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# Spacecraft design optimisation for demise and survivability

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## ABSTRACT

Among the mitigation measures introduced to cope with the space debris issue there is the de-orbiting of decommissioned satellites. Guidelines for re-entering objects call for a ground casualty risk no higher than $10^{-4}$. To comply with this requirement, satellites can be designed through a design-for-demise philosophy. Still, a spacecraft designed to demise through the atmosphere has to survive the debris-populated space environment for many years. The demisability and the survivability of a satellite can both be influenced by a set of common design choices such as the material selection, the geometry definition, and the position of the components inside the spacecraft. Within this context, two models have been developed to analyse the demise and the survivability of satellites. Given the competing nature of the demisability and the survivability requirements, a multi-objective optimisation framework was developed, with the aim to identify trade-off solutions for the preliminary design of satellites. As the problem is nonlinear and involves the combination of continuous and discrete variables, classical derivative based approaches are unsuited and a genetic algorithm was selected instead. The genetic algorithm uses the developed demisability and survivability criteria as the fitness functions of the multi-objective algorithm. The paper presents a test case, which considers the preliminary optimisation of tanks in terms of material, geometry, location, and number of tanks for a representative Earth observation mission. The configuration of the external structure of the spacecraft is fixed. Tanks were selected because they are sensitive to both design requirements: they represent critical components in the demise process and impact damage can cause the loss of the mission because of leaking and ruptures. The results present the possible trade off solutions, constituting the Pareto front obtained from the multi-objective optimisation.

## 1. Introduction

In the past two decades, the attention towards a more sustainable use of outer space has increased steadily. The major space-faring nations and international committees have proposed a series of debris mitigation measures [1,2] to protect the space environment. Among these mitigation measures, the de-orbiting of spacecraft at the end of their operational life is recommended in order to reduce the risk of collisions in orbit.

However, re-entering spacecraft can pose a risk to people and property on the ground. Consequently, the re-entry of disposed spacecraft needs to be analysed and its compliancy with international regulations has to be assessed. In particular, the casualty risk for people on the ground related to the re-entry of a spacecraft needs to be below the limit of $10^{-4}$ if an uncontrolled re-entry strategy is to be adopted [3,4]. A possible strategy to limit the ground casualty risk is to use a design-for-demise philosophy, where most (if not all) of the spacecraft will not survive the re-entry process. The implementation of design for demise strategies [5–7] may favour the selection of uncontrolled re-entry disposal options over controlled ones, leading to a simpler and cheaper alternative for the disposal of a satellite at the end of its operational life [6,7]. However, a spacecraft designed for demise still has to survive the space environment for many years. As a large number of space debris and meteoroids populates the space around the Earth, a spacecraft can suffer impacts from these particles, which can be extremely dangerous, damaging the spacecraft or even causing the complete loss of the mission [8–10]. This means that the spacecraft design has also to comply with the requirements arising from the survivability against debris impacts.

The demisability and survivability of a spacecraft are both influenced by a set of common design choices, such as the material of the structure, its shape, dimension and position inside the spacecraft. It is important to consider such design choices and how they influence the mission's survivability and demisability from the early stages of the mission design process [7]. In fact, taking into

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## Nomenclature

| | | |
|---|---|---|
| $a$ | Semi-major axis | Km |
| $A$ | Area | $m^2$ |
| $C$ | Speed of sound | m/s |
| $C_D$ | Drag coefficient | |
| $CF$ | Correction factor for component mutual shielding | |
| $C_m$ | Heat capacity | J/Kg K |
| $D$ | Diameter | m |
| $d$ | Lateral size of a component in the impact plane | m |
| $E_0$ | Maximum allowed displacement from the nominal ground track at the equator | Km |
| $\bar{F}_q$ | Motion and shape averaged shape factor for heat flux predictions | |
| $G$ | Universal gravitational constant | |
| $g_\varphi$ | Polar component of the gravitational acceleration | $m/s^2$ |
| $g_0$ | Gravitational acceleration at sea level | $m/s^2$ |
| $g_R$ | Radial component of the gravitational acceleration | $m/s^2$ |
| $h$ | Altitude | Km |
| $h_f$ | Heat of fusion | J/kg |
| $Isp$ | Specific impulse | s |
| $K1$ | Factor to account for the additional tank volume for the pressuring gas | |
| $K2$ | Factor to account for the separation between two tanks | |
| $l$ | Distance between two tanks | m |
| $L$ | Side length of the spacecraft | m |
| $L$ | Length | m |
| $m$ | Mass | Kg |
| $M_E$ | Earth's mass, $5.97 \times 10^{24}$ | Kg |
| $N$ | Total number of spacecraft components | |
| $n_t$ | Number of tanks | |
| $P_p$ | Penetration probability | |
| $r$ | Radius | m |
| $R_E$ | Earth's radius, 6371800 | m |
| $r_n$ | Nose radius | m |
| $S$ | Cross-section of the spacecraft | $m^2$ |
| $s$ | Stand-off distance | m |
| $SF$ | Tank pressure safety factor | |
| $t$ | Thickness | m |
| $t_m$ | Mission duration | years |
| $T_m$ | Melting temperature | K |
| $V$ | Velocity | m/s |
| $v$ | Volume | $m^3$ |
| $\alpha$ | Cone ejecta spread angle | ° |
| $\gamma$ | Flight path angle | ° |
| $\varepsilon$ | Emissivity | |
| $\theta$ | Impact angle | ° |
| $\lambda$ | Longitude | ° |
| $\rho$ | Density | $Kg/m^3$ |
| $\sigma$ | Stefan–Boltzmann constant, $5.67 \times 10^{-8}$ | $W/m^2/K^4$ |
| $\sigma_u$ | Ultimate tensile strength | MPa |
| $\sigma_y$ | Yield strength | MPa |
| $\varphi$ | Latitude | ° |
| $\phi$ | Debris flux | $1/m^2/yr$ |
| $\chi$ | Heading angle | ° |
| $\omega$ | Angular velocity | °/s |

### Subscripts

| | |
|---|---|
| *0* | Nominal orbit |
| *1* | Start orbit for the Hohmann transfer |
| *2* | Final orbit of the Hohmann transfer |
| *atm* | Atmosphere |
| *BLE* | Relative to the probability of penetrating an internal component |
| *C* | Relative to the critical diameter |
| *comp* | Relative to the probability of impacting a component after a first impact on the vulnerable zone |
| *decay* | Relative to decaying correction manoeuvres |
| *disp* | Relative to disposal manoeuvres |
| *e* | Earth |
| *ejecta* | Relative to the debris cone produced after impact |
| *f* | Fuel |
| *fin* | Final condition |
| *in* | Initial condition |
| *inc* | Relative to inclination change manoeuvres |
| *inj* | Relative to orbit injection errors |
| *mat* | Material |
| *p* | Debris particle |
| *s* | Spacecraft |
| *sec* | Relative to secular variations of the orbital parameters |
| *struct* | Relative to the impact on the external structure inside the vulnerable zone |
| *t* | Tank |
| *target* | Target component of the impact probability analysis |
| *tot* | Total |
| *VZ* | Relative to the vulnerable zone |
| *w* | Wall |

### Abbreviation

| | |
|---|---|
| DAS | Debris Assessment Software |
| LMF | Liquid Mass Fraction |
| BLE | Ballistic Limit Equation |
| SRL | Schafer–Ryan–Lambert |
| NSGA | Non-dominated Sorting Genetic Algorithm |
| PNP | Probability of no-penetration |

account these requirements at a later stage of the mission may cause an inadequate integration of these design solutions, leading to a delayed deployment of the mission and to an increased cost of the project. On the other hand, an early consideration of such requirements can favour cheaper options such as the uncontrolled re-entry of the satellite, whilst maintaining the necessary survivability and, thus, the mission reliability.

With these considerations, two models have been developed [11] to assess the demisability and the survivability of simplified mission designs as a function of different design parameters. Two criteria are presented to evaluate the degree of demisability and survivability of a spacecraft configuration. Such an analysis can be carried out on many different kinds of missions, provided that they can be disposed through atmospheric re-entry and they experience impacts from debris particles during their operational life. These characteristics are common to a variety of missions; however, it was decided to focus the current analysis on Earth observation and remote sensing missions. Many of these missions exploit sun-synchronous orbits due to their favourable characteristics, where a spacecraft passes over any given point of the Earth's surface at the same local solar time. Because of their appealing features, sun-synchronous orbits have high commercial value. Alongside their value from the commercial standpoint, they are also interesting for a combined survivability and demisability analysis. Sun-synchronous missions can in fact be disposed through atmospheric re-entry. They are also subject to very high debris