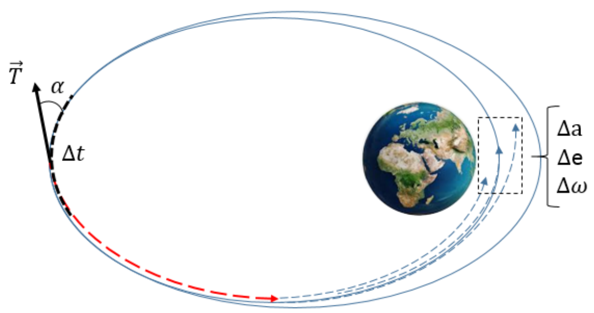

Figure 3. Last boost uncertainties in DRAMA dispersions entries.

Table 3. Rate of change of orbital elements with in-plane perturbations.

Parameter	f_t	f_n
$\frac{da}{dt}$	$\frac{2a^2}{\mu}v$	0
$\frac{de}{dt}$	$\frac{2avr(e+cos\theta)}{\mu(2a-r)}$	$\frac{rsin\theta}{2av}$
$\frac{d\omega}{dt}$	$\frac{2a^2}{\mu}v$	$\frac{2ae+rcos\theta}{2aev}$

Given representative characteristics for the last burn, the dispersions of the three orbital parameters are defined as the rate of change for the resulting thrusting acceleration, integrated over the region bounded by the thrusting arc centered at the apogee. The estimated dispersions are given in Table 4.

Table 4. Monte Carlo dispersion entries for controlled re-entry analysis based on representative last burn characteristics

Parameter	**Value**
Second-to-last Perigee	250 km
ΔV	60 m/s
Duration	26 min
Mean Thrust	50 N
Parameter	**Uncertainties**
Semi-major Axis	±10 km (Uniform)
Eccentricity	± 0.001 (Uniform)
Argument of Perigee	±1° (Uniform)

5 SPACECRAFT MODELLING

This section describes the assumptions taken to model the satellites in DRAMA. Different examples from missions analysed at OHB are provided in Figure 4 to Figure 6. Namely, an optical, a radar and a Synthetic Aperture Radar (SAR) spacecraft with a large deployable reflector.

The satellite systems are designed with components modelled as boxes, cylinders and spheres, each composed of a single material characterised by density, melting point, specific heat capacity, heat of fusion and emissivity. A main parent (Platform Module) is modelled with the equivalent dimensions of the satellite platform structure and a secondary main parent is modelled for the payload housing. Relationships “connected to” or “included in (parent)” are defined for each component. The payload housing is connected to the platform module parent, and is also the parent to the payload units.

As reported in the previous section where the methodology of the analysis is described, the survivability and casualty risk analysis is performed forcing the break-up of the parent at a pre-determined altitude. After break-up, the survivability of the external structural panels is no longer considered in the DRAMA tool. Therefore, structural panels are modelled separately and attached to their corresponding parent.

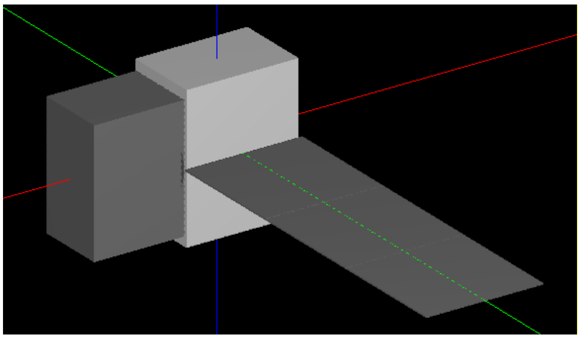

Figure 4. Example of optical spacecraft DRAMA model

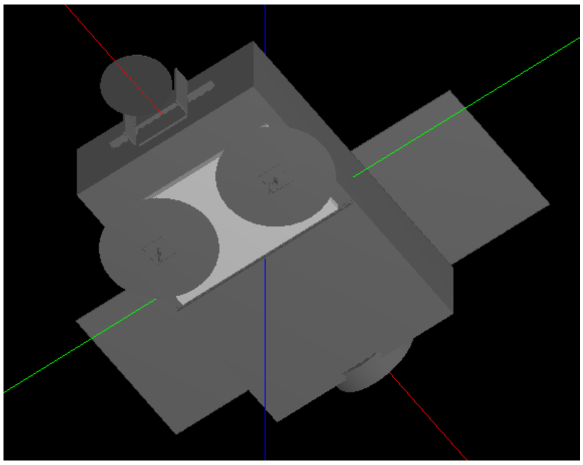

Figure 5. Example radar spacecraft model in DRAMA

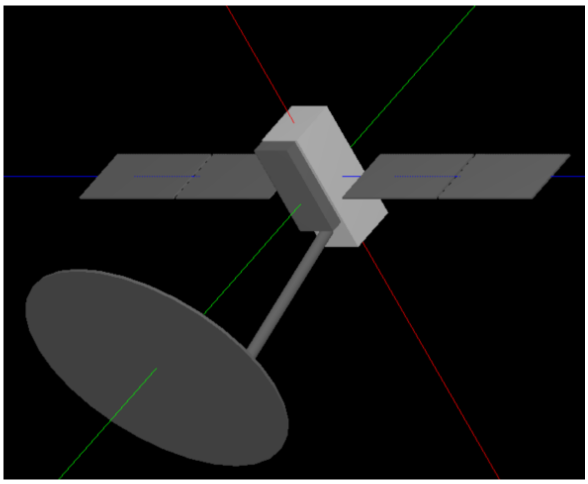

Figure 6. Example of reflector SAR spacecraft model in DRAMA

Efforts are made to model in great detail components with the most difficult materials to demise, as Titanium, Silica, CFRP and Steel. The simulation accounts for a high percentage of the total mass, components judged not critical for the casualty risk analysis are not included in the list. For instance, the thermal subsystem, with mainly aluminium components of weight lower than 0.2 kg, is deemed most likely to demise and is not included in the simulations. Electronic units integrate the different electronic packages such as FPGAs, mass memory modules, integrated circuits, circuitry, etc. into a mechanical frame. Thus, these are modelled with an aluminium housing of ~5 mm thickness and the remaining mass corresponds to the electronic packaging. Examples of materials used in electronic packaging are metals (thin metal interconnects on the integrated circuit made of Al or Cu), ceramics, polymers (silicones) and glasses. These components, with small size and mass, are easy-to-demise and are neglected in the model. To simulate the same dynamics of the main component, the corresponding mass of the internal components is added as a child in a dummy mass component. Even if it survives, it will not be added to the casualty risk budget. For instance, this criteria is applied to the PDHU and the OBC. Batteries are modelled in two separated parts, the chassis in Aluminium and the casing of the cells in Stainless Steel. For the cells, the mass of the chemical part is not considered, thus only the cell mass corresponding to the casing is considered (~23% of the cell mass, as specified by provider).

In contrast to the reaction wheel model in the DRAMA database, at OHB reaction wheels are modelled with two separated parts. It is understood the rotation mass in Steel is critical for the analysis, and thus it is separated from the Aluminium housing, parent to the rotation mass.

When possible, the default materials of DRAMA are used. The default properties in DRAMA 3 (from version 3.0.0 up to the most updated version 3.0.4) of the CFRP material model seem to be not representative for regular structural panels, which has a large impact on the casualty risk results of most ESA analysed missions as reported in Chapter 6, see Figure 10. Based on the operational experience from ESA [12], the default CFRP model is mostly useable for CFRP overwrap of tanks. The activation temperature of the resin in the CFRP DRAMA model is close to 3000 K, in contrast, in the data base of other object oriented tools such as DEBRISK [4], developed by CNES, CFRP properties are modelled as an equivalent metal with 700K as melting temperature. Moreover, the minimum facesheet thickness allowed in DRAMA is 2mm, whereas in common configurations, facesheet thickness depending of the number of plies can vary from 0.3 mm (4 plies) to 0.6 mm (8 plies).

Dedicated test campaigns are found in literature. Reference [6] shows that the Aluminium HC does not demise until the CFRP fibers ablation. Even if the resin first evaporates, the fibers act as thermal protection. On the contrary, [7] demonstrates that when the resin starts evaporating, the Aluminium inside increases temperature and starts demising. The study conducted at OHB [5] goes also in this direction, the CFRP facesheet detaches from the honeycomb at low temperatures and the honeycomb demises faster, see Figure 7. It is also observed the matrix vaporizes at rather low temperatures, leaving a char residue that acts as a matrix material for the dry fibres. Where the char is burned away, fibres become loose. The temperature at which the char leaves the fibres could be seeked to model a CFRP model in DRAMA as an equivalent metal, where the "melting temperature" is the threshold value at which fibres are released from the resin. Even if fibres survive re-entry, the kinetic energy at impact of the separated fibres would be sufficiently low to cause any injuries (e.g.: below the 15J threshold). Last, reference [8] concludes that the integrity of CFRP facesheets is maintained after re-entry, although these are very thin. Overall, results from these different dedicated test campaigns coincide with the understanding that CFRP fibers are highly resistant, although the Honeycomb (HC) is highly demisable when the charred CFRP facesheet detaches. Both understanding whether facesheet detaches at re-entry, and the casualty risk of the fibers is until now an open question.

For comparison analysis of demising results with respect to the CFRP DRAMA-based model, the CFRP model from DEBRISK is used in OHB analysis. The equivalent properties are given in Table 5.

Table 5. CFRP model based on DEBRISK [4]

Density [kg/m^3]	1600
Mean specific heat capacity (300 K - Tm) [J/kgK]	879
Melting Temperature [K]	700
Specific heat of melting [J/kg]	233
Mean emission coefficient (300 K - Tm) [-]	0.999

CFRP sandwich panels are modelled with three separated parts. CFRP facesheets are modelled as plates connected to the Aluminium honeycomb. However, the recently issued Guidelines for Analysing and Testing the Demise of Man Made Space Objects During Re-entry (DIVE) [9] suggest the modelling of the complete panel with Aluminium-HC, including the mass of the CFRP facesheets. It is highlighted the CFRP demise is an active research topic, as shown with this reported literature study. It then emphasizes the CFRP facesheets do not necessarily form a separate critical element.

Figure 7. Wind tunnel test results, Design for Demise Breadboarding Study, OHB System AG [5].

6 UNCONTROLLED RE-ENTRY SCENARIO

A Monte Carlo campaign is first conducted to determine the break-up altitude of the main parent, at which it releases all the included and connected components. The break-up condition is based on a thermal criterion (by default criterion in DRAMA 3.0.0); it occurs when the parent primitive is completely demised. The worst case scenario (mean-3σ break-up altitude) is selected to perform in a second step the survivability and casualty risk analysis, forcing this value as the trigger for the spacecraft fragmentation. In the example for Figure 8, the resulting break-up altitude is 83km. The survivability and casualty risk analysis for this mission was then performed with the break-up condition forced at this predetermined altitude.

In the break-up altitude analysis, a simplified model of the satellite is employed. Variables of interest for the S/C trajectory simulation and break-up of the main parent are identified:

- Spacecraft trajectory:
  - Initial conditions;
  - Total mass of the spacecraft;
  - Cross-section area and drag coefficient;
  - Atmospheric model and density;
- Break-up trigger:
  - Mass, material and shape of the main parent.

The simplified spacecraft model includes therefore the external appendages that increase the cross-section area (deployed solar arrays and payload housing) and dummy masses to model the spacecraft with the mass at End of Life (EOL); a dummy mass included as a child to the main parent and a dummy mass included as a child to the payload housing.

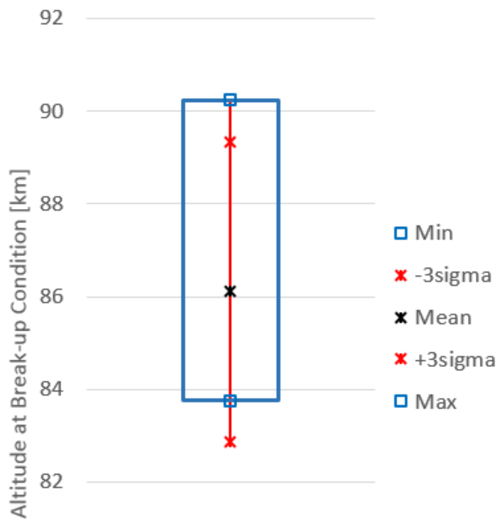

Figure 8. Break-up altitude results in the Monte Carlo campaign for spacecraft simplified model. Break-up condition is based on thermal criterion (Default criteria in DRAMA).

Results from the survivability and casualty risk analysis of different ESA Earth Observation missions, performed during the Phases 0/B1, are gathered in Figure 9. Optical missions result with a high casualty risk due the numerous surviving components from the payload, made in Silica, Titanium and Si3N4. Each optical unit is attached with a bracket to the optical bench. Bipods can even count as three components, with the two feet and the bracket separated by the beam in a different material. The number of surviving components is what highly increases the casualty area, rather than their size. On the contrary, the contribution of the payload to the casualty risk budget in radar missions is on the same order that the platform. The antenna feet in Titanium, as interface to the platform, are the main contributors. Surviving platform components are mainly the tank and the rotation mass of reaction wheels. For electric propulsion systems, the thrusters in Inconel and the Xenon flow controller in Titanium survive due to the larger size and mass in comparison with the regular chemical system.

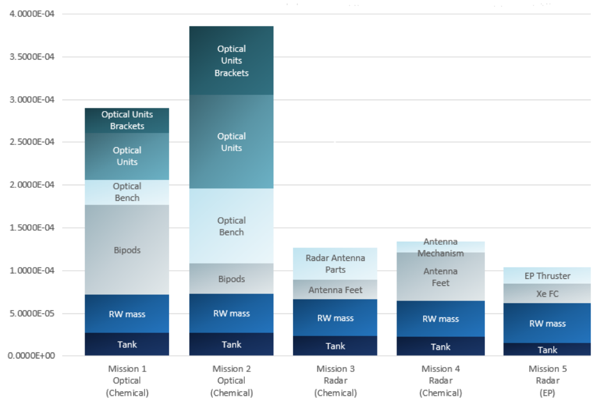

Figure 9. Casualty risk of different ESA EO Missions

These preliminary results comply with ESA guidelines in terms of modelling of the components. However, there is quite an uncertainty on the modelling of certain units, such as CFRP panels and electronic units. As reported in the previous section, CFRP model from DRAMA is not representative for structural panels. The reported results in Figure 9 correspond to the CFRP model defined in the previous section, which results to demise. The new design guidelines from ESA (DIVE, [9]) suggest to model CFRP panels in Aluminium. Therefore, the preliminary results are in line with ESA expectations.

There is quite an uncertainty on the modelling of certain units, such as CFRP panels. As reported in the previous section, CFRP model from DRAMA is not representative for structural panels. The reported results in Figure 9 correspond to the CFRP model defined in the previous section. Under these assumptions, structural CFRP panels do not survive re-entry. On the contrary, with the default CFRP model in DRAMA, most CFRP components survive re-entry. As shown in Figure 10, Mission 3 results compliant in the scenario that CFRP demises, but with the default CFRP DRAMA model, the contribution of CFRP lead to an increase of up to $4.5x10^{-4}$ casualty risk probability. It is of high interest to understand the real behaviour of CFRP panels at re-entry. Results from different dedicated test campaigns coincide with the understanding that CFRP fibers are highly resistant, although the HC is highly demisable when the charred CFRP facesheet detaches. Both understanding whether facesheet detaches at re-entry, and the casualty risk of the fibers is until now an open question.

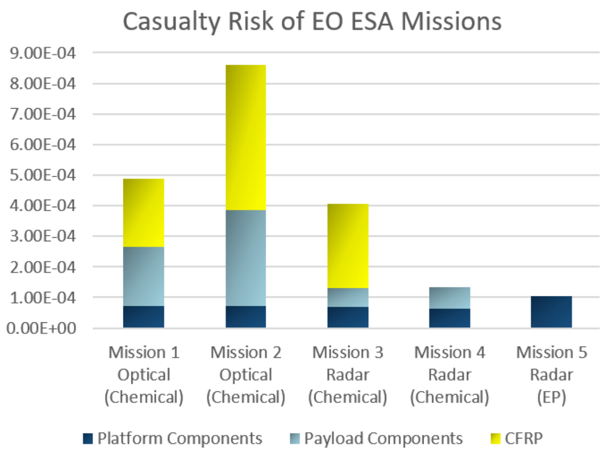

Figure 10. Casualty Risk of different ESA EO Missions with default DRAMA 3 CFRP model

Finally, the modelling of electronic units is also an open question in the survivability analysis. Modelling a box in aluminium, with the mass of the complete unit, is not representative, since the thickness of the housing is not higher than 1-2 mm. Instead, a more representative model is a box with the corresponding thickness of the housing, and the remaining mass modelled with the default Electronic Material in DRAMA. However, this highly demisable material might not be representative for electronic cards. The test campaign conducted in [8] shows that GFRP cards are less demisable than had been assumed, thus a deeper study of the demise of electronic cards is necessary.

7 DESIGN FOR DEMISE SOLUTIONS

Design for demise solutions (D4D) can be applied to reduce the casualty risk. There exist different solutions, with the aim of reducing the casualty area (e.g.: containment tethers), accelerating the break-up altitude (e.g.: new structural joining technologies) or replacing components with other of higher demisability materials.

For instance, a study conducted in [11] by “Rockwell Collins” introduces potential design changes on reaction wheels models. It has been proven that full demisability is achieved with minor modifications of the ball bearing unit or with an Aluminium rotation mass with no further modifications, at a demise altitude of 78 km. Reference [10] presents a preliminary design description of a containment tether for bipods to sustain the harsh re-entry environment. The proposed solution is a Tungsten tether with an external diameter of 10mm. Also the small modifications required on the bipods to accommodate the tether is presented. Reference [5] covers the testing on existing structural joining technologies to analyse the break-up altitudes at re-entry, differentiating between panels with aluminium and CFRP facesheets. Based on different worst-case test scenarios, it has been proved that cleated joints fail at altitudes above 90 km for both panel types.

In Mission 3, the use of D4D methods (A tether in tungsten containing the antenna feet, and demisable RW) allowed compliance to casualty risk threshold to perform an uncontrolled re-entry, see Table 6.

Table 6. Casualty risk reduction with D4D methods in Mission 3.

Design scenario	Total casualty area $[m^2]$	Total casualty probability
Nominal scenario	8.6057	1.2994E-04
+ D4D Tether (Antenna Feet)	7.0847	1.1795E-04
+ D4D Demisable RW	5.6483	8.5288E-05
+ D4D Tether (Antenna Feet) and Demisable RW	4.1273	7.3289E-05

8 CONTROLLED RE-ENTRY SCENARIO

In a controlled re-entry scenario, a Monte Carlo campaign is conducted to simulate the uncertainties of the final boost and estimate the debris footprint area. The methodology of the computation is provided in Chapter 4.

The re-entry trajectory shall be planed to target the debris impact over unpopulated areas (i.e. SPOUA: South Pacific Ocean Uninhabited Area). The final burn requires a large Delta-V demand and therefore, higher propellant mass. Thus, a analysis is performed to assess the highest perigee altitude for which the debris footprint area falls within the SPOUA target area. As seen from Figure 11 to Figure 14, for the mission analysed, a perigee altitude around 70km shall be targeted to guarantee the footprint size to fit within the SPOUA. For targeted perigee altitudes above 80km, a delayed re-entry may occurred, leading the debris footprint to fall at an undesired location. This is illustrated in Figure 15.

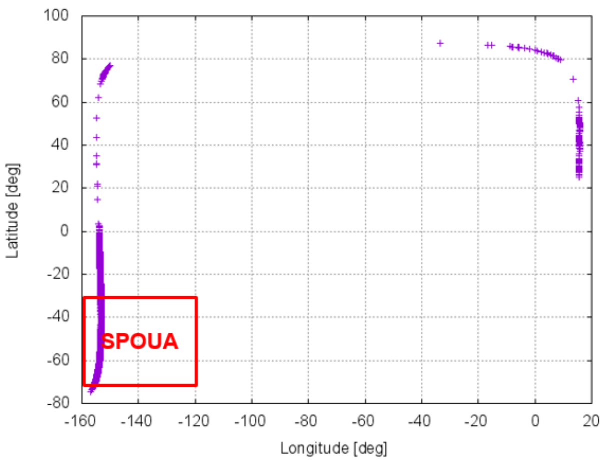

Figure 11. Footprint area of a controlled re-entry with 90km targeted perigee.

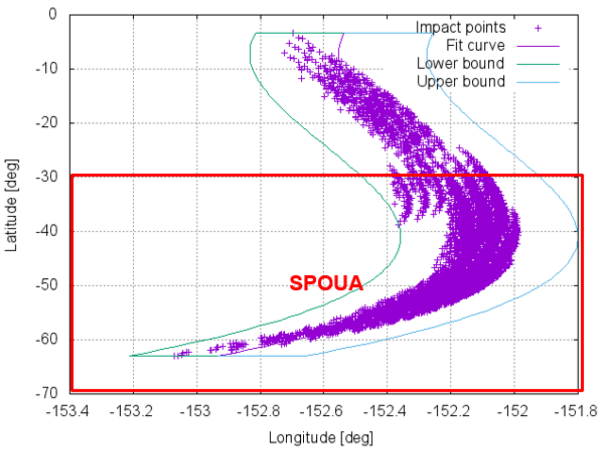

Figure 12. Footprint area of a controlled re-entry with 80km targeted perigee.

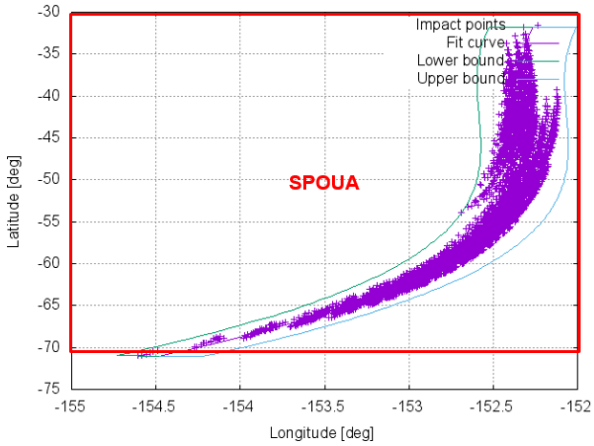

Figure 13. Footprint area of a controlled re-entry with 70km targeted perigee.

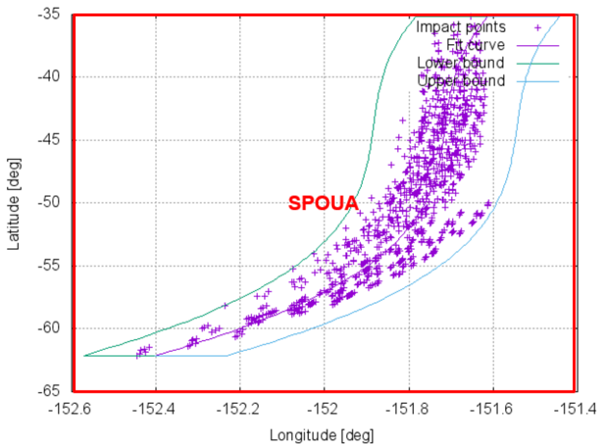

Figure 14. Footprint area of a controlled re-entry with 60km targeted perigee.

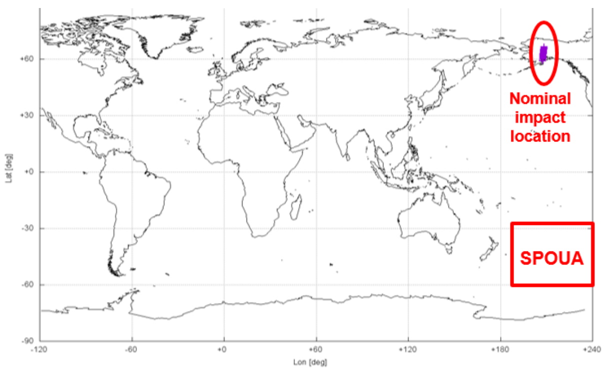

Figure 15. Delayed re-entry with a targeted perigee above 80km

9 CONCLUSIONS

The disposal strategy is driver for the selection of the propulsion system and the launcher. To minimize system complexity, a flow diagram to assure compliance with casualty risk and launcher performance is presented, which has been the basis for design decisions in recent

ESA EO missions at OHB. A chemical propulsion system is the preferred solution, although the launcher performance at the nominal altitude might be compromised. An electric propulsion system allows for increased performance, the satellite can be injected into a lower altitude and raise up to its nominal positions. However, this is not sufficient for the final burn of a controlled re-entry. A hybrid propulsion system can be considered for the final burn. Only then, if launcher performance are not sufficient, a different baseline launcher is considered.

DRAMA casualty risk results of different ESA EO missions analysed at OHB are presented. In dedicated Monte Carlo campaigns, break up altitudes between 84km and 90km are retrieved for the analysed ESA EO missions, in contrast to the 78km set in DRAMA by default. Optical missions result with a high casualty risk due the numerous surviving components from the payload. On the contrary, the contribution of the payload to the casualty risk budget in radar missions is on the same order that the platform. The default CFRP model from DRAMA survives re-entry, highly increasing the casualty risk budget and leading to non-compliances. The new ESA Design for Demise Verification Guidelines recommends the modelling of CFRP sandwich panels in Aluminium. Under these assumptions, the casualty risk is highly reduced, allowing in some missions to baseline an uncontrolled re-entry. Dedicated test campaigns coincide with the understanding that CFRP fibers are highly resistant, although the Honeycomb is highly demisable when the charred CFRP facesheet detaches. Both understanding whether facesheet detaches at re-entry, and the casualty risk of the fibers is until now an open question.

Design for demise solutions, such as containment tethers or demisable reaction wheels, are applied for missions non-compliant with the 1/10000 threshold. In some missions, these have allowed to baseline an uncontrolled re-entry strategy. In the impossibility of compliance, a controlled re-entry strategy is considered. For the missions analysed, a perigee altitude around 70km is shown to allow the debris footprint area fit within the SPOUA target area.

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