ESA ESTEC
Keplerlaan 1
2201 AZ Noordwijk
The Netherlands

Design for Demise Guidelines for Space Debris Mitigation

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Introduction

Today's space debris environment poses a safety hazard to operational spacecraft, as well as a hazard to the safety of people and property on the Earth. Since 1957, by February 2021 more than 6000 launches have placed ~45000 metric tons of objects into an orbit. From these, in at least 23000 re-entry events ~36000 tons have come down to Earth, in a mostly uncontrolled way. Most of the objects in these events are small pieces of so-called space debris, the remnants of in-orbit explosions and collisions. As they are generally small, these pieces of space debris disintegrate and burn up while passing through the atmosphere, posing only a minor risk on ground. Contrary to small pieces, large pieces of space debris such as spacecraft and launch vehicle orbital stages, which usually can have large geometric cross sections and masses of several tons, are deemed a risk. When not designed to survive through atmospheric re-entry, these space systems produce a significant amount of break-up fragments during re-entry, some of which can withstand the aerothermal heat flux and structural loads of the re-entry and hence generate ground impacts. These impacts in turn imply significant risk levels to the population and property on ground. Theoretical analyses and object retrievals suggest that 10 % to 40 % of the mass of a large spacecraft can survive and can impact on ground. By February 2020 over 80 re-entry events creating fragments on ground have been identified as coming from space. For example, it was reported that 3 space debris collided with houses and 1 with a person [RD06]. At the same time many more objects have re-entered the atmosphere but for example many fragments such as shards of metal, parts of reaction wheels are difficult to identify as belonging to former space objects. On average 1 large space system re-enters the Earth atmosphere in an uncontrolled way every fortnight, and this number is increasing quickly.

Re-entry events are more common in the recent years due to the increase in the number of satellites in LEO e.g. mega constellations. Space debris mitigation regulations, as mentioned, impose a maximum time limit on the orbital lifetime of space objects, they imply an earlier return to Earth for some object that can otherwise be abandoned in orbit. In turn, space debris mitigation requirements state that for atmospheric re-entries of space systems the probability of serious injury or death on ground (casualty risk) need not exceed a certain threshold per re-entry event (controlled or uncontrolled). Current knowledge suggests that for medium and large space systems, this requirement can be difficult to comply with if uncontrolled re-entry is performed. If compliance is not possible, a targeted controlled re-entry is performed in order not to exceed a risk level. The implementation of a controlled re-entry, however, has a large system impact. To avoid this, design for demise (D4D) provides an alternative engineering solution to reduce the number of surviving parts that reach the ground and to reduce the associated casualty risk. The cost and difficulty to make design changes to space systems increases as a project matures, so incorporating design for demise at an early stage of a project can provide the largest benefits.

Understanding how space systems break up during re-entry is highly complex due to the physics of aero-thermo-mechanical environment that the space object encounters during re-entry. Furthermore, there are limited in-situ observations and measurements available, and it is difficult and costly to faithfully reproduce re-entry conditions on the Earth. In the past years, numerous activities have taken place across the European Space Agency with the aim of maturing of application of the design for demise techniques as well as increasing the understanding of destructive re-entry events. These activities range from the development of re-entry models to material and equipment demise characterization in plasma wind tunnels or, more recently, the development of spacecraft equipment specifically designed for demise. Although further development is still needed, the developed knowledge base has reached a level of maturity that allows the analysis of the design drivers for the on-ground casualty risk and the definition of verification procedures for the evaluation of the systems and subsystems. Furthermore, the first system level modifications dedicated to design for demise are current being implemented in ESA on flight hardware [RD07].

1 Scope

This handbook provides guidance on the implementation of Design for Demise (D4D) technologies and methodologies to support compliance with requirements related to the reduction of on-ground casualty risk. It offers an overview of available techniques, their maturity levels, and practical implementation aspects, aiming to assist engineers and project teams in selecting and applying appropriate D4D solutions throughout the design process

This is done based on the so called "Design for Demise" (D4D) engineering paradigm. The requirements from ESSB-ST-U-004 [RD02] and ESSB-ST-U-007 [RD09] provide the context in which spacecraft and launch vehicle orbital stages, including element thereof are designed to limit the risk posed by the destructive re-entry of space objects to people and infrastructure on ground and in the air. With a focus on design for demise technologies and implementation in the space segment, this document complements guidelines on analysis and ground-based testing [RD04] and modelling for casualty risk verification [RD08]. In practical terms, Section 4 of this document creates the baseline for using the design for demise paradigm in ESA projects through the aforementioned standards. In Section 5, the current state of the art and lessons learned of design for demise techniques are covered from various angles.

The intended users of this handbook are any ESA project and its partners, but specifically system engineers responsible for complete space systems and domain experts, and all technical personnel involved, in the design of equipment and structures, which can contribute to fragments surviving an atmospheric re-entry.

This document has been prepared by the ESA Design for Demise Working Group involving experts from the relevant disciplines in the ESA Technical, Engineering and Quality (TEC) Directorate and the Space Safety Programme Office (OPS-S) in the ESA Operations (OPS) Directorate.

2 References

RD01	ECSS – Glossary of terms	ECSS-S-ST-00-01
RD02	ESA – ESA Re-entry Safety Requirements	ESSB-ST-U-004
RD03	ESA – ESA Space Debris Mitigation Compliance Verification Guidelines	ESSB-HB-U-002
RD04	ESA – DIVE - Guidelines for Analysing and Testing the Demise of Man-Made Space Objects During Re-entry	ESA-TECSYE-TN-018311, 20/04/2019
RD05	ECSS – Project planning and implementation	ECSS-M-ST-10
RD06	Recovered Space Debris	https://reentry.esoc.esa.int/home/recovereddebris, Consulted February 2020
RD07	Sentinel 1C&D: A novel approach to space debris mitigation	In proceedings of the 14th European Conference on Spacecraft Structures, Materials and Environmental Testing (ECSSMET), 2016
RD08	Re-entry Modelling Procedure	PR00051/D51, Issue 3, 02/12/2021
RD09	ESA - ESA Space Debris Mitigation Requirements	ESSB-ST-U-007
RD10	European Space maTerIal deMisability dATabasE (ESTIMATE)	https://estimate.sdo.esoc.esa.int/
RD11	DEMISABLE JOINT – Clean space industrial days 2021, Thales Alenia Space	https://indico.esa.int/event/321/contributions/6387/attachments/4344/6556/2021CSID_DemisableJoint_TASI-DLR.pdf
RD12	Airbus Defence & Space. (2016). TRP - Multidisciplinary Assessment of Design for Demise Techniques	ESA - Multi-Disciplinary Assessment of Design for DEMISE Techniques
RD13	Deimos. (2017). TRP - Multi-disciplinary Assessment of Design for Demise Techniques	ESA - Multi-Disciplinary Assessment of Design for DEMISE Techniques
RD14	Grassi L. (2016). Multi-disciplinary assessment of Design for Demise techniques. [Power Point presentation]. In ESA CleanSpace Industrial Days 2016.	https://indico.esa.int/event/128/attachments/736/904/02 D4DTAS-Cleansat Industrial daysv7.pptx.pdf
RD15	OHB. (2019). TRP - Multi Disciplinary Design and Breadboarding of Technologies for Early Break-up of Spacecraft During Re-entry.	Multi-Disciplinary Design and Breadboarding of Technologies for Early Break-up of Spacecraft During Re-entry | Nebula Public Library
RD16	Thales Alenia Space. (2016). TRP-Multidisciplinary Assessment of Design for Demise Techniques.	ESA - Multi-Disciplinary Assessment of Design for DEMISE Techniques

3
Terms and abbreviated terms

3.1 Terms defined in other documents

For the purpose of this document, the following terms and definitions from ECSS-S-ST-00-01 [RD01] apply:

a. **element**

b. **qualification**

c. **space system**

d. **verification**

e. **validation**

For the purpose of this document, the following terms and definitions from ESSB-ST-U-004 [RD02] apply:

a. **casualty,**

b. **casualty area,**

c. **casualty risk**

d. **fragment,**

e. **declared re-entry area,**

f. **safety re-entry area**

g. **re-entry,**

h. **destructive re-entry,**

i. **controlled re-entry,**

j. **uncontrolled re-entry**

For the purpose of this document the terms and definitions from ESSB-ST-U-007 [RD08] apply:

a. **space object**

b. **spacecraft**

c. **launch vehicle orbital stage**

3.2 Terms specific to the present document

3.2.1 critical elements

equipment and parts of a space object that are typically identified as surviving re-entry in system level re-entry analyses, or that have been confirmed by observations and recovery as surviving an atmospheric re-entry event

3.2.2 casualty area budget

list of critical elements and their associated casualty area, and or risk, as outcome of a probabilistic re-entry risk assessment

NOTE 1 A casualty area budget is built up from the casualty areas of surviving fragments. The area budget can also be converted to a risk budget following the definitions in ESSB-ST-U-004.

NOTE 2 The budget refers to the sum of the individual casualty areas, as requirements limiting the casualty risk imply a maximum casualty area available for a space object re-entry".

3.2.3 demise

result of an ablation process acting on elements, equipment, parts or components of a space object during an atmospheric re-entry event to the extent that the resulting fragments no longer pose a casualty risk.

[ESSB-ST-U-007]

NOTE 1 An element can be a whole space object (e.g., spacecraft, launch vehicle orbital stage) or part thereof (e.g., tank, reaction wheel, magnetorquer).

NOTE 2 The hyper-surface in a phase space which defines the region of full demise for an object is denoted as the demise surface (e.g., the altitudes and entry corridor parameters for which an equipment is fully demisable).

NOTE 3 A melt-based demise event occurs when an object has become molten and melted away. For metal alloys this is the baseline demise process.

3.2.4 design for demise

intentionally altering the design of a space object in such a way that it can improve the demise of its elements, equipment, parts or components'

[ESSB-ST-U-007]

3.2.5 equipment

component, sub-system, part, or element being combined to a space system

NOTE 1 In the scope of this document, equipment covers general sub-systems, e.g., reaction wheels, tanks, electronic components, but also structural joints.

NOTE 2 This is an extension of the definition provided in ECSS-S-ST-00-01 [RD01]

3.2.6 fragmentation

end of existence of a physical connection between spacecraft parts or elements caused by interaction with the atmospheric re-entry environment

NOTE 1 Fragmentation can be caused by a melting event. This process can be used at structural level, e.g., when a bracket melts and leads to a system level or equipment structural fragmentation.

NOTE 2 Fragmentation caused by a melting event cannot be used at material level, as melting events do not create fragments but only droplets. This is considered as an ablation process and hence constitutes a demise event.

3.2.7 re-entry corridor

set of all trajectories, considering uncertainties, which can possibly be followed by a general class of space objects, equipment, or parts thereof during an atmospheric re-entry event

NOTE 1 E.g. the re-entry corridor for typical Earth observation missions can be constructed as the trajectories of a non-demisable sphere, with area to mass ratio varied uniformly between 80 and 200, a variable inclination between 65 - 100 degrees, a true anomaly between 0 - 360 degrees, starting at a geodetic altitude of 120 km [RD04].

NOTE 2 The extraction of points part of the re-entry corridor, e.g., phase space defined at first order by velocity, altitude, and flight path angle, form the necessary atmospheric input conditions to assess the fragmentation and demise needs of a designed for demise system or equipment.

NOTE 3 Trajectories can serve specific needs. E.g., spacecraft internal equipment is from a design for demise perspective not concerned with higher area to mass trajectories, as the solar panels on the parent spacecraft no longer have a considerable influence on the trajectory at the point where the equipment is released to the atmosphere. On the contrary, design for demise techniques that aim to detach spacecraft parts at higher altitudes are less concerned with lower area to mass trajectories.

3.2.8 release altitude (of an equipment)

altitude defined by the fragmentation of an equipment from the space object or a larger equipment, and its full exposure to the atmospheric re-entry environment

3.2.9 structural element

element that ensures the structural integrity of the space object during its orbital lifetime, including launch environment, providing a mechanical interface between parts of the space system

NOTE 1 Examples of structural elements include sandwich panel, metallic bracket, insert, bolt.

NOTE 2 During re-entry, the failure of a structural element can lead to a fragmentation event (e.g., failure of main spacecraft panels' structural joints).

NOTE 3 For the purpose of this document, the structural elements definition includes all physical connections that can prevent the separation of the parts from each other, including harness, piping.

3.3 Abbreviated terms

The following abbreviations are defined and used within this document:

Abbreviation	Meaning
AIT	assembly integration and testing
CDR	critical design review
CFRP	carbon fiber (overwrap) reinforced polymer
CMC	ceramic matrix composites
CoG	centre of gravity
COPV	composite overwrapped pressure vessel
CTE	coefficient of thermal expansion
D4D	design for demise
EMC	electromagnetic compatibility
H/W	hardware
LEO	low Earth orbit
LCT	laser communication terminal
MLI	multi-layer insulation
MPG	magneto-plasma-dynamic plasma generator
MTQ	magnetorquer
PCDU	power control and distribution unit
PDR	preliminary design review
PRR	preliminary requirements review
PWT	plasma wind tunnel
ROM	rough order of magnitude
RW	reaction wheels
S/C	spacecraft
SADM	solar array drive mechanism
SAR	synthetic aperture radar
SMA	shape memory alloys
SRA	safety re-entry area
SRR	system requirements review
TBC	to be confirmed
TRL	technology readiness level

4
Design for Demise Formalisms

4.1 Introduction

The requirements for re-entry safety as defined in [RD02] state that *"for atmospheric re-entering space systems, the probability of serious injury or death on ground (casualty risk) shall not exceed 1 in 10000 for any re-entry event (controlled or uncontrolled)".* Current knowledge indicates that for medium and large sized space systems, or smaller systems with hard to demise components on board, this requirement presents difficulties to comply with if an uncontrolled re-entry is performed. As per standard [RD02], if the predicted casualty risk for an uncontrolled re-entry exceeds this value, an uncontrolled re-entry is not allowed, and a controlled re-entry is targeted in order not to exceed a risk level of 1 in 10 000. The implementation of a controlled re-entry, however, has a large system impact, requiring a high thrust to mass ratio and a substantial amount of propellant. This additional mass can also result in the need for a launcher upgrade resulting in a significant increase in the costs of the mission.

Design for Demise aims to reduce the casualty risk on Earth and thereby enable compliance with the requirement for a 1 in 10 000 casualty risk per re-entry. D4D techniques aim at the intentional design of space system hardware such that it ablates during an atmospheric re-entry. The range of techniques can be targeted at the system, equipment, and material levels. In particular, D4D methods are usually classified into two main categories, depending on whether they impact the entire spacecraft, i.e. *system level*, or focus on a specific sub-systems or structural elements, i.e. *equipment level*. Often the two levels are linked, as the fragmentation of the system is needed to expose equipment to the destructive re-entry environment. The assessment of the demise of space systems and equipment implies usage of computational tools, i.e. simulations, on-ground facilities testing, and re-entry flight experiments that can be highly specific to the system or equipment under analysis [RD04].

Compliance to the requirements from [RD02] can in practice be addressed by the adoption of D4D measures. Experience with implementing the casualty risk requirement, for D4D purposes or in case of general risk assessment for (un)controlled re-entry scenario, has led to exemplary strategies in further breaking down the requirement to suit the individual project's needs. Three broad classes of examples to improve programmatically the integration of D4D engineering have been identified and can be summarised as following:

- A casualty risk assessment is seldom established in a "one-off" activity but evolves with the maturity on the design of the space object. Consequentially, there is a best practice to establish the contributors to the casualty risk, where needed with margins, at the level of the equipment of the space object and track it through the project's lifetime.
- The applicable casualty risk threshold features a strict threshold but in practice simulations and the interpretation of demisability testing show significant uncertainties. Lessons learned from projects indicate that a constructive way to design a system or equipment to demise is to provide targets in altitude, or other orbital parameters, from which the demise of an element due to the aero-thermo-mechanical conditions associated with the trajectory has a desired likelihood.
- In practice, D4D assessments at system or equipment level rely on simulations to extrapolate ground-based testing to the flight regime. To ensure consistency between tests and simulations (which are often

extensive), a good practice is to ensure the immediate application of the test level outcome by evolving the simulation capabilities at project level. Due to the complex physics nature of the demise problem, different test facilities can give parallel answers. An in-depth overview is provided in [RD04].

The computational tools attempt to capture the physical processes during re-entry and thus are thoroughly based on the thermal properties of the typically used materials. Tests are performed using on-ground facilities, often attempting to reproduce the aero-thermal and mechanical phenomena occurring upon re-entry, thus enabling, to some extent, the validation of models used in the computational tools. As much as these means are complementary, there are still large gaps and approximations, which need further understanding and leads to the need for stochastics assessment of D4D techniques. In terms of implications, the impact of a demisability requirement and the resulting system change can have an impact on the other functionality and quality requirements of the system (e.g. material changes in a tank can affect the storage potential for a specific propellant).

4.2 Applicable requirements and D4D measures

In the following paragraphs, the formally applicable requirements are reviewed from a D4D perspective, with examples provided to illustrate how they have been addressed in the past. Two D4D relevant requirements from [RD02] and two from [RD09] are given as examples.

ESSB-ST-U-004, Issue 1 Rev. 0 / 5.1.a
Requirement text: *The space system shall be designed and operated such that the re-entry casualty risk does not exceed 10^{-4} for all re-entry events.* *NOTE 1 ESA space system can include, but is not limited to:* • *ESA spacecraft, including mated configurations of space debris remediation missions,* • *ESA launch vehicle orbital stage (e.g. an upper stages),* • *ESA unmanned steered vehicles,* • *ESA return items, either recoverable or non-recoverable (e.g. return capsules, re-entry recorders).* *NOTE 2 A re-entry event can be controlled or uncontrolled.* *NOTE 3 For ESA space systems for which the System Requirements Review was kicked-off after the entry into force of the ESA Space Debris Mitigation Policy for Agency Projects (2014), if the predicted casualty risk for an uncontrolled re-entry exceeds 10^{-4}, an uncontrolled re-entry is not allowed and a controlled re-entry assuring re-entry casualty risk less than 10^{-4} is only allowed.* *NOTE 4 The re-entry casualty risk of a space system that is intentionally partitioned before re-entry is obtained by summing up the re-entry casualty risk figures of all the individual parts. Therefore, partitioning of a space system, e.g. close to the time of re-entry, can be used to influence the overall re-entry casualty risk, but cannot be used to create multiple re-entry events needed to comply individually with the re-entry casualty risk requirement.*

D4D Measure:

The system level requirement identifies the maximum risk value per re-entry event, and clarifies the limitation that partitioning of the system can still lead to an aggregation of the risk for the purpose of the requirement verification. The design for demise engineering paradigm can be used in two ways.

Firstly, it can put the focus on ensuring that in case of destructive re-entry events, the system and individual equipment pieces demise in a well understood sequence to ensure that the system level risk remains below the threshold. This can be achieved by performing in-depth analysis and testing of the elements and equipment that can potentially create critical element when a risk is identified as part of the nominal procedural risk assessment.

Secondly, design for demise engineering paradigm can identify on a case by case basis which parts need to be introduced or the equipment that needs to be modified in order to achieve a noticeable risk reduction. This can range from replacing sub-systems like tanks from hard to demise Titanium baselines to other metal alloys with lower melting points, to introducing elements in a structure that induce fragmentation (e.g. by thermal expansion) when exposed to the atmospheric re-entry environment.

ESSB-ST-U-004, Issue 1 Rev. 0 / 5.2.1.a.1

Requirement text:

The re-entry of the space system, or elements thereof, shall not result in hazards to human population, harmful contamination of the Earth environment, and damages to assets, due to:

1. *Impacting fragments*

D4D Measure:

From the D4D engineering perspective, the requirement can be addressed by ensuring that there are no impacting fragments to pose a hazard by ensuring the full demise. Therefore the process to be followed is to:

1. Identify critical elements.
2. Analyse the effect of D4D techniques on the identified critical elements and the system level impacts.
3. Re-establish the critical element list after introducing D4D changes in the system.

ESSB-ST-U-007, Issue 1 Rev. 0 / 5.5.c
Requirement text: *The expected number of casualties per re-entry of a spacecraft or launch vehicle orbital stage, including elements thereof, shall be assessed probabilistically.* *NOTE 1 When a probabilistic assessment does not result in a multi-modal distribution with a mode above the threshold, a median approach is often used to validate the threshold against at system level. In case of a mode above the threshold a bespoke assessment is often applied to assess compliance.* *NOTE 2 To demonstrate the demise of an element, equipment, part or component of a space object, a 95 % confidence level is often adopted as function of the re-entry trajectories and demise events likely be encountered by the space objects. A set of minimum uncertainties for re-entry casualty risk analysis is specified in ESSB-HB-U-002-Issue 2, and further guidelines can be found in ESA-TECSYE-TN-018311.*
D4D Measure: The system level requirement specifies that the risk value per re-entry event needs to account for the uncertainties in the processes. Whether the design for demise technique is applied at system or equipment level, it is important to take a sufficient margin when establishing the casualty area such that uncertainties in the series of fragmentation events, or extreme possible re-entry trajectories within the re-entry corridor, do not lead to impacting fragments. The common D4D measure is to identify and implement sufficiently strict fragmentation events in the system or equipment for a wide re-entry corridor during the design phase. In view of an a-priori unknown re-entry casualty risk distribution, the median value is often considered as initial estimate for the verification step. In case of a large spread when comparing the tails of the distribution, an appropriate estimator is sometimes used. Further examples are given in Section 4.3 "Example D4D requirement break down #2".

ESSB-ST-U-007, Issue 1 Rev. 0 / 5.5.c
Requirement text: *d. Spacecraft part of a large constellation in Earth orbit re-entering shall either:* *1) Have an expected number of casualties per re-entry below 10^{-6}.*
D4D Measure: The requirement is an additional tightening of the threshold for a certain spacecraft type. The D4D measures are the same as for ESSB-ST-U-004, Issue 1 Rev. 0 / 5.1.a.

4.3 Examples of requirement break-down for D4D developments.

Re-entry safety risk requirements are objective-based, i.e. they prescribe the maximum allowed risk threshold but generally do not prescribe how the risk level needs to be achieved. The D4D paradigm is based on intentional alteration of the design of a space object or its equipment in such a way that it can improve the demise under the process of friction and due to the resulting heating during an atmospheric re-entry. This generally aims to reduce the area and mass of fragments created in this process to be small enough not to create a risk when impacting on ground. As described in depth in [RD03], this is key in the calculation of the casualty area, i.e. the average surface area of an impacting fragment combined with the average area of an unprotected humans under an impacting trajectory. D4D measures and techniques are deliberate changes on conventional designs to reduce the casualty area associated with impacting fragments.

In line with the recommendation in [RD03], a major system level decision is generally taken at the System Requirements Review (SRR) to address the re-entry risk requirements: to implement a controlled or uncontrolled re-entry sequence. D4D measures to reduce the casualty risk are thus most effective before SRR, in order to avoid significant changes to the spacecraft or launch vehicle design after. In order to deal with this, some projects have implemented the following:

- A budgeting logic, in terms of casualty area and risk per subsystem as a way of identifying the major contributors.
- Links between system and sub-system level, and
- Tracking the overall progress across the development stages.

D4D techniques imply fundamental changes to a space object's design and as such it is important that they are addressed early on, e.g. Phase A of mission development, to avoid major design changes later on. The following best practices can be used:

Example D4D requirement break down #1: tracking casualty risk budget across a project
Example requirement text: *The casualty risk of the system shall be presented as a casualty risk budget and updated at each project development milestone.*
D4D Measure: Following the mission phases defined in [RD05], a casualty risk budget can be established and traced as input toward the casualty risk requirement verification when D4D measures can be implemented. For the best practice below, casualty area is used as metric as it is independent of the re-entry epoch but can easily be converted in casualty risk when this epoch is known. <u>Phase 0</u> While it is acknowledged that the design of the susbsystem and system level geometry, and launch date, is not well known in a Phase 0, a casualty risk budget can be constructed along the following lines: • Establish a list of the masses, areas, and most common materials for each part of the intended space object. The associated casualty area can then be determined using object or component-oriented tools or a Rough Order of Magnitude (ROM) estimate, in accordance with [RD03].

- Broad assumptions on the fragmentation (e.g. fragmentation at 78 km for an uncontrolled re-entry from LEO) of the full space object can be made in order to derive the area. The use of a model can provide better insights when they implement a physics driven (e.g. thermal) break-up rather than an altitude induced one.
- potentially critical items envisaged for the system design are generally highlighted in the list, as is the equipement that contains them. For example, when the amount of mirrors in an optical design is not be known, the mirrors and associated lenses can be flagged as an area within the risk budget.

Phase A

With the draft system design budgets being established in the run up to a Preliminary Requirements Review (PRR), sub-system and system level design and accommodation aspects mature to the extends that a modelling of the space system for a re-entry risk analysis is possible. The casualty risk budget can be updated along the following lines:

- A list of all possible critical elements and elements that can possibly trigger fragmentation are derived from the system design with support of a re-entry simulation model (e.g. component-oriented).
- All identified critical elements have their casualty risk area identified under a probabilistic assessment, e.g. the maximum, minimum, and 90, 50, and 10 percentiles are extracted from the simulation data to give an overview of the impact of a specific element on the spacecraft's casualty risk.
- The sub-system containing critical elements that contribute significantly to the casualty risk budget can be used as input for design for demise technique or for identification of alternative sub systems that have been designed to demise.

Phase B1

At SRR the casualty risk budget can be presented to assess the need for a controlled re-entry based on a preliminary analyse of D4D solutions to reach the casualty risk threshold (if needed). The list of critical elements can be consolidated in general at SRR based on the re-entry break-up modelling. Often the models used at this stage of development are already suitable to predict the fragmentation events that drive the casualty risk budget. Optimisation or targets for the development of design for demise equipment can be extracted at SRR.

Phase B2

At the Preliminary Design Review (PDR), in case the decision at SRR was made to opt for a D4D solution or an uncontrolled re-entry in general, the casualty risk budget can be further matured based on the consolidated re-entry risk analyses. When a novel D4D technique is applied, it is the best practice to have a test and verification plan for Technology Readiness Level (TRL) raising (e.g. for material changes or fragmentation triggers), in line with other qualification criteria (e.g. material behaviour in case of a D4D tank). An extensive list of testing options to acompany a D4D development is presented in [RD04].

Phase C

The casualty risk budget generally finalised at Critical Design Review (CDR), in line with the system level design and confirmation that any D4D technique is mature enough for implementation. Generally at this stage, the casualty area is still used as metric, to enable updating the risk in later phases if the design of the space object does not alter.

Example D4D requirement break-down #2: Linking system level and equipement level casualty area budgets
Example requirement texts: *Structural elements shall fragment and guarantee the release of the included equipment in line with the demise requirements for these equipment.* *The re-entry corridor resulting from the system design shall be used to verify equipment level demise requirements that flow down from the system re-entry casualty risk requirement.* *The equipment shall demise when released at geodetic altitude of To Be Confirmed (TBC) km or above assuming an initial release temperature of TBC K, with a 5 % significance level for a TBC re-entry corridor.* *Given a range of physical conditions TBC at geodetic altitude range of TBC km, the equipment shall fragment at a 5 % significance level for a TBC re-entry corridor.*
D4D Measure: There is a direct link between the system level design of a spacecraft or launch vehicle and the criticality of an element contained in that system. For example, the amount of impacting fragments expected from a reaction wheel subsystem can vary between 0 and 4 as a function of the altitude at which the subsystem is exposed to the flow field of a destructive re-entry [RD04]. Moreover, this relationship with altitude is generally not monotonic, e.g. in the example of a reaction wheel the highest number of fragments in generally reached when it is exposed to the aerothermal flux at the moment the system undergoes peak heating between 60 km and 75 km, and less fragments are created outside this interval. To ensure a correct implementation of D4D strategies, requirements are linked and it is importat to have a casualty risk budget that accounts for D4D techniques applied at either system or equipment level. For example, an early break up of the system platform during the re-entry is usually beneficial, however depending on the demise behaviour of the specific equipment this can also result in an increased casualty risk. In the example requirements above, a system level link is created to a contained equipment. This can equally apply to a complex instrument and a contained element, such as lens in a telescope array. The use of such requirement generally occurs after Phase A, as in early phases the re-entry risk assessment can still be based on fixed fragmentation criteria such as reaching 78 km instead of physics models which are more suitable for D4D analyses. In case a D4D requirement is placed on the system or equipment level, the targeted critical element can be removed, or its risk reduced, from the casualty risk budget. A full set of example applications can be found in [RD04]. Specifying a significance level for a D4D requirement is in accordance for the requirement on probabilistic casualty risk assessment defined in [RD09]. I.e. a Monte Carlo analysis is in practice not solely executed on a maximum amount of samples, but rather executed until convergence of the percentiles under the uncertainties stated or derived is reached. The following two methods are often used to assess convergence: 1) Convergence plots of the estimators (value vs number of MC samples), or 2) The coefficient of variation of the estimator, i.e. the ratio of estimator θ standard deviation over estimator itself ($\sqrt{V[\theta]}/\theta$). In view of the actual problem at hand, the convergence is best demonstrated for each statistical quantity in use (e.g. mean, variance, median, confidence intervals), as having convergence for one of them does not imply convergence for all of them.

5
Design for Demise Techniques

5.1 Overview

As the demise physics is highly complex, and currently not fully predictable, ESSB-HB-U-003 handbook is based on lessons learnt and best practices that have been developed during various R&D activities [RD11], [RD12], [RD13], [RD14], [RD15] and [RD16]. They enable the definition of guidelines and criteria for demise verification at system, equipment, and material levels. The content of this Section focusses on the systematic overview of design for demise techniques and mature demisable technologies under development. In the past years several design for demise techniques have been proposed and investigated, to the extent that different methods have reached different level of maturity (e.g. TRL) and implementation readiness. However, the area of demise methods is still open for the inclusion of innovative concepts and growing fast.

The overall objective of Design for Demise is to reduce the casualty risk associated with impacting fragments of space object hardware (H/W) on ground population during an uncontrolled re-entry. In general, the casualty risk is driven by:

- The H/W, i.e. by part of the hardware surviving in terms of number of pieces and casualty.
- The human population density on ground, accounting for its evolution in time and geographical repartition.

The reduction of risk versus human population on ground is mostly driven by the mission parameters prior to the destructive re-entry, in terms of

- Orbit inclination.
- Re-entry date.
- Re-entry scenario.

The content of ESSB-HB-U-003 is mainly focused on the reduction of the risk related to spacecraft H/W. Other aspects related to the casualty risk analysis and risk reduction for population by other means not concerning the modifications of the spacecraft design H/W are out of scope of the current document and mainly addressed in the space debris mitigation handbook [RD03].

About spacecraft design changes, a top-level overview of D4D techniques is provided in ESSB-HB-U-003. In particular, the justifications behind the use of these techniques, and therefore the criticalities in terms of spacecraft demise and survivability, are presented in Section 5.1 considering the different layers of applicability: at material, equipment (and component), and system levels. While driven by applications on spacecraft, the same techniques can be applied to launch vehicle upper stages. Reversely, in Section 5.2, the design for demise implementation aspects is presented, addressing the lessons learned of the design for demise techniques and practical examples (demisable technologies) at different TRL, together with the implications coming from the design changes. The two sections can thus to a certain extend be overlapping in terms of content but provide a convenient and targeted entry point for different uses of ESSB-HB-U-003 depending on the information sought.

In the Figure 5-1, the current logic identified to reduce the casualty risk is presented, with focus on aspects related to the reduction of the risk coming from the spacecraft hardware (left side of the Figure 5-1). The coloured boxes summarise the current categorisation of design for demise techniques: in particular in red, techniques aiming at increasing the overall demise of the hardware, and in blue, the methods aiming at containing the fragments released during an atmospheric re-entry. In both cases, the objective is to reduce the total casualty area of the re-entering spacecraft.

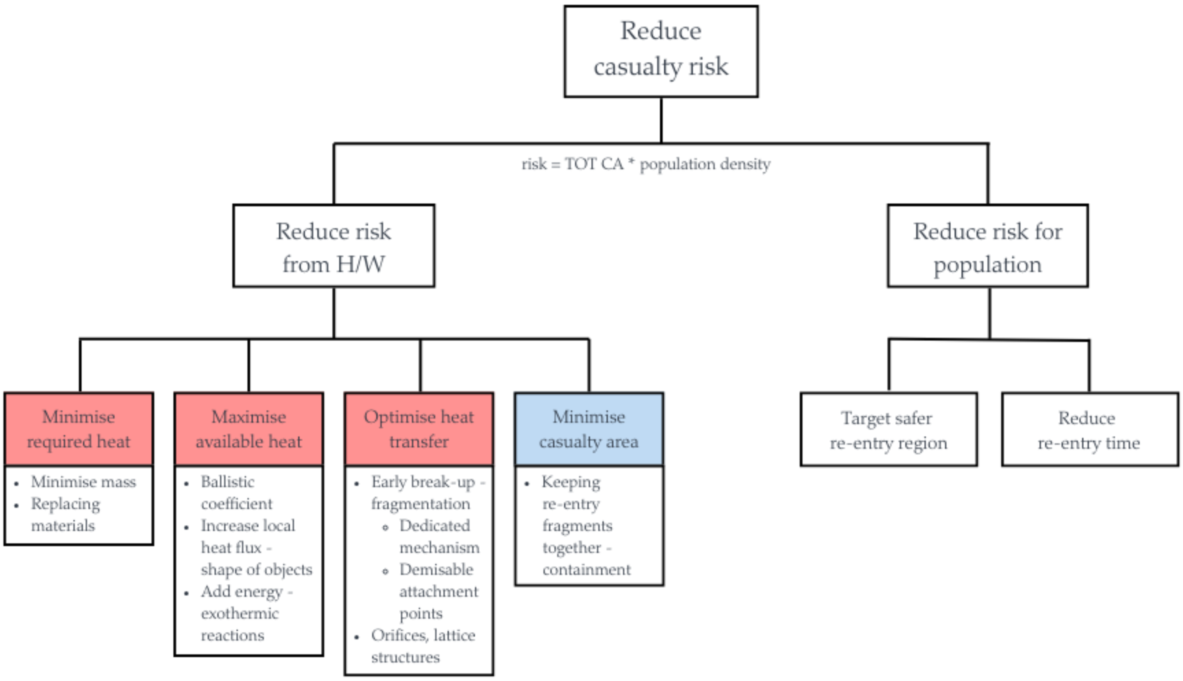


Figure 5-1: Casualty risk reduction conceptual scheme

The reduction of risk from H/W presented at Figure 5-1 is mostly achieved by:

- **Minimization of needed heat load for the ablation process**: This involves reducing the mass of the spacecraft or its equipment by decreasing thickness, altering the area-to-mass ratio, changing the manufacturing process, or replacing materials. For example, using materials with a lower melting point or lower emissivity can initiate the ablation process earlier. Alternatives include reducing the size of components or using layered structures, such as dividing a larger component into smaller elements when possible.
- **Maximization of available heat load to improve ablation of the entire system**: This can be achieved by changing the shape and mass distribution of the spacecraft and its components. For instance, placing critical internal components at the edges and corners of the structure can increase heat flux, affecting the ballistic coefficient or attitude during entry. Another approach is to add energy by planning and causing exothermic reactions during re-entry.
- **Optimization of heat transfer**: Internal equipment is often shielded from heat flux by external structures. Achieving an early break-up or incorporating strategic orifices, open structures, or lattice structures can expose internal components to heat flux earlier, enhancing their demise.
- **Minimization of casualty area**: Unlike the other methods, this approach focuses on reducing the overall casualty area by minimizing the number of fragments released during re-entry. This can be achieved

through containment techniques, ensuring that surviving fragments land as a single piece rather than multiple pieces. Techniques include regrouping, attaching, protecting, or encapsulating spacecraft elements.

Applying these techniques can impact the overall spacecraft and mission design, such as reducing casualty risk, changing mass, and affecting the chemical compatibility of materials. The following aspects are considered when these implications are generally known:

- **Demisability**: How much can the casualty area be reduced?
- **Applicability**: Can the technique be applied to many classes of spacecraft or only a few?
- **TRL (Technology Readiness Level)**: How close is the technique to being used? What is the likelihood of successful development?
- **Development cost**: How expensive is it to develop?
- **Recurring cost**: How much can it increase the cost of a mission, considering parts and design effort?
- **Systems impact and trade-offs**: Can the proposed technique lead to disadvantages such as higher mass, lower reliability, or shorter lifetime? Are significant changes in spacecraft design needed?

Some examples of the application of these techniques to technologies are presented in Figure 5-2.

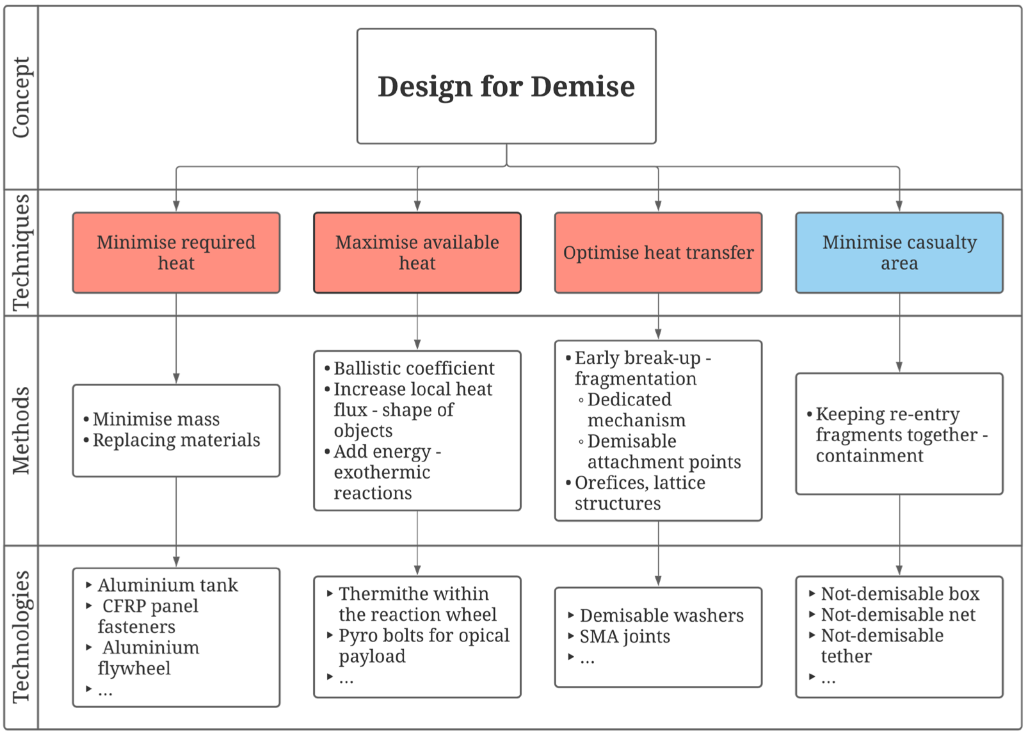


Figure 5-2: Design for demise – from the concept to technologies

5.2 Top Level Overview – Why?

5.2.1 Material Level - Identification

The demisability of materials is influenced by two main conditions:

a. **Heat and Melting Capability**: This refers to the ability of materials, particularly metals, to heat and melt quickly below an ablation limit. The three key parameters defining this limit are:

   1. **Melting Temperature (Tm)**: Preferred to be lower than ~1000 °C for D4D purposes.
   2. **Heat of Demise (Qdemise)**: Preferred to be lower than ~1000 W/Kg, implying a low melting temperature, low specific heat capacity, and low heat of demise.
   3. **Radiative Cooling**: Preferred to be low in terms of emissivity, implying high absorptivity of heat in the material.

      NOTE Materials that meet these criteria are promoted in the D4D material selection, while those exceeding these limits are ideally reduced or replaced with alternatives.

b. **Disintegration Capability**: This refers to the ability of materials to disintegrate and produce fragments with an impact velocity below an energy limit of 15 J. Materials in this category generally have a low ablation limit, such as Aluminium, or are brittle when exposed to the re-entry environment.

The main materials involved in hardware replacement for D4D purposes include:

- Stainless Steel.
- Titanium.
- Tungsten.
- Ceramics: SiC (Silicon Carbide), Si_3N_4 (Silicon Nitride).
- Ceramic Matrix Composites (CMC): C/C (Carbon/Carbon), C/SiC (Carbon/Silicon Carbide).
- Carbon Fiber Reinforced Polymer (CFRP).
- Glass Fibre Reinforced Plastic (GFRP).

Stainless steel and Titanium alloys have a high ablation limit and are often used in space object design for mechanical structural reasons. While few concrete alternatives exist, high hardness Aluminium alloys and high-performance thermoplastics (including those charged with magnetic metallic fibers) are available. Innovative materials like graphene offer interesting electrical properties, but their structural mechanical properties differ significantly from stainless steel.

Ceramics and CMCs are often chosen for their high mechanical structural or high-temperature range properties, sharing similarities with Titanium in terms of thermal stability or expansion properties. Additionally, materials such as INVAR, Beryllium, Nickel, and Inconel can pose a risk of casualty on the ground, depending on their application.

5.2.2 Material Level - Replacement

For mechanical resistance dimensioning and engineering, the main drivers are stiffness, which is the capability to withstand loads with minimal deformation, and stress limit, which is the capability to withstand loads with minimal material section. These design drivers for material selection are influenced by mass, specifically material density, according to the following ratios:

- Stiffness over Density.
- Stress Limit over Density.

A less stiff or strength-resistant material can be replaced by lighter ones with more section to withstand the same loads. This technique achieves the same mechanical resistance at the same mass through increased section but reduced material density. However, this approach is not favourable for small parts, as increasing the thickness of non-hollow parts can affect their overall dimensions.

Other parameters to consider for material selection for space parts include:

- Mechanical hardness or fracture sensitivity.
- Coefficient of Thermal Expansion (CTE) for stability or high-range operational temperature.
- Electric and Magnetic Properties (conduction or insulation).
- Thermal properties (conductivity, insulation).
- Radiation (shielding capability or radiation compatibility).
- Process compatibility (welding, machining).
- Fluid compatibility (investigated per fluid on a case-by-case basis, with alternative materials potentially enhanced via resistive coating to increase compatibility without losing demisability)

Below is a list of some materials that can substitute the critical ones mentioned in Section 5.2.1:

- **Alu-Li alloys**: Aluminium-lithium alloys can challenge recent carbon fibre structures due to their lower density and higher modulus.
- **Al-Be Alloys (AlBeMet)**: An Aluminium alloy with excellent structural qualities at very low density, similar to Beryllium alloys. It has a comparable melting temperature and lower density than Aluminium, with equivalent endurance and CTE of stainless steel. This material has been used in space optical applications, but its thermal parameters for Qdemise need clarification.
- **Al-Si Alloys**: These materials have controlled CTE, based on Al-Si alloys with 12 °C – 70 °C of Si, allowing CTE between 5 ppm/°C – 20 ppm/°C. They offer good mechanical strength at high temperatures, high thermal conductivity, low heat capacity, and good electrical conductivity, making them potential Titanium replacements for CTE reasons.
- **Al compounds with fibres (Al-C, Al-SiC, Al-CNT)**: These innovative materials achieve attractive mechanical characteristics by adding fibres to aluminium metallurgy. Their demise characteristics are driven by the aluminium matrix, and they can benefit from recent fibre technology advancements.
- **High-Performance Thermoplastics (PEEK, PAEK, PEI, PEK)**: Useful for mechanical parts, thermal resistance, magnetic, and conductive parts. Further development is needed to confirm suitable applications for mechanical resistance.
- **Innovative Fibres (Kevlar, Carbon, Nextel, Spectra, CNT, BNNT)**: These fibres can be re-used to enhance aluminium or other materials for overwrapping or mixing in a metallurgic compound, providing a different order of magnitude in mechanical resistance to mass ratio. However, their TRL and application process are still low.

- **Thermites**: A pyrotechnic composition of metal powder, fuel, and metal oxide that undergoes an exothermic reaction when ignited. This technology is being investigated for D4D for a reaction wheels ball bearing unit, currently at TRL 4.
- **Mg Alloys**: Magnesium alloys, known for their highly exothermic behaviour, can be grouped with titanium to provide high mechanical resistance and easier demise.

5.2.3 Equipment Level

The critical elements listed here have been identified through various studies and mission-level casualty risk assessments. They have been identified based on different levels of simulation tools recommended through [RD03], ranging from component-oriented tools (DRAMA, SAM) to spacecraft-oriented ones (SCARAB, PAMPERO) actively used in European industry. This list is often used to guide a casualty risk budget starting at Phase 0.

- **Propulsion tanks**: These tanks, if made of Titanium or CFRP overwrapped, can survive re-entry due to their voluminous nature, resulting in a large casualty area. Even "demisable" aluminium tanks can survive if shielded by the spacecraft structure for too long.
- **Reaction wheels**: These often survive re-entry but are smaller in size. Typically, there are four on a spacecraft, making them more critical than a single tank in a casualty risk budget. Their aluminium housing is easily demised, but the flywheel can survive re-entry (with some ablation) in the case of medium and heavy flywheels (mass > 1 kg). The ball bearing unit and the shaft are also considered critical items. Changing the material or exposing the flywheel to external aerothermal flux early can lead to its demise.
- **Magnetorquers (MTQs)**: Depending on their size and accommodations, small MTQs are expected to demise, whereas larger ones can resist re-entry if shielded. Like reaction wheels, there are typically three units on board a spacecraft. The aluminium housing and copper wiring usually demise, but the core can survive re-entry with ablation in the case of medium and heavy cores (mass > 1,5 kg).
- **Mechanisms**: Mechanisms like Solar Array Drive Mechanisms (SADM) or Instrument pointing mechanisms are likely to partially survive re-entry. Often, only a very small part inside the mechanism core (e.g., the gears) is predicted to impact the ground.
- **Optical payloads**: Made of ceramics or optical glasses, these are major critical elements. Ceramics are very heat-resistant and survive re-entry. Zerodur mirrors and large lenses in optical glass (e.g., BK7) are likely to partially survive.
- **Large SAR antenna parts**: Particularly in the central panel protected by the platform during the early phases of destructive re-entry, these can also survive.
- **Large metallic parts**: Large pieces of primary structure, manufactured in Stainless Steel, Invar, or Titanium, can also survive.
- **Batteries**: Historically, NiH_2 batteries were a hazard. Modern Li-ion batteries, with higher fidelity models, are shown to demise. A recent activity showed that Li-ion batteries with small ABSL cells are expected to completely demise from most release altitudes of interest, while large cells show lower demisability, posing a risk. The fragmentation is driven by GFRP, and it is necessary to investigate larger cells to provide demisability improvement ideas.
- **Laser Communication Terminal (LCT)**: This optical inter-spacecraft communication system can be large and heavy, made of poorly demisable materials, with a large casualty area.

- **Small Balance Masses**: Masses less than 20 kg made of aluminium are demised during re-entry; heavy balance aluminium masses (mass > 20 kg) and those made of critical materials such as steel are prone to survive.
- **Star Tracker sub-components**: Critical elements such as the focal plane array housing and optical barrel, both made of titanium or high-end lenses, can survive. A recent activity showed that large star trackers are expected to land in one piece, while small ones have a probability to demise.
- **Electronics cards**: Made of GFRP, these materials are significantly less demisable than Aluminium and likely to reach the ground. Demisability for these components is difficult to assess due to GFRP's very low demisability but also very low strength when hot, leading to shape changes and material tearing where forces act. It is expected that cards survive re-entry. Large components such as transformers also pose a potential risk.

The most important parameters for the demisability and survivability of a critical item are:

- Material.
- Mass and dimensions.
- Ballistic coefficient and shape.
- Accommodation in the spacecraft.
- Altitude of exposure and separation altitude.

Table 5-1 summarizes the list of most identified critical elements and provides additional critical items found on-ground in some simulations. The reasons impacting the criticality of the elements are also presented.

Table 5-1: list of potential critical objects

Object	Criticality	Reason
Battery	LOW	Cells have steel can and large number of cells
Electronics Card	MEDIUM	High GFRP failure temperature
Fill & Drain Valve	HIGH	Titanium part
Gyroscope	HIGH	Titanium housing
Magnetorquer	LOW	Magnetic core (higher melting point than steel)
Tank	HIGH	Titanium material
Reaction Wheel Shaft	MEDIUM	Steel material; multiple objects
Reaction Wheel Flywheel	HIGH	Steel material; multiple objects
Solar Array	LOW	Low ballistic coefficient
Structural Panels	LOW	Low ballistic coefficient (demise in component-based model)
Mirrors (Zerodur)	MEDIUM	Zerodur material
Mirrors (SiC)	HIGH	Ceramic material
Thrusters	MEDIUM	Inconel material
Optical payload fixings	HIGH	Invar and titanium materials
Solar Array Drive Mechanism	HIGH	Steel central shaft

Star Tracker	HIGH	Internal titanium parts
Lenses	HIGH	Silica material

The starting point to identify the more efficient D4D techniques to improve demisability of critical elements is to clearly identify their reason of survivability. In Table 5-2 and Table 5-3, each common critical element is summarized with the main reasons of survivability and the consequent possible approaches and implementation strategies that can be adopted to improve their demise.

Table 5-2 : List of platform critical elements and demise approach

Platform Critical element	Reason of survivability	D4D technique	Implementation strategy
Tank	Heat of demise	Minimise needed heat	Equipment level
		Reduce the number of fragments	System level
Reaction wheels	Heat of demise	Minimise needed heat	Equipment level
		Reduce the number of fragments	System level
Magneto-torquers	Late exposure and high mass	Minimise needed heat	Equipment level
		Optimise heat transfer	System level
		Reduce the number of fragments	System level
Flexure / struts / bracket / mounting / frames	Heat of demise	Minimise needed heat	Equipment level
		Reduce the number of fragments	System level
Power Control and Distribution Unit (PCDU) and big electronic box	Heat of demise (internal small components) Late exposure and high mass	Minimise needed heat	Equipment level
		Optimise heat transfer	Equipment level System level
		Reduce the number of fragments	System level
		Reduce below 15 J	Equipment level
Balance mass	Heat of demise Late exposure and high mass	Minimise needed heat	Equipment level
		Reduce below 15 J	Equipment level
Mechanisms / SADM	Heat of demise Late exposure and high mass	Minimise needed heat	Equipment level
		Optimise heat transfer	System level
		Reduce the number of fragments	System level
	Heat of demise	Minimise needed heat	Equipment level

Platform Critical element	Reason of survivability	D4D technique	Implementation strategy
Thrusters		Reduce the number of fragments	System level
Star Trackers	Heat of demise	Minimise needed heat	Equipment level
		Reduce the number of fragments	System level
Fill and Drain valves	Heat of demise	Minimise needed heat	Equipment level
Gyroscopes	Heat of demise	Minimise needed heat	Equipment level
Harness	Modelization granularity	Not critical	
Batteries	Modelization granularity	Not critical, depending on the cells size	
CFRP panels	Aerothermodynamics Model	Not critical	

Table 5-3 : List of Payload critical elements and demise approach

P/L Critical element	Reason of survivability	D4D approach	Implementation strategy
Optical Payloads (Optical benches (Beryllium, Silicon Carbide, Super Invar)) **Telescope mirror (Beryllium)** **Titanium frames (bipods, flexures and other optical payload fixings)** **Baffle**	Heat of demise Late exposure and high mass Model granularity	Reduce heat load to demise the fragment	Equipment level
		Increase the heat rate	System level
		Reduce the number of fragments	System level
		Reduce below 15 J	Equipment level
Coded instrument masks	Radiative survival	Reduce heat load to demise the fragment	Equipment level
		Reduce below 15 J	Equipment level

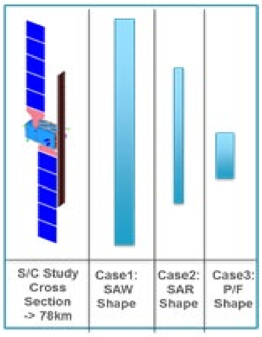
5.2.4 System Level

5.2.4.1 Introduction

During atmospheric re-entry, the system plays a crucial role in determining the initial release conditions of both external and internal equipment and parts after the main platform structure breaks up. The ballistic coefficient of the platform, along with the joint connections between the main structure and sub-systems and equipment, as well as the connections within the subsystems, are fundamental in the fragmentation and demise process of the entire spacecraft. Therefore, it is important that the system ensures the break-up and exposure of the equipment to the heat flux to facilitate their demise.

5.2.4.2 Ballistic coefficient influence

The ballistic coefficient is generally considered at the spacecraft level because the early re-entry phase determines the speed at break-up, which in turn affects the heat intensity received by all internal elements. For example, past analyses of different spacecraft ballistic coefficients on a large LEO platform, considering various cross sections (platform alone, with SAR panels, and with the addition of solar panels), have shown that a higher ballistic coefficient of the parent spacecraft benefits all the child hardware elements. Specifically, a higher demise altitude for the platform results in a higher ablation rate on the critical elements, as demonstrated in Figure 5-3.



		(GS shape)	(SAR shape)	(S/C shape)
		Case 1	Case 2	Case 3
S/C Outputs	Reference are/cross section (m^2)	40	10	5
	Velocity at 78 km (m/s)	6980	7560	7720
	Ballistic coefficient at 78 km	35	150	270
	Downrange at 78 km (m)	19960	47180	75420
Al full reference sphere	Diameter (cm)	10	15	15
	Demise altitude (m)	55420	56260	59120
Ti full reference sphere	Diameter (cm)	2	4	4
	Demise altitude (m)	63790	64240	66660

Figure 5-3 : Example of the comparison of different ballistic coefficients on the demise process

As the ballistic coefficient of the parent spacecraft can be a significant driver, it is important to carefully consider and optimise it when used as a D4D technique. The following rule of thumb are derived to improve demisability:

- Higher mass and lower cross section:
  - Compact platform with high internal architecture density,
  - Few external appendages (or released before re-entry).
- Lower Aerodynamic Drag and coefficient:
  - Bespoke aerodynamic shape, or
  - Design for a cylinder shape instead of box (the former has a lower ballistic coefficient).
- Low Tumbling & Spin:
  - Stabilised attitude to induce directional heating.

Consequently, to improve this ballistic coefficient, the intention is to release all appendages and elements providing drag and no added mass can be investigated and then the technological mechanisms to be activated by the re-entry heat can be considered as well. However, implementing solution have e.g. solar arrays remain attached for longer changes the trajectory, as they act as aerodynamic drag devices.

5.2.4.3 Anticipated preheating

Considering that internal equipment inside a spacecraft is submitted to aerothermal exposure only after the main spacecraft break-up, a D4D technique is to anticipate this heating and receive more heat and sooner. By design:

- Release early and intentionally critical elements into the aerothermal flux.

- Design deliberate fragmentation to guide the spacecraft break-up or controlled dismantlement.
- Embark venting holes or aperture devices.

5.2.4.4 Break-up altitude and controlled dismantlement

Better exposure of internal elements can be achieved through designing for in-orbit fragmentation or even intentional break-up at higher altitudes (Figure 5-4). A strut-based mechanical architecture, using tightening clamp band devices to join panels to the central tube, can reduce the number of interfaces released during an intentional break-up. However, it is important to approach this technique with caution. It is important that all elements released earlier demonstrate their ability to demise if released at a given altitude. For example, due to their own ballistic coefficient, the elements can slow down significantly, which can degrade their ablation capability.

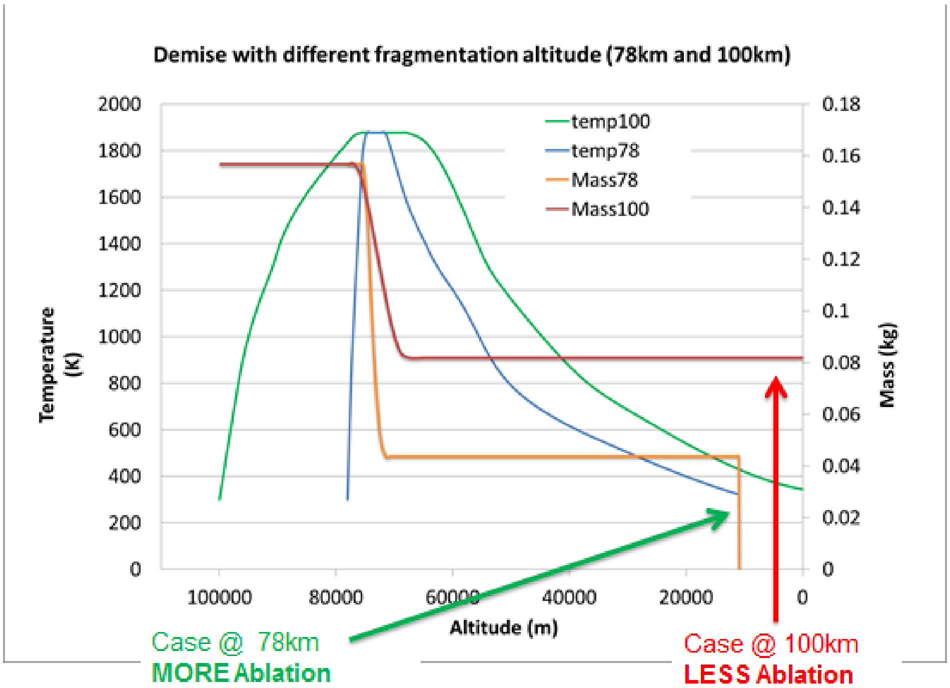


Figure 5-4: Example of a comparison of break-up altitudes between 78 km and 100 km on an example spacecraft

5.2.4.5 Venting and Aperture Holes – Internal Pre-Heating

A higher duration to the aerothermal flux and exposure can be achieved by the introduction of venting holes. These apertures can be either be permanent, i.e. included in the panel design when authorized by structure and radiation protection constraints or released releasing and jettisoning or orienting a whole panel during the re-entry. Devices can be built using glued panel sections or thermally actuated (shape memory alloys) washer expanding at high temperatures. Such technologies, outside the D4D application, have reached TRL 9.

5.2.4.6 Internal Structural Frame – External "Easy Demise" Panels

Internal Structural frames seem to have several advantages as allowing early-demise panels "Plastic-like" for use for S/C external shape (just implementing the solar arrays support or very small parts or brackets).

Those panels demise easily and expose the entire internal frames and equipment early to aerothermal flux with the following benefits:

- Reduce Cross section at high altitude 100 km - 80 km with few mass removed.
- Increase Ballistic Coefficient and then increase heat intensity.
- Reduce shadowing effect and increase view factor for equipment heating.

5.2.4.7 Spacecraft mass reduction

This technique is “normal work” in space design but appears as a D4D technique as well, given the benefit provided by a higher ballistic coefficient.

5.2.4.8 Containment techniques

Design for containment is sometimes listed as a D4D technique, working on reducing the number of impacting fragments after a destructive break-up rather than the existence of such fragments. Parts of the system or equipment are designed to sustain the re-entry environment and maintain all elements together without losing fragments. While this technique has been TRL 9 for nuclear power sources deployed in orbits, there have been test campaigns to apply solutions at equipment level. However, TRL 3 has not been achieved yet for a practical application on equipment level and significant development risks remain to demonstrate the concept in the relevant environment.

5.2.4.9 System equipment selection

This D4D technique intends to promote the reduction of entries in a casualty risk budget by replacing standard spacecraft components with equivalent D4D versions. Currently, one to one replacement is still under development and not of the shelf available.

5.2.4.10 Moving critical components at the edge

Moving critical elements to regions of the spacecraft where they encounter aerothermal flux from the start of entry means they can get the most effect of the available heat.

5.3 Applications and implications – How?

5.3.1 Overview

While the previous section outlined the reasons for applying Design for Demise, this section focuses on the techniques applicable to the critical elements identified earlier, assessing the benefits and effects they can bring. Various techniques for applying Design for Demise have been hypothesized. At a top level, these techniques can be categorized by whether they affect the entire spacecraft or are intended to reduce the risk from specific equipment. However, the categorization followed here is based on how the techniques are intended to improve demisability. It is important to note that this categorization is somewhat artificial, as applying one technique often influences other categories as well. To enhance the demise of a spacecraft and its components, one can minimize the heat needed for them to melt and ablate, maximize the heat available during re-entry, or optimize the heat transfer within the spacecraft, as explained in the introduction of Section 5.

As an example, Table 5-4 contains the assessment of the elements covered in Section 5.2 on a large LEO spacecraft for Earth observation. An example of associated analysis at Phase B1 maturity level is worked out in Annex A.

Table 5-4: D4D techniques with assessment of their effectiveness on a large LEO spacecraft

Level	Technique	Effectiveness
System	Rearrange components to put tank outside of structure	Low
System	Rearrange components to put reaction wheels outside of structure	Medium
System	Rearrange components to put MTQ outside of structure	High
System	Change overall shape to give a better trajectory	Low
System	Apply changes to give a better attitude during entry (e.g. shape or balance)	Medium
System	Apply changes to give a better attitude during entry by moving the Centre of Gravity (CoG)	Medium
System	Spin for stability	Low
System	Add demise attitude controllers (passive and aerodynamic, not active)	Low
System	Use a stronger connection to keep Solar Array attached for longer, leading to improving the trajectory or stabilisation, or inducing a spin	Low
System	Apply aerodynamic changes to encourage greater heat transfer	Medium
System	Cause the spacecraft to explode early in re-entry	Low
System	Eject components prior to re-entry	Low
System	Promote early breakup by weakening structure using activated devices e.g. frangibolts	High
System	Promote early breakup by weakening structure early in re-entry (for example breakup elements activated by heat, shape memory alloys, relocating joints outside the structure)	High
System	Promote early breakup by weakening structure by using components that age and degrade	High
System	Promote early breakup by using demisable elements (e.g. brackets and screws made from demisable materials)	High
System	Promote early breakup by using thermally weak elements	High
System	Utilise air pressure to encourage break up	Low
System	Use oxygen enrichment through ram air devices	Medium
System	Use easily removable outer panels e.g. actively removed or attached using frangible bolts or easily demisable bolts	High
System	Use burst disc and holes in the outer panels to let the aerothermal flux in	High

Level	Technique	Effectiveness
	Break-out patches in the outer panels to let the aerothermal flux in	High
	Apply heat conductivity changes to heat up the contents to 500 K on previous orbits	Low
	Remove Multi-Layer Insulation (MLI) at end of the satellite’s life	Low
	Use weakly connected Optical Solar Reflectors that detach early	Medium
	Contain non-demisable parts so they land as one object, not several	High
	Arrange timeline so re-entry takes place during solar maximum	Low
General unit	Arrange for objects to break up into small parts with impact energy below 15 J (e.g. battery cells)	High
	Reduce mass (of any particular object, e.g. bulkhead) by perforation	High
Propellant tank	Use steel tank	Low
	Use aluminium tank	High
	Use composite overwrap on titanium	Low
	Use composite overwrap on steel	Low
	Use composite overwrap on aluminium (see NOTE)	High
	Use electrical propulsion (smaller tank with less strength needed)	Medium
	Design propulsion system to have a less problematic tank, e.g. less corrosive propellant, or storing it at lower pressure	Low
	Use devices to break up tank e.g. shape charge	Low
	Use multiple smaller tanks	Low
	Use different shape tank (e.g. whether a cylinder is better than a sphere)	Low
	Encourage tank to stay attached to heavier objects	Medium
	Use aluminium flywheel	High
	Change shape of wheel e.g. holes in central area and more or less spokes	Medium
	Change shape of wheel e.g. bigger or smaller radius, fatter or thinner wheel	Medium
	Use composite wheel: Steel structure + demisable counterweights	High

Level	Technique	Effectiveness
Reaction wheel	Reduce number or size of the wheels	Low
	Locate on outside of spacecraft to give it more time to demise	Medium
	Change shape of ball bearing unit e.g. upper threaded ring	Medium
	Reduce shielding effect of casing	Low
	Use glued connections (e.g. between the rotor flange and magnet carrier and/or within the ball bearing unit)	Medium
MTQ	Use different core materials	High
	No core	High
	Design more open structure (e.g. reduce external housing, remove potting material, remove internal housing)	High
	Change of shape (e.g. longer and thinner or shorter and fatter)	Medium
	Change mass distribution between wiring and core	High
	Locate on outside of spacecraft - more time to demise	High
Balance masses	Held together multiple smaller masses with straps	High
	Use layered balance masses to minimise the needed heat	Low
	Rearrange components to avoid the need for a balance mass	Medium
	Use demisable material e.g. Aluminium	High
	Modify the shape to be more demisable (e.g. spheres vs cuboids)	Low
Payload elements	Make choice of different materials	Medium
	Shape the payload to encourage more demisability	Low
	Make a component from a larger number of smaller elements	Medium
	Use glued connections for payloads with mirrors	Medium
	Reduce mass and increase heating adding flow holes in bipods	High

NOTE: Demise tests conducted in a recent activity included in [RD10] indicated that non-demising overwrap materials maintain structural integrity under re-entry conditions. Specifically, a resilient resin used in these tests formed a protective layer, effectively preventing ablation of underlying composite strips. Similarly, in a previous activity documentation reveals that while the liner melted consistently, the composite overwrap remained intact. Given these findings, it can be prudent to note that overwrap selection can significantly impact demise effectiveness. Relying solely on an overwrap over aluminium can therefore be insufficient for achieving intended demise outcomes. It is important that the implementation of the abovementioned techniques is compliant with the space debris mitigation guidelines and, in any case, avoid the generation and propagation of debris in Earth orbit.

5.3.2 Minimize the needed heat

5.3.2.1 Introduction

The first method to enhance the demise of a spacecraft or of its equipment, is to minimise the heat needed for it to melt, ablate and fragment into pieces. This can be achieved in two main ways. Of course, with a lower object mass, less total heat is needed for demise. Another way to achieve this is changing the materials that constitute the object, for example to ensure a lower melting temperature and lower emissivity.

5.3.2.2 Minimize the mass

5.3.2.2.1 Introduction

Minimising the mass reduces the amount of material that it is needed to ablate during the re-entry. Usually, minimising the mass is part of the design of a spacecraft because it reduces the cost of a mission. In this Section, an example of equipment with potential for mass reduction is described.

5.3.2.2.2 Reaction wheels state of the art

Reaction wheels (RW) are used in space for three-axis attitude control. The RW usage is based on an electric motor attached to a flywheel, which, when its rotation speed is changed, causes the spacecraft to counter-rotate proportionately through conservation of angular momentum. Momentum depends on rotational inertia and rotation speed. Therefore, the mass of a RW can be reduced, if the speed is increased, without changing the key performance parameters of the wheel. However, increasing the speed of a RW, which usually operates up to 6000 rpm, can impact the lifetime of the wheel. Therefore, it is necessary to perform life tests, which are known to be potentially quite longer and therefore expensive.

5.3.2.3 Replacing materials

5.3.2.3.1 Introduction

To tackle the challenge of hardly demisable materials, studies were conducted to identify the most critical materials. The driving parameters were high melting temperatures and high specific heat of fusion (or specific enthalpy of fusion) needed for the demise, joined in the so called Q_{demise} factor introduced in Section 5.2.1.

The critical materials identified are notably stainless steel, beryllium, titanium, tungsten, super invar and silicon carbide. These materials are mostly present in tanks, which are traditionally made of titanium, and in reaction wheels and magnetorquers, which include steel components. For all these components, alternative materials have been proposed in literature and sometimes practically tested in the framework of D4D. Due to their requirements on thermal stability, optical payloads often include ceramics, which are another category of highly non-demisable materials. However, currently no demisable alternatives have been found to substitute ceramics and therefore other D4D solutions are applied.

5.3.2.3.2 Tanks: state of the art

Tanks are the most common kind of impacting fragments that have been confirmed to occasionally reach the surface of the Earth [RD06]. They were identified as a critical element because they are usually made either of Titanium, which has a very high melting temperature and specific enthalpy of fusion, or of a metallic liner covered with a composite overwrap of infinitely wounded fibres which is hardly demisable.

First, it is important to divide the tanks into two main categories, depending on the propulsion system that they support: high pressure tanks, used for electric propulsion and usually made of a Composite Overwrapped Pressure Vessel (COPV), and liquid propellant tanks, traditionally metallic and used for mono and bi-propellant propulsion systems.

The demisability of the COPV tanks was well investigated. The main challenge is related to the fact that the thermal characteristics of the composite layer are not well known, and this makes it difficult to model it and to identify an improved solution based on simulation. In particular, the composite overwrap does not have a specific melting point after which it can be gone. On the contrary, usually the carbon fibres gradually peel off, only if the fibres are not infinitely wounded. Therefore, the main conclusion was that, even if an Aluminium liner can melt inside the COPV, it is not clear how the material can escape the composite overwrap. Moreover, it is challenging, if not impossible, to extrapolate the results related to one type of composite to apply them to a different one.

Currently, there are ongoing activities that focus on enhancing the demise by using different manufacturing processes for the COPV tanks at TRL 5. Another activity that is ongoing is investigating the materials and processes needed to obtain a fully non-metallic gas tank at TRL 4. It is expected that non-metallic tanks can offer advantages in terms of reduced mass, lower manufacturing costs and lower lead times. The main challenge related to the development of this technology is the ability to meet the permeability requirements without employing a metallic liner.

Regarding liquid propellant tanks, the usual solution proposed to enhance their demisability is a change in the material. To substitute the commonly used Titanium alloy ($TiAl_6V_4$), which has a high melting temperature and heat capacity, with Aluminium alloys, such as Al-Cu, Al-Mg and Al-Li. The main advantage of the first one was its maturity. The Al-Mg alloy is compatible with green propellant, but its manufacturability was not demonstrated yet. Al-Li had good structural properties, and it has allowed for thinner tanks, but its manufacturability was also not yet well known.

When focusing on green propellants, a thermoplastic liner with a carbon composite overwrap has been proposed. However, it presents the same issues pointed out for COPV tanks and potentially leak problems because of the lack of a metallic liner.

Before 2020, only the tank shell was modelled and analysed, while now a more comprehensive assessment is performed, including interfaces and Propellant Management Devices, which can be demisable as well. Currently, various volume ranges have been identified and distinguished. Large tanks, with a volume range between 100 l and 220 l are currently being investigated, while demisable alternatives for small ones, with a volume range between 30 l and 50 l, are being researched as well.

5.3.2.3.3 Reaction wheels: state of the art

Analysis showed that medium and heavy RWs can survive re-entry. Moreover, spacecraft typically employ four RWs at a time, which means that a demisable alternative can greatly benefit their overall casualty risk. On top of this, RWs include various components, and the most internal ones are shielded by the external ones during the re-entry. In particular, the earlier the housing demises, the earlier the internal components are exposed to the heat flux and the more they demise.

Among the solutions, the most promising are to change the materials of the most critical sub-components, such as the flywheel or parts of the ball bearing unit. However, RWs are complex mechanisms, which means that some changes in their design can take a longer time to be qualified, especially if a life test is needed.

Lastly, changing the material of the flywheel without changing the size of the RW leads to an extensive increase of speed and torque need, which in turn implies a re-designed external drive electronics, with a great power increase.

5.3.2.3.4 Magnetorquers: state of the art

MTQ were identified as critical elements because they are made of various sub-components that are hosted one inside the other. Therefore, the most internal elements, such as the core, are exposed to the aerothermal flux late. Moreover, they are often mounted directly on the spacecraft panels and therefore are shielded by them until their mounting feet break. This means that they are fully exposed to the heat flux only late during the re-entry.

Possible solutions were investigated to enhance the demisability of MTQs, and the most promising ones aimed at exposing the core as early as possible during the flight, since the core material itself cannot be changed as it is strictly linked to the functionality of the MTQ. Therefore, the material of the mounting feet was changed to support an earlier separation, and the housing material was changed to guarantee an earlier exposure of the core. An inductive coil as small as possible can also enhance the demisability. Another option that was investigated was to split the core in juxtaposed cylinders.

5.3.3 Maximize the available heat

5.3.3.1 Introduction

Another option to enhance the demisability of a spacecraft during the re-entry phase is to provide additional energy to it. Of course, it is important to channel such energy into the melting and demisability processes, avoiding its dispersion in unwanted ways.

5.3.3.2 Ballistic coefficient

A variation in the ballistic coefficient can impact the trajectory profile (i.e. velocity and flight path angle). A higher ballistic coefficient makes the trajectory steeper, while a lower one makes it more gradual. However, when trying to tweak the parameters that define the ballistic coefficient, two quantities to be considered: the peak heat flux, which is important to achieve the melting point, and the total heat needed to achieve full demise. For example, decreasing the cross-sectional area increases the ballistic coefficient and, in turn, the heat peak flux. However, this corresponds to a faster re-entry, which potentially does not reach the total heat needed to fully demise the spacecraft. At the same time, decreasing the cross-sectional area also reduces the heat losses. To conclude, changing the ballistic parameter impacts on various other parameters and maximising the demise becomes in this case a complex optimisation challenge. For this reason, this method has not been implemented yet at system level, but topological change, e.g. to brackets and other structural components, and under investigation at equipment level.

5.3.3.3 Increase local heat flux

5.3.3.3.1 Introduction

The heat flux around a spacecraft is not uniform. Indeed, local heat flux is known to be higher at corners and edges. Therefore, a method that has been identified to enhance spacecraft demise is to change shapes locally, to trigger in such points an earlier start of the demisability process. The effect that certain shapes have on the local heat flux has been observed both in simulations and tests and can be applied. Notably, it is this same effect that make design for containment based on nets or small structures difficult to achieve in practice.

5.3.3.3.2 Introduce a heat source

Exothermic reactions can be triggered to increase the available energy during the re-entry. For example, they can be used to achieve interface separation or severing of harnesses and pipes and test have been conducted. Exothermic reactions, e.g. based on Thermite, have been investigated to enhance demisability of certain critical components, such as reaction wheels at TRL 4, but it is important to assess their effects at system level. This is also the case for Al-Fe_2O_3 based mixtures, which are inserted into joints, so they separate using exothermic reactions.

5.3.3.3.3 Reaction wheels: state of the art

The usage of exothermic reactions to enhance the demise of RW was trailed. The RWs were merely chosen to test the concept of exothermic reactions, but the conclusions can be extrapolated to other technologies. These tests demonstrated the thermite ignition in the relevant environment, Plasma Wind Tunnel (PWT), and the release of a significant amount of energy. However, the impact on the demise of the test samples was limited. There were issues related to a sub-optimal thermite composition, leading to unreliable ignition and even partial ignition, but also with an insufficient quantity of thermite for the selected test sample. Moreover, an overly complicated test sample, although close to an actual flight application, introduced additional unknowns in the test. Lastly, the formation of slag was identified and its influence on the test result was hard to assess.

In conclusion, these tests identified the need for further studies and concepts that can allow to properly channel the energy released to support the melting process, because otherwise most of it can be lost. The selection of the amount and placement of thermite to support demisability during re-entry is a difficult optimisation problem that cannot be tackled in the context of the past studies and that needs to be addressed in the future.

5.3.4 Optimize the heat transfer

5.3.4.1 Introduction

It is well known that during the re-entry phase, the usage of the heat that can serve the purpose of triggering the fragmentation and demise of spacecraft is not often optimised. In fact, for selection of a certain spacecraft architecture and configuration when designing a space mission, it is important to consider various other constraints. Previous missions that did not employ any D4D technology often expose the most internal pieces of equipment to the heat flux when it was too late for them to be demised before reaching the ground.

By achieving an earlier exposure, for example through the triggering of an earlier break-up of the external structure or of an early separation between different elements, the casualty area of each fragment can be greatly reduced.

5.3.4.2 Early break-up

5.3.4.2.1 Introduction

The earlier internal elements are exposed to the heat flux, the more they demise. Therefore, various ways to achieve a so-called early break-up have been envisaged and investigated. Currently fragmentation is generally observed to happen at an altitude of approximately 75 km - 85 km depending on the size of a spacecraft.

Various options to increase the fragmentation altitude were investigated, including composite inserts, Shape Memory Alloys (SMA) cylinders and bonded cleats. Some of these concepts are based on weakening the joints that hold together the various spacecraft panels, so that they fail earlier on during the re-entry phase. SMA are materials that expand when heated up and therefore can be triggered to apply a rupturing force on the joints that need to be broken during re-entry. Of course, it is important that all these concepts still guarantee the performances needed for the launch and operational lifetime of the spacecraft. Composite inserts showed no gains in demise. Bonded cleats provide a low-cost solution but showed only small demisability gains.

Demisable inserts in two parts and SMA cylinders were selected as the most promising technologies. The former is a simple replacement of the current insert technology. They do not store any energy themselves, but due to their enhanced melting process provide the chance for the forces present during the re-entry to separate the panels. They also have a low system impact and complexity. SMA cylinders expand when heated and therefore introduce a force to assist in the panel release. This concept has a high TRL, but also a high system impact. The increase of mass and cost was estimated to be proportional to number of joints.

However, at current fragmentation altitudes the MLI is usually significantly destroyed due to the mechanical forces, or to the interaction with atomic oxygen in the atmosphere during the mission lifetime and orbital decay phase. Therefore, it is not needed to consider its effect in current break-up simulations. However, when envisaging a break-up at higher altitudes, where the mechanical stresses on the spacecraft and the oxygen quantity in the atmosphere are lower, it is generally account for that the MLI can still be in place. Its presence can prevent the heating of demisable joints or SMA actuators and thus the early fragmentation.

The behaviour of various kind of joints under re-entry conditions were investigated as well. The film adhesive used to connect sandwich panel facesheets to the honeycomb core are likely to peel relatively early, resulting in loss of structural stiffness. Therefore, panel failure is not a gradual melt, but a relatively fast process. The failure of CFRP panels is expected to be delayed relative to aluminium equivalents, due to the high heat resistance and low in-depth conductivity of CFRP. Instead, the epoxy potted sandwich panel inserts imply a slower failure of the joints, because the heat needs to soak in the epoxy potting material, which has very high failing temperatures. However, the inserts release from CFRP facesheets much more easily than from those constructed of aluminium. This is due to bending failures around the hole which can be induced by thermal stresses.

An analysis of titanium bolts through aluminium brackets were conducted. These kinds of joints did not fail under the thermal stresses that were tested. However, the aluminium inserts threads and the aluminium brackets both deform significantly. This induces loosening of the bolts, which can result in substantial loads being applied in a dynamic environment. Based on this experience a more detailed and patented (EP3227184A1) design for a concept of a demisable washer was established [RD11]. This concept is built in a material that reaches its melting point earlier than traditional alternatives and earlier than the other joint assembly items, a demisable washer can trigger early fragmentation. Near its melting temperature, the washer structural performances are very low, thus it can be either broken by the structural loads or disintegrated by ablation. Once the washer has demised, the cleats can have a mutual shift, due to the proper hole in one of them, eventually leading to the joint dismantlement.

Generally, the SMA dismantle mechanism were also investigated. The concepts that were proposed as D4D techniques are:

- The SMA washers used to get frangible screws complied with all the requirements (high temperature requirement for passive capability, heaters, and thermal sensors for active capability). They were also very easy to manage, very versatile and they had the lowest development risk.
- SMA inserts to release the screws from inside panel inserts were the most innovative and efficient compromise for panel release. Certain temperatures in the dedicated inserts can promote screws release. This solution can fit the usual panel inserts and eventually be removable and replaceable. However, it needed to be demonstrated with tests and was linked to development risks.
- SMA cutting cords constituted a concept that had the capability to break structural parts but asked for further investigation. Moreover, it was the least mass efficient solution of the ones available.
- SMA Sleeve can be used to dismantle struts, bars, booms for external appendages or Payload modules. It was proved that this concept can work for most applications given that the sizing of the elements is designed to have low-level stress inside SMA.

The first two concepts were identified as the most promising. The materials that have been used for the various SMA concepts were categorised depending on their activation temperature: low temperature (Ti-Ni), high temperature (AlCu-X), very high temperature (TiNi-X). AlCu-X were the most suitable for the various temperature ranges needed by different applications.

5.3.4.2.2 Reaction wheels: state of the art

The dismantlement or separation of the various components can enhance the demisability of reaction wheels, but it is hard to perform in practice. In particular, the separation of RW electronics increases the demisability. The dismantlement of the flywheel or of the core slightly improves demisability, but its feasibility is still to be confirmed, while the dismantlement of the internal core parts improves the demisability, but it is complex and hard to implement.

5.3.4.2.3 Balance masses state of the art

In early studies, heavy balance masses were found to be prone to survive re-entry. A solution that was proposed was to develop layered balance masses combined with a passive release system of the layers. According to simulations, the balance masses where this solution is put into practice are expected to be always completely demisable.

However, it is important to remember that balance masses are different for every mission and can normally be easily adapted. Therefore, for the time being, it was deemed to not be convenient to investigate further this demisability solution, since it is not generic and applicable to all cases.

5.3.4.3 Orifices and lattice structures

This kind of structure or holes can be included in the outer panels of a spacecraft to allow for the heat to reach the internal elements faster during the re-entry phase. This cannot compromise the structural properties of the spacecraft itself, which need to successfully withstand various loads throughout its operational lifetime. This method has been proposed, but has not yet been implemented on system level, but lattice structures have been demonstrated to be beneficial for demise at equipment level.

5.3.5 Minimize casualty area

5.3.5.1 Introduction

Currently, the only method identified to minimise the casualty area without enhancing demise is containment. Indeed, this technique aims at reducing the number of fragments that land, therefore reducing the overall casualty area, rather than reducing the individual casualty area of each one of them. However, reducing the total casualty area by keeping fragments together can result in an increment of the impact kinetic energy, possibly making it higher than the safety threshold (15 J), therefore making re-entering debris dangerous. Thus, it is important to further investigate potential benefits and implications of these techniques. Also, the demonstrability of the containment technique has been an issue.

5.3.5.2 Containment

Unlike other techniques, containment does not seek to encourage components to demise, but instead to reduce the total casualty area due to undemised components. For some spacecraft, the casualty requirement can be met if all the undemisable debris lands as one item rather than several separate ones. Although implementation is not be easy, as rearranging the components to keep the critical ones together affects the mass properties and thermal engineering, doing so, can significantly reduce the total casualty area. As it was already anticipated, containment is usually applied to components for which more demisable alternatives have not been found yet, such as optical payload. Those include high melting temperature materials that cannot be changed due to its functional and performance requirements.

Containment activities revealed challenges in the material selection and the impact of re-entry conditions on structural integrity: Titanium is unsuitable for thin containment structures due to its low resistance to structural compromise under high heat, Tungsten requires further oxidation studies to assess containment durability under re-entry conditions, Silicon Carbide's thermal shock susceptibility limits its application to benign geometries where abrupt temperature changes can be managed, and aluminium oxide and Nextel 440 were tested and showed a susceptibility to thermal shock and low melting points that indicated them as impractical for re-entry containment.

Concepts studied for containment include bolts and tethers, which experienced higher-than-anticipated overheating relative to larger components in the tests. This highlighted the need for models incorporating local length scale to predict the demise behaviour accurately. Elements like thermal guards and optical benches focused on adapting flight-qualified materials for containment, and others such as tethers and cages need testing to achieve a mature TRL. Common failure points identified in the testing are the connections (e.g., bolts for thermal guards or tethers). It was also highlighted that no model for a containment net or cage can be accepted without direct test support of the specific concept.

5.3.5.3 Optical payload state of the art

Payloads, and in particular optical instruments, have many design constraints that need to be fulfilled and that are hard to be transcended while applying D4D. Notably, some materials, such as ceramics, cannot (yet) be substituted at all which are needed because of their thermal stability, and the glasses and mirrors that are used for the lenses.

A secondary challenge that has been identified when trying to apply D4D techniques to optical instruments is that the most undemisable component, such as ceramics and glass, are not well characterised. Indeed, the ceramic breakage is hard to predict, while glass can have a viscous behaviour that is difficult to model. If not contained, other options that were considered were design for fragmentation, i.e. divide lenses into multiple smaller components that can easily separate during re-entry, and the usage of pyrotechnic devices, such as pyro-bolts.

For the optical payloads supports, improved demisable concepts have been identified for elements such as bipods. Using hollow bipods (Additive Manufacturing that also allows to reduce mass) with large holes to increase the heat flux has been demonstrated to be a highly effective D4D technique.

5.4 Considerations for Test Facilities

This section intends to guide the reader through finding the testing aspects commonly executed to support the D4D analyses where no previous test data exits. As a guideline, there is a strong correlation between the equipment or material to be analysed and the capabilities of existing test facilities (an overview is given in Annex B). An in-depth worked-out example of the use of test facilities and resulting modelling aspects that can be used to support the verification of D4D techniques are provided in [RD04].

The demise process targeted by D4D techniques results from various physical phenomena, and the conditions of interest can vary significantly depending on where in the re-entry corridor the technique is applied. An example of the characteristic events during the re-entry can be found in DIVE [RD04]. The main events during an atmospheric re-entry, their driving parameters (e.g. heat flux, flow effects), and their impact on internal, structural, or external equipment are highlighted. This overview is completed with Figure 5-5, which shows the convective heat flux and dynamic pressure profiles as a function of altitude of a re-entry obtained when assuming an undemisable sphere. It can be noted that for a shallow re-entry, i.e. flight part angle at 120 km below 1 degree and representative for uncontrolled re-entries, generally the maximum heat flux happens significantly before the maximum dynamic pressure.

Ground test facilities are however not capable of faithfully simulating all the effects occurring during a re-entry. Firstly, current facilities limit the generation of fully representative hypersonic flows during re-entry either in scope and duration and second, the tested sample cannot be fully representative of a re-entering spacecraft due to limited testing section size and lack of dynamic motion. This limitation can be partially alleviated using several facilities to capture representative conditions of interest. Unavoidably, testing gaps remain and are listed in Annex B. Figure 5-6 shows the main classes of facilities of interest for assessing the demise of equipment and materials during an atmospheric re-entry.

Given the limitation of test facilities, and the limitation of simulation models [RD04], the tests can be targeted to single phenomena, e.g. detecting the physical conditions at which a steel cylinder breaks as function of its radius of curvature in a PWT instead of having flight qualified ball bearing unit of a reaction wheel. It is advisable to obtain material properties for demisability through test, before testing equipment where said material are a major constituent to avoid issues in numerically rebuilding test from a facility in simulation tools that often lack extensive material databases.

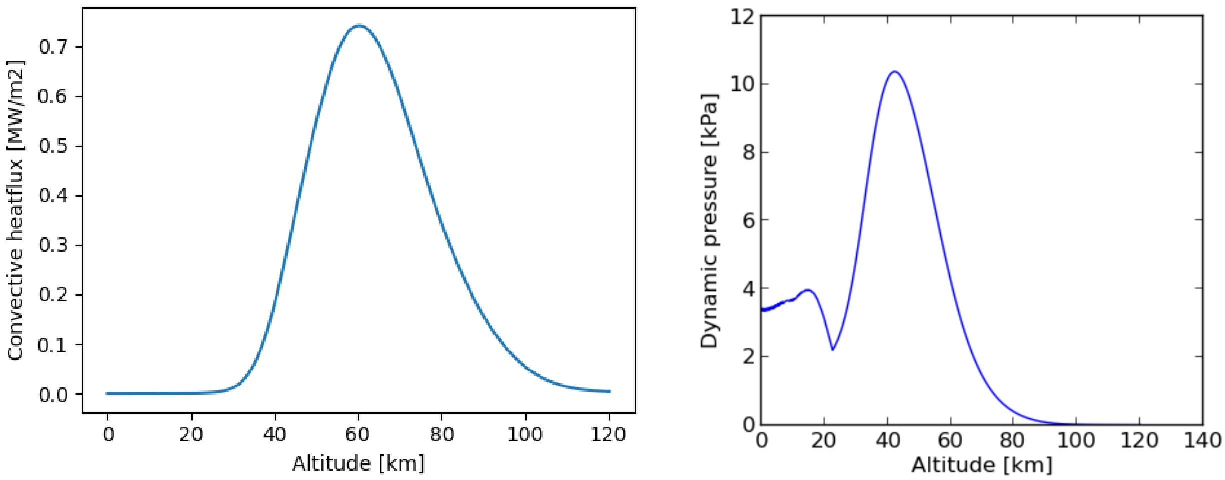


Figure 5-5: Convective heat flux and dynamic pressure profiles of a typical shallow uncontrolled re-entry from a circular orbit

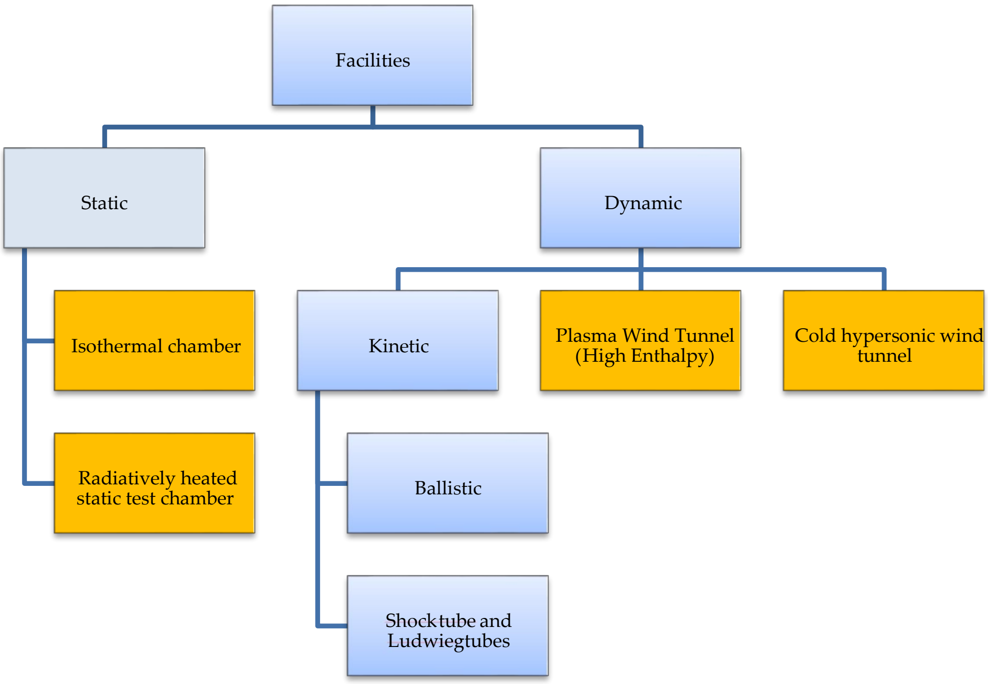


Figure 5-6: Test facilities available in Europe (in yellow the ones of interest within the scope of D4D techniques)

The primary interest for D4D technique testing is in high enthalpy testing facilities where a high heat flux in a partially representative flow-field can be applied to generate a representative material response. This is useful for assessing the demise of materials and equipment, but facilities that can directly manage system level test are not available. For such cases, (e.g. the testing of the demisability of joints), it is important that tests are carefully designed within the facilities' constraints.

However, where fragmentation events are not expected to be driven by melt, but are mechanical in nature — for example, the breakup of solar arrays or structural panels due to aerodynamic loads at lower altitudes — or where events are expected to occur before flow effects become more significant, static radiatively heated facilities where mechanical loads can be applied can provide more relevant data.

It is noted that there is currently some evidence of mechanical loads having an influence on the fragmentation behaviour of equipment and system level demise, but with limited impact when uncertainties are considered as they are noted close to when thermal driven failure causes fragmentation as well. Furthermore, if a specific fragmentation mechanism is dependent on high heat flux in a particular location of a particular shape or potential local viscous effects, then the heat flux distribution using infra-red thermography in a cold hypersonic wind tunnel (typically blow-down facility) is used. Note, however, that the flow energy is not necessarily be representative of the high enthalpy flow effects, but this is in line with the recommendation to target fragmentation modes when constructing a test due to overall limitation on representativeness of ground facilities.

The main aspects of the different facility types are summarised in Table 5-5. The facilities are presented, with an insight on their capabilities and limitations as well the driving parameters that can be expected to be verified in the test, for the event of interest, e.g. if the event to test is mainly temperature and mechanical loads driven, with negligible impact of heat flux effects and flux effects, the most suitable choice is running the test in an isothermal chamber. This approach is further detailed in Annex B.

The facilities can only provide a partial flow-field representation. Therefore, several facilities can be used, as the combination of their capabilities can provide a more optimal outcome. Figure 5-7 shows how the outputs from cold hypersonic wind tunnel tests, in terms of heat flux mapping, e.g. heat flux coefficient, aerodynamic coefficients, can be used as inputs for plasma wind tunnel and static facility tests. This combination allows to assess the appropriate test conditions a-priori, leading to an accurate test re-building of the models, during the post-test analysis phase.

Table 5-5: Main aspects of different test facility types

Facilities	Driving Parameters	Test Complexity	Capabilities and Limitations
Isothermal chamber	Temperature, Mechanical loads	Low	• Capture events which are temperature driven • No dynamics in terms of heating profile • Easy mechanical load application • Mechanical fragmentation observations • High temperature material property database generation
Radiatively heated static test chamber	Temperature, Mechanical loads, Heat flux effects	Low/ Medium	• Representative heating and timescale (trajectory simulation) • Easy mechanical load application • Suitable to test external fragmentation at high altitude and low flux • Internal fragmentation under load • Mechanical fragmentation observations • Low Heat flux level
Cold hypersonic wind tunnel	Mechanical loads, Heat flux effects	High	• Heat flux and heat transfer coefficient mapping • Aerodynamic database generation • Mechanical loads obtained from dynamic motion • No demise observations

Facilities	Driving Parameters	Test Complexity	Capabilities and Limitations
High enthalpy wind tunnel (Plasma Wind Tunnel)	Temperature, Heat flux effects, Flux effects	High	• Representative oxidizing plasma environment • Not representative dynamic pressure and/or flow velocity • Useful for demise observations. • Representative heat flux distribution. Steps in heat flux can capture demise threshold. • Representative thermochemistry • Not easy application of mechanical loads • The heat flux profile can only be re-built at discrete trajectory points. • High Heat flux level

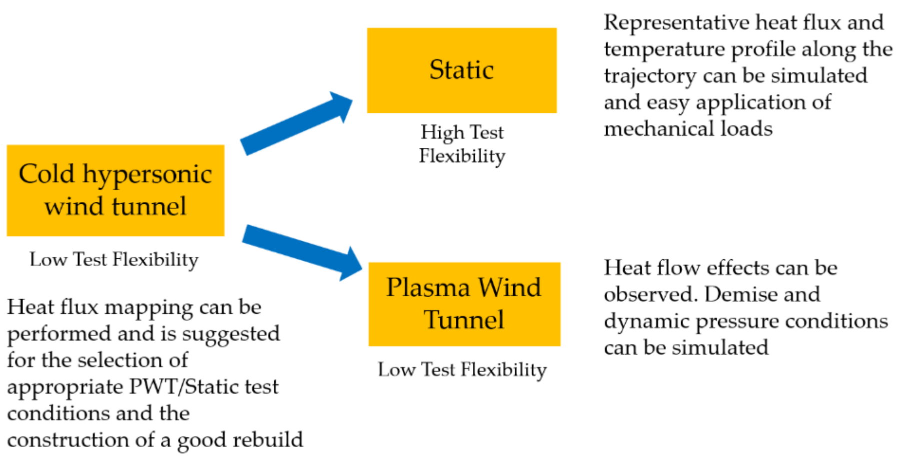


Figure 5-7: Potential logic for the set-up of a test campaign for D4D technique

6 Handbook update

It is recognised that that the field of a design for demise is still a field evolving, with only a limited set of implementations reaching TRL 9. However, there is an increasing amount material and equipment test data available and first mission implementing the techniques are in orbit [RD10]. At the time of writing, the influence of mechanical effects, e.g. tearing and loss of strength, on the demise of equipment is only just being demonstrated in relevant facilities and the aspect of mimicking the actual motion during a destructive break-up, e.g. dynamics pressure and rotational variations instead of static samples, to gain new insights is just starting. These developments have the potential to improve the understanding of the physics and guide the design of new D4D measures. This handbook can be updated as newer information becomes available.

Annex A
System level D4D for a Large LEO Platform

A.1 Introduction

By means of example, a full design for demise assessment was performed on a large LEO platform, applying the techniques from Section 5 on a spacecraft modelled on a hybrid between Sentinel-2 and Sentinel-3. Relevant D4D techniques were taken forward after a Phase A casualty risk budget, and subsequently critically assessed from a multi-disciplinary point of view. During a Phase B, the system-level impact on performance, cost and schedule can be considered for a mission when assessing the D4D impact. Re-entry modelling is generally used to assess the reduction in the casualty risk that can be achieved by each technique, and to update the casualty risk budget accordingly. The following aspects are generally deemed representative for a large LEO mission focussing on Earth observation tasks.

A.2 Rearranging components

A.2.1 Reaction Wheels outside the structure

Mounting the wheels on the outside of as spacecraft can lead to improved demisability as the wheels experience a higher exposure to the aerothermal flux during re-entry. Several system impacts and challenges were identified, including the need for the reaction wheels to be well supported, typically in a stiff corner, and accommodated symmetrically around the launcher axis (due to relatively high mass) with the correct relative attitude. This can in turn imply less demisable attachment points. Thermal and Assembly Integration and Testing (AIT) issues need to be considered. Although reaction wheels are able to cope with the radiation environment outside the spacecraft, there can potentially be an issue for the drive electronics, avoidable separating them from the wheel itself and accommodating them in the inside. This separation limits the choice of existing H/W dramatically. Achieving a high aerothermal flux exposure in itself is challenging, requiring the component to be as little protected by surrounding parts of the spacecraft as possible.

A.2.2 MTQ outside the structure

As the magnetic torquers are also potentially critical elements, an external mounting, can increase the exposure during re-entry. The systems-level impacts of moving the MTQ outside are expected to be minor:

a. Depending on the mission there can be Electromagnetic Compatibility (EMC) interferences with antenna or other instruments.

b. Neither shock from the launcher separation nor the higher radiation dose are expected to be a problem; the variable operating temperature needs to be assessed further. In addition, the rearrangement can lead to a simplified AIT.

The recurring costs of using the technique are small, and technique is based on existing hardware and technologies that are adjusted to improve the demisability, so no real technology, new developments are expected to be needed, with a high recurrence rate on similar platform. In this case, the MTQ do not constitute a critical element and can be removed from the casualty risk budget.

A.3 Outer panel-free platform design

In an outer panel-free platform design is a design in which all outer panels are removed from the spacecraft. Regarding demisability, a design without outer panels (some or all) has the same potentially beneficial effect on individual components as putting them outside of the main body of the spacecraft. However, the systems-level impacts of such a design are rather large, with a significant effort needed to bring the technique to a sufficient TRL and significant reduction in accommodation space for units.

A.4 Closure panel-free platform design

The closure panel-free design is a limited version of the outer panel-free design where only some of the panels are removed. The closure panels can be a subset of the outer panels which do not accommodate any units. Quite often, these panels also do not carry significant loads, and the closure panels are only attached in the end of the assembly process and are therefore easier to leave out, keeping the system impact on a lower level. Removing the closure panels increases the exposure of all components from the beginning of entry, which is overall positive for demisability, with the exact impact on any specific spacecraft being very dependent on its individual layout and aerodynamic properties.

Due to the similarities to the outer panel-free design the two concepts add similar constraints on load carrying structure, accommodation of units and protection and shielding e.g. against radiation and micrometeoroids and space debris. However, in the closure panel-free design the remaining panels can fulfil the standard panel functions at least partly. The AIT can become simpler, as there are fewer panels to be integrated and the late access increases. On the other hand, integration of units on other panels is potentially more complex.

A.5 Break-out patches within structure panels

Options for creating so-called breakout patches in the panels are possible. A breakout patch can be realised through a hole in the panel, which can be left open or covered by MLI. Another possibility can be to reduce the panel core density or the facesheet thickness in areas where the occurring stress do not exceed certain limits or where no units are accommodated. Due to the extensive manufacturing process, reduced density areas are more complex and cost intensive, whereas the range of application and the system impacts are comparable to the ones with the cut-outs. The demisability can be slightly worse, as cut-outs directly transfer the heat flux to the internal components and in the second option a (weakened) panel still exists.

Cut-outs and holes are already used for missions in terms of mass optimisations, and so there are no great difficulties in principle with the technique, although having them for demisability reasons means putting them in locations where they reduce the thermal control or radiation shielding, leading to a requirement for additional shielding and MLI. No significant new technology developments are needed.

A.6 Demisable structure joints

Targeting a development to demisable structural joints can be used to induce an earlier structural fragmentation, which again can lead to a higher exposure of the internal components. Several concepts therefore have been developed and patented. The system impacts for the demisable structure joints technique are very low, as the technique does not imply huge changes to the spacecraft platform in comparison to a state-of-the-art design. One potential weak point is the stiffness of the structure due to the use of the joint materials, which can limit the range of applicability given the pointing requirements. Due to the fact that the joints are additional components it is expected for the recurring costs to increase. It is important to test and analyse the materials for demisable structure joints to achieve the necessary TRL. In case some useful materials can be identified at least a qualification campaign is needed to reduce the risk (for example through looking at the performance lifetime of the suggested options).

A.7 Removable panels

Panels which are actively removed, generally before the start of re-entry, can be used to open the structure and to increase the exposure of internal components during re-entry. One possibility to realise this goal can be to use shape memory alloys that can be activated by temperature, air pressure or possibly even humidity. Bi-metal springs that open the panels actively before passivation are another option. For removable panels which are triggered by the conditions during entry, detailed modelling of the spacecraft design and the entry conditions are needed to determine when the panel is removed and establish and AIT plan for the non-standard environment.

There are some challenges that is important to control in the process. If the panels are removed before passivation, the next step is to decide whether the harness also stays attached or is cut off during the removal process. Environment driven removal adds complexity and can have the same effect than demisable joints, which are activated by the re-entry heat directly. Due to the higher complexity and the added risk and mass the removable panel technique causes a high system impact.

Depending on the individual mission scenario the system impact can vary and therefore require specialized approaches to deal with the occurring problems. A cost increase is expected, as the overall spacecraft design is more complex, and there are additional mechanisms and extra components in comparison to actual state of the art designs.

Annex B
Test Facilities Description

B.1 Static Test Facilities

Static test facilities usually consist of a vacuum chamber with means to heat the test sample to temperatures that are relevant to re-entry conditions. This information can then be used on the models to give more accurate predictions. Examples for these facilities ESA member or associated countries, non-exhaustive, are the Re-entry Simulation Chamber at AAC ((Aerospace and Advanced Composites), Austria, MESOX (Moyen d'Essai Solaire d'OXydation) heating facility at PROMES (Procédés, Matériaux et Énergie Solaire), France.

The instrumentation typically used in these facilities is for temperature measurements, such as pyrometers. Emission spectroscopy can also be used to determine ionised gas concentration and temperature profiles.

In general, the facilities are useful to obtain information about material properties at high temperatures, and the thermochemistry of the surface of the sample if the facility permits. In the specific cases, mechanical loads (static and dynamic) can also be applied, making it possible to study the thermo-mechanical behaviour of the sample.

General applications include thermal shock testing, thermal cycling and gas cycling tests, thermo-mechanical test, and combinations thereof. In other facilities, the thermochemistry of the surface can be studied by exposing the surface to oxidising plasmas. The most obvious limitation for D4D technique testing is that there is no aerothermal flux velocity. However, it is usually possible to control the temperature and pressure of these chambers independently and with high accuracy, as opposed to plasma wind tunnels.

B.2 Dynamic Test Facilities

B.2.1 Cold Hypersonic Wind Tunnel

Hypersonic flow is a flow for which speeds are much larger than the local speed of sound. In general, hypersonic flow is defined as the flow at Mach 5 or greater at which changes in the physical properties of the flow, i.e. ionization and dissociation begin to occur. A test facility designed or considered for hypersonic testing simulates the typical flow features of this flow regime. These flow features include thin shock layer, entropy layer, viscous interaction and most importantly high total or stagnation temperature of the flow.

This category of facilities also includes shock tubes, which are presented hereafter for completeness. A shock tube is a cold hypersonic wind tunnel, in which hypersonic flows can be produced for periods of time ranging in the order of milliseconds to a few seconds. The shock tube consists of a long tube that is connected to a convergent divergent nozzle at the downstream end and are separated from an evacuated dump tank in the upstream end, by diaphragm or fast acting valve. The throat area of the nozzle is significantly smaller than the cross-section of the tube. As the process starts, the separation is removed, i.e. the diaphragm is raptured or the valve is rapidly opened, a shock wave propagates towards the dump tank (initially at lower pressure) and an expansion wave propagates towards the nozzle and tube (initially at higher pressure). This expansion wave sets a subsonic flow towards the nozzle and tube. The hypersonic flow is then achieved by the subsonic flow being accelerated through the nozzle. The hypersonic conditions are maintained until the wave reaches the end of the tube and is reflected back to the nozzle. In this period of time, it is not possible to reproduce the heat conditions an object experiences in re-entry, however a flow field with Reynolds and Mach numbers which are relevant to orbital re-entry conditions can be produced. These are useful to study the aerodynamics of the object and the shock behaviour. For hypervelocity re-entry conditions (e.g. re-entry from Moon, Mars, Lagrange points) real gas effects need to be considered even if it is not feasible to mimic them in a cold hypersonic wind tunnel.

For optical measurements, it is possible to use infrared and vacuum ultraviolet spectrometry (both emission and absorption, time and or spatially resolved), shadowgraphy, and Schlieren.

Other instrumentation comprises high-speed pressure transducers and probes to measure static and total pressures. Thin-films and coaxial thermocouples for fast-response temperature and heat transfer measurements. Pressure and temperature sensitive paints. Force balance to measure forces and momentums on the test model.

B.2.2 Plasma Wind Tunnel

In a plasma wind tunnel, a continuous stream of plasma is created and passed through a nozzle and into the test section. These kind of wind tunnels can be classified according to the method used to create the plasma, which affects the characteristics of it and its recommended use.

Firstly, PWT can be equipped with magneto-plasma-dynamic plasma sources (MPG). They are used to simulate re-entry conditions with high specific enthalpies and low-pressure levels. They are useful to simulate the first phase of re-entry, where the temperature of the sample reaches its maximum.

The plasma can also be created using an arcjet, making it possible to test at high dynamic pressure levels. They have a more limited reachable specific enthalpy, but higher pressures can be obtained together with higher Mach numbers. The gases reach high temperatures and hypersonic speeds when they pass through an arc jet between an anode and a cathode. This kind of PWT is better suited to study the late phase of re-entry trajectories.

The last kind is the inductively heated plasma wind tunnels. This type of plasma wind tunnels is used to generate plasma flow that does not create any unwanted chemical reaction in front of the test sample, with the objective to study its catalytic behaviour. For this reason, inductively heated plasma wind tunnels have no electrodes. In addition, they can be operated on more reactive gases, which can be useful to simulate the atmospheres of non-Earth planets.

Usually, the instrumentation is used for thermal and optical measurements. For thermal measurements, the possible instrumentation includes thermocouples, infrared pyrometers and radiometry. Pyrometers are remote-sensing thermometers, and radiometry is just another technique to measure the temperature of an object. For optical measurements, it is possible to use emission spectroscopy, or laser induced fluorescence in addition to high definition and infrared cameras. Infrared thermography can therefore be used as well. An important parameter that can be measured is the emissivity of the test sample.

A critical limitation is that there is no wind tunnel that can emulate all the phases of re-entry having to choose between testing with a representative total pressure (arc heater) or a representative enthalpy (MPG) for the peak heating phase.

In addition, usually the way the heat flux is applied to the test sample does not correspond exactly to the way it happens in flight. An example of an implementation is based on using the stagnation point heat flux profile which is integrated over the re-entry trajectory, and this value is used to compute the average constant heat flux. That is applied over a determined amount of time in order to subject the sample to the same amount of energy. This way the entire process during the trajectory cannot be resolved but the final result can be representative. Several different combinations of time duration and average heat flux can be used, but it is important that this last value is above a certain amount in order to reach melting temperatures.

Another important limitation has to do with the way the sample is placed inside the test section. Usually, it is supported by a holder and is held still throughout the test, while during re-entry the object is tumbling, which makes the heat flux received roughly 4 times lower than the stagnation point heat flux. Additionally, the support for the test sample can use as a heat sink during the test, making the results less representative. Therefore, to minimise this effect, the use of thermal insulation is recommended. Finally, the flow velocities achieved do not match the ones experienced during re-entry. For this reason, testing is conducted via Local Heat Transfer Simulation, by which the enthalpy, stagnation pressure and velocity gradient at the stagnation point are reproduced.

Steady state atmospheric entry test facilities for aerothermal testing at flight-relevant heating conditions, commonly referred to as PWT, are meaningful as a facility for the duplication of a boundary layer environment that is representative of local conditions generated from hypersonic compression in front of an object entering and interacting with an atmosphere.

The environmental conditions ensuring this boundary layer similarity comprise an equivalence of the local mass-specific enthalpy, the total pressure and the velocity gradient at the boundary layer edge. The latter condition is typically met by scaling the effective nose radius of the test article to mimic the given real flight scenario. Scaling approaches exist that enable a direct, meaningful scaling and read across of stagnation point heating conditions between a given flight reference and an accordingly calibrated PWT test scenario, such that boundary layer conditions near a surface can be considered fully representative within the stagnation area. This ensures local thermochemical similarity and results in representative and comparable gas-surface interactions and heat transfer.

The relative deficiency in kinetic energy of the typically sub-hypersonic simulated flow contributing to compression heating is compensated by an artificial increase of the flow enthalpy through the facility's plasma generator. Through this approach, the definition of Mach and Reynolds analogies as employed in classical fluid mechanics is not useful and the resulting values are misleading. Whereas the localised boundary layer conditions are accordingly inaccurate aside the stagnation point, they can still be relevant and conducive towards a verification of thermo-structural responses of non-trivial objects to entry conditions (e.g. nose structures or mock-ups) and can serve to verify numerical simulation tools via correspondingly implemented test cases.

B.2.3 Ballistic range facilities

Ballistic range is a kind of test facility in which the test models are launched at desired velocity. The aerodynamic properties of the flying models are then measured during its flight.

B.3 Testing Gaps

Notwithstanding the general usability of test facility, some D4D techniques rely on or are affected by events that can currently not be faithfully reproduced in ground facilities.

- On-ground test facilities can only offer partial duplication of the re-entry conditions. While some of the limitations can be palliated with a combination of several test and facilities (PWT and statics facilities), currently, some specific phenomena taking place during a re-entry cannot be duplicated on ground mostly due to test section size limitation, partial duplication of the flow properties and the impossibility to duplicate the attitude motion and forces applied on a component or spacecraft. Specifically, the limitations include: a system level wind tunnel testing: the size of the component being tested limits the capability to observe large scale event such as buckling, full panel failure and shielding at system level.

- Joint thermo-mechanical breakup can hardly be observed due to the impossibility to properly jointly duplicate the forces (in particular, the gravity, inertial effects and aerodynamic forces) acting on the object, heat flux and attitude motion on the time scale long enough to observe thermo-mechanical breakups. Static loading in static facilities and PWTs has been achieved.
- Early re-entry events (typically above 80 km) are complex to duplicate due a limited number of rarefied flow testing facilities. This limitation prevents the complete validation of numerical models and early breakup design for demise solutions in early re-entry phases.
- The duplication of the attitude motion of a component under re-entry flow conditions, current testing facilities are not able to duplicate the attitude 3 DoF attitude motion to derive the averaged heat flux and inertial forces. Some rotational motion has been achieved in PWTs.

Design for demise techniques that rely heavily on the phenomena mentioned above to demonstrate improved demise need to be validated with particular care, considering the large uncertainties associated with such complex events, for which our understanding and testing capabilities remain limited.

B.4 Facility Overview

There are different types of facilities that allow D4D testing in ESA member states or associated countries (see Table B-1 and Table B-2). A, non-exhaustive, collection of dynamic entry test facility types and the plasma source type available for plasma wind tunnel facilities are presented in [RD04].

Table B-1: Overview of dynamic entry test facility types in ESA member states or associated countries

Facility type	Plasma Wind Tunnels	Shock Tubes and Ludwieg Tubes *(kinetic)*	Ballistic Facilities *(kinetic)*
Examples in ESA member or associated countries	• L2K & L3K (DLR, DE) • PHEDRA (ICARE, FR) • Plasmatron (VKI, BE) • PWK1-4 (IRS, DE) • SCIROCCO & GHIBLI (CIRA, IT) • SIMOUN (Airbus, FR) • HSST (UMIST, GB)	• ESTHER (IPFN, PT) • HEG (DLR, DE) • HWK (ZARM, DE) • Longshot (VKI, BE) • T6 (Oxford, GB) • TH2 (Aachen, DE) • ST-DC & HST2 (UMIST, GB) • Shock Tube (ISL, FR) • TCM2 (IUSTI, FR)	• Terminal Ballistics facility (DREV, CA) • Terminal Ballistics facility (ISL, FR)
Test duration	~ 1 h	~ 1 ms	~ 100 ms
Total pressures	< 1,7 MPa (SCIROCCO, constricted arc plenum pressure)	< 150 MPa (before expansion)	< 480 GPa (assumption of isentropic kinetic energy transfer)
Mach number	< 12	< 25	< 20
Reynolds number	$10^3 – 10^4$ m^{-1}	< 10^7 m^{-1}	< 6,5 x 10^6 m^{-1}
Sample size	< 0,3 m (typical), < 2 m (SCIROCCO)	< 0,5 m	< 0,5 m (typically, cm range)

Facility type	Plasma Wind Tunnels	Shock Tubes and Ludwieg Tubes *(kinetic)*	Ballistic Facilities *(kinetic)*
Sample mass	(limited by size)	(limited by size)	< 10 kg (typically, g range for hypersonic velocities)
Specific enthalpy	< 220 MJ/kg (air) < 5 GJ/kg (H_2)	< 25 MJ/kg	< 30 MJ/kg
Primary drawbacks (as compared e.g. to flight experiments)	• No Reynolds or Mach number analogy • Academic reduction of reference object and trajectory needed • Challenges concerning representation of large-scale structure-material interactions (e.g. for anisotropic materials) • Trade-off usually needed between representative specific enthalpy and representative total pressure	• Very short test durations • Unsuitable for material heating response tests • Typically, thermochemically nonrepresentative test gases distort boundary layer shaping	• Very short test durations • Unsuitable for material heating response tests
Suitable applications	• Material response characterisations (demise phenomenology, catalysis at relevant wall temperatures) • Joint break-off characterisations • Limited verification of thermos-structural responses of small-scale components or mock-ups • Thermochemistry verification cases for ground risk assessment simulation tools	• Simple aerodynamic break-up (e.g. "break-off") experiments (multi-body) shock layer formation and interaction • Aerodynamics verification cases for ground risk assessment simulation tools	• Potentially relevant when combined with shock tube or Ludwieg tube for break-up ("breakoff" experiments

Table B-2: Overview of plasma source type available for plasma wind tunnel facilities in ESA member states or associated countries

Types of Plasma Sources	Arc heaters (a.k.a. Thermal Plasma Generators or TPG)	Magneto-plasma dynamic generators (MPG)	Inductively heated Plasma Generators (IPG, a.k.a. Radio Frequency Generators or IPG)
Subtypes	• Hollow electrode / Huels type arc heater, e.g. L2K • (DLR, DE), SIMOUN • (Airbus, FR), HEAT (ALTA, IT) • Constrictor-design thermal arcjet generator, e.g. L3K • (DLR, DE), • SCIROCCO (CIRA, IT) • Central cathode design Thermal Arcjet, e.g. RB3 in • PWK4 (IRS, DE), F4 (ONERA, FR) • Hot cathode generator • 3-phase AC generator	• Self-field MPG (SFMPG), e.g. RD5/6/7 in • PWK1 (IRS, DE) • Applied-field MPG (AFMPG), used in Russia only for entry testing	• Inductive plasma generator (IPG) / radio-frequency generator (RFG), e.g. IPG3/4/7 in PWK3 (IRS, DE), Plasmatron • (VKI, BE)
Advantages	• High pressure range • Comparatively high • Mach numbers • locally representative aeromechanics (stagnation area)	• High enthalpy range • Fairly high plasma purity locally representative thermochemistry (stagnation area)	• Full working gas flexibility due to electrode-less design • High plasma purity • Moderately high pressure and enthalpy ranges
Constraints	• Limited mass-specific enthalpies especially at high-pressure conditions • Typically, some plasma contamination through electrodes	• Limited maximum • pressure range	• Only moderate enthalpy and pressure ranges

Types of Plasma Sources	Arc heaters (a.k.a. Thermal Plasma Generators or TPG)	Magneto-plasma dynamic generators (MPG)	Inductively heated Plasma Generators (IPG, a.k.a. Radio Frequency Generators or IPG)
Best applications for investigation of uncontrolled re-entry aspects	• Late phase (e.g. post-break-up) entry trajectory simulation • (material demise) • Investigation of localised aeromechanical effects	• Early phase (e.g. prebreak-up) entry trajectory simulation • High-energy entry trajectory simulation • (energy similarity) • Investigation of localised high enthalpy thermochemistry (oxidation and catalysis)	• Detailed surface thermochemistry investigation, e.g. catalysis and oxidation

Annex C Simulation support for a casualty area budget

The whereas the level of confidence in a casualty area budget is dependent on the maturity of the space object's design, similarly the complexity of the space object's model is dependent upon the maturity of the design and the phenomena which are needed to be captured in order to demonstrate a D4D technique. As primary tracker for design maturity, the notion of mission development phase is used [RD05]. For the purpose of this guide, two modelling approaches are re-used: The "Single Primitive Model" (see Table C-1) and "Component based Model" (see Table C-2) [RD03]. In this Annex C, Component based models include "Spacecraft Oriented Models" from [RD03] as they implement the same functionality. The characteristics of these two models make use of contained-in and connected-to proprieties which a needed as a minimum to represent D4D techniques.

In the contained-in property it is important to consider that objects representing parts of a space object in a re-entry break-up simulation can be divided into Parent or Children, being the second protected from the atmosphere by their Parent. The Children are released to the flow if their Parent demises, or if one of the several trigger criteria are met during the simulation. This is the simplest manner to account for different materials in a component.

For the connected-to property, connected parts (Children or other Parents) can impact as a single object or as multiple objects depending on the failure of the joints connecting them. Joints fail either when one of the components connected by the joint demises, or if one of several trigger criteria are met. This also allows different parts of different materials to be represented.

Table C-1: Modelling of a Single Primitive Model at different stages and levels

Single Primitive Model			
Sub-Category	**System Level**	**Equipment Level**	**Material Level**
One stage model	Parent spacecraft modelled as single primitive shape of the correct ballistic coefficient. Complete set of children components are released in one or more catastrophic fragmentation events.	Each component modelled as single primitive shape. Contained-in propriety used to model nested components.	Different materials in a component are modelled using a contained-in propriety with primitives of different material.
Two stage model (In case of appendages, such as solar arrays)	A primitive shape parent with the ballistic coefficient of the spacecraft with appendages removed is connected to single-shaped primitives representing the appendages. The overall system has the same ballistic coefficient of the original spacecraft. Complete set of children components are released in one or more fragmentation events.		

Table C-2: Modelling of a Component Based Model at different levels

Component Based Model		
System Level	**Equipment level**	**Material level**
The spacecraft modelled as a set of connected components.	Components modelled using one or more primitive shapes connected to each other. Contained-in propriety is used to model nested components.	Different materials in a component are modelled using a contained-in and/or connected-to propriety with primitives of different material.

The following guidelines are provided to support the user in the selection of the model to use to follow a D4D development:

- The "Component Based Model" or "Spacecraft Oriented" approach is preferred as this leads to a more comprehensive and refined model of the vehicle.
- If an approved model for a piece of equipment exists for a D4D technique, it is recommended for use. This recommendation extends to both "Single Parent Model" and "Component Based Model".
- If any component demonstrates to demise due to its being located on the outside of the structure, the use a "Component Based Model" spacecraft model with the part connected to the outside of the structure is recommended.
- If fragmentation is demonstrated to occur early, the use of a "Component Based Model" spacecraft model is recommended.
- The use of "Component Based Model" is recommended to demonstrate any dependence of demise on the heating before the release of components from the intact spacecraft.

Typically, complete information for the modelling of critical elements during the early project phases are not available. To overcome this limitation, it is recommended to perform a critical evaluation of the spacecraft components and to allocate budget for a preventive casualty area for each those. In fact, for each mission, the maximum casualty area is bounded by the limitation on the casualty area and by the population density on the impacting latitude, which itself can be obtained by the orbit inclination [RD03]. An example procedure for the accompanying modelling develops in two parallel but dependent lines, one focusing on the estimation of the allocated casualty area, the other focusing on the model development for the re-entry risk analyses, as explained in the following:

Phase 0:

Model development

A first assessment of the casualty area can be made using a single parent model with a fix break-up altitude. In case of Controlled or Highly Elliptic re-entry, this can also include the modelling of potentially critical components in low demisable materials (as silicon carbide), as conservative approach.

Casualty area allocation:

All the spacecraft components are preliminary evaluated to obtain a first estimation of the casualty area to allocate for each of them taking into account known critical elements and considering that the sum of the casualty areas across all space object components is limited accordingly. This first evaluation highlights which components are expected to be more critical already at this phase and help to establish preliminary limits on the design at equipment level. E.g. a Telescope: this component has requirements in terms of resolutions and the number of optical lenses is not expected to be decided in preliminary phases, but estimating its overall casualty area as system level is helpful to define a maximum number of lenses suitable, i.e. equipment level, to comply with the casualty area requirements, since lenses are made in silica and this is one of the critical materials.

PRR - Phase A:

Model development

At this development phase, the risk assessment can still be made using single primitive models with a stochastic variation for the break-up altitude or other re-entry parameters. If the design, or parts thereof, contain elements that are beyond phase A level maturity, such as when re-using a spacecraft bus or instrument, it is possible to already use component-based models to avoid extra modelling iterations. Stochastic analysis on the survivability of each component, defined as the percentage of simulations in which the component survived with respect of the overall number of simulations, is obtained as results of the stochastic approach.

Casualty area allocation:

The allocated casualty area for each of the component is refined accordingly with the more detailed information available on the model itself and proportionally accounting for the survivability of the component, according with the following scheme:

- 100 % of the allocated casualty area can be considered if the component has more than 90 % survivability.
- Proportional allocated casualty area is considered if the component survivability belongs to the range of values 10 % - 90 %. For example, if the component survivability is 45 %, the scaled allocated casualty area is 45 % of the originally proposed value.
- 0% of the allocated casualty area is considered if the component survivability is lower than 10 %.

The sum of these scaled allocated casualty area is then compared with the limit values for the casualty area, in order to estimate is the design is aligned with the allowed risk. Initial assessment of D4D techniques can start in this phase.

The component level casualty area allocation is generally reported in aggregated form at equipment level for the casualty area budget.

Starting from SRR – Phase B:

Model development

At this development phase, the risk assessment can be made using component-based model with modelled break-up criteria.

Casualty area allocation:

The allocated casualty is refined according with the recommended procedure explained for the PRR- Phase A in Table C-3.

Table C-3: Spacecraft Model, Fragmentation Criteria and Casualty area Allocation evolution through initial mission phases

	Spacecraft Model	Fragmentation Criteria	Casualty area Allocation
Phase 0	Single Primitive	Single Fixed Altitude	Preliminary estimation based on critical elements
PRR - Phase A	Single Primitive	Stochastic Altitude Variation	Refined estimation, proportional to component survivability
SRR – Phase B	Component-based	Modelled	As for PRR – Phase A.