Multi-objective optimisation for spacecraft design for demise and survivability

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Summary

The paper presents the development of a multi-objective optimisation framework to study the effects that preliminary design choices have on the demisability and the survivability of a spacecraft. Building a spacecraft such that most of it will demise during the re-entry through design-for-demise strategies may lead to design that are more vulnerable to space debris impacts, thus compromising the reliability of the mission. The two models developed to analyse the demisability and the survivability are presented and used as the objective functions of the multi-objective optimisation of the external structure of a simplified spacecraft configuration. The results are presented in the form of Pareto fronts and selected solutions are used to test a spacecraft configuration with internal components in order to better assess the quality of the solutions.

Keywords: *design-for-demise, survivability, optimization, re-entry, vulnerable zones, shielding*

1 Introduction

In a period where the evolution of the space environment is causing increasing concerns for the future of space exploitation and sustainability, the design for demise philosophy has gained an increased interest. However, satellites designed for demise still have to survive the space environment polluted by space debris. Within this context, we are developing a method to evaluate the effect of preliminary design choices on the survivability and on the demise of a spacecraft configuration since the early stages of the mission design. A multi-objective framework is applied to study the effect of the design choices on both the demisability and the survivability. In the paper an analysis of the external structure of a satellite as a function of the material, the shielding type, and the dimensions is presented. In addition, different mission scenarios are considered.

2 Demise and Survivability Models

To compute the demisability and the survivability of simplified spacecraft configuration two models have been developed[1]. The demisability model analyses the re-entry of a spacecraft and assesses its demisability determining which components will survive the extreme conditions encountered during re-entry. The survivability model assesses instead the vulnerability of a spacecraft configuration against the impacts from space debris computing the penetration probability on the outer structure and on its internal components.

2.1. Demisability Model

The demise model consists of an object-oriented code[2]. The model uses a simplified representation of the spacecraft configuration achieved reducing the satellite design and its components into primitive shapes (e.g. spheres, boxes, cylinders, and flat plates). The definition of the satellite architecture is subdivided into three levels: Level 0, Level 1, and Level 2. The Level 0 represents the main structure of the spacecraft. Level 1 objects are used to represent the external panels of the main structure, which can detach before the break-up if their melting temperature is reached before the break-up altitude. Level 2 geometries instead represent the inner components. The inner components are not exposed to the heat flux until the main spacecraft breaks up. The breakup occurs at a predefined altitude (standard is 78 km), after that the inner component are released and their re-entry is simulated. The re-entry trajectory simulation uses a three degree of freedom dynamics considering the effect of the zonal harmonics of the Earth's gravity filed and aerodynamic drag, using the 1976 U.S. Standard Atmosphere.

The attitude motion of the object is not directly computed but assumed as random tumbling. The adoption of motion and shape averaged drag coefficients allow the determination of the pressure forces on each component[3, 4]. The computation of the thermal load uses the Detra-Kemp-Riddel correlation[5] and a set of motion and shape averaged shape factor to adjust the heat load to the specific shape considered[4, 6, 7]. The materials have temperature independent properties, which have been obtained from the database of the Debris Assessment Software (DAS). The demise of an object is analysed with a lumped mass model where the temperature of the object remains uniform over the entire volume. After reaching the melting temperature, the object starts losing mass at a rate that is proportional to the net heat flux on the object and to the heat of fusion.

2.2. Survivability Model

In the survivability model, the spacecraft structure is schematised with a panelised representation. To each panel we assign a material and geometrical properties such as the type of shielding, the wall thickness. The survivability model uses the same geometrical elementary shapes of the demise model to represent satellite structures, in order to keep the two models comparable. Beside the geometrical schematisation of the satellite, a representation of the space environment is also needed. This is obtained using the European Space Agency (ESA) software MASTER-2009[8]. MASTER-2009 provides a set of 2D and 3D flux distributions as a function of the impact azimuth, impact elevation, impact velocity, and particle diameter. Then, we subdivide the space around the satellite in a set of angular sectors and associate to each sector a vector element containing the average of the impact flux, impact direction and impact velocity. These vectors are used to compute the critical diameter corresponding to each panel of the structure using Ballistic Limit Equations (BLEs). Once obtained the critical diameter, the corresponding critical flux allows the computation of the penetration probability using Poisson statistics. The penetration probability ($P_{j,i}^{p}$) is computed for each vector flux on every panel of the structure and then we compute the overall penetration probability (P_p) with Eq. (1)

$$P_p = 1 - \prod_{j=1}^{N_{panels}} \left(\prod_{i=1}^{N_{fluxes}} \left(1 - P_{j,i}^{p}\right) \right) \tag{1}$$

where N_{panels} and N_{fluxes} are the number of panels in which the structure is schematised and the number of vector flux elements, respectively

The vulnerability of internal components is assessed using the concept of vulnerable zones[9] and a probabilistic approach. The vulnerable zone consists of an adjusted projection of an inner component onto the outer spacecraft structure. This area represents the portion of the external structure that, if impacted by a particle, could also lead to the impact of the inner component to which the relevant vulnerable area is associated.

The penetration probability on a component is then evaluated as the product of three different probabilities.

$$P_p = P_{struct} \cdot P_{comp} \cdot P_{BLE} \tag{2}$$

where P_{struct} is the probability of space debris impacting the spacecraft external structure inside the vulnerable zone assigned to the specific spacecraft component; P_{comp} is the probability that the downrange fragment cloud will hit the component; and P_{BLE} is the probability that the projectile in this cloud perforates the component wall. Eq. (1) is still apllied to the computation of the penetration probability substituting the panels with the vulnearble area ssociated to the component.

3 Multi-objective Optimisation

The demisability and survivability model are used to study the effect that different design choices can have on the survivability and the demisability of a spacecraft. As panels compose the external structure of a satellite, their different characteristics can lead to diverse re-entry scenarios and can affect the survivability of the satellite itself. These external panels can in fact detach from the main structure of the spacecraft before the actual break-up occurs. In this case, the internal components can be exposed to the heat flux even before the break-up, improving their demisability. The amount of heat that the internal components can receive can be computed as follows.

$$\dot{q}_{\text{int}} = \sum_{i=1}^{N_{panels}} \left(\int_{h_d}^{h_b} \dot{q}_i \right) \tag{3}$$

where q_{int} is the heat flux the internal objects can receive because of the early detachment of external panels, h_b is the break-up altitude, h_d is the detachment altitude of the i-th panel considered, and $\dot{q}_i$ is the heat flux on the i-th panel.

However, to have a panel detach as early as possible, means that it has to usually be thinner and made of lighter materials. This will in turn influence the survivability of the spacecraft itself, possibly leading to more vulnerable configurations. Moreover, even the relationship between the position of the panels and their properties can be important because of the directional nature of the debris fluxes. To study how these two requirements can influence the preliminary design choices for the external configuration of a satellite, a multi-objective optimisation problem is formulated whose fitness functions (f_1, f_2) are as follows

$$\begin{aligned} f_1 &= PNP = 1 - P_p \\ f_2 &= \frac{q_{\text{int}}}{q_{\max}} \end{aligned} \tag{4}$$

Where PNP is the probability of no penetration and q_{max} is the maximum value of the internal heat flux for the solutions inside the search space of the optimisation problem.

As the nature of the optimisation problem is non-linear and requires the use of mixed types of variables, i.e. the dimensions of the panels are continuous whereas the material, the type of shielding are discrete variable, classical gradient based optimisation methods are not a viable option. Genetic algorithms were instead selected for the current problem. In particular, the Non-dominated Sorting Genetic Algorithm[10] (NSGA II) was the choice for the selection algorithm.

4 Results and Discussion

As a preliminary result we present a Pareto front for a simplified satellite structure. The satellite is box shaped, with length, width and height respectively of 3.2 m, 2.5 m, and 2.5 m. The spacecraft has a nominal sun-synchronous orbit with an altitude of 800 km and an inclination of 98°, and a mission lifetime of 7 years. For the demisability part, we chose standard re-entry conditions, with an entry altitude of 120 km, entry velocity of 7.3 km/s, and a flight path angle of 0°.

As the problem of optimising the characteristics of all the external panels of the satellite against two objective functions is not trivial we decided to start with a simple approach. We fixed the shielding type to single wall and the material of all the panels to Al-6061-T6, and let the optimiser change the thickness independently for all the panels. The search space for the panel thickness was set between 0.1 mm and 8 mm. The resulting Pareto front can be observed in Figure 1. The underlying plot shows in addition the average thickness of the structure panels for the different solutions in the Pareto front. Is interesting to observe a sort of ladder trend in the average thickness. This can be explained by the fact that after a certain thickness the panels cannot detach before the break-up altitude, thus not influencing the demisability anymore. When a panel reaches such thickness the optimiser will tend to increase its thickness to the maximum possible value in order to get the maximum survivability without affecting the demisability.

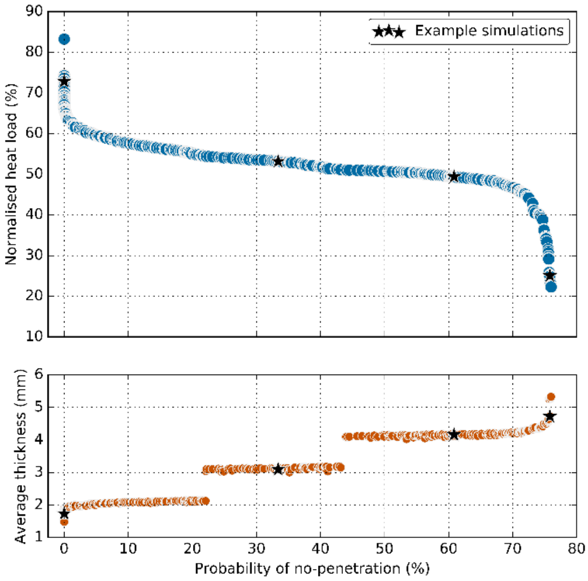

Figure 1: Pareto front for a box shaped satellite in a sun-synchronous orbit with a mission duration of 7 years.

In Figure 1 are also highlighted four solutions. These solutions have been compared using a *test component* inside the satellite main structure. We used an Al-6061-T6 cubic component with a side length of 0.8 m, a thickness of 4 mm, and a mass of 20 kg. To test the effect on the demisability, it is assumed that the test box absorbs all the heat computed through Eq. (3). The consequent temperature increase is a measure of the quality of the solution. On the survivability side, the penetration probability (P_p) on the box located in four different position inside the spacecraft have been evaluated. Four locations (close to the front, bottom, top and left faces of the structure) have been chose to consider the effect of the position and of the directional nature of the debris fluxes. The results can be observed in Table 1

Table 1: comparison between the example solutions.

Sim.	ΔT	Pp – front	Pp – back	Pp - top	Pp - left
1	479.9	1.924	0.163	0.215	0.837
2	350.4	1.913	0.152	0.205	0.318
3	325.4	1.906	0.145	0.197	0.315
4	165.3	1.906	0.145	0.197	0.314

It is possible to observe that the ΔT changes significantly for the different solutions. On the other hand, the penetration probability only have slight changes. Only for the case close to the left side a significant change of 0.5% is noticeable. This is due to different reasons. First, all the solutions obtained have a front face with the maximum allowed thickness (8 mm). This strongly influences the survivability of every internal component as the front face is the most exposed to impacts. In addition, the fluxes on the back and top faces are quite small, making them less influenced by the variations in the shielding parameters. The left side instead suffer a non-negligible flux and as the optimiser searches for more survivable solutions it increases the thickness thus causing the changes observable in Table 1.

In conclusion, despite the competing nature of the survivability and the demisability, when looking at the external structure of a satellite, it is possible to find solutions that will increase the exposure of internal component to the heat flux before the break-up to favour the demise. This seems to be possible without affecting too much the survivability of the internal component, which can exploit the directional nature of the debris fluxes to avoid a significant decrease of their survivability. Of course this are only very preliminary results and further and more complete analyses will be necessary and are currently in process to take into account multiple materials and shielding types.

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