PROGRESS IN HYBRID SPACECRAFT/OBJECT ORIENTED DESTRUCTIVE RE-ENTRY MODELLING USING THE SAM CODE

J. Beck[1], I. Holbrough[1], J.Merrifield[2], N.Joiner[2], S.Bainbridge[2]

[1] *Belstead Research Ltd., 387 Sandyhurst Lane, Ashford, TN25 4PF, UK, Email: james.beck@belstead.com*
[2] *Fluid Gravity Engineering Ltd., 1 West Street, Emsworth, PO10 7DX, UK, Email: jim.merrifield@fluidgravity.co.uk*

ABSTRACT

The SAM destructive re-entry tool has been designed to capture the most useful aspects of the spacecraft oriented and object oriented approaches to create a hybrid approach which is also suitable for statistical Monte Carlo assessments. The model philosophy is to capture the physical phenomenology, using experimental data wherever possible, and particular emphasis has been placed on the spacecraft fragmentation and material modelling. Application of this hybrid approach is described in two contexts; a toy spacecraft where a limited number of simulations were performed and vital lessons were learned, and four optical payloads where full 1000 simulation Monte Carlo runs were performed and the key aspects of spacecraft oriented modelling were demonstrated to be captured successfully.

1 INTRODUCTION

If space debris mitigation guidelines are followed, there is likely to be an increase in the frequency of spacecraft and upper stages re-entering the atmosphere. For the majority of spacecraft, some fragments will reach the ground providing a casualty risk, often due to the high performance materials which are used. The space debris mitigation guidelines also limit the casualty risk acceptable for uncontrolled re-entries. As the need to consider a controlled re-entry can become a cost driver, especially if a step in launcher class is required, there is a growing interest in designing spacecraft with reduced casualty risks. This relatively new discipline is known as design-for-demise.

Destructive re-entry analysis codes assessing casualty risks are generally separated into two approaches. The object-oriented approach models the spacecraft components as a set of simple shapes, essentially a debris catalogue, and considers a catastrophic breakup event at a fixed altitude, set to 78km in most cases. The trajectories and resultant heating to the objects are modelled in three-degrees-of-freedom (3dof) using equivalent sphere correlations for the tumble-averaged aerodynamics and aerothermal heating. The component materials are generally modelled as equivalent metals with a melting point and a latent heat of fusion [1].

The second approach is significantly more geometrically complex, with the entire spacecraft modelled using a six-degree-of-freedom (6dof) approach. The surface is triangulated, and demise is considered at the triangular element level. The physical modelling includes a predictive fragmentation model, which is usually based on a pure melt criterion, and thermal conduction [2]. In general, modified Newtonian aerodynamics is used, which provides a good estimate of the hypersonic aerodynamics on complex shapes. The aerothermodynamic heating is much more difficult to predict, and the favoured modified Lees approach has significant difficulties on concave shapes, elongated shapes, and any shape where there are multiple length scales present. For simple shapes, especially of high aspect ratio, the 3dof correlations have been found to provide better approximations of the actual heating relative to experimental data [3]. Generally, the equivalent metal materials models are also used.

The advent of design-for-demise has demonstrated that the current physical modelling of the spacecraft demise does not necessarily capture the correct material behaviour or breakup processes necessary to understand whether particular design changes will be effective in reducing the casualty risk [4]. This is mainly due to the fact that it is very difficult to obtain validation data on destructive re-entry. There is basic data on real re-entries, but this is at a level where only a basic correlative verification can be performed and thus this data can only really be used to tune basic correlations such as the fragmentation altitude criteria [5]. For detailed model validation, some material testing has been performed, but it is only recently that significant effort has started to be made to provide data against which material models and fragmentation models can be assessed in a reasonable way [6, 7]. Furthermore, comparison of the aerodynamics and aerothermodynamic heating with more complex Computational Fluid Dynamics (CFD) codes has only recently started to be considered [8].

2 SPACECRAFT AEROTHERMAL MODEL

The Spacecraft Aerothermal Model (SAM) was developed as a destructive re-entry tool for the prediction of casualty risk with aspects of both object oriented and spacecraft oriented approaches. The initial version of this tool, where the user is strongly constrained, allows a full 6dof representation of the vehicle with attitude dependent aerodynamics and aerothermodynamics. This version

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includes a number of options for a catastrophic fragmentation, inclusive of altitude and dynamic pressure based criteria. The fragments can then be propagated in 6dof, or a switch to the 3dof representation of object oriented codes can be performed.

Specific material models for conductive metals and a unique heat balance integral (HBI) implementation for insulating and ablating materials are used [9], as well as improved heating correlations to standard shapes, providing more accurate results than the standard ORSAT correlations used in many codes [10].

An expert mode where a spacecraft is constructed as a set of components connected by joints to an arbitrary level of complexity was also devised, with a set of predictive fragmentation criteria based on the failure of the joints. These criteria can be purely temperature based, or purely force based, but experience and testing has suggested a combination of these criteria is necessary to obtain results in line with basic testing [4]. With the modelling of aerodynamics and heating to arbitrary shapes throughout these simulations, the sequential failure of joints results in a simplified spacecraft oriented representation of the fragmentation process. One major advantage of the fragmentation and demise being based at the component level is that this limits the number of geometric configurations which can be reached [11]. This results in a spacecraft oriented approach which is applicable to Monte Carlo studies as many repeat configurations for which the aerodynamics and aerothermal heating are already calculated will be reached, and this part of the calculation is the most computationally expensive.

3 MODELLING DETAILS

3.1 General Philosophy

The philosophy driving the SAM tool construction is to properly represent the phenomenology of destructive re-entry in order to provide a reasonable estimate of the casualty risk. A key aspect of this is to balance the fidelity of the modelling by using a similar complexity of geometric and physical modelling. Whilst the geometric modelling in SAM is simpler than in other spacecraft oriented codes, the selection of important physical models to properly represent the destructive re-entry process has been given significant priority.

The understanding of the physical processes has driven the architecture of the SAM spacecraft model, and the representation of the destructive re-entry phenomena. Given the large uncertainties in a number of the models, and the large uncertainties inherent in a destructive process, the architecture has been designed to be accommodating to uncertainty analyses. This is considered a high priority, and only very limited uncertainty analyses exist in the field [12] until recently.

In order to construct physical models which have a reasonable validation, with the difficulties in validation having been described above, a bottom-up approach is used. Ideally, each sub-model can be validated against test data, data from higher fidelity codes such as CFD or Finite Element Analysis (FEA), and literature data. It can then be identified where validation data exists and where the models are poorly validated, and thus uncertainties are higher. There are still knowledge gaps in key areas, and testing programmes to improve the understanding of the phenomenology are a subject of current and future work which is necessary in order to have a tool capable of genuine design-for-demise.

3.2 Building a SAM Spacecraft

There are a number of trade-offs which require consideration in the construction of a spacecraft for use in a SAM analysis. The complexity of the spacecraft oriented model is driven by the number of components which are considered. Theoretically, there is no limit to this, but the current code is able to run 1000 simulations using 20-25 components within a reasonable time. Ongoing improvements to the database calculation methodologies to re-use more previously calculated data are expected to result in a significant increase in performance and thus the number of components which will be used in a standard model. Links between components are identified and a component network maps, such as that shown in Fig. 1 is constructed.

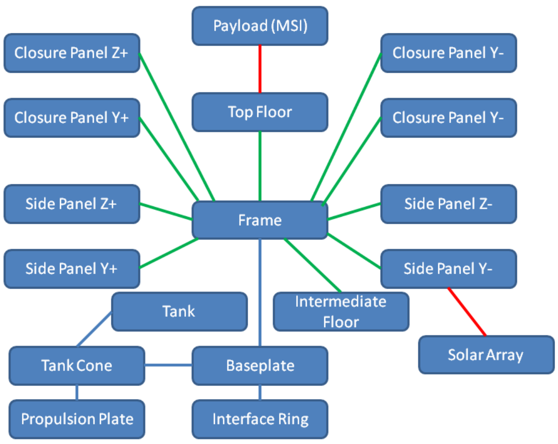

Figure 1. Toy Spacecraft Component Network

The next consideration is to ensure that the critical objects for casualty risk are identified, and included within the simulation. It is important to note that some of these can be relatively small items, such as lenses, which might be neglected if the modelling is driven by purely geometric considerations.

Once the components are selected, the spacecraft model is assembled using an unstructured mesh as shown for the Sentinel-2 Multi-Spectral Instrument (MSI) in Fig. 2. This unstructured mesh is used in the calculation of the aerodynamics and aerothermal heating.

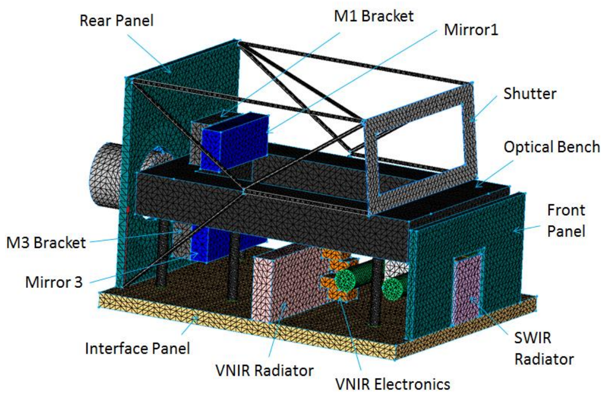

Figure 2. SAM model for Sentinel-2 MSI instrument

It should be noted that although this model is geometrically representative, it does not attempt to be an exact representation. Essentially, the accuracy of current spacecraft oriented model heating algorithms is sufficiently poor that there is no benefit in constructing a more precise geometric model. SAM has the capability to import Computer Aided Design (CAD) models if required, but this has not yet been used in practice for destructive re-entry calculations.

The SAM fragmentation model is based on the separation of the components via the failure of the joints linking them together. The current version of SAM requires a tree structure for the components as in Fig.1 and Fig.3. Fig. 3 also shows the different joint types which can be selected in the SAM model. Careful assessment of each joint is performed in order to find the weakest link. Often bolted joints are connected using inserts, or feet which connect to the main component adhesively. It is important that the expected failure is identified, and in many cases it is not the nominal 'joint' which will fail at high temperature.

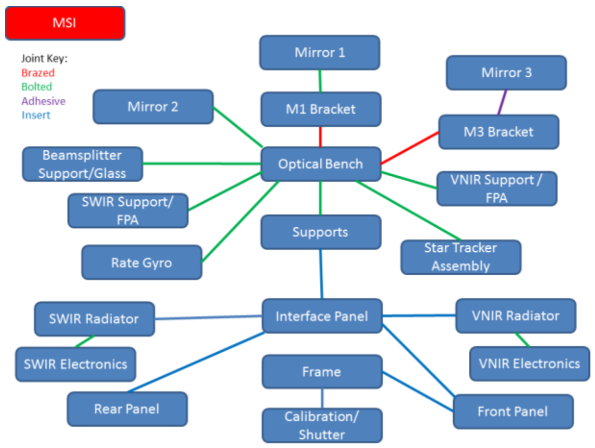

Figure 3. Sentinel-2 MSI Component Network

Some of the components modelled at the spacecraft level contain a number of parts of interest. Critical objects of interest include batteries and magnetic torquers as well as focal plane assemblies in optical instruments. Therefore, within the SAM code, these are given particular attention.

Work performed in the ESA design-for-demise studies [3] has demonstrated that the prediction of the casualty risk is highly dependent upon the modelling of the component, but that the use of a 6dof model provides essentially equivalent results to a 3dof tumble average model. Therefore, once a component has separated from the spacecraft, the SAM model switches to a 3dof representation of the component. This allows significant flexibility in modelling as well as a more efficient solution procedure.

The power of this flexibility is demonstrated in two models which have been used in ESA studies. The first is a battery model, where the predicted casualty risk from a range of different models was assessed [3]. The final battery model, which is recommended for use in other studies uses the following demise phenomenology:

- Failure of the cuboid aluminium housing
- Failure of the Glass Fibre Reinforced Plastic (GFRP) covering the battery cells
- Release of the individual cylindrical cells, modelled as a steel can, a copper layer and an internal aluminium layer.

It is useful to note that although a 3dof model is used, the significant phenomenology can be included with suitable nesting and layering of component parts.

A second example of a complex component is the beamsplitter assembly on the MSI instrument. This is separated into four parts, each with a separate material and heating input. Once the aluminium structure has melted, the remaining parts – two glass panels and the silicon carbide support – are released. These components are all highly likely to reach the ground separately, so less careful modelling would be highly likely to underpredict the casualty risk.

From this description, it is evident that the SAM destructive re-entry simulation can be considered a unique hybrid spacecraft/object oriented approach. It captures the important aspects of a spacecraft oriented model, namely a predictive fragmentation model and the heat-up of components prior to their release from the spacecraft, whilst maintaining a balance of fidelity between the geometric and physical representativeness.

3.3 Aerodynamics and Aerothermodynamics

For spacecraft fragments where there are multiple components due to the presence of an intact joint, a 6dof representation of the fragment is used. In this case, the spacecraft continuum aerodynamics is calculated using the Modified Newtonian approach, which is expected to be reasonable in hypersonic flows. An improved pressure field could be obtained by use of more sophisticated methods, but given that the fidelity of the aerothermodynamic models are substantially lower, this is considered acceptable. Free molecular aerodynamics is calculated using the methodology of Schaaf & Chambre [13]. An example of the meshes used for the aerodynamics calculations is given in Fig. 4.

The 6dof representation of the continuum aerothermodynamics is provided by the Modified Lees approach which was originally designed as a conservative engineering estimate for spheres and sphere-cone heatshields [14]. As such, this inclination based methodology is not designed for arbitrary shapes and provides very poor heat fluxes to cavities, concave shapes, and any shape with multiple length scales. A good example of this is a magnetic torquer, where the Lees model underestimates the heating by almost a factor of two on a cylinder where the radius is 16 times smaller than the length [3]. The free molecular heating is simply calculated as the incident energy flux (product of dynamic pressure and velocity) and an accommodation factor.

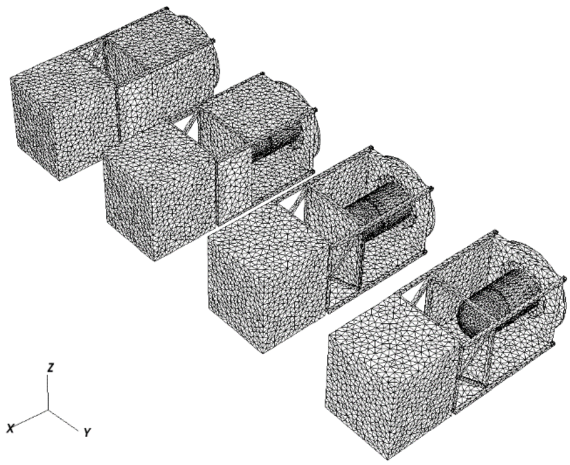

Figure 4. Aerodynamic Grids on Some Example Spacecraft Configurations

Full attitude dependent databases are calculated for each spacecraft configuration encountered during the re-entry simulation as the spacecraft fragments. These databases are stored, and in subsequent simulations, the code enquires whether the current configuration databases have been calculated and re-uses them if they already exist. This way, a few thousand simulations can be performed using a few hundred aerodynamic configurations. This efficiency is important, because aerodynamic and aerothermodynamic database calculations that include the effect of shadowing, are the rate determining step of the simulation. A simulation where no new databases are required to be calculated takes under 1 minute for the whole destructive re-entry calculation. Each database currently takes tens of minutes, although work is ongoing to make this process significantly more efficient.

The aerothermodynamic heating database is calculated per heating point, with each component having at least one, and potentially a significant number of heating points. Although, the modelling is usually simplified to one heating point per material used, this is not a necessary restriction. Once again, the quality of the heating algorithm makes geometric optimisation irrelevant. Indeed, geometric precision can give the impression that the simulation is providing a high fidelity solution when this cannot be the case given the quality of the physical modelling.

Further, as the heating is provided per heating point, there is an implicit assumption that the heating over the heating point is approximately uniform such that the lateral conduction within a heating point is accounted for. In depth conduction is also calculated for insulators. Currently, conduction between heating points, and between components, is not considered, but this is planned to be added within an ongoing activity.

Once all the joints on a component have failed, and it is released from the spacecraft, it is modelled in 3dof, and so tumble average aerodynamic and aerothermodynamic heating coefficients are required. The aerodynamics is calculated using the same modified Newtonian approach, but the aerothermodynamic heating is calculated using the running length models for simple shapes presented by Merrifield [10], which have been demonstrated as an improvement on standard methodologies in comparison with CFD simulations. For efficiency, both the aerodynamics and aerothermodynamics have been correlated such that the calculation for a given primitive shape is fast.

Given the relatively large uncertainties on the aerothermodynamic heating, this is a parameter for which uncertainties are considered in Monte Carlo assessments.

3.4 Fragmentation Modelling

One of the most important differences between object oriented and spacecraft oriented models is the capability to predict fragmentation of the spacecraft. In SAM, this is achieved through failure of the joints which link together the components. These joints can fail through a number of mechanisms in SAM, which are both temperature and force based.

A set of tests were performed on adhesive, potted and bolted joints as part of the ESA CleanSat building block [4] has demonstrated that joints can fail at relatively low temperatures which are significantly below the melt point of aluminium. Therefore, although SAM can employ a melt based fragmentation criterion, this is not used in practice as it is not consistent with the available data. Using the test data, the SAM fragmentation model currently considers fragmentation when:

- An adhesive joint is exerted to a force of 250N at a temperature above 150^0C
- A potted insert is exerted to a force of 500N at a temperature above 450^0C
- A bolted joint is exerted to a force of 5000N at a temperature above 550^0C

The tests suggest that the joint failure is also a function of the heat soak into the joint, but there is not currently sufficient data to construct a reasonable model for this effect. There is an expectation that revision of the current models, including a heat soak effect, will be achieved in an ongoing ESA study. Also under consideration for further improvement is consideration of the nature of the force applied, with different failure criteria in tension, compression and shear.

Once a joint has been considered to have failed, the spacecraft component network (see Fig. 1 or Fig. 3) is assessed to determine the resulting fragments. The components making up these fragments are then identified, and the relevant aerodynamic and aerothermodynamic databases loaded or calculated. If a component is alone, it is switched to a 3dof representation.

For example, during the fragmentation of the Sentinel-2 MSI payload shown in Fig. 2, the fragmentation pathway can proceed via the configurations shown in Fig. 5 and Fig. 6.

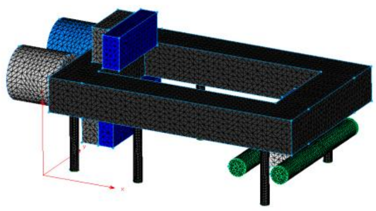

Figure 5. Partially Fragmented Sentinel-2 MSI Model

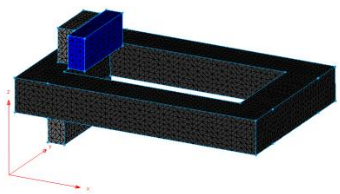

Figure 6. Further Fragmentation of Sentinel-2 MSI Model

The fragmentation models are clearly one of the weakest aspects of current spacecraft oriented destructive re-entry modelling, and thus relatively large uncertainties are used on the fragmentation criteria in Monte Carlo assessments.

3.5 Material Modelling

One of the distinct features of SAM is the emphasis placed on the material modelling. The use of equivalent materials for components such as batteries or electronic boxes is prevalent in the field, but SAM does not use such models. The key reason for this is that it is possible to obtain unrepresentative results using this methodology.

For example, consider the scan mirrors on the Sentinel-3 Sea and Land Surface Temperature Radiometer (SLSTR) payload. These have beryllium mirrors and titanium mechanisms, but there is a reasonable amount of aluminium structure. Using an equivalent material for the whole assembly results in a single object surviving to the ground – or indeed no surviving objects dependent upon the relative mass of aluminium considered. In SAM, where it is important to consider the effect of a number of materials in the composition, the component is modelled with a number of separate heating points, at least one for each material of interest. With separate heating points for the different materials in the scan mirrors, and the aluminium material separating the titanium and beryllium materials, there are always two landed objects providing a higher casualty risk.

The SAM material models consider separate in-depth conduction models for conductive materials and insulating materials. Bulk heating is used for conductors, and an HBI model for insulators [9]. Temperature dependence is considered for both specific heat capacity and thermal conductivity. As a result of a recent ESA materials study, the surface catalycity of the material is also considered, with flux reductions of about 20% being seen for titanium [7]. Sufficient data to construct SAM catalycity models for titanium, steel, aluminium and silicon carbide exist. Also within this activity, the emissivities of these materials were measured, with the values being higher than those most quoted in papers [15]. There is also some significant uncertainty on the emissivity values of materials. Therefore, the higher values have been used as the baseline values in SAM, and the emissivity has been selected as a parameter to be varied within Monte Carlo studies.

There are also a number of different possible regimes for the material failure in SAM. These are:

- Metal; these materials heat to the melt point and then the remaining heat is used as latent heat to melt the material. Demise is considered based on melt.
- Ablator; these materials undergo internal thermal decomposition which results in blowing and also undergo surface oxidation at sufficient

pressure and heat flux. Demise is considered at the point that the ablator is fully charred on the assumption that it is then weak and will break up under aerodynamic loads.

- Glass; these materials become less viscous as they heat up and eventually a liquid shear layer can be formed. This is removed by aerodynamic forces and demise is considered when the material is sufficiently low viscosity that the forces will cause a catastrophic shear failure.

Of the material models in existence, the most critical is that of Carbon Fibre Reinforce Polymer (CFRP). Many modern spacecraft are constructed of sandwich panels with CFRP facesheets, and some components also have CFRP honeycomb cores. Testing on CFRP materials has suggested that even when fully charred they are still very strong in tension, but very weak in compression [7]. This suggests that where there are hoop stresses, as in overwrapped propellant tanks, the carbon fibres might be expected to survive intact, but where compressive force are possible, they may not. There is currently insufficient data to construct a reliable general demise criterion for CFRP. In practice, SAM considers CFRP to be demised once it is fully charred, but this is a model which requires significantly more experimental verification to have a reasonable level of confidence in it.

As well as the modelling of the materials themselves, there is a question over how to model sandwich panels using a spacecraft oriented tool. The SAM model currently follows the standard spacecraft oriented approach of using low density aluminium for the sandwich structure, although some sensitivity analysis to the use of CFRP facesheets has been performed. It is interesting to note that substantially different casualty risks result from the use of CFRP facesheet models. As the sandwich panel provides the structural stiffness in the spacecraft, and that stiffness might be expected to be reduced once the adhesive bonding the facesheets to the honeycomb has reached sufficient temperature to have lost strength, it is not clear what happens to the panel integrity, especially where it is attached to relatively large masses. This is an open question, and has the potential to fundamentally affect the current understanding of the spacecraft fragmentation process. There is a clear need for some experimental data to inform sandwich panel demise modelling.

3.6 Key Knowledge Gaps

From the discussion of the construction of the SAM spacecraft and model, a set of key knowledge gaps which preclude the design of a truly representative spacecraft oriented destructive re-entry model have been identified. These are the most critical aspects where data is required in order to produce physically representative models which have some verification beyond simple correlative tuning, and are summarised below:

- **Aerothermodynamic heating to complex shapes with multiple length scales.** Some of this work can be performed with CFD tools, although a better validation would be obtained using cold wind tunnel tests on low conductivity models of compound shapes with the heat fluxes inferred from Infra-Red (IR) camera data.
- **Fragmentation phenomena.** The basic data available suggests that the current understanding of which are the important phenomena in spacecraft fragmentation is very limited, and therefore the basic parameters which need modelling are not well defined. Significant effort is required here to establish the demise process before attempts at well verified representative modelling can be attempted.
- **Sandwich panel failure.** As the initial fragmentation of the vehicle is currently understood to be down to the opening of the spacecraft via panel failure, this is of prime importance. It is not yet known whether the primary failure mode is melt, insert failure or direct panel failure, nor whether the stiffness reduction due to the adhesive bond temperature has an impact. Indeed, it is possible that different re-entries may have different failure modes.

4 APPLICATION OF THE SAM TOOL

The hybrid spacecraft/object oriented SAM methodology has been applied in two European Space Agency (ESA) projects, the first using a toy satellite to assess potential design-for-demise techniques, and the second in the concept and assessment of design-for-demise methodologies for optical payloads. Note that the focus of this paper is the understanding of the tool application and capability, and a discussion of the implications for the design of spacecraft [16] and optical payload is the subject of other papers.

4.1 Toy Spacecraft

In the first design-for-demise campaign in which the hybrid SAM model was used, the purpose was to get an indication of the effectiveness of proposed demise techniques on a simplified spacecraft oriented model. The toy spacecraft constructed, with the component map shown in Fig. 1, was approximately based on Sentinel-2. The full model is shown in Fig. 7, with different colours representing the separate components. The triangulation of the surface is shown in the different configurations of Fig. 4.

As this work took place prior to the fragmentation testing campaign [4], the fragmentation criterion was purely temperature based. Approximately 150 simulations were run at a range of initial orientations and spin rates, and with a range of demise techniques applied.

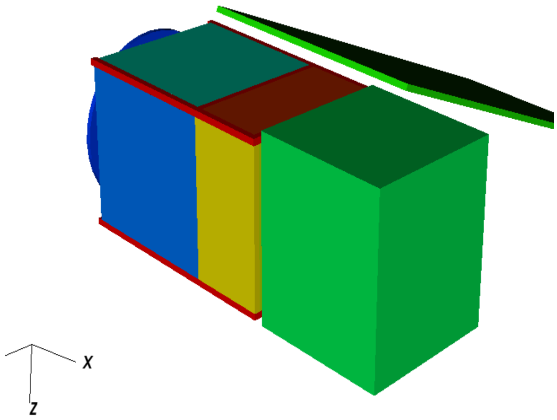

Figure 7. Toy Spacecraft

A number of important lessons were learned in the application of different models within spacecraft oriented tools, and of the performance of these tools. One technique assessed was the potential benefit of panel separation due to the failure of the inserts. There was a reasonable expectation that the inserts would fail prior to the melting temperature of the aluminium panels, which has been confirmed by the later test campaign [4]. As such, a parametric study on the panel release altitude as a function of the insert failure temperature was conducted. This is shown in Fig. 8, where the release altitudes of the four main panels from a set of three initial conditions are shown for each insert failure temperature considered.

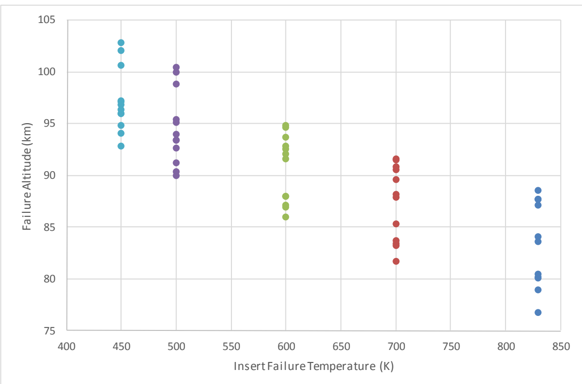

Figure 8. Fragmentation Altitudes Based on Insert Failures at Fixed Temperature

This demonstrates that there is a high sensitivity to the fragmentation model, and that some care is required when using a purely temperature based criterion. The failure at altitudes above 90km occurs where the forces on the spacecraft are very small indeed, and there is some uncertainty as to whether the components can separate when the forces are so low. This has led to the use of the force/temperature criteria described in Section 3.4 being used in subsequent work.

As the SAM model does not currently employ component-to-component conduction, an assessment was made of the importance of this aspect. This was done by considering a separate heating point for the structural panels and the equipment attached to them. In the standard case, the panels are heated, and there is no conduction to the equipment. In the full conduction case, as labelled on Fig. 9, there is complete heat soak between the panel and the equipment. The conduction to the equipment results in a very significant delay of the separation of the panels. Comparison of the results with the results from another spacecraft oriented code which includes conduction effects showed very good agreement with the standard, non-conducting, solution. Therefore, addition of a component-to-component conduction capability has not been considered a high priority and has not been employed in subsequent work.

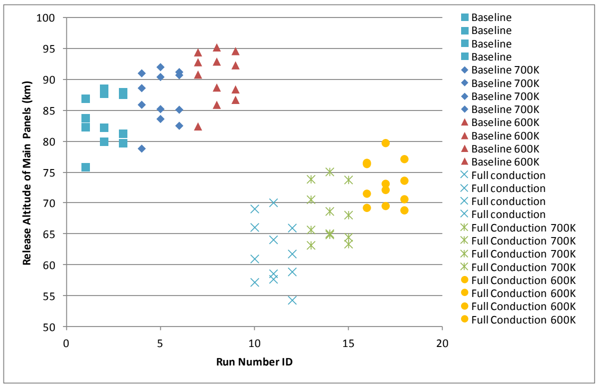

Figure 9. Fragmentation Altitudes Based on Insert Failures at Fixed Temperature with Full Conduction Model

One of the major uncertainties discussed is in the aerothermodynamic heating model. Within this work, this is most clearly seen during the fragmentation process when the panels have been removed, but the frame structure remains intact for some time. This frame structure is shown in Fig 10, and due to the small radii of curvature of the struts would be expected to heat up and melt rapidly. The fact that this is not predicted can be traced to use of the modified Lees approach. In this approach a single length scale is used, and this is determined by the projected area of the complete structure at the attitude of interest. It is evident that the predicted length scale will be substantially larger than the relevant length scale for heating of the struts, which is the strut radius. Therefore, the heating to the struts is significantly underpredicted. This highlights in a practical example the need for an improved continuum heating model for all spacecraft oriented tools.

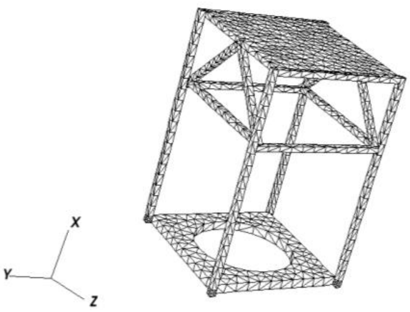

Figure 10. Persistent Frame Structure

Significantly more detail on the toy spacecraft results, and the design-for-demise techniques employed is given in [16].

4.2 Payload Demise

The second activity in which the SAM hybrid mode has been used is an ESA study on the design of optical payloads. In this work, the aim is firstly to understand the critical items from the payload, and secondly to devise and test demise techniques at the payload level in order to improve the demisability. The lessons learned from the earlier studies have been applied, as far as is practical, in this study.

4.2.1 Uncertainty Parameter Selection

From the modelling assessment in this paper it is evident that there is a high level of uncertainties in the fragmentation and demise processes, partly driven by a simple lack of knowledge of the true phenomenology of spacecraft breakup due to the lack of suitable test campaigns, and partly driven by the stochastic nature of the breakup processes themselves. Therefore, uncertainties should be a pre-requisite for destructive re-entry analyses.

As such, with the payload demise study, it was decided to run 1000 Monte Carlo runs for each payload in each configuration, whether an object oriented simulation or a hybrid spacecraft/object oriented run was being performed. In order to determine the uncertainties to consider within the Monte Carlo assessment, an initial parametric study was performed for each material type to find the most sensitive parameters. The selected uncertainty parameters for the different simulation types are given in Tab. 1.

Previous studies with spacecraft oriented codes have suggested that the demise is sensitive to the initial attitude, so this has been included with variation of the initial speed and flight path angle to account for inclination and oblateness effects. The modelling uncertainties selected reflect the particular issues discussed in this paper. The impact of the aerothermodynamic heating is relatively high, and it should be noted that ±20% is the minimum reasonable value – it is likely that the real uncertainty in the spacecraft oriented model is significantly higher.

Table 1. Uncertainty Parameters

Parameter	**Distribution**	**Range**
Aerothermodynamic Heating	Uniform	±20%
Speed	Uniform	7700m/s to 7850m/s
Flight Path Angle	Uniform	-0.05^0 to -0.5^0
Material Emissivity	Uniform	ε-0.2(1-ε) to ε+0.5(1-ε)
Initial Attitude	Uniform	Attack -180^0 to 180^0 Sideslip -90^0 to 90^0
Joint Fragmentation Criteria	Uniform	Fail temperature ± 100K Fail force ± 200N

It is worth noting that the fragmentation models employed here would be expected to produce significantly different results from other spacecraft oriented tools, especially where a purely melt-based fragmentation model is used.

4.2.2 Payload Model Construction

Within the study, four payloads have been selected for analysis. These are the Sentinel-2 MSI payload whose SAM model has already been presented in Fig. 2, the Pleiades High Resolution (HR) payload (model shown in Fig. 11), the Sentinel-3 SLSTR payload (model shown in Fig. 12) and the MetOp 3MI payload (model shown in Fig. 13).

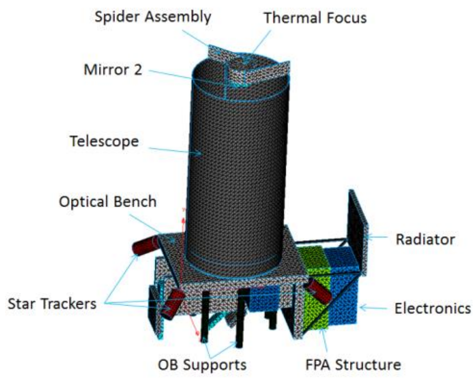

Figure 11. Pleiades HR SAM Model

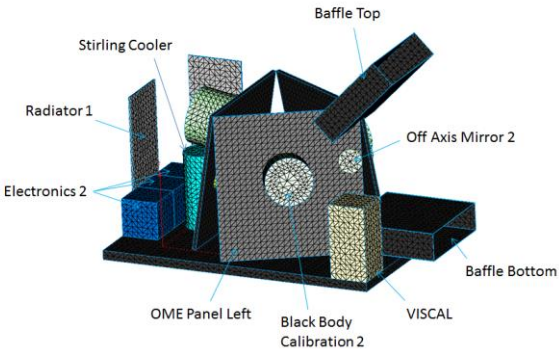

Figure 12. Sentinel-3 SLSTR SAM Model

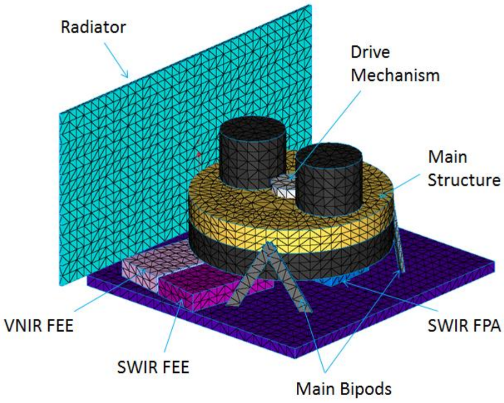

Figure 13. MetOp 3MI SAM Model

4.2.3 Simulations Performed

For each of these payloads, an initial object oriented, component level analysis was performed, again using 1000 Monte Carlo runs inclusive of a ±20% uncertainty on the nominal 78km fragmentation altitude. These results gave a first indication of the critical components for the casualty risk, but did not take account of the heating prior to the fragmentation event, nor the possibility of compound components reaching the ground. Running multiple cases suggests that the error on the mean casualty risk is of the order of 2% using this sample size.

This was then augmented by the 1000 run Monte Carlo assessments on the full spacecraft oriented models, using the hybrid approach. Inclusive of the simulation of the design-for-demise techniques, more than 60,000 spacecraft oriented simulations have been performed within the study. This allows a far fuller understanding of the statistical nature of the casualty risk than in studies using small numbers of simulations. Indeed, the authors are not aware of any other study where a Monte Carlo approach has been used in a re-entry casualty risk analysis with a spacecraft oriented model.

4.2.4 Results

Although the first three payloads are similar in size, mass and function, the demise characteristics are very different. The MSI payload casualty risk is driven by the use of silicon carbide for the optical bench and the mirrors. In the object oriented component level analysis, there are nine separate silicon carbide components, and two fused silica components which always reach the ground, resulting in the prediction of a very high casualty area.

This risk is reduced in the spacecraft/object oriented payload level analysis as a number of compound objects reach the ground where the joints have not failed between silicon carbide objects. Three different compound objects reach the ground through the 1000 simulations, and these are shown in Fig. 14. These objects land in 71%, 27% and 2% of the simulations respectively.

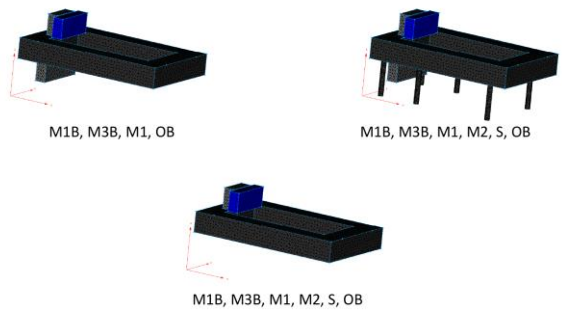

Figure 14. Different MSI Compound Objects Reaching Ground

It is important to note that three of the silicon carbide components have adhesive connections via bipods to the optical bench, and therefore no simulation results in fewer than six objects reaching the ground. This analysis predicts that the payload alone would exceed the casualty risk threshold. A casualty area of 7.5-8m^2 is approximately representative of the 10^{-4} casualty risk probability condition, depending on the orbit inclination.

Table 2. MSI Payload Casualty Area

Case	**Casualty Area (m^2)**
Component Level	12.93
Payload Level	8.33

The effect of the heating on the structural sandwich panels of the payload can also be identified as a contributor to the reduced casualty risk in the payload level simulation. Thus it can be seen that the major advantages of the spacecraft oriented models; predictive fragmentation, the possibility to land compound objects, and the heating of (partially shadowed) components prior to the fragmentation event are all captured by this approach.

The casualty risk from the Pleiades HR payload is also driven by objects constructed from materials of low demisability, namely the carbon-carbon telescope and Zerodur mirrors. In this case, there is one compound object landing in 97% of cases where the bolted joint connecting the spider assembly to the telescope does not fail. This contributes to the difference seen between the component and payload level simulations, but the larger contributor is the heating to the structural panels prior to breakup as this results in the structure being fully demised in the vast majority of the payload level runs, and only about half of the component level runs.

Table 3. HR Payload Casualty Area

Case	Casualty Area (m^2)
Component Level	8.52
Payload Level	6.41

Interestingly, in terms of design-for-demise, the most effective approaches to reduce the casualty area are different for these two payloads. For the MSI, it is possible to connect the silicon carbide elements together such that they are guaranteed to land as a single unit, which is not likely to be the case with the current configuration when careful attention is paid to the joints. For the HR payload, the undemisable items are well separated by demisable parts so this is not possible, and techniques at component level, such as material change, are necessary.

The demise characteristics of the SLSTR payload are different once again due to the critical items being constructed of demisable materials in most cases. This results in the aerothermodynamic heating prior to fragmentation being of key importance in the demise of a number of objects. This produces a substantial difference in the prediction of the casualty area between the component level and payload level analyses, even though no compound objects are predicted to land in any of the simulations. Indeed, the casualty area from the payload level analysis is almost exclusively due to parts constructed of undemisable materials.

Table 4. SLSTR Payload Casualty Area

Case	Casualty Area (m^2)
Component Level	10.01
Payload Level	3.40

For this payload, the casualty is substantially lower, which is partly due to the relative weakness of the joints which are almost exclusively insert based, and therefore can be expected to fail reasonably early. This highlights that the demise behaviour is again significantly different from the MSI and HR payloads even though this is a similar optical instrument.

The 3MI payload is distinct from the other three payloads considered in that it is refractive in nature. For this payload, the major impact on the casualty risk is from the titanium telescopes. As these contain a large number of fused silica lenses which are sufficiently large to create a casualty risk and will not demise when released from the telescopes during re-entry, the mean casualty risk is dependent upon the probability that the telescope housing will demise, releasing the lenses. As the titanium housing is relatively thin and contributes under half of the telescope mass, the ballistic coefficient of the telescope is sufficiently large that this can demise.

Table 5. 3MI Payload Casualty Area

Case	Casualty Area (m^2)
Component Level	4.54
Payload Level	3.34

In the component level analysis, this occurs in 27% of the simulations. At payload level, the heating prior to fragmentation results in an increased fraction, 37%, of the simulations resulting in the failure of the titanium housing and the release of the lenses. As can be seen in Tab. 5, this effect is not quite sufficient for the payload level mean casualty risk to exceed the mean component level risk due to the increased demisability of the structural parts due to the earlier heating, but the casualty area values are much closer in this case than in other cases. It is clear that containment of the lenses, by use of a less demisable material such as silicon carbide or carbon-carbon would reduce the casualty risk for this payload.

One of the clear outcomes of this work is that, in each case, the differences between the object oriented component level analysis and the spacecraft oriented payload level analysis can be easily understood. The impact of the partially shadowed earlier exposure to the aerothermodynamic heat flux, and the effect of joints remaining intact can be very clearly identified in the statistics.

5 CONCLUSIONS

The SAM destructive re-entry tool has undergone significant recent development, and a unique hybrid spacecraft/object oriented simulation mode has been devised which captures the key aspects of spacecraft oriented simulations in a manner which is able to re-use data such that Monte Carlo statistical assessments are tractable.

The philosophy of the modelling approach is to be faithfully representative of the physical phenomenology from the bottom-up, rooting models in experimental and higher fidelity CFD or FEA data, whilst acknowledging where the key gaps in the current understanding of some of the processes which mean that some level of uncertainty analysis must be considered. As such, particular emphasis has been placed on the fragmentation and material models employed.

Application of the tool to a set of toy spacecraft simulations has highlighted particular needs, resulting in an improved set of models which have been applied successfully in full Monte Carlo simulation campaigns to four optical payloads. The key aspects of the spacecraft oriented payload level assessment can be clearly identified.

6 ACKNOWLEDGEMENTS

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