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DOCUMENT

ESA Space Debris Mitigation Compliance Verification Guidelines

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Introduction

Launch vehicle orbital stages and spacecraft becoming non-functional, at the end of mission or because of accidental failures, as well as mission-related objects, contribute to the space debris population. Furthermore, any launch vehicle orbital stage or spacecraft can be involved in fragmentation events due to orbital collisions and break-ups. The resulting fragmentation debris pose a significant risk for short and long-term survivability of any other operational space mission. Launch vehicle orbital stages and spacecraft becoming non-functional, at the end of mission or because of accidental failures, as well as mission-related objects, contribute to the space debris population. Furthermore, any launch vehicle orbital stage or spacecraft can be involved in fragmentation events due to orbital collisions and break-ups. The resulting fragmentation debris pose a significant risk for short and long-term survivability of any other operational space mission, since a spacecraft or launch vehicle orbital stage can have a high probability of collision if it passes through a region of high debris density concentration. This high debris density concentration can occur after a break-up. A debris cloud exhibits large spatial and temporal changes in the concentration of the spatial density in space. In high-inclined LEO orbits, within a few days after the break-up, the debris becomes more uniformly distributed within the orbital plane, and the cloud reaches a state called the pseudo-torus. At a later point in time, the debris cloud expands and evolves into a shell distribution.

According to ESA DISCOS database, 240 break-up events of spacecraft and rocket bodies were recorded up to May 2014: 74 (30,8%) due to propulsion system; 8 (3,33%) due to battery; 11 (4,58%) due to collisions (which includes potential collisions with sub-catalogued objects); 86 (35,8%) due to unidentified causes; 61 (25,4%) deliberate break-ups.

Re-entering space debris also may represent a hazard to human population, air and naval traffic, and ground and sea assets. Currently every year a few hundred of catalogued objects, including spacecraft, launch vehicle orbital stages, and fragments re-enter the Earth atmosphere without any control. A few tens of these objects are large and heavy enough to survive an atmospheric re-entry. Typically about 10-40% of the mass can survive (depending on the object design, re-entry trajectory, atmospheric conditions) and parts or fragments can reach the Earth surface with high kinetic energy. Propellant tanks, high-pressure vessels, and motor cases made of Titanium or heavy components like reaction wheels are often likely to reach the ground.

In 2004, all major European space agencies agreed on the “European Code of Conduct for Space Debris Mitigation” (ASI, BNSC, CNES, DLR, ESA, 28/06/2004). In 2008, the first ESA Space Debris Mitigation Policy was released. This policy has been updated in 2014 with the ESA/ADMIN/IPOL(2014)2, which adopts the ECSS-U-AS-10C / ISO 24113 as standard for Space Debris Mitigation.

Space Debris Mitigation measures aim at reducing the probability of fragmentation events and collisions and minimize the probability of hazard occurrence in orbit and in case of re-entry.



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Scope

This handbook provides guidelines on verification methods and possible implementation of mitigation measures in support to ESA Projects to facilitate the compliance with the ESA Space Debris Mitigation (SDM) requirements defined by the ESA policy ESA/ADMIN/IPOL(2014)2. This document has been prepared by the ESA Space Debris Mitigation Working Group, coordinated by the Independent Safety Office (TEC-QI), involving experts from the relevant disciplines in the ESA Technical and Quality Management (TEC) Directorate, the Space Debris Office (HSO-GR) in the ESA Human Spaceflight and Operations (HSO) Directorate, and representatives from the other Programme Directorates.

This handbook provides as well a description of the analysis approaches and documentation to be prepared to demonstrate compliance with the requirements in the ESA Policy.

The intended users of this handbook are any ESA project and its partners, including ESA Directors, ESA Project Managers, Study Managers, PA Managers, System Engineers, experts and all technical personnel, which are involved in the design or operation control of space systems with respect to the implementation of the ESA SDM requirements. This document has also a tutorial nature since the implementation of the SDM requirements is relatively recent in the Agency.

This handbook will be updated on a regular basis. The first update is planned to take place one year after its initial release to include feedback from ESA and Industry users and the outcome of studies currently on going (e.g. studies in the frame of the Clean Space Initiative).



2

References

- [RD1] ESA/ADMIN/IPOL(2014)2 - Space Debris Mitigation Policy for Agency Projects - ESA Director General's Office, 28/03/2014.
- [RD2] ECSS-U-AS-10C - Space Sustainability - Adoption Notice of ISO 24113: Space Systems - Space Debris Mitigation Requirements - 10/02/2012.
- [RD3] ECSS-S-ST-00-01C - ECSS System - Glossary of Terms - 01/10/2012.
- [RD4] ECSS-E-ST-10-04C - Space Engineering - Space Environment - 15/11/2008.
- [RD5] ECSS-Q-ST-30-02C - Space Product Assurance - Failure Mode Effects (and Criticality) Analysis (FMEA/FMECA) - 06/03/2009.
- [RD6] ECSS-E-ST-32-02C Rev. 1 - Space Engineering - Structural Design and Verification of Pressurized Hardware - 15/11/2008.
- [RD7] ECSS-Q-ST-70C Rev.1 - Space Product Assurance - Materials, Mechanical Parts and Processes – 15/10/2014.
- [RD8] ISO 24113 - Space Systems - Space Debris Mitigation Requirements - 15/05/2011.
- [RD9] ISO 23339 - Space Systems - Unmanned Spacecraft - Estimating the Mass of Remaining Usable Propellant - 01/12/2010.

3

Terms, definitions and abbreviated terms

3.1 Terms from other standards

- a. For the purpose of this handbook, the terms and definitions from ECSS-S-ST-00-01C, ECSS-E-ST-32-02C, ECSS-E-ST-10-04C, and ECSS-Q-ST-30-02C apply, in particular for the following terms:
 - 1. deviation
 - 2. failure
 - 3. failure mode, effects and criticality analysis (FMECA)
 - 4. leak-before-burst (LBB)
 - 5. meteoroids
 - 6. pressure vessel
 - 7. reliability
 - 8. single point failure
 - 9. space system
 - 10. verification
 - 11. waiver
- b. For the purpose of this handbook, the terms and definitions from ESA/ADMIN/IPOL(2014)2 apply:
 - 1. approving agent
 - 2. casualty risk
 - 3. disposal
 - 4. disposal phase
 - 5. ESA space system
 - 6. end of life
 - 7. operational phase
 - 8. orbital lifetime
 - 9. space debris
 - 10. re-entry



- c. For the purpose of this handbook, the terms and definitions from ISO 24113:2011 apply:
1. break-up
 2. end of mission
 3. GEO Protected Region
 4. Geostationary Earth orbit
 5. launch vehicle orbital stage
 6. LEO Protected Region
 7. normal operations
 8. probability of successful disposal
 9. protected region
 10. spacecraft

3.2 Terms specific to the this document

3.1.1 area-to-mass ratio

cross-sectional area exposed into the flight direction divided by the total mass

3.1.2 casualty area

equivalent impact area that lead a casualty if a person is struck by a piece of fragment

NOTE Casualty area is defined as:

$$A_c = (A_i^{1/2} + A_h^{1/2})^2$$

where A_i is the average projected area of the fragment, and A_h is the cross-section of a human ($A_h = 0,36 \text{ m}^2$, as per NASA-STD-8719.14A - Process for Limiting Orbital Debris - NASA, 25/05/2012).

3.1.3 casualty expectancy

computed number of expected casualties in a form of injuries or deaths as outcome of a re-entry event

3.1.4 casualty probability

probability of serious injury or death

3.1.5 catastrophic collision

collision which can cause a destructive structural break-up of a space system

3.1.6 controlled re-entry

type of re-entry where the time of re-entry is controlled and the impact of debris is confined to a designated zone



3.1.7 declared re-entry area (DRA)

area on-ground where the re-entry debris are enclosed with a probability of 99% given the delivery accuracy

3.1.8 disposal orbit

final orbit acquired after end of mission for permanent disposal on which end of life occurs and which allows to achieve the required long-term clearance of protected regions

3.1.9 graveyard orbit

disposal orbit remaining outside the Protected Regions even under the influence of perturbations

3.1.10 interference with Earth orbit

occurrence of a spacecraft which can be or even reach an Earth orbit at any time during its orbital lifetime

3.1.11 mission-related object (MRO)

objects dispensed, separated, or released during a mission

NOTE The following is a not exhaustive list of examples of mission-related objects: launch vehicle connectors and fasteners (e.g. separation bolts, clamp bands), fairings (e.g. fairings and adapters for launching multiple payloads), covers (e.g. nozzle closures, lens caps, cooler covers), others (e.g. yo-yo weights and lines).

3.1.12 passivation

action to permanently deplete or make safe all on-board sources of stored energy in a controlled way in order to prevent break-ups

3.1.13 population density

number of inhabitants over a specified area divided by the extension of that area

3.1.14 presence in the LEO or GEO Protected Region after the operational phase

time from the first interference with the LEO or GEO Protected Region after the operational phase to the time of the last interference, which is not the accumulated residence time in the Protected Regions

NOTE Example of last interference with LEO and GEO Protected Regions after the operational phase are termination by atmospheric entry to a stable disposal.

3.1.15 re-entry probability

probability that a re-entry scenario occurs, even if not planned by nominal disposal orbit

NOTE Example of nominal disposal orbit is disposal on a HEO or on an orbit around Sun-Earth Lagrange Points.



3.1.16 safety re-entry area (SRA)

area on-ground where the re-entry debris are enclosed with a probability of 99,999% given the delivery accuracy

3.1.17 uncontrolled re-entry

type of re-entry where the time of re-entry and ground zone of impact are not controlled

3.3 Abbreviated terms

For the purpose of this handbook, the abbreviated terms from ECSS-S-ST-00-01C and the following apply:

Abbreviation	Meaning
AVURNAV	Avis Urgent aux Navigateurs (Notice to Mariners)
CDM	Conjunction Data Message
DRA	Declared Re-entry Area
EEZ	Exclusive Economic Zone
EMR	Energy-to-Mass Ratio
EOM	End of Mission
GNC	Guidance Navigation and Control
HVI	Hypervelocity Impact
JSpOC	Joint Space Operations Centre
MMOD	Micro Meteoroid and Orbital Debris
MRO	Mission-Related Object
NOTAM	Notice To Airmen
SDM	Space Debris Mitigation
SDMP	Space Debris Mitigation Plan
SDMR	Space Debris Mitigation Report
SPOUA	South Pacific Ocean Uninhabited Area
SRA	Safety Re-entry Area
SRM	Solid Rocket Motor
TLE	Two-Line Element

Space Debris Mitigation requirements compliance verification

4.1 Introduction

4.1.1 ESA Space Debris Mitigation Policy

This Handbook provides the guidelines for ESA projects to comply with the requirements of the ESA Space Debris Mitigation Policy (SDM) ([RD1]).

The objectives of the ESA Space Debris Mitigation policy are:

- a. To prevent uncontrolled growth of abandoned spacecraft and spent launch vehicle orbital stages with particular regard to preserve the LEO and GEO Protected Regions.
- b. To prevent debris generation as a result of intentional release of mission-related objects or break-up of space systems.
- c. To prevent accidental break-ups as a result of explosions of components storing energy on-board space systems.
- d. To prevent orbital collisions by performing collision avoidance maneuvers and disposal maneuvers to limit long-term presence of non-operational space systems in the Protected Regions.
- e. To limit casualty risk due to controlled or uncontrolled re-entry of space systems.

4.1.2 Space Debris Mitigation responsibilities within ESA

The responsibilities for the implementation of the SDM policy and the issue, review and approval of the Space Debris Mitigation Plan (SDMP) and Space Debris Mitigation Report (SDMR) are the following:

- a. The ESA Directors are responsible for the implementation of the Space Debris Mitigation Policy.
- b. The ESA Study Managers, Project Managers or Mission Managers are responsible for the preparation and maintenance of the SDMP and SDMR.
- c. The ESA Director in charge of Technical and Quality Management and the relevant ESA Programme Director are delegated by the ESA Director General (DG) the responsibility and authority to approve waivers to the requirements.
- d. The Head of the ESA Department in charge of Product Assurance and Safety (TEC-Q) is delegated by the ESA Director in charge of Technical and Quality Management the



responsibility for the management of the implementation of the Space Debris Mitigation Policy and the approval of the SDMP at SRR and SDMR at FAR.

- e. The ESA Inspector General (DG-I) is responsible for ensuring that the implementation of the applicable requirements and the SDMP and SDMR are reviewed in the frame of the Technical Project Reviews.
- f. The Head of the ESA Independent Safety Office (TEC-QI) is the Technical Authority delegated by the Head of the ESA Department in charge of Product Assurance and Safety for the maintenance of the Space Debris Mitigation Policy with the related requirements, the independent supervision of the verification of compliance, and the processing of waivers with the technical assistance and expertise of the Directorate in charge of Technical and Quality Management and the Space Debris Office (HSO-GR) of the Directorate in charge of Operations.

4.1.3 Historical overview of debris generated in orbit

Debris generated by fragmentation in orbit, both for intentional or accidental causes, currently represent more than 60% of the human-made objects orbiting around Earth.

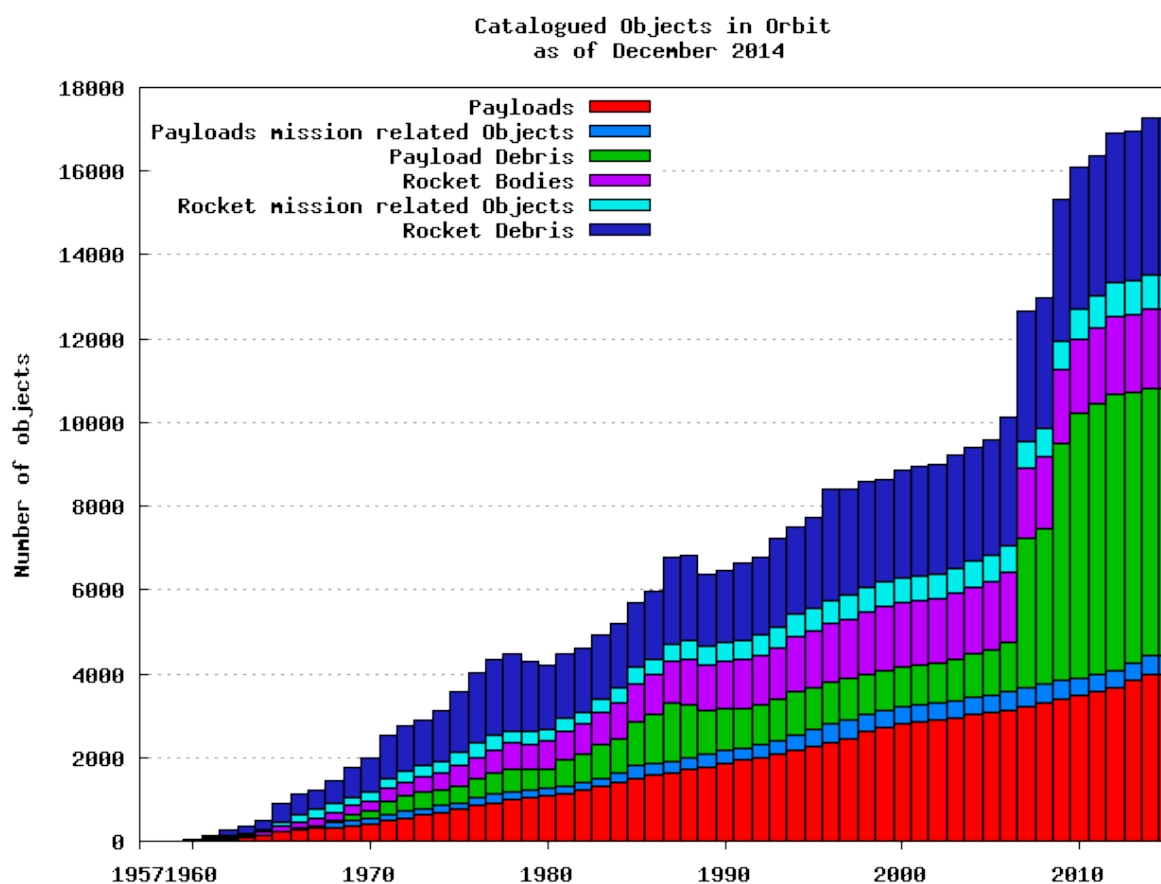


Figure 4-1: Monthly number of catalogued objects in Earth orbit by object type (ESA DISCOS database, Dec-2014)

The most relevant fragmentation events, involving large amount of debris generation, occurred just in the latest 10 years.



On 10th February 2009, the Iridium 33 (US operational communication satellite, 560 kg) and Cosmos 2251 (a Russian decommissioned communications satellite, 900 kg) collided at 790 km altitude with relative velocity of 11,6 km/s, which generated more than 2000 catalogued debris in LEO.

On 11th January 2011, a Chinese anti-satellite test involving the intentional destruction of the satellite Feng Yun 1C (957 kg) by a Dong Feng missile, at about 850 km altitude, generated more than 2700 catalogued debris in LEO.

Figure 4-1 shows the sharp increase of the debris population density due to the two events described above. The data are based on the US Space Surveillance Network, whose radars allow to track objects approximately larger than 10 cm in LEO and larger than 30 cm in GEO.

In general, orbital break-up and explosions of spacecraft and launch vehicle orbital stages can be originated by failure of components storing energy (e.g. propellant tanks, batteries, high-pressure vessels, self-destructive devices, flywheels, momentum wheels), hypervelocity impacts or intentional break-ups. Unfortunately, in the past, before the collision risk awareness became mature, intentional fragmentations were done under the wrong assumption to reduce the amount of debris surviving the re-entry of large space structures and in conjunction with on-orbit tests (e.g. the deliberate structural limits testing of the second flight of the NASA Saturn IVB stage in 1966). Up to May 2014, there were 61 deliberate break-ups in an orbit.

4.1.4 Structure of sections 4.2 - 4.20 of the Space Debris Mitigation Handbook

For each requirement, the structure of the sections is summarized in Table 4-1.

Table 4-1: Structure of the sections

Section Number	Section Title	Content
4.Y	Requirement	Text of the requirement (see NOTE)
4.Y.1	Rationale for the Requirement	Background information and justification for the adoption of the requirement
4.Y.2	Methods to Assess Compliance	Guidelines on how to demonstrate project compliance with the requirement
4.Y.3	Mitigation Measures	Possible implementation approaches to comply with the requirement
NOTE: Due to copyrights, the requirement text is to be found in the ISO 24113:2011.		



4.2 Requirement 6.1.1.1: mission-related objects release

ECSS-U-AS-10C / ISO 24113:2011 6.1.1.1	<i>See ISO Standard for text of requirement</i>
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4.2.1 Rationale for the requirement

The requirement aims at limiting the spacecraft and launch vehicle Mission-Related Objects (MROs), larger than 1 mm, which can be intentionally separated or unintentionally released during the launch and operational phases. This applies to objects released during normal operations in the LEO and GEO Protected Regions and objects released outside the Protected Regions, but with probability to pass through the Protected Regions during their orbital lifetime.

The requirement aims as well at limiting the risk of collision of MROs with the parent space system or any other operating space system since most of these mission-related objects are released with low relative velocity and, therefore, remain in close proximity to the operational orbit of the parent body.

4.2.2 Methods to assess compliance

The verification of compliance with the requirement should be based on review-of-design, to identify any mission-related object released from the space system during normal operations. This should include, but not limited to:

- a. Launch vehicle MROs: i.e. orbital stages (e.g. upper stages, kick-motor stage), connectors and fasteners (e.g. separation bolts, clamp bands), fairings, adapters for launching multiple payloads.
- b. Spacecraft MROs: i.e. covers (e.g. nozzle closures, lens caps, cooler covers), tethers, yo-yo weights and lines.

4.2.3 Mitigation measures

Possible mitigation measures to minimize the release of MROs are:

- a. To avoid release of spacecraft and launch vehicle MROs, which are not strictly functional to the orbit injection.
- b. To design spacecraft and launch vehicles such that unintentional release of MROs is minimized.
- c. To select materials and basic system technologies (e.g. tanks, surface materials, structures) able to be resistant to environmental degradation (e.g. due to radiation exposure, atomic oxygen erosion, thermal cycling) before atmospheric re-entry or GEO disposal maneuvers.



4.3 Requirement 6.1.1.2: mission-related objects on-orbit presence

ECSS-U-AS-10C / ISO 24113:2011 6.1.1.2	<i>See ISO Standard for text of requirement</i>
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4.3.1 Rationale for the requirement

The requirement aims at reducing the risk of debris collision by limiting the presence of MROs in the LEO and GEO Protected Regions.

The limit of 25 years for the presence of debris in LEO is widely internationally accepted as initially proposed by the Inter-Agency Space Debris Committee (IADC) Guidelines for Space Debris Mitigation (IADC-02-01, Rev. 1, 01/09/2007) and adopted by large number of space agencies.

4.3.2 Methods to assess compliance

The verification of compliance with the requirement should be based on review-of-design and analysis, to identify any debris released from the spacecraft and launch vehicle orbital stages during normal operations and demonstrate the limited presence on orbit. The analysis should include for each mission-related object the following:

- a. Definition of the physical characteristic, including shape, dimensions, (e.g. through drawings 2D drawings or 3D CAD files), mass, material (including heat treatment and manufacturing processes).
- b. Determination of the epoch and orbital state vector at the release event.
- c. Determination of the ejection velocity from the parent body (Delta-v).
- d. Orbit propagation analysis (see Annex A) to demonstrate that the presence in LEO is limited to 25 years and the presence outside the GEO Protected Regions is at least 100 years starting from the release epoch.

4.3.3 Mitigation measures

Possible mitigation measures to minimize the presence of MROs in the LEO and GEO Protected Regions are to design MROs with reduced orbital lifetime and limiting the release by design and operational procedures.



4.4 Requirement 6.1.1.3: launch mission-related objects release

ECSS-U-AS-10C 6.1.1.3	<p><i>If space debris are released into Earth orbit during normal launch operations, then the number of space debris released, other than those covered by 6.1.2, shall not exceed:</i></p> <ul style="list-style-type: none"> <i>a. One, for the launch of a single spacecraft.</i> <i>b. Two, for the launch of multiple spacecraft.</i>
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4.4.1 Rationale for the requirement

The requirement aims at limiting the number of orbital stages used in the launch vehicles. For the launch of co-passenger spacecraft, it also aims at limiting the number of adapters or other MROs, which are going to be released.

4.4.2 Methods to assess compliance

The verification of compliance with the requirement should be based on review-of-design, to show that the planned number of launch vehicle orbital stages, adapters, and other MROs is within the allowed quantity.

4.4.3 Mitigation measures

Possible mitigation measures to minimize the presence of MROs in the LEO and GEO Protected Regions are the reduction of the orbital lifetime of the released MROs by design and operational procedures and the selection of appropriate launch vehicles.

4.5 Requirement 6.1.2.1: pyrotechnic particle release

ECSS-U-AS-10C / ISO 24113:2011 6.1.2.1	<i>See ISO Standard for text of requirement</i>
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4.5.1 Rationale for the requirement

The requirement aims at preventing the release of randomly dispersed small particles (> 1 mm) in Earth orbit. These particles, although small, can damage operational space systems.

4.5.2 Methods to assess compliance

The verification of compliance with the requirement should be based on review-of-design or test, to demonstrate that the selected pyrotechnic devices are qualified not to generate particles larger than 1 mm and disperse them in the outer space.

4.5.3 Mitigation measures

Possible mitigation measures to minimize the release of small particles generated by pyrotechnic devices include the selection of compliant devices or space system design, which ensure that the particles generated are contained within the structure of the space system.



4.6 Requirement 6.1.2.2: solid rocket motors particle release in GEO

ECSS-U-AS-10C 6.1.2.2 / adapted from ISO 24113:2011 6.1.2.2	<i>Solid rocket motors shall be designed and operated so as to avoid releasing solid combustion products larger than 1 mm into the GEO Protected Region.</i>
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4.6.1 Rationale for the requirement

The requirement aims at avoiding release of combustion products larger than 1 mm from Solid Rocket Motors in the GEO Protected Region since these particles can damage operational space systems.

4.6.2 Methods to assess compliance

The verification of compliance with the requirement should be based on review-of-design or test, to show that the SRMs used in the space system have been qualified for not releasing particles larger than 1 mm. The qualification should cover all applicable mission operational conditions, including release of materials immediately after the termination of motor firing (e.g. from liners, nozzles).

4.6.3 Mitigation measures

Possible mitigation measures to minimize the release of large combustion products in the GEO Protected Region include avoiding the use of SRMs for orbital operations. Liquid propulsion systems or metal-free propellants should be used instead.

4.7 Requirement 6.1.2.3: solid rocket motors particle release in LEO

ECSS-U-AS-10C / ISO 24113:2011 6.1.2.3	<i>See ISO Standard for text of requirement</i>
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4.7.1 Rationale for the requirement

The requirement aims at avoiding release of combustion products larger than 1 mm from Solid Rocket Motors in the LEO Protected Region since these particles can damage operational space systems.

4.7.2 Methods to assess compliance

The verification of compliance with the requirement should be based on review-of-design or test, to show that the SRMs used in the space system have been qualified for not releasing particles larger than 1 mm.

4.7.3 Mitigation measures

Possible mitigation measures to minimize the release of large combustion products in the LEO Protected Region include avoiding the use of SRMs for orbital operations. Liquid propulsion systems or metal-free propellants should be used instead.



4.8 Requirement 6.2.1: intentional break-ups

ECSS-U-AS-10C / ISO 24113:2011 6.2.1	<i>See ISO Standard for text of requirement</i>
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4.8.1 Rationale for the requirement

The requirement aims at preventing any deliberate generation of space debris in Earth orbit.

4.8.2 Methods to assess compliance

The verification of compliance with the requirement should be based on adopting a mission plan which does not involve any intentional fragmentation in orbit.

4.8.3 Mitigation measures

Intentional fragmentations in orbit should not be planned.

4.9 Requirement 6.2.2.1: break-up probability threshold

ECSS-U-AS-10C / ISO 24113:2011 6.2.2.1	<i>See ISO Standard for text of requirement</i>
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4.9.1 Rationale for the requirement

The requirement aims at reducing the risk of accidental break-up and avoid generation and propagation of high density debris clouds in Earth orbit.

4.9.2 Methods to assess compliance

The verification of compliance with the requirement should be based on performing a Failure Mode and Effects Critical Analysis (FMECA) ([RD5]) for each critical component storing energy and demonstrate that the integrated probability of break-up for credible failure modes (excluding impacts with debris and meteoroids) is less than 10^{-3} .

4.9.3 Mitigation measures

Possible mitigation measures to prevent or reduce the risk related to accidental break-up include the selection of components and subsystems with low probability to cause explosions and design the space systems which do not break-up due to explosion of one of its components.



4.10 Requirement 6.2.2.2: break-up probability assessment

ECSS-U-AS-10C / ISO 24113:2011 6.2.2.2	<i>See ISO Standard for text of requirement</i>
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4.10.1 Rationale for the requirement

See section 4.9.1

4.10.2 Methods to assess compliance

See section 4.9.2.

4.10.3 Mitigation measures

Possible mitigation measures to prevent or reduce the risk related to accidental break-up include the selection of components with low probability to cause explosions.

For pressure vessels, it is recommended:

- Design and verification in accordance with the applicable standards (e.g. ECSS-E-ST-32-02C, ECSS-E-ST-10-04C, ANSI/AIAA S-081A-2006, ANSI/AIAA S-080-1988); the load spectra considered in the design should include all loads up to EOL including the disposal phase.
- Assessment of thermal effects, environment effects, and effects at system level and adoption of safety design requirements over all mission phases up to EOL.
- Verification by analysis that propellant dissociation (if present) does not represent a hazard at sub-system level which can lead to accidental explosion before EOL.

4.11 Requirement 6.2.2.3: passivation

ECSS-U-AS-10C / ISO 24113:2011 6.2.2.3	<i>See ISO Standard for text of requirement</i>
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4.11.1 Rationale for the requirement

The requirement aims at preventing accidental break-ups of the space system in Earth orbit after the completion of its mission operations and post-mission disposal by depleting energy or making safe all on-board sources of energy (passivation). For example, events possibly leading to a break-up are:

- Explosions or bursts of pressure vessels (e.g. propellant tanks, pressurized tanks), which can result, for example, from exothermal dissociation of propellant due to heating, mixture of hypergolic propellants due to leaks, pressure build-up, penetrating hypervelocity impacts, material degradation due to thermal cycling, atomic oxygen, ultraviolet radiation, corrosion, and Stress Corrosion Cracking (SCC).
- Explosions or bursts of battery cells, which can result, for example, from cell degradation, exothermal chemical reactions, short-circuit, overcharge, overdischarge, overpressure, corrosion, and Stress Corrosion Cracking (SCC).
- Mechanical ruptures of active rotating components.



In the current interpretation of the requirement, space systems performing disposal by controlled re-entry at EOM are not required to have passivation capabilities. In this case risk aspects are covered by the requirements on disposal reliability and re-entry safety.

It is also understood that passivation is not required to be Single Point Failure tolerant.

4.11.2 Methods to assess compliance

The verification of compliance with the requirement should be based on review-of-design, analysis, or test, to:

- a. Identify all components that store residual energy and need to be depleted or made safe during the disposal phase (typical components are listed in Annex E).
- b. Describe the implemented measures for depleting or making safe the stored energy for each component taking into account the environment after the passivation.
- c. Provide rationale for acceptance of components that cannot be fully depleted (residual energy, explosion probability, and effects due to failures of hypervelocity impacts).
- d. Include dynamic simulations to verify that the operations (e.g. venting) do not result in unpredictable attitude or orbit.

4.11.3 Mitigation measures

Possible mitigation measures to minimize risks related to stored energy after the disposal phase are summarized in Annex E.

4.12 Requirement 6.3.1.1: disposal reliability threshold

ECSS-U-AS-10C / ISO 24113:2011 6.3.1.1	<i>See ISO Standard for text of requirement</i>
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4.12.1 Rationale for the requirement

The requirement aims at ensuring by design a high probability of performing the disposal of the space system, and, therefore, minimize the risk to remain in the LEO or GEO Protected Regions after EOL.

For the cases of controlled re-entry, the requirement can also contribute to ensure that there is low probability of debris falling over populated areas resulting in the increase of the expected casualties.

It is highlighted that the requirement addresses the probability of successful disposal at the time the disposal is executed, which is different on a mission success probability requirement that typically has a nature of a forecast not facing a failure causing the loss of the mission. The disposal reliability requires to maintain a probability of 0,9 for a successful disposal up to the point when the disposal is executed, i.e. it requires to react on in-flight anomalies affecting the disposal function. Therefore, solely predicting a disposal reliability of equal or higher than 0,9 during the development phase does not allow to verify the compliance with the requirement. Nevertheless, predicting the disposal reliability accounting for the nominal mission duration is an important task to ensure a high chance of not having to abort the mission because the required disposal reliability cannot be maintained anymore the mission.



Note that the reliability at EOM can be considered a conservative prediction of the disposal reliability and no different estimation is necessary along the development phase under the following conditions:

- a. The disposal function is a subset of the space system bus functions used to perform the mission.
- b. The potential for an early termination of the mission should not be accounted.
- c. The duration of the disposal is negligible with respect to the mission duration.

4.12.2 Methods to assess compliance

The verification of compliance with the requirement during the development phase should be based on analysis, to forecast the probability of successful disposal at the time the disposal is planned (see Annex F).

4.12.3 Mitigation measures

Possible mitigation measures to ensure the compliance with the requirement include:

- a. To take into account the requirement on EOL disposal together with an EOL strategy in the early phases of the space system development.
- b. To re-evaluate the reliability of successful disposal during the life of the space system in order to take into account evolution of the reliability of the functions and the availability of the on-board resources required for the disposal phase such that the EOL strategy can be timely modified to ensure a 0,9 probability of successful disposal. Methods for assessing the propellant mass availability are summarized in Annex D.
- c. To include in the mission plan the possibility to terminate the mission before its nominal end if the availability of the functions and resources for EOL disposal go below what has been planned at the beginning of the mission (e.g. large use of propellant, degradation of subsystems).
- d. To re-design the system such that not to have Single Point of Failures (SPFs) that may result in loss of the mission and generation of space debris.



4.13 Requirement 6.3.1.2: disposal reliability assessment

ECSS-U-AS-10C / ISO 24113:2011 6.3.1.2	<i>See ISO Standard for text of requirement</i>
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4.13.1 Rationale for the requirement

The requirement aims at providing the method to compute the 0,9 probability of successful disposal defined in the requirement 6.3.1.1 of ECSS-U-AS-10C / ISO 24113.

This requirement should drive the actual disposal reliability of the space system in orbit during the mission to monitor and maintain compliance with the defined acceptable level (0,90), i.e. verification of the compliance as an operational requirement such that the space system has a 90% chance to successfully perform the disposal whenever point along the mission to dispose the space system is reached.

The requirement is under consideration for a reformulation in the frame of ECSS in order to define a figure of the probability of successful disposal as a non-conditional probability.

4.13.2 Methods to assess compliance

See section 4.12.2

4.13.3 Mitigation measures

See section 4.12.3.

4.14 Requirement 6.3.1.3: disposal reliability constraints

ECSS-U-AS-10C / ISO 24113:2011 6.3.1.3	<i>See ISO Standard for text of requirement</i>
* illustration is given in Annex B of ISO 24113:2011	

4.14.1 Rationale for the requirement

The requirement aims at ensuring that the disposal operations plan is compatible with the system reliability at EOL.

4.14.2 Methods to assess compliance

See section 4.12.2.

4.14.3 Mitigation measures

See section 4.12.3.



4.15 Requirement 6.3.2.1: GEO clearance

ECSS-U-AS-10C / ISO 24113:2011 6.3.2.1	<i>See ISO Standard for text of requirement</i>
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4.15.1 Rationale for the requirement

The requirement addresses spacecraft and launch vehicle orbital stages operating in the GEO Protected Region and it aims at clearing the GEO Protected Region from space systems once they have completed their mission.

The requirement applies as well to space systems operating outside the GEO Protected Region, in case these space systems can drift into orbits that cause permanent or periodic presence in the GEO Protected Region (e.g. due natural orbit evolution from an HEO or Lagrange Points).

4.15.2 Methods to assess compliance

The verification of compliance with the requirement should be based on the space system operation plan, review-of-design, and analysis, including:

- a. The planned EOL maneuvers, providing information on the EOL space system status and parameters, and demonstrating the availability of a reliable propulsion system and propellant mass budget adequate to manoeuvre the space system to the disposal orbit.
- b. The orbital trajectory propagation after the EOL planned maneuvers for at least 100 years after the completion of the EOL maneuvers (see Annex A).
- c. For orbits around Earth-Sun Lagrange Points, as a minimum the probability to interfere with the GEO Protected Region for longer than 100 year after the end of the operational phase (e.g. by using Monte Carlo methods within a reasonable parameter space, or justified assumption and analyses).

4.15.3 Mitigation measures

4.15.3.1 General

Possible mitigation measures to comply with the requirement include the transfer of the space system from GEO to a graveyard orbit that satisfies as minimum the requirement 6.3.2.2 of ECSS-U-AS-10C / ISO 24113.

4.15.3.2 Mitigation measures for temporary presence in the GEO Protected Region

There can be cases where space systems operating completely outside the GEO Protected Region, at EOL or due to an intentional disposal action, can drift into orbits that cause permanent or periodic presence in the LEO or GEO Protected Regions.

Possible mitigation measures to comply with the requirement include:

- a. To lower the apogee below the GEO Protected Region (e.g. lower than 550 km from GEO), which will avoid later interference with GEO, but without interfering with lower existing or assigned operational orbits.



- b. To select a disposal orbit on which natural perturbations lead to a permanent clearance of the GEO Protected Region and any other operational orbit (in GTO, MEO, HEO) after the end of the operational phase.
- c. To dispose the space system into heliocentric orbits with no revisit closer than 1,5 million km to Earth within the next 100 years (e.g. if orbiting around the Lagrangian Point L2).

4.15.3.3 Mitigation measures for possible interference with other operational orbits

For the EOL of a space system on any Earth-bound (including orbits around the Earth-Sun Lagrange Points), efforts should be made such that the disposal orbit and the resulting orbital evolution does also not lead to interference with known operational satellite constellations, e.g. in MEO, as:

- a. GLONASS: ca. 19000 km – 19200 km geocentric altitude
- b. GPS: ca. 20000 km – 20500 km geocentric altitude
- c. BEIDOU: ca. 21400 km – 21700 km geocentric altitude
- d. GALILEO: ca. 23200 km – 23300 km geocentric altitude

For space systems operated on circular MEOs (such as the Galileo orbit) or potentially interfering with any other operational orbit in GTO, MEO, HEO, possible mitigation measures include:

- a. To re-orbit the space system to a higher altitude with sufficient clearance from the MEO operational regions, and, as well as the GEO Protected Region, such that intersection is never occur (or, at least, for 100 years).
- b. To de-orbit the space system to a lower altitude with sufficient clearance from the MEO operational regions such that intersection is never occur (or, at least, for 100 years).
- c. To maneuver the space system onto an eccentric orbit where natural effects lead to a maximum eccentricity increase such that a re-entry can be reached within a reasonable amount of time.
- d. To select a disposal orbit on which natural perturbations can lead to a permanent clearance of the operational orbits in GTO, MEO, HEO after the end of the operational phase.
- e. To dispose the space system into heliocentric orbits with no revisit closer than 1,5 million km to Earth within the next 100 years (e.g. if orbiting around the Lagrangian Point L2).

4.16 Requirement 6.3.2.2: GEO disposal maneuvers

ECSS-U-AS-10C / ISO 24113:2011 6.3.2.2	<i>See ISO Standard for text of requirement</i>
<p><i>*C_r is the solar radiation pressure coefficient (dimensionless)</i></p> <p><i>**A/m is the ratio of the cross-section area (in m^2) to dry mass (in kg) of the space system</i></p>	

4.16.1 Rationale for the requirement

The requirement aims at providing the criteria to allow clearance of the GEO Protected Region and compliance with the requirement 6.3.2.1 of ECSS-U-AS-10C / ISO 24113 for spacecraft and any launch vehicle orbital stage.



The requirement is derived from IADC recommendations (2007), which take into account: clearance of 235 km as the sum of the upper altitude of the GEO Protected Region (200 km) and the maximum descent of a re-orbited spacecraft due to lunisolar and geopotential perturbations (35 km); the factor ($1000 * C_r * A/m$), which reflects the effect of the solar radiation pressure depending on the physical properties area and mass of the satellite, and the solar radiation pressure coefficient (C_r); maximum eccentricity of 0,003, to ensure highest perigee altitude if there is much uncertainty in the estimated quantity of residual propellant, to minimise the deviation between the apogee and perigee altitudes, and to increase the stability of the disposal orbit from lunisolar perturbation.

4.16.2 Methods to assess compliance

See section 4.15.2.

4.16.3 Mitigation measures

See section 4.15.3.

4.17 Requirement 6.3.3.1: LEO clearance

ECSS-U-AS-10C / ISO 24113:2011 6.3.3.1	See ISO Standard for text of requirement
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4.17.1 Rationale for the requirement

The requirement aims at limiting the presence in LEO of space systems that have ended their mission and, therefore, in order to reduce the risk of collision which can generate large quantities of debris and ultimately render some orbital regions completely unusable.

The historical practice of abandoning spacecraft and launch vehicle orbital stages at EOM has allowed more than 2 million kg of debris to accumulate in LEO. Catastrophic collisions among these objects and feedback collisions (invoked by the fragments of collisions) lead to an uncontrolled increase in the object number, which can render some orbital regions in LEO completely unusable (Kessler syndrome).

The 25 years limit is derived from IADC guidelines as a result of studies on the forecast of evolution of the population of space systems and debris in LEO. It is a compromise between the limitation in the growth of the debris environment over the next 100 years and the cost burden to programs and projects for implementing measures to limit the presence on orbit after EOM.

For any orbit both totally included or even temporarily intersecting the LEO Protected Region, the maximum allowed stay (25 years) comes into effect from the time of the first interference with the LEO Protected Region, i.e. as soon as the launch stage is released, until the time of the last interference, i.e. terminated by atmospheric re-entry or reached a disposal orbit outside the LEO and GEO Protected Regions.

4.17.2 Methods to assess compliance

The verification of compliance with the requirement should be based on the space system operation plan, review-of-design, and analysis, including:



- a. The planned EOL maneuvers, providing information on the EOL space system status and parameters, and demonstrating the availability of a reliable propulsion system and propellant mass budget adequate to manoeuvre the space system to the disposal orbit.
- b. The orbital trajectory propagation after the EOL planned maneuvers after the completion of the EOL maneuvers (see Annex A).
- c. For orbits around Earth-Sun Lagrange Points, as a minimum the probability to interfere with the LEO Protected Region for longer than 25 year after the end of the operational phase (e.g. by using Monte Carlo methods within a reasonable parameter space, or justified assumption and analyses).

4.17.3 Mitigation measures

4.17.3.1 General

Possible mitigation measures to limit the presence of space systems in LEO to 25 years are listed in the requirement 6.3.3.2 of ECSS-U-AS-10C / ISO 24113.

4.17.3.2 Mitigation measures for temporary presence in the LEO Protected Region

Possible mitigation measures to comply with the requirement include:

- a. To select a disposal orbit on which natural perturbations lead to a permanent clearance of the LEO Protected Region within 25 years after the end of the operational phase.
- b. To dispose the space system into heliocentric orbits with no revisit closer than 1,5 million km to Earth within the next 100 years (e.g. if orbiting around the Lagrangian Point L2).

4.18 Requirement 6.3.3.2: LEO disposal maneuvers

ECSS-U-AS-10C / ISO 24113:2011 6.3.3.2	<i>See ISO Standard for text of requirement</i>
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4.18.1 Rationale for the requirement

See section 4.17.1.

4.18.2 Methods to assess compliance

See section 4.17.2.

4.18.3 Mitigation measures

The requirement identifies various approaches to limit the presence of space systems in LEO at EOM.

In general, the most energy-efficient way to comply is to shorten the orbital lifetime via a manoeuvre to an orbit from which natural decay can occur within 25 years after the end of the operational phase.

The retrieval of a space system and return to Earth by means of an external chaser vehicle (Active Debris Removal) is an option that currently cannot be considered yet as a feasible baseline solution for all spacecraft and launch vehicle orbital stages due to high operational costs and low technology readiness.



4.19 Requirement 6.3.4.1: re-entry casualty risk acceptance

ECSS-U-AS-10C / ISO 24113:2011 6.3.4.1	See ISO Standard for text of requirement
ESA/ADMIN/IPOL(2014)2 Section 2	<p>a) For ESA Space Systems for which the System Requirements Review has already been kicked off at the time of entry into force of this Instruction (28/03/2014), casualty risk minimisation shall be implemented on a best effort basis and documented in the Space Debris Mitigation Report.</p> <p>b) For ESA Space Systems for which the System Requirements Review has not yet been kicked off at the time of entry into force of this Instruction (28/03/2014), the casualty risk shall not exceed 1 in 10000 for any re-entry event (controlled or uncontrolled). If the predicted casualty risk for an uncontrolled re-entry exceeds this value, an uncontrolled re-entry is not allowed and a targeted controlled re-entry shall be performed in order not to exceed a risk level of 1 in 10000.</p>

4.19.1 Rationale for the requirement

This requirement addresses with a maximum risk figure each spacecraft, launch vehicle stage, and MRO, which can re-enter in controlled or uncontrolled way. ESA as “approving agent” has set the maximum acceptable casualty risk of 10^{-4} (computed as casualty expectancy) for each re-entry event of space systems procured under ESA Programmes, including spacecraft, launch vehicles stages, inhabited or robotic vehicles ([RD1]).

4.19.2 Methods to assess compliance

The verification of compliance with the requirement should be based on analysis, to assess the re-entry casualty risk as described in Annex C.

The analysis should cover all mission-related objects which can re-enter, i.e.:

- All objects with permanent presence in the LEO Protected Region.
- All objects with periodic presence in the LEO Protected Region.
- Objects on HEOs, if an orbital lifetime analysis reveals a non-zero re-entry probability for a timeframe of 100 years after the end of the operational phase.
- Objects on orbits around Sun-Earth Lagrange Point or on heliocentric orbits with periodic vicinity to the Earth, if an analysis reveals a non-zero re-entry probability for a timeframe of 100 years after the end of the operational phase.

4.19.3 Mitigation measures

Possible mitigation measures to ensure compliance with the requirement include designing the space system such that demise during re-entry is maximized.



4.20 Requirement 6.3.4.2: re-entry casualty risk assessment

ECSS-U-AS-10C / ISO 24113:2011 6.3.4.2	<i>See ISO Standard for text of requirement</i>
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4.20.1 Rationale for the requirement

A specific re-entry casualty risk analysis (see Annex C) is required to determine the risk to human population due to the destructive re-entry of any human-made object.

Clarification: when a space system is deliberately or unintentionally partitioned close to the time of the re-entry event, the risk figures for the resulting parts should be summed up, i.e. a break-up of the space system can be used to influence the overall re-entry-risk figure, but it should not be used to intentionally create multiple re-entry events and treat them as separate cases.

4.20.2 Methods to assess compliance

The verification of compliance with the requirement should be based on:

- a. For an uncontrolled re-entry of a space system:
 1. Analysis (see Annex C) to demonstrate that the maximum casualty risk is below 10^{-4} .
- b. For a controlled re-entry of a space system:
 1. Analysis (see Annex C) to demonstrate that the maximum casualty risk is below 10^{-4} .
 2. Analysis (see Annex C) to demonstrate that the Safety Re-entry Area (SRA) does not extend over inhabited regions, does not impinge on the state territories and the territorial waters without the agreement of the relevant authorities.

NOTE 1 As per ESA/ADMIN/IPOL(2008)2 - ESA Space Debris Mitigation for Agency Projects - ESA Director General's Office, 01/04/2008, requirement OR-07: "In case the total casualty risk is larger than 10^{-4} , uncontrolled re-entry is not allowed. Instead, a controlled re-entry should be performed such that the impact footprint can be ensured over an ocean area, with sufficient clearance of landmasses and traffic routes."

NOTE 2 Territorial waters, i.e. 12 nm (22,2 km) from coastline, are considered to be part of national territories.

NOTE 3 The sovereign state should be informed in case of interference with its Economic Exclusive Zone (EEZ), i.e. 200 nm (370,4 km) from coastline.

NOTE 4 In the frame of the ATVs re-entries, the South Pacific Ocean Uninhabited Area (SPOUA) has been identified as the largest unpopulated area to target those controlled re-entries (for ATV defined by the longitude range from 185° East, i.e. -175°deg, to 275°East, i.e. -85°, and latitude range from 29° South to 60° South).



NOTE 5 Preserving zones classified as Marine Protected Areas for environment safeguard can be a constraint to take into account.

3. Notice to the relevant authorities overseeing the affected air and sea space authorities to supply them, in a timely manner and in-line with their procedures, with the technical information they need in order to issue NOTAM (Notice To Airmen) and AVURNAV / NAVAREA (Notice to Mariners) messages (e.g. NOTAM warning for ATV-4 re-entry).

For objects released during the launch phase and falling back without orbiting around Earth (e.g. launch vehicle stages, fairings), the verification of compliance is ruled by the launch safety regulations applicable to the launch service.

NOTE 1 In particular, according to Article 20 of the «Arrêté du 31 mars 2011 relatif à la réglementation technique en application du décret no 2009-643 du 9 juin 2009 relatif aux autorisations délivrées en application de la loi no 2008-518 du 3 juin 2008 relative aux opérations spatiales - 31/05/2011», France applies for launch safety the following requirements:

“1. For the cumulative catastrophic damage risks, the launch operator must meet the following quantitative objectives, expressed as a maximum allowable probability of causing at least one casualty (collective risk):

a) Lift-off risk

- *$2 \cdot 10^{-5}$ for the entire launch phase, including consideration of degraded launch system situations and fall-back of elements designed to separate from the launcher without being placed in orbit,*
- *10^{-7} by nominal fall-back of those elements designed to separate from the launcher without being placed in orbit, in accordance with paragraph 1 of article 23 of this order.”*

4.20.3 Mitigation measures

Possible mitigation measures to reduce the casualty expectancy are:

- a. To design the space system such that the most components and materials can demise during re-entry. This can imply, for example, preference for materials with low melting temperature (e.g. Aluminium instead of Titanium, stainless steel, Tungsten) or appropriate space system architecture favouring structural demise.
- b. To perform a controlled re-entry over uninhabited areas.

5

Space Debris Mitigation documentation

The Space Debris Mitigation documentation includes:

- a. The Space Debris Mitigation Plan (SDMP), which should be provided from the Preliminary Requirements Review (PRR), documenting how the compliance with the Space Debris Mitigation requirements is intended to be achieved. For the table of content of this document see Annex G.
- b. The Space Debris Mitigation Report (SDMR), which should be provided from the Preliminary Design Review (PDR), documenting the implementation and verification of the Space Debris Mitigation requirements, to be updated at each major project review and submitted for approval at time of the Acceptance Review (AR). For the table of content of this document see Annex H.

The process for the issue, review and approval of the SDMP and SDMR is summarized in Table 5-1.

Table 5-1: Space Debris Mitigation documentation process

Doc.	Review/ Phase	Content	ESA Technical Authority Approval / Review	Ref.
SDMP	PRR	Preliminary plan for the implementation and verification of the SDM requirements	Review	Annex G
SDMP	SRR	Final plan for the implementation and verification of the SDM requirements	Approval	Annex G
SDMR	PDR	Implementation and verification of the SDM requirements	Review	Annex H
SDMR	CDR	Update of the implementation and verification of the SDM requirements	Review	Annex H
SDMR	QAR	Update of the implementation and verification of the SDM requirements	Review	Annex H
SDMR	FAR	Update of the implementation and verification of the SDM requirements	Approval	Annex H
SDMR	FRR	Update of the implementation and verification of the SDM requirements	Review	Annex H
SDMR	Before Mission Change commitment	Update of the implementation and verification of the SDM requirements	Approval	Annex H
SDMR	EOM	Update of the risk analysis at de-commissioning	Approval	Annex H
SDMR	Prior to re-entry	Update of the risk analysis prior to re-entry	Approval	Annex H

Annex A

Orbit propagation analysis

A.1 Objectives

An orbit propagation analysis is required to provide the estimation of the time spent in orbit after the operational phase with regard to the violation of the Protected Regions for:

- a. Spacecraft after the operational phase and disposal maneuvers
- b. Launch vehicle orbital stages after the operational phase and disposal maneuvers (if applicable)
- c. All MROs (by the time of the release).

The orbit propagation analysis should be documented such that to provide:

- a. Description of the methodology of the computation
- b. Description of the model assumptions and uncertainties
- c. Description of the initial or boundary conditions
- d. Determination of the orbital trajectory propagation vs. time
- e. Determination of the presence in the LEO or GEO Protected Regions or MEO Operational Regions.

A.2 Methodology

A.2.1 General

Numerical or analytical determinations of the orbit propagation of an mission-related object are very sensitive to the model complexity and assumptions. In order to perform an orbit propagation analysis for the disposal phase guidelines are here provided to cover all relevant aspects.

The inputs for the orbit propagation analysis should be defined and modelled according to the following criteria:

- a. Disposal orbit parameters (section A.2.2)
- b. Ejection velocity (Δv) for Mission-related Objects (section A.2.3)
- c. Atmospheric drag (section A.2.4)
- d. Atmospheric density (section A.2.5)



- e. Earth gravitational attraction (section A.2.6)
- f. Lunisolar attraction (section A.2.7)
- g. Solar activity and geomagnetic index (section A.2.8)
- h. Solar radiation pressure (section A.2.9)
- i. Object cross-sectional area (section A.2.10)
- j. Object drag coefficient (section A.2.11)
- k. Object mass (section A.2.12)
- l. Object ballistic coefficient (section A.2.13)
- m. Solar radiation pressure reflectivity coefficient (section A.2.14)
- n. Propagation time and frequency (section A.2.15)
- o. Result accuracy margin (section A.2.16))
- p. Tool(s) use and acceptance (section A.2.17).

Depending on the violation of which Protected Region (i.e. LEO or GEO) the assessment is aimed to verify, an appropriate or conservative level of the model accuracy and a minimum set of assumptions should be taken into account for the propagation of the orbit after the operational phase. The assumptions and accuracy should be appropriate depending on the type of orbit, e.g. LEO, MEO, HEO, GTO, GEO, Lagrangian Points. In particular, GTOs have the most complex dynamical properties on propagation among the Earth orbits due to their high eccentricity, wide range of inclinations and semi-major axes covered, and third body perturbations, involving resonance effects, which need a statistical approach including several Monte Carlo simulations in order to find the most reliable trajectory propagation.

The output of the analysis should be:

- a. Orbital trajectory parameters propagation
- b. Time spent in the LEO or GEO Protected Region after the operational phase.

A.2.2 Disposal orbit parameters

The disposal orbit can be any Earth-bound orbit (including Sun-Earth Lagrange Point orbits) that the space system has after the end of the operational phase. Hence, this is the orbit after all EOL measures have been completed (including passivation and its effect onto the disposal orbit) and the space system has been fully decommissioned. Any additional potential active effects on the orbit (such as outgassing, residual pressure release) can be ignored. The orbit should be estimated with all six parameters and the associated epoch.

A.2.3 Ejection velocity (Delta-v) for MROs

For MROs released from spacecraft or launch vehicle stages (parent object), the ejection velocity (Delta-v) should be determined and applied to the initial conditions for the trajectory propagation. The following guidelines are proposed for the ejection velocity assumption:

- a. If the release direction is unknown, a worse case direction (acceleration into flight direction) should be assumed.



- b. An impulsive release maneuver can be assumed.
- c. The initial orbit should be computed by vector addition of the parent object osculating orbital state with the release velocity vector.
- d. The resulting osculating state should be converted into a singly average (over true anomaly) orbital state.

A.2.4 Atmospheric drag

The atmospheric drag (F_{drag}) formula is:

$$\vec{F}_{drag} = -\frac{1}{2}\rho V A_{drag} C_D \vec{V}_r \quad [A-1]$$

where:

- ρ atmospheric density
- A_{drag} cross-sectional area for atmospheric drag
- C_D drag coefficient
- V_r relative velocity between the object and the atmosphere

The atmospheric drag is relevant to determine the trajectory for LEO and GTO orbits.

A.2.5 Atmospheric density

The following atmosphere density models are recommended (see also [RD4]):

- a. NRL-MSISE-00
- b. Jacchia-Bowman 2006 (JB-2006) / Jacchia-Bowman 2008 (JB-2008)

The use of atmosphere models that were designed to fit a select altitude range (e.g. the exponential atmosphere model) or models that do not accommodate solar activity variations should be avoided as they are not sufficiently accurate.

It should be noted that the model accuracy of prediction of atmospheric density and other parameters is limited by the complex behaviour of the atmosphere, and the causes of variability. The primary influence on accuracy of the model density output is the accuracy of the future predictions of solar and geomagnetic activity used as inputs, rather than the accuracy of the specific model in representing the density as a function of solar and geomagnetic activity.

NRL-MSISE-00 has a density uncertainty of 15% for mean activity conditions and 100% for short term and local scale variations; within the homosphere the uncertainty is below 5%.

JB-2006 has a density uncertainty of 10-15% within the thermosphere, depending on altitude; for extreme conditions (very high solar or geomagnetic activities), the uncertainty can considerably increase due to the lack of corresponding measurement data; the total density can have $\pm 100\%$ variation at 400-500 km for some activities and locations.



A.2.6 Earth gravitational attraction

The Earth gravitational attraction based on JGM-3 (Joint Earth Gravity Model) is recommended with appropriate accuracy depending on the type of orbit (see also [RD4]). As a minimum, it is recommended the following approximation:

- a. LEO, MEO, HEO, GTO, GEO vicinity:
 1. zonal harmonics including J2, J3, J4, J22
 2. zonal harmonics up to J15 for orbits with inclination close to the critical inclination (63,4)
- b. Operational GEO (very close to 35786 km altitude):
 1. zonal harmonics up to 4th degree and 4th order, including J2, J3, J4, and J22

A.2.7 Lunisolar attraction

The third body lunar and solar attraction should be taken into account with appropriate accuracy when involving the following orbits (see also [RD4]). As a minimum, it is recommended the following approximation:

- a. LEO:
 1. expansion of perturbation potential up to 2nd order
- b. MEO, HEO, GTO:
 1. expansion of perturbation potential up to 2nd order
- c. GEO:
 1. expansion of perturbation potential up to 2nd order

Lunar and solar attraction is quite relevant for sunsynchronous or quasi-sunsynchronous orbits, higher LEO orbits, high eccentric LEO orbits, GTO and GEO orbits.

A.2.8 Solar activity and geomagnetic index

Solar activity (solar flux and geomagnetic index) has effect mainly on the orbital lifetime in LEO. The solar flux $F_{10,7}$, i.e. the solar flux at a wavelength of 10,7 cm in units of 10^4 Jansky (1 Jansky equals 10^{-26} Wm⁻²Hz⁻¹) and geomagnetic index A_p , i.e. the index to describe fluctuations of the geomagnetic field (range 0-400), should be used as much accurate as possible to reflect the real case. The following approaches can be adopted:

- a. Best last updated predictions:
A modified McNish-Lincoln method is used to estimate the future behaviour of the current sunspot cycle by adding to the approximated 13-month smoothed sunspot number of all past cycles (using activity proxies provided by the National Oceanic and Atmospheric Administration, NOAA) a correction term which is derived from the current cycle's deviation from the smoothed mean cycle. Such predictions are available as output of ESA's SOLMAG.
- b. Best Case (BC) and Worst Case (WC):
The method is based on an arbitrary value for the so-called confidence interval. From the space system's operator point of view a BC is referred to a shorter lifetime and therefore a high solar activity, while the opposite is the case for the WC. In order to derive the solar activity for a



given confidence, the analyst has to find the underlying probability density function for the physical process behind each solar cycle.

- c. Constant solar and geomagnetic activity:
The method is based on assuming mean values solar flux and geomagnetic index over the epoch and time of the propagation.
- d. Repeated solar cycle:
It is based on repeated cycles for the solar flux taking from the ECSS solar cycle per [RD4] (Annex A.1: Solar activity values for complete solar cycle) which provides a table with minimum, mean, and maximum daily and 81-daily averaged values for $F_{10,7}$ for each month of solar cycle 23; the values are averaged over 30-day (1 month) intervals.
- e. Monte Carlo Sampling with at least 5 sampled cycles:
The method is based on the sampling of a randomly drawn solar cycle out of available observed data from 5 preceding solar cycles.
- f. Equivalent solar flux
In order to have sensitivity with the solar activity an equivalent solar flux has been defined as a constant value depending on altitude and ballistic coefficient such that 50% of the simulations imply an orbital lifetime lower or equal to 25 years. The equivalent constant solar flux as derived in with the following conditions:

$$\begin{cases} F_{10,7} = 201 + 3,25 \ln \left(\frac{A_{avg} C_D}{M} \right) - 7 \ln(H_a) \\ A_p = 15 \end{cases} \quad [A-2]$$

where:

$F_{10,7}$	solar flux in sfu
A_{avg}	object average cross-sectional area in m ²
C_D	object drag coefficient (2,2)
M	object mass
H_a	apogee altitude in km
A_p	geomagnetic index (average $A_p = 15$ as per [RD4])

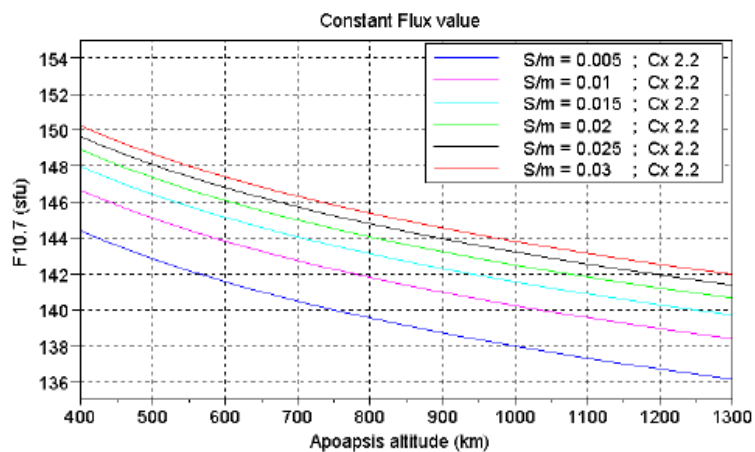
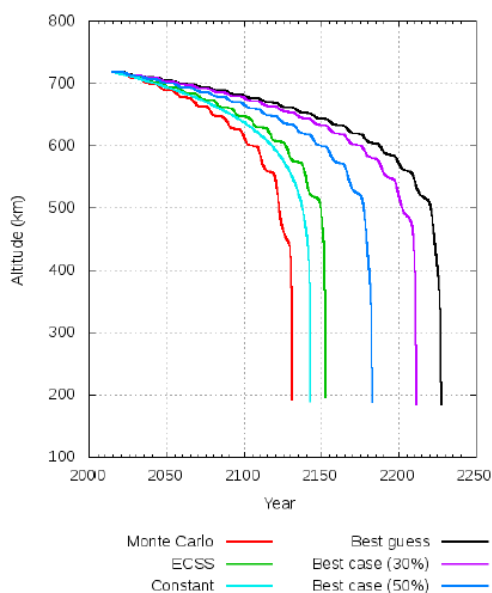


Figure A-1: Equivalent solar flux vs. orbit altitude for different values of the ballistic coefficient (area-to-mass ratio, S/m ; drag coefficient, C_x)



**Figure A-2: An example of orbital lifetime assessment for different solar cycle scenarios:
Cryosat-2, initial orbit of 710 km x 726 km, 92° inclination, epoch 01/01/2015, cross-section
3,4 m²**

A.2.9 Solar radiation pressure

The solar radiation pressure force (F_{srp}) is computed as:

$$\vec{F}_{srp} = C_r P_0 A_{srp} \left(\frac{R_0}{R} \right)^2 \vec{u} \quad [A-3]$$

where:

- C_r solar radiation pressure reflectivity coefficient
- R Sun-object distance
- R_0 Reference distance (Earth-Sun mean distance = 1 AU)
- P_0 solar radiation pressure at R_0
- A_{srp} cross-sectional area for solar pressure radiation (see section A.2.10)
- \vec{u} unit vector of the Sun-object direction

The solar radiation pressure is significant for orbits with a strong coupling with the J2 perturbation. Solar radiation pressure perturbation is typically computed considering a cylindrical Earth shadow.

A.2.10 Object cross-sectional area

The cross-sectional area of an mission-related object is a parameter necessary to compute the atmospheric drag and the solar radiation pressure. Nevertheless, these forces cannot necessarily act along the same direction, and, therefore, the cross-sectional area definition cannot be the same. The cross-sectional area is orthogonal to the direction along which the force to be computed is acting, i.e.:

- a. The cross-sectional area used for the determination of the atmospheric drag (A_{drag}) is the object area projection on the plane orthogonal to the flow direction.
- b. The cross-sectional area used for the determination of the solar pressure radiation (A_{srp}) is the object area projection on the plane orthogonal to direction of the Sun.

The determination of cross-sectional area should properly cover all uncertainties that cannot be predicted with sufficient accuracy, i.e. mission-related object orbital state and attitude. Therefore, a set of values of the cross-sectional area should be identified and used for several numerical propagations in order to cover all possible scenarios without neglecting the worst cases. For example, the following considerations should be taken into account:

- a. Object geometrical configuration at the beginning of the assessment, i.e. release time of a MRO, end of the operational phase of a spacecraft or launch vehicle stage.
- b. Object attitude, i.e. stabilization, uncontrolled stabilization, gravity gradient stabilization or aerodynamic stabilization effects, random tumbling, or any other damping effects.

Since the nature of the objects released in orbit is non-functional, they are expected to be uncontrolled. Under certain conditions uncontrolled objects can be gravity gradient stabilized or aerodynamically stabilized.

An analysis should be performed to determine the expected mid- and long-term attitude state after loss of control:



- a. If enough justification can be presented for one of the following two attitude modes and if they can be accurately quantified, they can be assumed for the next steps of the analysis:
 1. Stable (inertial or with respect to Earth orientation)
 2. Rotation around one axis with known and constant motion vector
- b. If, as in most cases, the type of motion is unknown and longer-term predictions on the rotation axis are noisy also damping effects become very difficult to quantify during the development phase, and, therefore, it should be assumed that the object is:
 1. Randomly tumbling

A tool to determine the cross-sectional area of complex geometries is implemented in the DRAMA tool (CROC) and in NASA's DAS.

A.2.11 Object drag coefficient

A wrong assumption of the drag coefficient can lead to errors in the orbital lifetime duration even of 10%. The drag coefficient of a spacecraft or a launch vehicle stage is determined through:

- a. Experimental analysis in wind tunnels, if available
- b. Integral solution of analytical equations (i.e. integration over the body surface of normal and tangential momentum exchanged between the flow and the body)
- c. Summation of 6 or more simple-sided plates
- d. Direct Simulation Monte Carlo (DMSC).

If an accurate estimation of the drag coefficient for the specific geometry, altitude, solar activity, and flow regime is missing, an average value with a margin should be taken into account, e.g. 2,2 is acceptable for long-duration orbital lifetime.

A.2.12 Object mass

The mass (M) of mission-related object at time or phase of the prediction should be considered. This includes the object dry mass plus eventual residual fluids (e.g. unused propellant). If the value of the mass at the time of prediction is not known with sufficient certainty, a reasonable margin (e.g. $\pm 20\%$ at PRR, 10% at PDR, and 5% at CDR) should be taken into account.

A.2.13 Object ballistic coefficient

The ballistic coefficient (M) is defined as:

$$B = \frac{M}{A_{drag} C_D} \quad [A-4]$$

where:

- M mass (see section A.2.12)
- A_{drag} cross-sectional area for drag (see section A.2.10)
- C_D drag coefficient (see section A.2.11)

A.2.14 Solar radiation pressure reflectivity coefficient

The solar radiation pressure reflectivity coefficient (C_r) is a parameter used to compute the solar pressure radiation force. The determination of the reflectivity coefficient mainly depends on the larger areas (e.g. solar panels) and decrease with ageing.

If C_r has not been determined with sufficient accuracy, a conservative value should be assumed with respect to the violation of the LEO or GEO Protected Region. The following typical values are used:

- a. LEO, MEO, HEO, GTO:

- 1. $C_r = 1,2$

- b. GEO:

- 1. $C_r = 1,5$

A.2.15 Propagation time and frequency

The following settings are recommended:

- a. Propagation time of at least 200 years, unless re-entry occurs before.
- b. Frequency of the orbital states of at least 1 per day.

A.2.16 Result accuracy margin

Since the uncertainties on the physical parameters and assumptions on the models, an error can affect the accuracy of the determination of orbital lifetime and presence in the LEO or GEO Protected Regions. Therefore, the final value should be considered with an appropriate error margin, which can be even higher than 10%, if the analysis is based on too few simulations or poor or rough assumptions.

A.2.17 Tool(s) use

Tools for orbit propagation analysis are typically based on numerical solution of 3D differential equations for orbital dynamics.

The following tools are currently used to perform an orbit propagation analysis as implementing a sufficient solution methodology and are endorsed by ESA:

a. DRAMA/OSCAR

Use of other different tools is also possible. It is required that, a priori, the selected tool needs to be discussed with ESA.

A.2.18 Empirical simplified look-up

An example of the orbital lifetime prediction as function of the initial altitude at the equator and the mass-to-area ratio is presented in Figure A-3, which is based on a numerical propagator considering the MSIS-90E Thermosphere Model (Hedin), an 8th order and degree gravity model, lunisolar perturbations, solar radiation pressure, and SOLMAG solar activity predictions (best last update prediction). The start epoch is the 1st January 2015. The re-entry is assumed as soon as the perigee altitude is below 120 km. Nevertheless, note that it is a simplified diagram and should only be used for rough assessments.

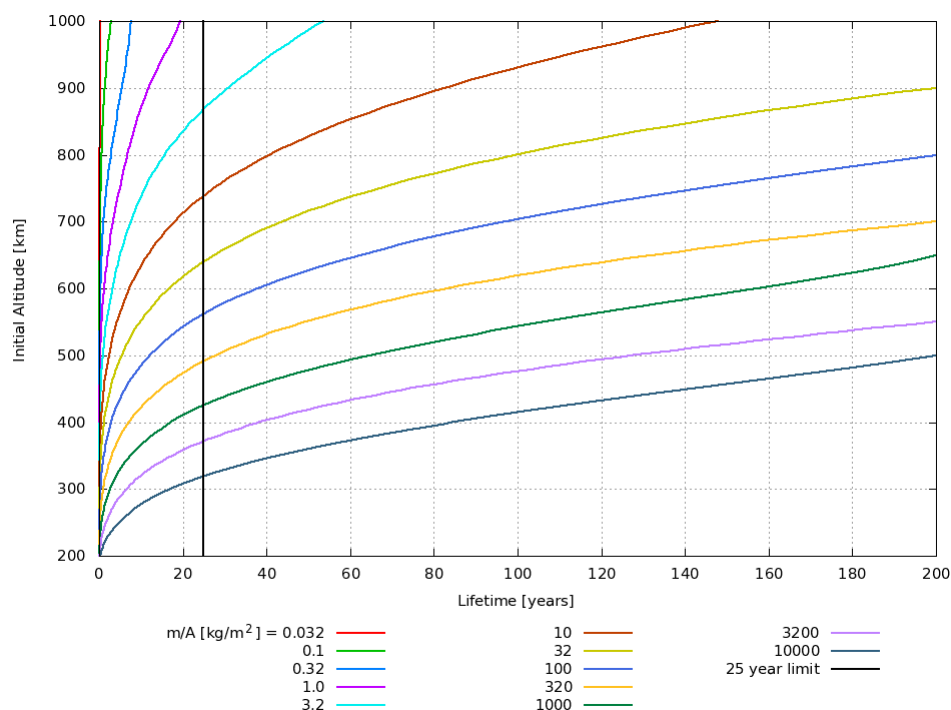


Figure A-3: Orbital lifetime and (initial) geodetic altitude for an object decaying from a circular orbit as function of mass-to-area ratio

Annex B

On-orbit collision risk and vulnerability analysis

B.1 Objectives

Although there are no specific requirements that explicitly addresses the probability of accidental collisions, there are general guidelines available:

- a. UN Space Debris Mitigation Guidelines of the Committee on the Peaceful Uses of Outer Space (United Nations (UN), 2010):

Guideline 3: Limit the probability of accidental collision in orbit

In developing the design and mission profile of spacecraft and launch vehicle stages, the probability of accidental collision with known objects during the system's launch phase and orbital lifetime should be estimated and limited. If available orbital data indicate a potential collision, adjustment of the launch time or an on-orbit avoidance maneuver should be considered.

- b. IADC Space Debris Mitigation Guidelines (IADC-02-01, Rev. 1, 01/09/2007):

5.4 Prevention of On-Orbit Collisions

In developing the design and mission profile of a spacecraft or orbital stage, a program or project should estimate and limit the probability of accidental collision with known objects during the spacecraft or orbital stage's orbital lifetime. If reliable orbital data is available, avoidance maneuvers for spacecraft and co-ordination of launch windows may be considered if the collision risk is not considered negligible. Spacecraft design should limit the consequences of collision with small debris which could cause a loss of control, thus preventing post-mission disposal.

An On-orbit collision risk and vulnerability analysis is required to estimate the collision risk and eventual mitigation measures (e.g. avoidance maneuvers) of:

- a. Spacecraft
- b. Spacecraft Tether Systems (if applicable)
- c. Launch Vehicle Orbital Stages (if applicable).

The on-orbit collision risk and vulnerability analysis should be documented such that to provide:

- a. Description of the methodology of the computation
- b. Description of the model assumptions and uncertainties
- c. Description of the initial and boundary conditions



- d. Determination of the catastrophic collision threshold
- e. Determination of the probability of catastrophic collision against large objects or debris
- f. Determination of the collision avoidance maneuvers to reduce the probability of catastrophic collision with large objects or debris
- g. Determination of the probability of catastrophic collision with small debris and meteoroids
- h. Determination of the probability of damage or failure due to collisions (vulnerability).

B.2 Methodology

B.2.1 General

The probability of an accidental break-up due to an impact or collision against an orbiting object is always not negligible. Collisions with space debris or meteoroids can cause:

- a. Spacecraft or launch vehicle stage break-up, i.e. catastrophic collision.
- b. Spacecraft or launch vehicle stage failure, e.g. propellant tank rupture or leakage, critical damages to attitude and control sensors or actuators, solar arrays, power lines.

There are two types of particles constituting a hypervelocity-impact (HVI) risk to space systems:

- a. Meteoroids, i.e. cometary or asteroidal fragments.
- b. Human-made orbital debris, including larger trackable objects and smaller non-trackable particles.

B.2.2 Catastrophic collisions threshold

The main parameter to determine whether a collision is catastrophic, i.e. generating fragments, or non-catastrophic is the energy-to-mass ratio (EMR):

$$EMR = \frac{\frac{1}{2} M_p V_{imp}^2}{M_t} \quad [B-1]$$

where:

- M_p projectile mass (i.e. an orbiting and uncontrolled debris)
- V_{imp} impact velocity (i.e. relative velocity between the projectile and target)
- M_t target mass (i.e. the spacecraft or launch vehicle stage)

The threshold for a catastrophic collision $(EMR)_{cc}$ is assumed to be:

$$EMR \geq (EMR)_{cc} = 40 \text{ J/g}$$

To compute the impact velocity (V_{imp}) against a debris, the worst case should be considered, which can occur when the target has a head-on (frontal) collision in the perigee against a projectile. In this case the impact velocity is the sum of the velocity at perigee of the target plus the circular velocity of an object orbiting at the perigee altitude of the target:



$$V_{imp} = \sqrt{\frac{2\mu}{r_p} - \frac{\mu}{a}} + \sqrt{\frac{\mu}{r_p}} \quad [B-2]$$

where:

μ gravity constant (i.e. the product of the gravitational constant and planet mass; for Earth $\mu = 3,98604415 \cdot 10^{14} \text{ m}^3\text{s}^{-2}$, [RD4])

r_p perigee distance

a semi-major axis

Once the impact velocity is computed, it is possible to extract a threshold mass for the projectile:

$$M_p = \frac{2 \cdot (EMR)_{cc} \cdot M_t}{V_{imp}^2} \quad [B-3]$$

To convert this threshold mass to a threshold size, a standard projectile with the following characteristics should be used:

- a. Area-to-mass ratio: $(A_p/M_p)_{std} = 0,01 \text{ m}^2/\text{kg}$
- b. Spherical surface with diameter D: $A_p = \pi/4 \cdot D_p^2$

Following this criterion, two different levels of events can occur, which require two different types of analysis, respectively:

- a. if $D_p \geq \sqrt{\frac{8}{\pi} \left(\frac{A_p}{M_p} \right)_{std} \frac{(EMR)_{cc} M_t}{V_{imp}^2}} \rightarrow$ Catastrophic Collision Analysis
- b. if $D_p < \sqrt{\frac{8}{\pi} \left(\frac{A_p}{M_p} \right)_{std} \frac{(EMR)_{cc} M_t}{V_{imp}^2}} \rightarrow$ Non-catastrophic Collision Analysis

As an example, a meteoroid with V_{imp} around 20 km/s (typical velocity value) and mass of 0,01 kg (maximum meteoroid mass according to MASTER-2009) implies a catastrophic collision for impact with spacecraft with mass below 50 kg, and a non-catastrophic collision for spacecraft with mass above 50 kg.

B.2.3 Collision cross-sectional area

The collision cross-sectional area (A_{Coll}) of a space system (spacecraft or launch vehicle stage) is the envelope of the maximum projected area of the space system and the area of the impacting object (meteoroids or debris). For complex shaped objects, it can be determined by considering the circular area with diameter equal to the maximum object size:

$$A_{Coll} = \frac{\pi}{4} (D_t + D_p)^2 \quad [B-4]$$

where:

D_t space system (spacecraft or launch vehicle stage) maximum diameter

D_p debris or meteoroids diameter



B.2.4 Probability of catastrophic collision with large objects

This assessment should allow to understand if active collision avoidance maneuvers are required to reduce the risk of collision with space debris and large mission-related objects. Space debris and large objects category includes at least all objects which can be identified and tracked with radar observation capability, conventionally assumed to have a size (D_p):

$$D_p \geq 0,01 - 0,02 \text{ m}$$

The assessment should include the following steps:

- Definition of the phase (e.g. launch phase, operational phase, disposal phase).
- Definition of the orbit state vector (as per mission design for the phase under analysis).
- Definition of the phase duration over the orbit (as per mission design for the phase under analysis).
- Definition of the threshold size of debris likely to cause a catastrophic collision (as per criterion provided in section B.2.2).
- Use of the space debris flux as per “Business as Usual” for future predictions.
- Definition of the position uncertainty level.
- Determination of the annual collision probability ($P_{c,yr}$):

$$P_{c,yr} = \sum_{j=1}^N F_j \frac{\pi}{4} (D_t + D_{p,j})^2 \quad [\text{B-5}]$$

where:

- j index for the debris and meteoroids diameter range $[D_{p,j} D_{p,j+1}]$
- F_j flux of debris and meteoroids in the diameter range $[D_{p,j} D_{p,j+1}]$
- D_t space system (spacecraft or launch vehicle stage) maximum diameter
- $D_{p,j}$ debris or meteoroids diameter

- Determination of the collision probability over the phase duration.

B.2.5 Collision avoidance maneuvers against large objects

Performance of collision avoidance maneuvers is, basically, in charge of the operator of the space system. Nevertheless, it needs appropriate propulsion system capability and propellant mass, which require an assessment in the design phase.

The assessment should include the following steps:

- Definition of the phase (e.g. launch phase, operational phase, disposal phase).
- Definition of the orbit state vector (as per mission design for the phase under analysis).
- Definition of the propulsion system able to perform collision avoidance and returning maneuvers (as per system design).
- Definition of the propellant mass available at time of the required manoeuvre (included in the mass budget).



- e. Definition of the time between an event prediction and event occurrence (2 days).
- f. Definition of the position uncertainty level.
- g. Definition by the project of the collision avoidance strategy (e.g. 0,5 km as allowed minimum distance or specific risk threshold).
- h. Determination of the expected number of collision avoidance maneuvers per year.
- i. Determination of the total expected number of collision avoidance maneuvers over the phase duration.
- j. Estimation of the amount of propellant for collision avoidance and returning maneuvers with sufficient margins.

Each space system operator should define its philosophy, policy, and strategy for collision avoidance. The philosophy for collision avoidance should be described in the system specification to avoid the risk of lack of function or propellant, including:

- a. A basic concept for collision avoidance, i.e. determination of allowable criteria for collision probability, apply functions, propellant allocation.
- b. Collision detection measures, including self-analysis, or analysis provided by accepted providers, e.g.:
 - 1. USSTRATCOM TLE, whose data are public (or other TLE providers, if available).
 - 2. CDM (Conjunction Data Messages), whose data are distributed by JSpOC (Joint Space Operations Centre) directly to the operators (or other CDM providers, if available).
- c. Criteria for notification, i.e. conjunction distance, probability of collision.
- d. Criteria for conducting avoidance maneuvers.
- e. Strategy to access contact points to plan coordinated avoidance maneuvers, data exchanging rules.

B.2.6 Probability of catastrophic collision with small debris and meteoroids

Typically spacecraft or launch vehicle stages with mass smaller than 50 kg can be destroyed or damaged even by micrometeoroids or small debris (MMOD), e.g. explosion fragments, collision fragments, MRO, NaK droplets, SRM slag, SRM dust, paint flakes, ejecta particles, Multi-Layer Insulation (MLI) particles with size smaller than 0,02 m.

The assessment should include the following steps:

- a. Definition of the phase (e.g. launch phase, operational phase, disposal phase).
- b. Definition of the orbit state vector (as per mission design for the phase under analysis).
- c. Definition of the phase duration over the orbit (as per mission design for the phase under analysis).
- d. Definition of the threshold size of MMOD likely to cause a catastrophic collision (as per criterion provided in section B.2.2).
- e. Determination of the impact strength of the material and estimation of the thickness of the layers of material for shielding impacts.

- f. Use of a reference MMOD flux model as specified in [RD4].
- g. Definition of the position uncertainty level.
- h. Determination of the annual collision probability ($P_{c,yr}$):

$$P_{c,yr} = \sum_{j=1}^N F_j \frac{\pi}{4} (D_t + D_{p,j})^2 \quad [B-6]$$

where:

- j index for the debris and meteoroids diameter range $[D_{p,j} D_{p,j+1}]$
- F_j flux of debris and meteoroids in the diameter range $[D_{p,j} D_{p,j+1}]$
- D_t space system (spacecraft or launch vehicle stage) maximum diameter
- $D_{p,j}$ debris or meteoroids diameter
- i. Determination of the number of catastrophic impacts and collision probability over the phase duration.

B.2.7 Probability of damage or failure due to collisions

This assessment should provide the vulnerability level of the spacecraft or launch vehicle stage against the impact with space debris or meteoroids. This is also in line with the standard ECSS-Q-ST-70C Rev.1, which states:

5.1.14 Micrometeoroids and debris

- a. *The effect of impacts by micrometeoroids and debris on materials shall be reviewed and assessed on a case by case basis.*
- b. *Use of materials shall comply with safety evaluation and assessment results concerning design and application criteria or details.*
- c. *Micrometeoroids and debris analysis and test procedures shall be subject to the approval by the customer.*

The assessment should include the following steps:

- a. Definition of the operating parameters and architecture design.
- b. Definition of the impact survivability requirement with minimum Probability of “No Failure” (PNF_{min}) for each critical component, which is correlated with the Probability of “No Penetration” (PNP), and depends on the survivability of the space system against debris and meteoroids impacts in order to accomplish successful post-mission disposal. The PNF of all critical components are inputs contributing to the determination of the probability of successful disposal.
- c. Identification of the components critical for disposal and the surface of the component that, when damaged by impact, and cause the component to fail (i.e. critical surfaces).
- d. Determination of the at-risk surface area for the critical surface of each critical component, for example, through the following guidelines:



1. Determination of the parts of the critical surface that are the predominant contributor to failure (e.g. the parts that have the least protection from orbital debris and meteoroid impact):
 - (a) In the case the critical surface is equally protected by other spacecraft components the at-risk area is the total area of the critical surface.
 - (b) In the case some parts of the critical surface are less protected from impact than other parts, the at-risk area is the surface area of the parts of the critical surface most exposed to space.
2. For example, if an electronics box is attached to the inside of the outer wall of the vehicle, the at-risk area is the area of the box on the side attached to the outer wall. If the electronics box is attached to the exterior of the outer wall of the vehicle, the at-risk area can be the total area of the box, excluding the side attached to the outer wall.
3. Correction of the identified at-risk areas according to the orientation of the surface with respect to the spacecraft attitude.
4. Identification of the ballistic limit, i.e. the impact-induced threshold of failure (typically, the critical size at which perforation occurs), for each critical surface.
5. Determination of the impact-induced Probability of “No Perforation” (PNP) and “Probability of No Failure” (PNF) for each critical surface until the end of the disposal phase (assuming that the space system is properly disposed in a graveyard orbit and passivated).
6. Identification of components, material damage tolerance (e.g. toughness), layer thickness that help to shield the critical surfaces if $PNF \geq PNF_{min}$ (e.g. pressure vessels, electronic devices, sensors can require shields, bumpers or other protection measure).
7. Determination of the expected number of impacts likely to cause damages or failures.
8. Determination of the contribution to the probability of successful disposal due to the PNF combination of all critical components.

B.2.8 Tether systems

Tethers are flexible long and narrow structures with two of the dimensions much smaller than the third extended longer from a spacecraft. The potential to damage operating spacecraft can be larger than is expected solely from the tether mass and the cross-sectional area.

The probability of collision with large objects or meteoroids and small debris should be assessed with a specific analysis for the tethers using the same methodology in section B.2.4, B.2.6 and B.2.7 both for the time the tether is left in space (i.e. operational phase, disposal phase).

The collision cross-sectional area of a tether ($A_{Coll,T}$) should be determined as:

$$A_{Coll,T} = D_{Ti}L \quad [B-7]$$

where:

D_{Ti} tether diameter + diameter of orbital debris/meteoroid

L tether length



In case of severing a deployed tether, the tether can be assumed to be cut in two equal halves. The average cross-sectional area is the cross-sectional area of the tether plus the cross-sectional area of one end mass; the mass of each tether fragment is one-half of the tether plus the mass of the respective end mass.

B.2.9 Tool(s) use

DRAMA/ARES tool is currently used to determine the probability of catastrophic collision with space debris and is endorsed by ESA.

DRAMA/MIDAS tool is currently used to determine the probability of damage and failures due to collisions with space debris or meteoroids (vulnerability) and is endorsed by ESA.

Use of other different tools is also possible. Nevertheless, the selected tool needs to be discussed with ESA.

Operational conjunction screening can be performed based on two types of surveillance data:

- a. USSTRATCOM TLE, whose data are public (or other TLE providers, if available).
- b. CDM (Conjunction Data Messages), whose data are distributed by JSpOC (Joint Space Operations Centre) directly to the operators (or other CDM providers, if available).

CDM data are much better accurate and have covariances of about 50 times smaller than TLE. Nowadays, the use of CDMs is the only recommended way of performance collision avoidance analysis. However, as the provision of CDMs is not necessarily guaranteed, the analysis should be done using TLE data.

Annex C

Re-entry casualty risk analysis

C.1 Objectives

A re-entry casualty risk analysis is required to estimate the risk to cause human injuries or fatalities due to the mission. A re-entry casualty risk analysis is required for:

- a. Spacecraft
- b. Launch vehicle orbital stages
- c. All MROs

The Re-entry Casualty Risk Analyses should be documented such that to provide:

- a. Description of the methodology of the computation
- b. Description of the model assumptions and uncertainties
- c. Description of the initial or boundary conditions
- d. Determination of the re-entry probability ($P_{re-entry}$)
- e. In case of a controlled re-entry, determination of the failure conditions
- f. Determination of the geometrical and physical characteristics of all fragments surviving a re-entry and impacting on Earth, including:
 - 1. Size
 - 2. Shape
 - 3. Average cross-section (assuming randomly tumbling)
 - 4. Mass
 - 5. Material
 - 6. Casualty area
 - 7. Velocity
 - 8. Kinetic energy
- g. Determination of the casualty risk via the casualty expectancy (E_c)
- h. Determination of the Declared Re-entry Area (DRA) and Safety Re-entry Area (SRA)



C.2 Methodology

C.2.1 Re-entry probability

The calculation of the casualty risk to human population should take into account the probability of a re-entry scenario to occur ($P_{re-entry}$). This is particularly important for disposal orbits where a re-entry is not necessarily envisaged (e.g. disposal on a HEO or on an orbit around a Sun-Earth Lagrange Points). Any mission-related object falls within one of the following cases:

- $P_{re-entry} = 1$ for objects with permanent or periodic presence in the LEO Protected Region.
- $P_{re-entry} = 0$ for objects on HEOs, orbits around Sun-Earth Lagrange Points and heliocentric orbits with periodic vicinity to the Earth, if a precise long-term orbit propagation does not indicate a re-entry within a timeframe of 200 years after the end of the operational phase.
- $0 \leq P_{re-entry} \leq 1$ for objects where the initial conditions for long-term propagation cannot be precisely predictable, and a statistical approach (e.g. making use of Monte-Carlo techniques) should be used to assess accurately $P_{re-entry}$ for a timeframe of 200 years after the end of the operational phase.

C.2.2 Uncertainties for nominal controlled or off-nominal uncontrolled re-entry

The re-entry casualty risk analysis should be performed for each relevant mission scenario with sufficient confidence to cover all re-entry uncertainties:

- Nominal case, e.g. controlled or uncontrolled re-entry.
- Off-nominal cases, e.g. degraded controlled re-entry and uncontrolled re-entry due to failures.

The uncertainties for the nominal and off-nominal cases should be identified and taken into account depending on the space system design and operations. For example, the following dispersion parameters have been considered for the ESA ATV controlled re-entries:

- a. Position at last boost ignition: ± 3 km
- b. Burn Start Time: ± 5 s
- c. Delta-v realisation dispersion: Gaussian, 1σ (e.g. ± 5 %)
- d. Thrust level dispersion: Uniform, $[-5,4\%; 13,2 \text{ %}]$
- e. Thrust pitch angle: Gaussian, 3σ (e.g. 2°)
- f. Atmospheric density dispersion: Uniform, $\pm 20\%$
- g. Drag coefficient dispersion: Uniform, mean $2,2 \pm 25 \text{ %}$ (before atmospheric entry)
- h. Vehicle mass dispersion: Gaussian (depending on residual fuel)
- i. State vector at 120 km geodetic altitude: ± 3 km
- j. Break-up or explosion altitude dispersion: Gaussian, mean = 78 km, $3\sigma = 6$ km
- k. Off-nominal scenarios should be identified and considered in case of spacecraft boost failure at re-entry. The following error should be at least taken into account:



1. Error on Delta-v (burn time):
 - (a) Nominal Delta-v -30 %
 - (b) Nominal Delta-v +30 %
2. Error on Thrust Level:
 - (a) Nominal Thrust Level -50 %
 - (b) Nominal Thrust Level +60 %
3. Error on Thrust Pitch Angle:
 - (a) Nominal Pitch Angle -50°
 - (b) Nominal Pitch Angle +50°

The quantities mentioned from C.2.2a. to C.2.2k. need to be quantitatively defined and reviewed with respect to each project since, in general, they depend on the Fault Tree analysis and corresponding vehicle dynamics in the failure cases, e.g. thruster open failure, pressure drop, on-board computer reboot with different spacecraft moment of inertia, thruster.

C.2.3 Initial conditions

The re-entry trajectory should be defined including the following:

- a. Epoch, initial orbital state vector from the end of the operational phase (beginning of the disposal phase) as per assessment in Annex A.
- b. Planned disposal maneuvers, including epoch, initial orbital state vector, target state vector, boosts magnitude (Delta-v) and direction, maneuvered and ballistic phases durations as per assessment in Annex A.
- c. Epoch, initial orbital state vector at atmospheric entry, e.g. between an altitude of 120 km and 130 km, as per assessment in Annex A.
- d. Attitude as per re-entry scenario, reasonably justified, i.e. uncontrolled random tumbling, controlled stabilization, gravity gradient stabilization, atmospheric drag stabilization.

C.2.4 Object-oriented approach for fragmentation model

So-called object-oriented re-entry survivability analysis tools (see Figure C-1) are used to compute the casualty area and impact location of surviving fragments with the use of some simplifying assumption:

- a. The major spacecraft break-up altitude is pre-determined and leads to the release of all components or, alternatively, to the release of compounds with their own release conditions for sub-components. The prediction of the break-up altitudes can be based on valid physical considerations, similitudes, or probabilistic assessments.
- b. All released components are pre-determined and have simplified shapes (typically spheres, plates, cylinders, boxes).
- c. All released components are considered randomly tumbling and melt from the outside layer-by-layer, hence maintaining their shape type.

- d. The trajectory analysis of all fragments considers translational motion only (3° of freedom).

The following tools are based on the object-oriented approach:

- a. DRAMA/SESAM/SERAM (ESA; Institute of Aerospace Systems, Technische Universität of Braunschweig; DEIMOS Space S.L.U.)
- b. ASTOS/DARS (ESA; Astos Solutions GmbH)
- c. DAS (NASA)
- d. ORSAT (NASA)
- e. DEBRISK (CNES)

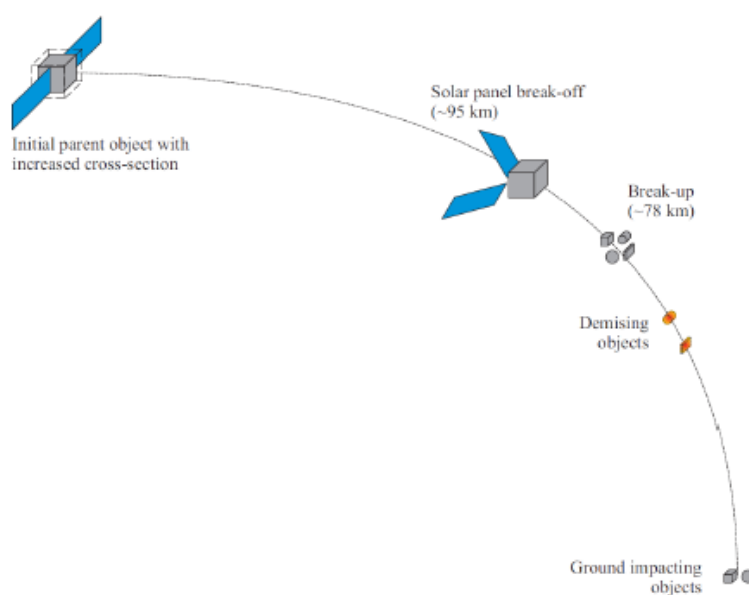


Figure C-1: Object-oriented tool concept

The object-oriented tools should be configured with a model of the space system that includes:

- a. Mass and shape of the parent body and (if present) external solar arrays
- b. Break-up altitude of external solar array (if present)
- c. Main break-up altitude of parent body (and subsequent compound break-up altitudes if available)
- d. Description of all subsystems and components, including:
 - 1. Selection of shape type (sphere, box, plate, cylinder)
 - 2. Dimensions
 - 3. Materials
- e. Material properties of all relevant components, including at least:
 - 1. Density
 - 2. Heat capacity
 - 3. Melting temperature

4. Heat of melting
5. Emissivity
- f. Justification for any assumption or simplification in the model with respect to the real structure.

C.2.5 Spacecraft-oriented approach for fragmentation model

Spacecraft-oriented tools take into account the spacecraft geometry and moments of inertia in a full-force and torque six degree of freedom analysis (see Figure C-2). A highly detailed model of the spacecraft is broken down in discrete volume panels to form the starting point of the analysis. In the subsequent simulation, aerothermal loads and heat transmission by convection, conduction and radiation, as well as aerodynamic and dynamic forces and structural loads are considered for each volume panel. Changes to the geometry due to the failure of a panel, and the consequences on the attitude and further demise and destruction process are considered. This highly deterministic approach makes spacecraft oriented codes adequate and relevant tools to study the influence of spacecraft design changes on the on-ground casualty with a high degree of realism. This process is known as “design for demise” and plays an important role for missions, which have been decided for an uncontrolled re-entry. These tools are also suited to clarify critical issues like the probability for explosive break-ups, detailed footprint analysis for controlled re-entries or the effect of critical components on the re-entry (pyrotechnics, coupled structures, large external components). SCARAB (ESA; Hypersonic Technology Göttingen) is based on the spacecraft-oriented approach.

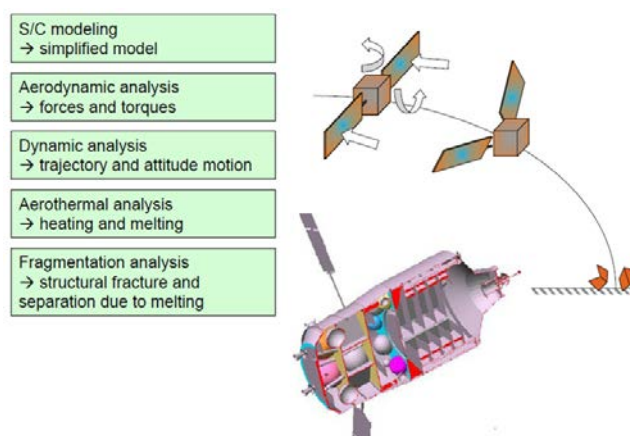


Figure C-2: Spacecraft-oriented tool concept

The re-entry model of the spacecraft –oriented tool should be as much as possible representative of the real mission-related object geometry and the aerothermodynamics, mechanical, and structural behaviour.

The re-entry model definition of the spacecraft-oriented tool should include:

- a. Overall assembly dimensions and 2D drawings with readable or measurable dimensions and positions or 3D CAD file.
- b. Detailed description of all subsystems and components, including:
 1. Shapes
 2. Dimensions
 3. 2D drawings with readable/measurable dimensions or 3D CAD file



4. Mass
 5. Center of mass
 6. Moments of inertia
 7. Materials
 8. Maximum structural loads for major connection elements
- c. Material properties of all relevant components, including at least:
1. Density
 2. Specific heat capacity
 3. Melting temperature
 4. Melting heat
 5. Emissivity
 6. Other mechanical properties relevant to the fragmentation approach, e.g. elasticity module, Poisson module, ultimate tensile stress.
- d. Justification for any assumption or simplification in the model with respect to the real structure.

C.2.6 Earth population density

Earth population density data is required to assess the casualty expectancy. Depending on the type of re-entry, i.e. uncontrolled from a circular orbit or highly eccentric orbit, or controlled, an average or local value is required. Population density should reflect as better as possible the expected epoch of the re-entry such that to take into account the population growth trend.

Earth population density data should be based on:

- a. Best estimation for the re-entry date updated at the time of the current issue of the SDMR
- b. Median projection to the re-entry date
- c. Data resolution of at least $0,25^\circ \times 0,25^\circ$.

It should be possible to derive latitude dependent population density values $\rho_p(\varphi, \Delta, \varphi)$ easy (i.e. density summed up along a latitude band $\Delta\varphi$) (as an example see Figure C-3) .

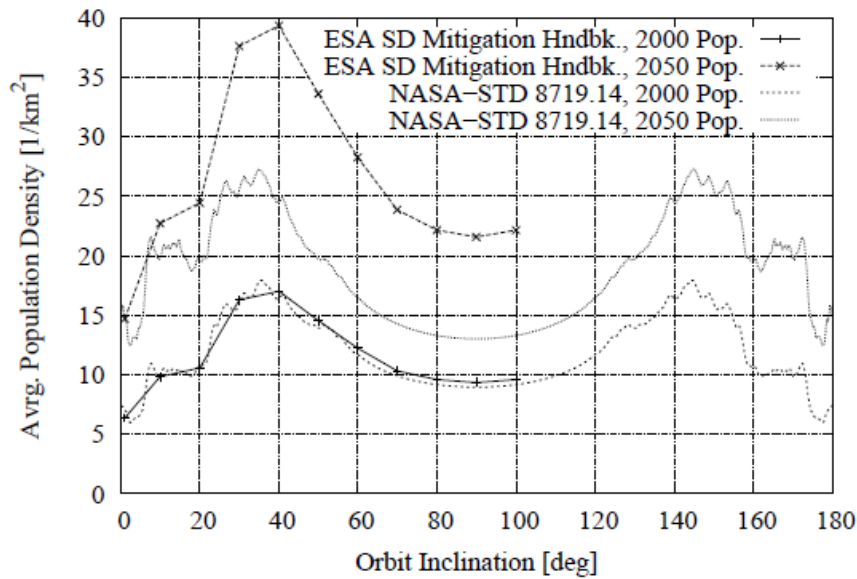


Figure C-3: An example of assessment of the average population density for a circular orbit re-entry as a function of the orbit inclination

There are many different population estimations and population growth forecast models. Available population models are, for example, the following:

- UN Population data
- Gridded Population of the World (GPW)
- DRAMA/SERAM implemented model based on GPW

C.2.7 Ground impact probability (1D)

C.2.7.1 General

When the average population density is required to determine the casualty risk for an uncontrolled re-entry, the ground impact probability should also be taken into account since the impact latitude is a function of the orbit inclination (i). The average population density as a function of the orbit inclination can be derived by calculating the weighted sum of longitudinal averaged population density distribution over the whole latitude range. The ground impact probability distribution $P_i(i, \varphi, \Delta, \varphi, \omega)$ should be used as follows to determine the average population density ρ_p :

$$\rho_p = \rho_p(i, \Delta, \varphi, \omega) = \sum_{\varphi=-\frac{\pi}{2}}^{\varphi=+\frac{\pi}{2}} \rho(\varphi, \Delta\varphi) P_i(i, \varphi, \Delta\varphi, \omega) \quad [\text{C-1}]$$

where ρ is the population density along the latitude, φ is the latitude, $\Delta\varphi$ is a margin around the latitude φ , and ω is the argument of perigee at epoch of atmospheric capture (dependency only for re-entry along eccentric orbits), and $P_i(i, \varphi, \Delta, \varphi, \omega)$ is such that:

$$\sum_{\varphi=-\frac{\pi}{2}}^{\varphi=+\frac{\pi}{2}} P_i(i, \varphi, \Delta\varphi, \omega) = 1 \quad [\text{C-2}]$$

Depending on the eccentricity of the re-entry orbit, appropriate formulations of the ground impact probability should be used (see section C.2.7.2 and C.2.7.3).

C.2.7.2 Ground impact probability for circular re-entry orbits

An analytical solution of the probability that an uncontrolled re-entry from a near circular orbit of inclination $i \in (0, \pi)$ occurs in a latitude band of width $\Delta\varphi$, centred at latitude $\varphi \in (-\pi/2, \pi/2)$, is (ESA), see Figure C-4:

$$P_i(\varphi, \Delta\varphi; i) = F(\varphi, \Delta\varphi) - \frac{1}{\pi} \arcsin\left(\frac{\sin(\varphi - \Delta\varphi/2)}{\sin(i)}\right) \quad [\text{C-3}]$$

$$F(\varphi, \Delta\varphi; i) = \begin{cases} \frac{1}{\pi} \arcsin\left(\frac{\sin(\varphi + \Delta\varphi/2)}{\sin(i)}\right) & \text{if } \varphi \leq i - \Delta\varphi/2 \\ \frac{1}{2} & \text{if } i - \frac{\Delta\varphi}{2} < \varphi \leq i + \Delta\varphi/2 \end{cases} \quad [\text{C-4}]$$

An alternative equivalent solution was also found by NASA, which is valid for $\varphi \leq i$ and $0 < i \leq \pi/2$:

$$\begin{aligned} P_i(i, \varphi, \Delta\varphi) &= \frac{1}{\pi} \arcsin\left(\frac{\sin(\varphi)}{\sin(i)}\right) \Big|_{\varphi-\Delta\varphi/2}^{\varphi+\Delta\varphi/2} \\ &= \frac{1}{\pi} \left[\arcsin\left(\frac{\sin(\varphi + \Delta\varphi/2)}{\sin(i)}\right) - \arcsin\left(\frac{\sin(\varphi - \Delta\varphi/2)}{\sin(i)}\right) \right] \end{aligned} \quad [\text{C-5}]$$

In any case, an mission-related object coming from an orbit with inclination i re-enter in the latitude range $[-i, i]$, with the highest ground impact probability occurring at the extreme of this interval.

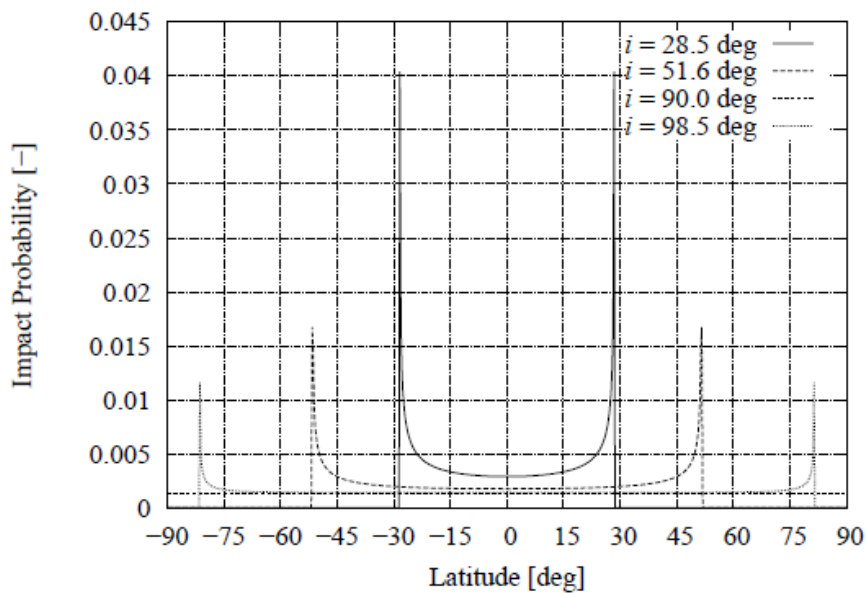


Figure C-4: Ground impact probability as function of latitude



C.2.7.3 Ground impact probability for eccentric re-entry orbits

Uncontrolled re-entries from highly eccentric orbit (without any prior circularisation due to atmospheric drag effects) can occur when the re-entry is not driven by the effects of atmospheric drag, but by third body (Moon, Sun) orbit perturbations (lunisolar perturbations) acting mainly on the apogee part of the orbit. This can only occur for eccentric orbits with significant geocentric apogee altitudes (e.g. some Molniya orbits, HEOs as for Integral, Cluster-II).

The main relevant effect of these lunisolar perturbations for re-entries is periodic lowering of the perigee until complete atmospheric capture of the space system. In contrast to atmospheric drag, the lunisolar perturbations are well predictable, and also the interaction with the atmosphere (until complete atmospheric capture) is typically short compared to the revolution time. The epoch of atmospheric capture is thus predictable with an accuracy of a few revolutions for years ahead. The lunisolar perturbation determines that the re-entry is likely to occur near the location of the perigee. The geographic latitude of the re-entry is thus determined by the argument of perigee. The geodetic longitude of the re-entry is determined by the revolution number at which atmospheric capture occurs.

C.2.8 Re-entry trajectory

The re-entry trajectory should be determined including:

- a. Time history of the state parameters, including perigee altitude, apogee altitude, inclination, right ascension of ascending node, argument of perigee, true anomaly, altitude, longitude, latitude, velocity, flight path angle, azimuth angle from the end of the operational phase to ground impact.
- b. Ground track of the re-entry trajectory.

C.2.9 Explosion probability assessment

An explosion probability assessment should be able to determine:

- a. Residual quantity of fuel(s) and pressurized gas or liquids after the disposal phase and before the entry and fragmentation in atmosphere.
- b. Probability of explosion with respect to altitude and fragmentation process during re-entry.

C.2.10 Characteristics of surviving fragments

The physical characteristics of all expected fragment surviving re-entry should be determined, including:

- a. Maximum size of the fragment along three main orthogonal directions
- b. Mass of the fragment (total and by material, if multi-material)
- c. Material of the fragment
- d. Fragment casualty area
- e. Fragment impact velocity
- f. Fragment impact flight path angle

- g. Fragment impact kinetic energy
- h. Fragment impact location (latitude, longitude) for deterministic simulations (for global distribution of all fragments, see Re-entry Casualty Area Determination, section C.2.13)
- i. Fragment Floating or non-floating capability over water or oceans.

C.2.11 Fragment casualty area

The casualty area of a surviving fragment k ($A_{C,k}$) leading to a casualty if a person is struck, conventionally with impact kinetic energy greater than 15 J (e.g. NASA-STD-8719.14A - Process for Limiting Orbital Debris - NASA, 25/05/2012), is defined as (see Figure C-5):

$$A_{C,k} = [\sqrt{A_{i,k}} + \sqrt{A_h}]^2 \quad [C-6]$$

where:

A_i average projected area of the fragment surviving the re-entry

A_h cross-section of a human, which is conventionally defined equal to 0,36 m² according to the NASA Safety Standard NSS 1740.14

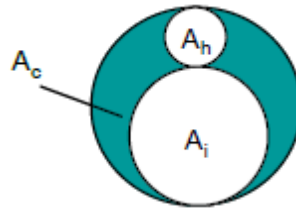


Figure C-5: Casualty area definition

C.2.12 Total casualty area

The total casualty area (A_C) for the re-entry is the sum of the casualty area of all surviving fragments ($A_{C,k}$):

$$A_C = \sum_{i=1}^N A_{C,k} \quad [C-7]$$

C.2.13 Casualty expectancy

Risk is an undesirable situation or circumstance that has a likelihood of occurring and a potential negative consequence. Re-entry casualty risk is determined through the probability to cause serious injury or death. As a probability, the risk is by definition ≤ 1 . Since the variable number of casualties (N) is discrete and the computation of the probability implies a sum (integration) over the space over which the probability is distributed, this corresponds to the expected number of casualties ($E = N$), i.e. to an expectancy which can even allow values > 1 . The computation of the risk profile requires



knowledge on the underlying (discrete) probability distribution function, which is difficult to determine. If the probability (P) of at least one casualty needs to be lower than a given value it is not required to project the full risk profile. The Markov's Inequality gives an upper limit for the probability distribution:

$$P(N \geq a) \leq \frac{E}{a} \quad [\text{C-8}]$$

where a is an integer number supposed to be equal to 1, which implies:

$$P(N \geq 1) \leq E = N \quad [\text{C-9}]$$

In the practice, the re-entry casualty probability can be approximated by the re-entry casualty expectancy since N is expected to be low. Nevertheless, in general, such approximation is not strictly exact since the value of a probability cannot be larger than 1, while an expectancy can be larger than 1.

The methodology to perform risk assessment is slightly different in the controlled and uncontrolled cases due to the uncertainty on the impact point associated to the uncontrolled re-entries. In a controlled case, it is possible to directly relate the impact point, the characteristics of the surviving fragments and the total population density at the impact point, while in an uncontrolled re-entry all impact locations in the latitude range $[-i, i]$, where i is the orbit inclination, are possible and each of the impact points has different impact probabilities.

A ground impact probability distribution function can be analytically obtained as a function of the latitude (see C.2.7). This function is used in combination with the population density distribution data to create a weighted average population density which is used together with the total casualty area of all surviving fragments to obtain the casualty probabilities.

It is important to note that the casualty risk requirement holds for the whole mission duration even if a controlled re-entry is already planned. The probability of a successful controlled re-entry is, therefore, playing an important role in the analysis. The probability of failing to perform a controlled re-entry will be weighted with its consequence (i.e. the casualty risk for an uncontrolled re-entry). In turn, the probability of performing a successful controlled re-entry will be weighted with the resulting casualty risk. A functional Fault Tree should be identified such to quantify the combined casualty risk of the nominal controlled re-entry and off-nominal re-entry cases, including degraded controlled re-entry and uncontrolled re-entry.

The sum of all weighted expectancies for all scenarios per mission to be compared to the requirement of 1:10000 and should be fulfilled for any disposal strategy (controlled or uncontrolled re-entry).

The re-entry casualty expectancy is computed as follows, depending on controlled or uncontrolled re-entry case:

- a. The re-entry casualty expectancy for uncontrolled re-entry ($E_{C,unc}$) is the product of the total casualty area A_c due to all surviving fragments and the latitude dependent population density (inhabitants or surface) weighted with the ground impact probability $P_i(i, \varphi, \Delta\varphi)$ or $P_i(i, \varphi, \Delta\varphi, \omega)$ depending on the orbit eccentricity (section C.2.8), which is a function of the orbit inclination i , the latitude step size $\Delta\varphi$, and the argument of perigee at the epoch of atmospheric capture ω :

$$E_{C,unc} = A_c \rho_p(i, \Delta\varphi, \omega) \quad [\text{C-10}]$$

- b. The re-entry casualty expectancy for controlled re-entry ($P_{C,con}$) is the sum of the products of each fragment casualty area and the local population density (inhabitants or surface):

$$E_{C,con} = 1 - \prod_{k=1}^N \left(1 - \sum_n \sum_m (P_{i,k})_{n,m} (\rho_p)_{n,m} (A_{c,k})_{n,m} \right) \quad [C-11]$$

where the index k is for fragment, the indices n and m are for area bins, $(P_{i,k})_{n,m}$ is the local ground impact probability of the k -th fragment in the (m,n) bin, $(\rho_p)_{n,m}$ is the local population density in the (m,n) bin, and $(A_{c,k})_{n,m}$ is the casualty area of the k -th fragment in the (m,n) bin.

- c. The re-entry casualty expectancy for a failed controlled re-entry ($E_{C,unc,fail}$) is the product the re-entry casualty expectancy for uncontrolled re-entry (bullet a)) and the probability to have failures compromising the controlled re-entry (P_f):

$$E_{C,unc,fail} = E_{C,unc} P_f = A_c \rho_p(i, \varphi, \Delta\varphi) P_f \quad [C-12]$$

- d. The re-entry casualty expectancy for a space system, which is not nominally required to be disposed by re-entry, but, which has, anyway, a non-zero probability to approach re-entry conditions ($E_{C,prob,re-entry}$), e.g. disposal on a HEO or on an orbit around Sun-Earth Lagrange Points, is the product of the casualty expectancy for an uncontrolled re-entry and the re-entry probability ($P_{re-entry}$):

$$E_{C,prob,re-entry} = E_{C,unc} P_{re-entry} = A_c \rho_p(i, \varphi, \Delta\varphi) P_{re-entry} \quad [C-13]$$

- e. The combined re-entry expectancy ($E_{C,comb}$) which takes into account all possible re-entry scenarios should be determined as follow:

$$E_{C,comb} = E_{C,nom} R_{nom} + \sum_{k=1}^N E_{C,off-nom,k} P_{off-nom,k} \quad [C-14]$$

where $E_{C,nom}$ is the casualty expectancy for the nominal controlled re-entry, R_{nom} is the reliability to perform the nominal controlled re-entry, $P_{off-nom,k}$ is the probability to have the k -th non-nominal case (e.g. degraded controlled re-entry, or uncontrolled re-entry due to failures or unplanned re-entry, e.g. for disposal on a HEO or on an orbit around a Sun-Earth Lagrange Points), and $E_{C,off-nom,k}$ is the casualty expectancy associated to the k -th non-nominal case.

Extensive human casualty studies have examined the probability of injury or death from falling debris for a variety of impacting kinetic energies to humans. A kinetic energy threshold criterion of 15 J is widely accepted as the minimum level for potential injury to an unprotected person.

C.2.14 Declared Re-entry Area (DRA) and Safety Re-entry Area (SRA)

The Declared Re-entry Area (DRA) and the Safety Re-entry Area (SRA) should be computed following several simulation runs (Monte Carlo), which should be based on the dispersions of the relevant variables to cover all uncertainties of the model (see section C.2.2), where the amount of runs should yield stable confidence intervals (see Figure C-6)):

- The Declared Re-entry Area (DRA) should delimit the area where the debris should be enclosed with a probability of 99% given the delivery accuracy.
- The Safety Re-entry Area (SRA) should delimit the area where the debris should be enclosed with a probability of 99,999% given the delivery accuracy.

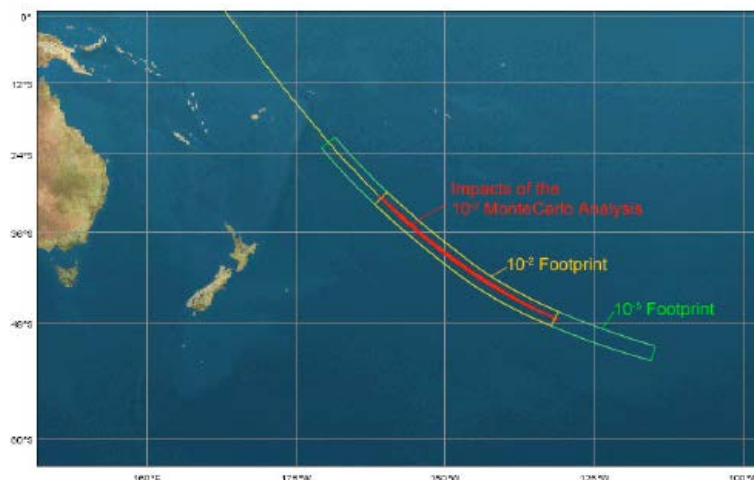


Figure C-6: Example of DRA (10^{-2} Footprint) and SRA (10^{-5} Footprint)

C.2.15 Rough order of magnitude approach for casualty risk

A very rough approach to assess the re-entry casualty area, which can be useful at a very early stage of a project when the space system design is still mostly undefined, is discussed in this section. It is derived from previous re-entry assessment using high-fidelity models for re-entries from circular orbits. The results have been statistically fitted with simple polynomials as a function of inclination i , dry mass and re-entry epoch t_{re} (Figure C-7).

This can be used in conjunction with a population density model based on the Gridded Population of the World (GPW) v3 for the year 2000, and by applying latitude dependent growth factors for the predicted population growth. Considering this evolution, the related casualty cross-section threshold can be computed for a given risk level (e.g. 10^{-4}) and re-entry scenario (Figure C-8, Figure C-9).

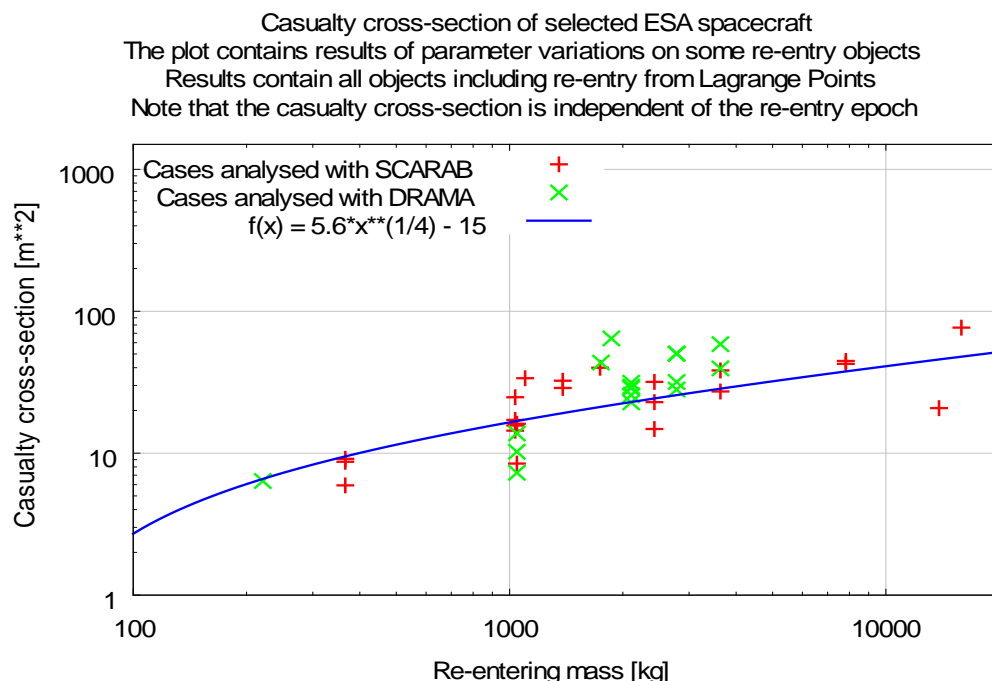


Figure C-7: Fit of historical re-entry assessment for the casualty cross-section as a function object mass

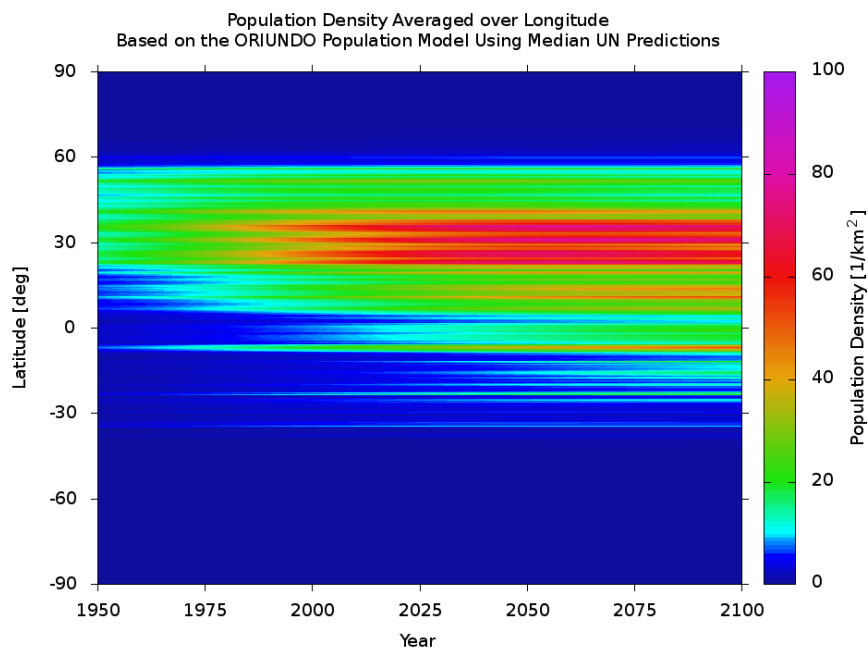


Figure C-8: Latitude-dependent predicted evolution of the population density, using median UN predictions for the population growth (Rev. 2010)

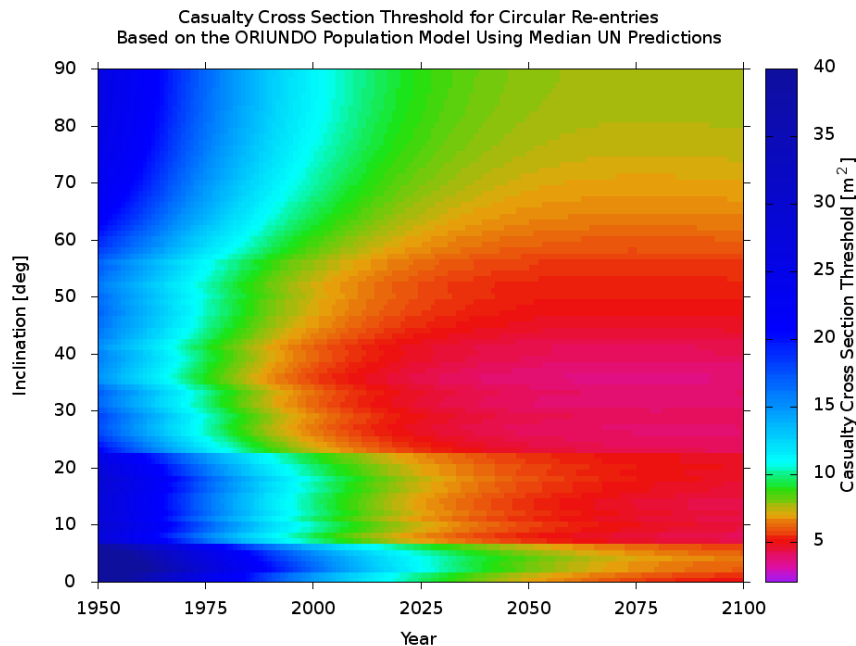


Figure C-9: Casualty cross-section threshold for a 10^{-4} casualty risk and uncontrolled re-entries from circular orbits, using median UN predictions for the population growth (Rev. 2010)

C.2.16 Tool(s) use

Tools for re-entry casualty risk analysis need to consider trajectory dynamics, aerothermodynamics and its interaction with the object geometry, as well as the distribution of the human population on ground. DRAMA is made available by ESA and is accepted to perform a re-entry casualty risk analysis.

For a more refined analysis, independent cross-checks, or in-depth investigation of particular re-entry phenomena (e.g. explosive break-ups, complex geometric structures or other particularities), the use of other tools is also possible. It is required that, a priori, the selected tool needs to be discussed with ESA.

C.2.17 Project phasing for re-entry casualty risk analysis

The need to perform a controlled re-entry can have substantial impact on the design of a space system. It is, thus, necessary, that this need is identified early in the development to trigger the right design decisions and save cost. First indications can already be obtained during the mission definition phase. This then enables a decision as to whether or not a controlled re-entry is part of the mission being made at the end of the mission SRR. Once this decision is made and an uncontrolled re-entry is planned “design for demise” measures can be used to mitigate the re-entry casualty expectancy, which are implemented and verified at PDR.

Accordingly, multi-step approach is required (see Table C-1).

Table C-1: Re-entry casualty risk analysis process

Review / Phase	Action		
Phase 0	1. First assessment of $P_{re-entry}$ for the mission orbit (C.2.1). 2. First assessment of E_c for an uncontrolled re-entry (two options): a. Rough order of magnitude assessment of $E_c = f(inclination, Epoch, dry\ mass)$ (C.2.15) (for circularised LEO re-entry orbits). b. First assessment of E_c (C.2.13) using likely re-entry conditions (C.2.3) and a first space system model using object-oriented tools (for all orbits) (C.2.4).		
	Result: First indication of $P_{re-entry} \cdot E_c$ (controlled re-entry to be considered if $P_{re-entry} \cdot E_c > 10^{-4}$)		
PRR Phase A	1. Refined assessment of $P_{re-entry}$ for the mission orbit. 2. Refined assessment of E_c for an uncontrolled re-entry using a more elaborate space system model and object-oriented tools (C.2.4)		
	Result: Preliminary decision on the re-entry approach (controlled or uncontrolled)		
	Uncontrolled re-entry		Controlled Re-entry
	Circular re-entry (C.2.7.2)	Eccentric Re-entry (C.2.7.3)	First sizing of debris fall-out footprint (C.2.14)
SRR Phase B	1. Final assessment of $P_{re-entry}$ for the mission orbit (C.2.2). 2. Final assessment of a space system model in object-oriented tools, including uncertainty quantification (C.2.5) 3. Establishment of a model in a spacecraft-oriented tools (for confirmation) (C.2.6), determination of explosion likelihood and consideration of these effects (C.2.10), and conclusion on passivation measures.		
	Uncontrolled re-entry		Controlled Re-entry
	Circular re-entry (spacecraft-oriented tool)	Eccentric Re-entry (spacecraft-oriented tool)	1. Preliminary assessment of SRA and DRA. 2. Preliminary assessment of reliability figures and failure modes. 3. First assessment of $E_{C,comb}$ (C.2.15).
	Result: Final decision on the re-entry approach (controlled or uncontrolled)		

Review / Phase	Action				
PDR Phase B	1. Refinement of the model for the spacecraft-oriented tool				
	Uncontrolled re-entry		Controlled Re-entry (C.2.15)		
	Circular re-entry If $P_{re-entry} \cdot E_C > 10^{-4}$, mitigation options: 1. Implementation and verification of “design for demise” measures; 2. Passivation to prevent explosive break-ups.	Eccentric Re-entry If $P_{re-entry} \cdot E_C > 10^{-4}$, mitigation options: 1. Implementation and verification of “design for demise” measures; 2. Passivation to prevent explosive break-ups; 3. Modification of the disposal strategy with a different re-entry latitude.	$E_{C,unc,fail}$ Mitigation options: 1. Improvement of system reliability; 2. Implementation and verification of “design for demise” measures; 3. Passivation to prevent explosive break-ups.	$E_{C,nom} R_{nom}$ Mitigation options: 1. Lower re-entry perigee; 2. Alternate target area; 3. Passivation to prevent explosive break-ups.	$E_{C,off-nom} P_{off-nom}$ Mitigation options: 1. Improvement of system reliability; 2. Passivation to prevent explosive break-ups.
CDR Phase C	Uncontrolled re-entry		Controlled re-entry		
	As before		As before		
FAR Phase D	Uncontrolled re-entry		Controlled re-entry		
	As before		As before		
Mission Change	Uncontrolled re-entry		Controlled re-entry (C.2.15)		
	Circular re-entry 1. Verify that changes to re-entry epoch (on-ground population growth) do not lead to a violation the requirement.	Eccentric re-entry 1. Verify that changes to re-entry epoch do not lead to re-entry latitude with higher population densities.	$E_{C,unc,fail}$ 1. Verify that P_f has not reached critical levels.	$E_{C,nom} R_{nom}$ 1. Verify that R_{nom} has not reached critical levels.	$E_{C,off-nom} P_{off-nom}$ 1. Verify the $P_{off-nom}$ has not reached critical levels.

Review / Phase	Action			
EOL	Uncontrolled re-entry	Controlled re-entry		
	<ol style="list-style-type: none"> 1. Monitor re-entry and predict re-entry epoch and location; 2. Notify national alert centers and supply them with the prediction results; 3. Confirm the re-entry. 	Uncontrolled re-entry <ol style="list-style-type: none"> 1. Monitor re-entry and predict re-entry epoch and location; 2. Notify national alert centers and supply them with the prediction results; 3. Confirm the re-entry. 	Controlled re-entry <ol style="list-style-type: none"> 1. Inform sea traffic authorities for NAVAREA messages at least 6 days before; 2. Inform air traffic authorities for NOTAM messages at least 2 days before. 	Degraded Controlled re-entry <ol style="list-style-type: none"> 1. Same actions as for controlled re-entry.

Annex D

Propellant gauging methods

Possible methods to gauge the propellant mass for any phase during the mission are given in ISO 23339:2010, originally coming from ESA SP-398, Aug 1997: "Huffenbach et. Al, Comparative Assessment of Gauging Systems and Description of a Liquid Level Gauging Concept for a Spin Stabilised Spacecraft, Proceedings of the Second European Propulsion Conference, 27-29 May 1997, and reproduced in Table D-1.

Table D-1: Examples of estimation methods

Method	Used for	Measurement Principle	Advantages	Disadvantages	Heritage / Development Status
pVT	3 axis + spinner	Measurement of the tank temperature and pressure and calculation of the tank ullage volume and thereby the remaining propellant mass by applying the gas law	No additional equipment and low cost	Decreasing accuracy towards EOL, low accuracy with conventional pressure transducer	Used on many spacecraft
Thermal Knocking	3 axis + spinner	Heating of the propellant tank and measurement of its thermal response which is related to the propellant load	No additional equipment	Low accuracy, high calibration efforts and long operational gauging times	Used on OLUMPUS, EUROSTAR
Gas Injection	3 axis	Transfer of a known amount of pressurant gas into the propellant tank and measurement of the pressure and temperature increase to determine the ullage volume and thereby the remaining propellant mass	Good accuracy at EOL	Complex system, modification of propulsion system required, needs high accuracy pressure transducer, high calibration effort and high costs	Ground qualification of an Engineering Model Gauging Device; Operational use of Hughes Propellant Gauging System; In-flight demonstration of Foreign Mass Injection Method and Periodic Volume Stimulus Method
Liquid Levelling	Spinner	Measurement of liquid level in tanks of satellites where propellants are settled due to spin acceleration	Simple system with very high accuracy	Limited to spinning satellites	Used on MSG (capacitive measurement), Hughes (Δp measurement)



Method	Used for	Measurement Principle	Advantages	Disadvantages	Heritage / Development Status
Bookkeeping	3 axis + spinner	Calculation of propellant consumption during each manoeuvre by on-ground recording of all manoeuvre data (e.g. pulse duration, pulse mode, thruster temperature), making additionally use data of individual thruster calibration test firings on ground	Simple system with non no additional equipment needed	Inflight calibration required, operational effort on ground	Used on many spacecraft
Flometer	3 axis	Integration of mass flow rate measurements during operations	High accuracy sensor	Sensor still to be developed, low accuracy for pulse firings	CNES funded spatialization of a thermal flowmeter failed in 1990 due to sensitivity to gravity vector, thermal environment and small gas bubbles in flow; ESA/ESTEC development of a ultrasonic flowmeter
Liquid Levelling	3 Axis	Measurement of liquid level in tank during thruster operations	Accuracy improvement feasible	Low achieved accuracy, works only during thruster firings of minimum thrust level and duration	Used on Apollo, Shuttle OMS.



Annex E

Passivation methods

Table E-1 summarizes passivation measures that can be used for the most common components storing energy. This table should, anyway, not be considered a priori as exhaustive.

Table E-1: Passivation measures

Subsystem	Component	Passivation Measures	Critical Issues and Remarks
GNC	Attitude Control Sensors and Actuators	1. Disconnect from power supply sources	<ul style="list-style-type: none"> To add a dedicated GNC mode.
GNC	Cold Gas Thrusters	1. Disconnect from chemical supply sources	<ul style="list-style-type: none"> To add a dedicated GNC mode.
GNC	Control Moment Gyros	1. Disconnect from power supply sources 2. De-spin/stop rotating parts	<ul style="list-style-type: none"> Mobile parts can lead to mechanical ruptures due to fatigue. To add a dedicated GNC mode.
GNC	Reaction or Momentum Wheels	1. Disconnect from power supply sources 2. De-spin or stop rotating parts	<ul style="list-style-type: none"> Mobile parts can lead to mechanical ruptures due to fatigue. To add a dedicated GNC mode.
Mechanism	Any rotating or movable part	1. Fix and block relative movements	<ul style="list-style-type: none"> Mobile parts can lead to mechanical ruptures due to fatigue.
Mechanism	Electro-explosive devices	1. De-activation if not useful any more 2. Disconnection from power supply sources	
Mechanism	Pyrotechnic devices	1. De-activation if not useful any more 2. Disconnection from power supply sources	
Power	Batteries	1. Self-protection 2. Discharge 3. Disconnect from solar array, power bus or any charging source 4. Depressurization of the cells (if needed)	<ul style="list-style-type: none"> The discharge of batteries (safe complete discharge to 0 V or to a safe state of charge ensuring no break-up risk) and their subsequent disconnection from charging circuits is a preferred option. Passivation of the solar array can be sufficient since leading to completely battery discharge (initially via the power bus loads and, then, also via the leakage current of control electronics connected to the battery). The batteries can also be left with a permanent electrical drain to prevent recharging. The passivation device should be robust enough to cope with the long lifetime required and the harsh environment at EOL (e.g. loss of temperature control, radiations) to avoid losing passivation after some time. If not possible to eliminate all energy or disconnect the batteries, the batteries can be self-protected provided that the absence of break-up risk can be guaranteed (to be supported by risk assessment, analysis or tests under the environmental conditions after passivation). Small batteries (e.g. for cubesats) can be protected in containers to limit debris propagation in case of explosion (to be demonstrated by analysis based on the maximum stored energy and structural

Subsystem	Component	Passivation Measures	Critical Issues and Remarks
			resistance of the container).
Power	Fuel Cells	<ol style="list-style-type: none"> 1. Self-protection 2. Discharge 3. Disconnect from solar array, power bus or any charging source 4. Depressurization of the cells (if needed) 	<ul style="list-style-type: none"> • See batteries.
Power	Power Conditioning and Distribution Unit (PCDU)	<ol style="list-style-type: none"> 1. Disconnect from power sources 2. Switch-off all possible circuits 	
Power	Solar Array	<ol style="list-style-type: none"> 1. Disconnect from power bus or batteries 2. Short-circuit 	<ul style="list-style-type: none"> • See batteries.
Propulsion	Pipelines	<ol style="list-style-type: none"> 1. Venting (as far as possible) 2. Scavenging of residual propellants actively (though pressurization) or passively (by slow evaporation) 3. Demonstration of low probability of rupture 	<ul style="list-style-type: none"> • Ventings should not cause uncontrolled accelerations, attitude, or orbit changes. • Residual propellant should be scavenged without generating solid particles greater than 10 µm. • Mixing of fuel and oxydizer should be avoided. • Blockage due to freezing propellants or venting fluids should be avoided. • The risk of explosion of pipelines, which are not connected to pressure vessels with high stored energy, can be demonstrated to be minor due to high design factors and low volume involved.
Propulsion	Pressurized Gas Tank	<ol style="list-style-type: none"> 1. Venting (as far as possible) 2. Depressurization at least down to a level such that no bursts can occur due over-pressure or over-temperature or to HVI 	<ul style="list-style-type: none"> • Depressurization should not generate new hazards (e.g. structure embrittlement). • Venting should not cause uncontrolled accelerations, attitude, or orbit changes. • Explosions due to over-pressure or over-temperature should be prevented by design during any phase of the mission, for example, through relief valve mechanisms. • In case residual gas cannot be drained, the following conditions should be ensured: no bursts should occur as a result of a penetrating impact; design of tanks and efficiency of thermal protection should inhibit pressure build-up that can cause tank burst (e.g. relief valve mechanisms). • The behaviour of pressurized vessels under HVI should be demonstrated.

Subsystem	Component	Passivation Measures	Critical Issues and Remarks
Propulsion	Propellant Tank	<ol style="list-style-type: none"> 1. Venting (as far as possible) 2. Depletion burn(s) 3. Depressurization at least down to a level such that no bursts can occur due over-pressure or over-temperature or to HVI 	<ul style="list-style-type: none"> • Depressurization should not generate new hazards (e.g. structure embrittlement). • Venting and depletion burns should not cause uncontrolled accelerations, attitude, or orbit changes. • Venting and depletion burns should not affect other space systems or increase the likelihood of fragmentation. • Spin-up of the vehicle or inadvertent mixing of vented hypergolic propellants should be prevented. • Venting and depletion burns operations should prevent collisions with known objects. • Leak-before-burst tank designs, although beneficial, are not sufficient to prevent explosions in all scenarios (depressurization is still required). • Pressure vessels with pressure-relief mechanisms do not need to be depressurized if it can be shown that no plausible scenario exists in which the pressure-relief mechanism is insufficient. • In case residual propellant cannot be drained, the following conditions should not occur: explosive reactions of the propellant as a result of a penetrating impact; exothermal dissociation of the propellant due to tank heating; leak that can cause the mixture of hypergolic propellants; pressure build-up that can cause tank explosion (e.g. to be prevented through thermal protection).
TC	Telemetry	<ol style="list-style-type: none"> 1. Switch-off telemetry transmitter with monitoring RF signal 	
Thermal Control	Heat Pipes	<ol style="list-style-type: none"> 1. Demonstration of low probability of rupture 	

Annex F

EOL disposal reliability

F.1 Objectives

This Annex describes proposed guidelines for the assessment of the probability to successfully perform the EOL disposal, further on referred to as “EOL disposal reliability”.

F.2 Methodology

F.2.1 General

A successful disposal can be achieved by a space system through performing the following activities:

- a. Assessment, during the development phase, of the probability of successfully performing the EOL disposal operations to achieve compliance with design-to requirements.
- b. Assessment of the actual disposal reliability in-orbit along the mission to monitor and maintain compliance with the defined disposal reliability requirements.

The principal contributors to the EOL disposal probability are:

- a. The reliability of the on-board systems to perform successfully the required functionality at the required performance.
- b. The probability to not break-up by an internal explosion or burst.
- c. The probability to not break-up by collision with other space systems, debris and meteoroids.
- d. The probability to load and maintain sufficient consumables, mostly propellant, on board to perform all required disposal operations.
- e. The probability of the operator to monitor the space system health status and plan and execute correctly all disposal maneuvers.

Margins and uncertainties should be taken into account for:

- a. Execution of all nominal maneuvers.
- b. Execution of predictable collision avoidance maneuvers.
- c. Execution of maneuvers to approach the target disposal orbit (e.g. assuming maximum disposal mass at EOL and worst case (-3σ) propulsion performance).
- d. Execution of the maneuvers for controlled re-entry, if planned.
- e. Execution of potential specified mission extensions.



- f. Worst case propellant residuals (static and dynamic).

F.2.2 EOL disposal reliability assessment during the development phase

The prediction of the probability of successfully performing the EOL disposal operations during the development phase should be performed by estimating the following quantities:

- a. Mission reliability ($R_{Mission}$), i.e. the probability to perform successfully the mission.
- b. Mission and disposal reliability ($R_{Mission+Disposal}$), i.e. the probability to accomplish successfully both the mission and the disposal.
- c. Disposal reliability ($R_{Disposal}$), i.e. the conditional probability to have the successful disposal assumed the successful mission.

$$R_{Disposal} = \frac{R_{Mission+Disposal}}{R_{Mission}} \quad [F-1]$$

$R_{Mission}$ and $R_{Mission+Disposal}$ need to be estimated accounting for the probability of failure or accidental break-up preventing disposal due to:

- a. Internal explosions as per FMECA or specific assessment.
- b. Collision with other orbital objects likely to cause break-up (as per Annex B).

If it is demonstrated that the space system design follows best practice design rules, e.g. compliant with the applicable ECSS standards with respect to tank pressurization, thermal design propellant vapour segregation, battery charge/discharge control, MMOD protection the contribution of the break-up probability can be negligible.

In the case that the mission reliability is not available, the calculation can be performed with the conservative assumption that the mission reliability is 1.

F.2.3 EOL disposal reliability in-orbit assessment

The assessment of the actual disposal reliability of the space system in-orbit along the mission to monitor and maintain compliance with the defined disposal reliability requirement should be performed because:

- a. Reliability predictions cannot cover systematic faults that were not detected prior to the launch and can evolve to system failures once activated under the specific operational or environmental conditions that the spacecraft faces in orbit. Such faults can be design, manufacturing, assembly and integration errors that pass undetected through all inspections and tests. Since they cannot be reflected in the reliability predictions they represent an unknown and undetermined add-on to the EOL unreliability.
- b. A space system can experience a random failure on equipment required for EOL operations during its mission.
- c. The performance of life limited equipment needs to be monitored during the operational phase to determine the need to possibly terminate the nominal mission at an early date or, conversely, assess the possibility to extend a mission. Generally mechanical and life limited items degrade gradually and show observable symptoms. For example, a degrading reaction wheel can show an increased friction torque or torque instabilities, which can be observed by telemetry.

- d. The health of a space system can be monitored to identify unanticipated degradation faster than expected. This can be either an early loss of redundancy or degradation in performance of equipment needed for EOL operations. In such a case the EOL reliability is adversely affected and need to be re-evaluated to define the further mission planning to control the risk of generating space debris in LEO or GEO Protected Regions (e.g. see Figure F-1).
- e. For the re-assessment of the EOL reliability after an in orbit anomaly, it is important to gain sufficient confidence that an observed anomaly is not subject to a common cause, potentially affecting multiple equipment parts of the space system and thus lowering the effectiveness of redundancies. Typical common causes are manufacturing or material deficiencies affecting a manufacturing lot or higher degradation of equipment performance by environmental conditions.
- f. In addition to controlling the risk in case of in orbit anomalies, confirming the good health of space system disposal functions in orbit allows to extend a mission beyond its nominal life, should it be requested.

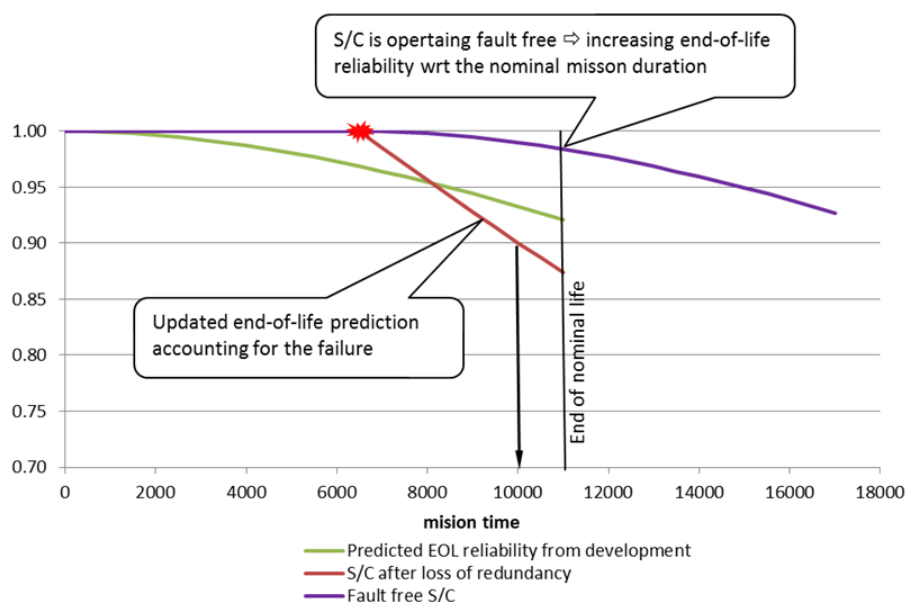


Figure F-1: Principle of EOL reliability updating along the space system life

In order to allow an efficient re-assessment of the EOL reliability it is recommended to build the EOL reliability prediction models during the development phase such that they can be used as risk monitors along the life of the spacecraft. These models can rest either with the engineering support to mission operations or with the mission operations team directly.

A project can decide to perform an EOL Probabilistic Risk Assessment (PRA) instead of an EOL reliability assessment. Such PRA models account typically also for causes that do not strictly belong to the reliability domain and are, thus, more sophisticated.



Annex G

Space Debris Mitigation Plan (SDMP)

document description

G.1 Scope

This document, called Space Debris Mitigation Plan (SDMP), should provide the plan for the implementation and verification of the Space Debris Mitigation requirements. The document should be issued at Preliminary Requirements Review (PRR) for approval at the System Requirements Review (SRR).

The SDMP should contain the information required in section G.2.

G.2 Table of contents

- a. Introduction**
- b. Scope**
- c. References**
 - 1. Applicable Documents
 - 2. Reference Documents
- d. Terms, definitions and abbreviated terms**
 - 1. Terms and definitions
 - 2. Abbreviated terms
- e. Programme management overview**
 - 1. Identification of the ESA Directorate responsible for the project or mission
 - 2. Identification of the ESA Study Manager or Project Manager responsible for the preparation and maintenance of the SDMP
 - 3. Identification of any non-ESA participation in any phase of the mission
 - 4. Chronology of ESA project reviews and issues of the SDMP and SDMR

**f. Mission overview**

1. Description of the mission objectives
2. Schedule of the mission milestones from launch through EOM, including the timeline of the planned orbital maneuvers
3. Description of the mission profile, including type of orbit (e.g. LEO, MEO, HEO, GEO, L1, L2) and orbital parameters
4. Description of the mission requirements for orbit injection, maintenance, and disposal (e.g. Sun-synchronous, ground track, latitude, formation flying)

g. ESA space system description

1. Summary description of the space system (i.e. launch vehicle, spacecraft, inhabited or robotic vehicles), e.g. including spacecraft platform, payload instrumentation, and all appendices (e.g. solar arrays, antennas, instrument or attitude control booms)
2. Mass budget at launch and EOM, including all propellants and fluids with details known at the present phase
3. Description of the propulsion system
4. Description of the power system
5. Description of the Attitude and Orbit Control System (AOCS)
6. Description of the Guidance, Navigation and Control (GNC) system

h. Procured launch service description

1. Identification of the launch service (if the launch vehicle has not been identified or procured yet, available information about launch constraints and launch vehicle opportunities)
2. Description of the launch vehicle mission profile, including all separation stages with all parking, transfer, and mission orbital parameters (apogee, perigee, inclination)
3. Identification of the launch vehicle capabilities concerning de-orbiting of upper stages
4. Launch vehicle stages mass budget, including propellant before and after operation, if available
5. Description of the propulsion systems for each stage (e.g. solid, liquid), if available

i. Space Debris Mitigation implementation and verification plan

1. Space system mission-related objects (MROs)
 - (a) Identification and description of any MRO (>1 mm) expected to be released at any time after launch, including type, dimensions, mass, and material
 - (b) Rationale for release of each MRO
2. Space system fragmentation and explosion risk
 - (a) Identification of all potential causes of break-up during deployment and mission operations
3. Space system small particles release



- (a) Design approach to avoid release in orbit of SRM particles larger than 1 mm
 - (b) Design approach to avoid release in orbit of pyrotechnic particles larger than 1 mm
- 4. Space system on-orbit collision risk assessment
 - (a) Identification of the methodology to assess on-orbit collision risk
 - (b) Identification of a strategy to prevent orbital collisions through avoidance maneuvers
- 5. Space system disposal
 - (a) Description of the disposal options (e.g. re/de-orbit to a graveyard orbit, controlled or uncontrolled re-entry)
 - (b) Identification of all subsystems or components required to accomplish any disposal manoeuvre
 - (c) Preliminary analysis of the presence in the LEO or GEO Protected Regions after the disposal maneuvers Annex A)
 - (d) Tether systems disposal plan (if applicable)
- 6. Space system passivation
 - (a) List of the components storing energy (e.g. Annex E)
 - (b) List of the components, which needs to be passivated at the end of the operational phase
 - (c) Description of the design measures and operations implemented for the passivation of the required components
 - (d) List of the component, which cannot be passivated at the end of the operational phase and rationale
 - (e) Available options (e.g. platform modification or changes) and related additional impact on project cost, effort, and schedule in order to allow passivation of components which have not been planned to be passivated
- 7. Space system re-entry
 - (a) Preliminary re-entry casualty risk analysis of the spacecraft and any other MRO likely to re-enter (Annex C), including the list of components with mass, dimensions, shape, and material information used by the analysis
 - (b) Identification of the re-entry scenario, including nominal and degraded cases
 - (c) Identification of the system functions that contribute to the controlled re-entry, if planned
 - (d) Identification of the methodology planned to be used for the final re-entry casualty risk analysis (Annex C)
- 8. Space system hazardous materials
 - (a) Identification of hazardous materials on-board (e.g. toxic, radioactive)



j. ESA space system compliance and verification

1. Preliminary Compliance and Verification Matrix, including identification of the applicable SDM requirements, statement of planned compliance with the SDM requirements, planned approach for compliance, and reference documents (e.g. Table G-1)

k. Launch service compliance

1. Assessment of the compliance with the SDM requirements, if the launch service has been already selected

As per [RD1]: “For the procurement of launch services for ESA space systems, all reasonable efforts shall be made to ensure the use of launchers which are compliant with the space debris mitigation technical requirements in the standard.”

Table G-1: Example of preliminary compliance and verification matrix

Req. Id. (see Note 1)	Requirement Text (see Note 2)	Compliance Status (C/PC/NC/NA) (see Note 3)	Verification Method(s) (see Note 4)	Planned Approach (see Note 5)	Reference (Doc., section, pag.) (see Note 6)	Close-out event (see Note 7)
6.1.1.1	See ISO Standard for text of requirement	C	ROD	No release of debris is identified by review-of-design.	XXX, pag. Y, section Z	
6.1.1.2	See ISO Standard for text of requirement	C	ROD, A	An analysis as per Annex A is performed to determine the presence in the LEO or GEO Protected Region for any released space debris.	XXX, pag. Y, section Z	
6.1.1.3	If space debris are released into Earth orbit during normal launch operations, then the number of space debris released, other than those covered by 6.1.2, shall not exceed: a. One, for the launch of a single spacecraft. b. Two, for the launch of multiple spacecraft.	C	ROD	Objects released during launch operations are identified and minimized by review-of-design since only one object is expected to be released.	XXX, pag. Y, section Z	
6.1.2.1	See ISO Standard for text of requirement	NA	ROD, T	The spacecraft does not use pyrotechnic devices.	XXX, pag. Y, section Z	
6.1.2.2	Solid rocket motors shall be designed and operated so as to avoid releasing solid combustion products larger than 1 mm into the GEO Protected Region.	NA	ROD, T	The spacecraft does not use solid rocket motors.	XXX, pag. Y, section Z	
6.1.2.3	See ISO Standard for text of requirement	NA	ROD, T	The spacecraft does not use solid rocket motors.	XXX, pag. Y, section Z	
6.2.1	See ISO Standard for text of requirement	C	ROD	The mission plan does not involve intentional fragmentations.	XXX, pag. Y, section Z	
6.2.2.1	See ISO Standard for text of requirement	C	A, T	The accidental break-up probability will be determined by analysis or tests.	XXX, pag. Y, section Z	
6.2.2.2	See ISO Standard for text of requirement	C	A, T	FMECA of components likely to cause explosions or break-ups will be performed.	XXX, pag. Y, section Z	
6.2.2.3	See ISO Standard for text of requirement	C	ROD, A, T	All stored energy will be depleted at the end of the operational phase.	XXX, pag. Y, section Z	

Req. Id. (see Note 1)	Requirement Text (see Note 2)	Compliance Status (C/PC/NC/NA) (see Note 3)	Verification Method(s) (see Note 4)	Planned Approach (see Note 5)	Reference (Doc., section, pag.) (see Note 6)	Close-out event (see Note 7)
				Components, which will not be passivated, will not involve risk of explosions and break-ups of the spacecraft such to generate debris.		
6.3.1.1	<i>See ISO Standard for text of requirement</i>	C	A	An analysis to demonstrate the probability of successful disposal will be performed (e.g. Annex F).	XXX, pag. Y, section Z	
6.3.1.2	<i>See ISO Standard for text of requirement</i>	C	A	An analysis to demonstrate the probability of successful disposal will be performed (e.g. Annex F).		
6.3.1.3	<i>See ISO Standard for text of requirement</i>	C	A	An analysis to demonstrate the probability of successful disposal will be performed (e.g. Annex F).	XXX, pag. Y, section Z	
6.3.2.1	<i>See ISO Standard for text of requirement</i>	NA	ROD	The spacecraft does not operate in the GEO Protected Region since it only operates in LEO.	XXX, pag. Y, section Z	
6.3.2.2	<i>See ISO Standard for text of requirement</i>	NA	A	The spacecraft does not operate in the GEO Protected Region since it only operates in LEO (in case of presence in the GEO Protected Region, an analysis as per Annex A will be performed).	XXX, pag. Y, section Z	
6.3.3.1	<i>See ISO Standard for text of requirement</i>	NC	ROD, A	The spacecraft is not able to leave the LEO Protected Region within 25 years according to a preliminary analysis as per Annex A.	XXX, pag. Y, section Z	
6.3.3.2	<i>See ISO Standard for text of requirement</i>	C	ROD, A	A design and operational approach will be implemented to clear the LEO Protected Region. An analysis as per Annex A will be performed to verify the consistency of the approach.	XXX, pag. Y, section Z	

Req. Id. (see Note 1)	Requirement Text (see Note 2)	Compliance Status (C/PC/NC/NA) (see Note 3)	Verification Method(s) (see Note 4)	Planned Approach (see Note 5)	Reference (Doc., section, pag.) (see Note 6)	Close-out event (see Note 7)
6.3.4.1	See ISO Standard for text of requirement	C	A	A preliminary analysis as per Annex C is performed.	XXX, pag. Y, section Z	
6.3.4.2	See ISO Standard for text of requirement	C	A	A preliminary analysis as per Annex C is performed. An uncontrolled re-entry strategy is chosen as the re-entry casualty risk is less than 10^{-4} .	XXX, pag. Y, section Z	
1. Identification of the requirement ECSS-U-AS-10C / ISO 24113:2011 as per ESA/ADMIN/IPOL(2014)2. 2. Text of the requirement. 3. Status of planned compliance: Compliant (C), Partial Compliant (PC), Not Compliant (NC), Not Applicable (NA). 4. Verification methods: Test (T), Analysis (A), Review-of-design (ROD), Inspection (I). 5. Description of the planned approach to demonstrate compliance with the requirement (if the requirement is not applicable, provide a justification). 6. Reference to any documentation that demonstrates compliance with the requirement (e.g. section of the SDMP, reports, analysis, RFD/RFWs). 7. Information on when is planned the verification close-out, e.g. PDR, CDR.						



Annex H

Space Debris Mitigation Report (SDMR) document description

H.1 Scope

This document, called Space Debris Mitigation Report (SDMR), should provide the status of the compliance with the Space Debris Mitigation requirements, including verification methods, reports, and close-out, starting from the Preliminary Requirements Review (PDR) through all the main project phases and reviews and submitted for the final approval at the Flight Acceptance Review (FAR).

The SDMP should contain the information required in section H.2.

H.2 Table of contents

- a. **Introduction**
- b. **Scope**
- c. **References**
 - 1. Applicable documents
 - 2. Reference documents
- d. **Terms, definitions and abbreviated terms**
 - 1. Terms and definitions
 - 2. Abbreviated terms
- e. **Programme management overview**
 - 1. Identification of the ESA Directorate responsible for the project or mission
 - 2. Identification of the ESA Project Manager or Mission Manager responsible for the preparation and maintenance of the SDMR
 - 3. Identification of any non-ESA participation in any phase of the mission
 - 4. Chronology of ESA project reviews and issues of the SDMP and SDMR
- f. **Mission overview**
 - 1. Description of the mission objectives



2. Schedule of the mission milestones from launch through EOM, including the timeline of the planned orbital maneuvers
3. Description of the mission profile, including type of orbit (e.g. LEO, MEO, HEO, GEO, L1, L2) and orbital parameters
4. Description of the mission requirements, operations, and means for orbit injection, maintenance, and disposal (e.g. Sun-synchronous, ground track, latitude, formation flying)

g. ESA space system description

1. Summary description of the space system (e.g. launch vehicle, spacecraft, inhabited or robotic vehicles), e.g. including spacecraft platform, payload instrumentation, and all appendices (e.g. solar arrays, antennas, instrument or attitude control booms.)
2. Mass budget at launch and EOM, including all propellants and fluids
3. Description of the propulsion system
4. Description of the power system
5. Description of the Attitude and Orbit Control System (AOCS)
6. Description of the Guidance, Navigation and Control (GNC) system as applicable

h. Procured launch vehicle description

1. Identification of the launch vehicle service (provider, launch vehicle, launch site)
2. Description of the launch vehicle mission profile, including all separation stages with all parking, transfer, and mission orbital parameters (apogee, perigee, inclination)
3. Identification of the launch vehicle capabilities concerning de-orbiting of upper stages
4. Launch vehicle stages mass budget, including propellant before and after operation, if available
5. Description of the propulsion systems for each stage (e.g. solid, liquid) , if available

i. Space Debris Mitigation implementation and verification

1. Space system mission-related objects (MROs)

List of MROs including:

- (a) Characteristics of any MRO released at any time after launch, including:
 - Object type
 - Object dry mass and fuel or fluids mass (if foreseen)
 - Object materials
 - Object dimensions
 - Object drawings
- (b) Rationale for release of each MRO and possible effects on debris generation
- (c) Time of release of each MRO with respect to launch time
- (d) Release or ejection velocity of each MRO



- (e) Expected orbital parameters (apogee, perigee, inclination) of each MRO after release
- (f) Analysis to determine the trajectory propagation and expected presence in the LEO or GEO Protected Regions of each MRO (Annex A)
- 2. Space system fragmentation and explosion risk
 - (a) Identification of all potential causes of break-up during deployment and mission operations
 - (b) Summary of failure modes and effects analyses of all credible failure modes which can lead to an accidental explosion
 - (c) Analysis to determine the probability of accidental break-up
- 3. Space system small particles release
 - (a) Review-of-design or test to verify that all engines do not release particles larger than 1 mm
 - (b) Review-of-design or test to verify that pyrotechnic devices do not release particles larger than 1 mm
 - (c) Assessment of the type and quantity of small particles larger than 1 mm expected to be released during normal operations, including:
 - Type of particles, e.g. SRM slags, pyrotechnic particles
 - Size, mass and density (ranges) of the particles
 - Conditions and orbit where released
- 4. Space system on-orbit collision risk assessment
 - (a) Assessment of the collision risk for the spacecraft, including:
 - Analysis to determine the probability of catastrophic collisions with objects or debris over the launch, operational, and disposal phases (Annex B)
 - Analysis to determine the probability of damage or failure due to collisions over the launch, operational, and disposal phases (Annex B)
 - (b) Assessment of the collision risk for tether systems (if applicable), including:
 - Analysis to determine the probability of catastrophic collisions with objects or debris over the launch, operational, and disposal phases (Annex B)
 - Analysis to determine the probability of damage or failure due to collisions over the launch, operational, and disposal phases (Annex B)
 - (c) Analysis to determine the number of collision avoidance maneuvers of the space system over the operational phase and the required amount of propellant mass and Delta-v
- 5. Space system disposal
 - (a) Description of the disposal option (e.g. re/de-orbit to a graveyard orbit, controlled/uncontrolled re-entry)



- (b) Identification of all subsystems or components required to accomplish any disposal manoeuvre
 - (c) Plan of the maneuvers required to accomplish the disposal phase, including engine type, thrust level, maneuvered or non-maneuvered sequence and duration, initial orbit parameters, intermediate or transfer orbits parameters, final orbit parameters, propellant mass, and Delta-v
 - (d) Analysis to determine the presence in the LEO or GEO Protected Regions after the disposal maneuvers (Annex A), including:
 - Object physical and geometrical parameters and related dispersion margins:
 - Cross-sectional area
 - Drag coefficient
 - Mass
 - Ballistic coefficient
 - Solar radiation pressure reflectivity coefficient
 - Model assumptions:
 - Atmospheric density, if relevant
 - Earth gravitational attraction
 - Lunisolar attraction
 - Solar activity (solar flux and geomagnetic index)
 - Initial conditions:
 - Orbital parameters and epoch at the end of the operational phase
 - Orbital parameters and epoch after the disposal maneuvers
 - Tool and methodology used
 - Justification for the used methodology and assumptions
 - Estimation of the presence in the LEO or GEO Protected Regions
 - (e) Analysis to determine the reliability of EOL disposal operations (Annex F)
 - (f) Tether systems disposal status (if applicable)
6. Space system passivation
- For each component storing energy (e.g. Annex E), specify:
- (a) Component type and subsystem
 - (b) Number of items
 - (c) Design measures allowing passivation
 - (d) Operational procedure allowing passivation
 - (e) Residual type and quantity of energy after the passivation operations
 - (f) Rationale if design measures or operational procedures do not allow full passivation



- (g) Explosion risk and potential effects on space debris generation, if passivation is not fully accomplished
 - (h) Available options (e.g. platform modification or changes) and related additional impact on project cost, effort, and schedule in order to allow passivation of the components which have not been planned to be passivated
7. Space system re-entry
- (a) Identification of the system functions that contribute to the controlled re-entry, if planned
 - (b) Identification of the re-entry scenario and related Fault Tree leading to nominal controlled, degraded controlled, and uncontrolled re-entry, and associated on ground risk
 - (c) Analysis to determine the re-entry casualty risk of the spacecraft and any other MRO likely to re-enter, based on the most updated re-entry conditions (Annex C), including:
 - Object physical and geometrical assumptions at re-entry i.e. last orbit before fragmentation events:
 - Model and design of the assembly
 - Design details of components (shape, sizes, mass, material, accommodation)
 - Model assumptions:
 - Atmospheric density
 - Earth gravitational attraction
 - Solar activity (solar flux and geomagnetic index)
 - Earth population model
 - Ground impact probability
 - Fragmentation model
 - Controlled or uncontrolled re-entry approach
 - Initial conditions
 - Orbital parameters and epoch at re-entry, i.e. last orbit before fragmentation events
 - Attitude at re-entry
 - Tool and methodology used
 - Justification for the used methodology and assumptions
 - Results:
 - Physical properties of each surviving fragments (size, shape, mass, material)
 - Dynamic properties of each surviving fragment (impact velocity, kinetic energy)
 - Casualty area of each surviving fragment



- Total casualty area
 - Casualty expectancy
 - Declared Re-entry Area (DRA) and Safety Re-entry Area (SRA)
 - Floating or non-floating fragments
- (d) Notification Plan for re-entry, i.e. schedule and procedure for the issue of the NOTAM, if controlled re-entry is planned
8. Space system hazardous materials (to be covered by a specific policy)
- (a) Summary of the hazardous materials contained on the space system, including:
- Chemical and commercial name of the material
 - Description of how material is hazard to humans (e.g. explosive, carcinogen, toxic, radioactive)
 - Material state (gas/liquid/solid/powder) and mass/volume and pressure at launch
 - Material state (gas/liquid/solid/powder) and mass/volume and pressure at the operational phase
 - Material state (gas/liquid/solid/powder) and mass/volume and pressure at EOM/end of passivation
 - Material state (gas/liquid/solid/powder) and mass/volume and pressure at released in the atmosphere at re-entry
 - Material state (gas/liquid/solid/powder) and mass/volume and pressure expected to survive at re-entry
- j. ESA space system compliance and verification**
1. Compliance and Verification Matrix, including identification of the applicable SDM requirements, status of the compliance with the SDM requirements, justification, close-out documents, and close-out status (e.g. Table H-1)
 2. List of events which can cause violation of the requirements and relevant consequences (e.g. description and characteristics of debris generated)
- k. Launch service compliance**
1. Assessment of the of the compliance with the SDM requirements
- As per [RD1]: *“For the procurement of launch services for ESA space systems, all reasonable efforts shall be made to ensure the use of launchers which are compliant with the space debris mitigation technical requirements in the standard.”*

Table H-1: Example of compliance and verification matrix

Req. Id. (see Note 1)	Requirement Text (see Note 2)	Compliance Status (C/PC/NC/NA) (see Note 3)	Verification Method(s) (see Note 4)	Justification (see Note 5)	Close-out Reference (Doc., section, pag.) (see Note 6)	Close-out Status (see Note 7)
6.1.1.1	See ISO Standard for text of requirement	C	ROD	No release of debris is identified by review-of-design.	XXX, pag. Y, section Z	Closed
6.1.1.2	See ISO Standard for text of requirement	C	ROD, A	No release of debris is identified by review-of-design.	XXX, pag. Y, section Z	Open
6.1.1.3	If space debris are released into Earth orbit during normal launch operations, then the number of space debris released, other than those covered by 6.1.2, shall not exceed: a. One, for the launch of a single spacecraft. b. Two, for the launch of multiple spacecraft.	C	ROD	Objects released during launch operations are identified and minimized by review-of-design since only one object is released.	XXX, pag. Y, section Z	Closed
6.1.2.1	See ISO Standard for text of requirement	NA	ROD, T	The spacecraft does not use pyrotechnic devices.	XXX, pag. Y, section Z	Closed
6.1.2.2	Solid rocket motors shall be designed and operated so as to avoid releasing solid combustion products larger than 1 mm into the GEO Protected Region.	NA	ROD, T	The spacecraft does not use solid rocket motors.	XXX, pag. Y, section Z	Closed
6.1.2.3	See ISO Standard for text of requirement	NA	ROD, T	The spacecraft does not use solid rocket motors.	XXX, pag. Y, section Z	Closed
6.2.1	See ISO Standard for text of requirement	C	ROD	The mission plan does not involve intentional fragmentations.	XXX, pag. Y, section Z	Closed
6.2.2.1	See ISO Standard for text of requirement	C	A, T	The accidental break-up probability is determined by analysis or tests.	XXX, pag. Y, section Z	Closed
6.2.2.2	See ISO Standard for text of requirement	C	A, T	FMECA of components likely to cause explosions or break-ups is performed.	XXX, pag. Y, section Z	Closed

Req. Id. (see Note 1)	Requirement Text (see Note 2)	Compliance Status (C/PC/NC/NA) (see Note 3)	Verification Method(s) (see Note 4)	Justification (see Note 5)	Close-out Reference (Doc., section, pag.) (see Note 6)	Close-out Status (see Note 7)
6.2.2.3	See ISO Standard for text of requirement	C	ROD, A, T	All components storing energy are passivated at the end of the operational phase. Components, which are not passivated do not involve risk of explosions and break-ups of the spacecraft such to generate debris.	XXX, pag. Y, section Z	Open
6.3.1.1	See ISO Standard for text of requirement	C	A	An analysis to demonstrate the probability of successful disposal is performed (e.g. Annex F)	XXX, pag. Y, section Z	Closed
6.3.1.2	See ISO Standard for text of requirement	C	A	An analysis to demonstrate the probability of successful disposal is performed (e.g. Annex F)	XXX, pag. Y, section Z	Closed
6.3.1.3	See ISO Standard for text of requirement	C	A	An analysis to demonstrate the probability of successful disposal is performed (e.g. Annex F)	XXX, pag. Y, section Z	Closed
6.3.2.1	See ISO Standard for text of requirement	NA	ROD	The spacecraft does not operate in the GEO Protected Region since it only operates in LEO.	XXX, pag. Y, section Z	Closed
6.3.2.2	See ISO Standard for text of requirement	NA	ROD, A	The spacecraft does not operate in the GEO Protected Region since it only operates in LEO.	XXX, pag. Y, section Z	Closed
6.3.3.1	See ISO Standard for text of requirement	NC	ROD	The spacecraft is not able to leave the LEO Protected Region within 25 years as per Annex A.	XXX, pag. Y, section Z ; RFD/RFW XXX	Open

Req. Id. (see Note 1)	Requirement Text (see Note 2)	Compliance Status (C/PC/NC/NA) (see Note 3)	Verification Method(s) (see Note 4)	Justification (see Note 5)	Close-out Reference (Doc., section, pag.) (see Note 6)	Close-out Status (see Note 7)
6.3.3.2	<i>See ISO Standard for text of requirement.</i>	C	ROD, A	A design and operational approach is implemented to clear the LEO Protected Region. An analysis as per Annex A is performed to verify the consistency of the approach.	XXX, pag. Y, section Z	Closed
6.3.4.1	<i>See ISO Standard for text of requirement</i>	C	A	An analysis as per Annex C is performed.	XXX, pag. Y, section Z	Closed
6.3.4.2	<i>See ISO Standard for text of requirement</i>	C	A	An analysis as per Annex C is performed. An uncontrolled re-entry strategy is chosen as the re-entry casualty risk is less than 10^{-4} .	XXX, pag. Y, section Z	Closed
<p>1. Identification of the requirement ECSS-U-AS-10C / ISO 24113:2011 as per ESA/ADMIN/IPOL(2014)2.</p> <p>2. Text of the requirement. (NOTE: Due to copyrights, the requirement text is to be found in the ISO 24113:2011.)</p> <p>3. Status of the compliance: Compliant (C), Partial Compliant (PC), Not Compliant (NC), Not Applicable (NA).</p> <p>4. Verification methods: Test (T), Analysis (A), Review-of-design (ROD), Inspection (I).</p> <p>5. Justification of the requirement applicability or not applicability and rationale for the statement of compliance.</p> <p>6. Reference to any documentation that demonstrates compliance with the requirement (e.g. section of the SDMR, reports, analysis, RFD/RFWs).</p> <p>7. Close-out status: open or closed.</p>						



Annex I

Space Debris Mitigation Request for Deviation (RFD) / Waiver (RFW) form

The form for a Request for Deviation (RFD) or Request for Waiver (RFW) is provided in the next page. The process is the following:

- a. A RFD or RFW is issued by the ESA Study Manager, Project Manager, or Mission Manager. The RFD is issued to authorize departure from a specified requirement prior to the production phase. The RFW is issued to authorize departure from a specified requirement during the production phase.
- b. The RFD/RFW is reviewed in the frame of Technical Project Reviews, which provide endorsement and recommendation. In case a RFD/RFW is generated outside project reviews, the RFD/RFW will be assessed by the Space Debris Mitigation Review Panel chaired by the Independent Safety Office (TEC-QI).
- c. The RFD/RFW is submitted to the ESA Technical Authority, i.e. the ESA Independent Safety Office (TEC-QI) for processing.
- d. The ESA Technical Authority, i.e. the Independent Safety Office (TEC-QI) is responsible for the processing of the RFD/RFW, with the technical assistance and expertise of the Directorate in charge of Technical and Quality Management, and the Space Debris Office (HSO-GR) of the Directorate in charge of Operations.
- e. The disposition of the RFD/RFW, if accepted, is done jointly by the Director of the Programme and the Director of the Technical and Quality Management Directorate (D/TEC).



ESA PROJECT NAME SPACE DEBRIS MITIGATION REQUEST FOR DEVIATION (RFD) / WAIVER (RFD)		Page 1 of X
1. RFD/RFD Number: XXX - Issue Y		2. Date: dd/mm/yyyy
3. Requested Type: <input type="checkbox"/> Deviation (RFD) <input type="checkbox"/> Waiver (RFD)		
4. Title of RFD/RFD:		
5. ESA Project Submittal:		Directorate:
6. Applicable Requirement: ECSS-U-AS-10C - Requirement XXXX		
7. Description of the Non-compliance:		
8. Rationale for Acceptance:		
9. ESA Technical Project Review Board Recommendation:		
10. ESA Project Manager (XXX-XX) Signature: _____ Date: xx/xx/xxxx		
11. Technical Authority Recommendations:		
12. Recommendation for <input type="checkbox"/> Approval <input type="checkbox"/> Not Approval		
ESA Technical Authority - Head of Independent Safety Office (TEC-QI) Signature: _____ Date: xx/xx/xxxx		
13. Approval		
ESA Programme Director (D/XXX) Signature: _____ Date: xx/xx/xxxx	ESA TEC Director (D/TEC) Signature: _____ Date: xx/xx/xxxx	



Table I-1 provides explanations about how to fill in the fields of the RFD/RFW form.

Table I-1: Description of the field content of the RFD/RFW form

Field	Content
1.	The ESA Project provides the reference number of the RFD/RFW.
2.	The ESA Study Manager, Project Manager, or Mission Manager indicates the date of the issue of the RFD/RFW.
3.	The ESA Study Manager, Project Manager, or Mission Manager indicates the type of request, i.e. deviation or waiver. <i>Note: Deviation is an a priori decision whereas waiver is an a posteriori decision with respect to the production phase (ECSS-S-ST-00-01C).</i>
4.	The ESA Study Manager, Project Manager, or Mission Manager indicates a title of the RFD/RFW.
5.	The ESA Study Manager, Project Manager, or Mission Manager indicates the Project name and the Directorate to which the project belongs.
6.	The ESA Study Manager, Project Manager, or Mission Manager indicates the applicable requirement violated by the non-compliance. The applicable document is ECSS-U-AS-10C.
7.	The ESA Study Manager, Project Manager, or Mission Manager provides a technical description of the non-compliance. If necessary, detailed description should be given in attachment.
8.	The ESA Study Manager, Project Manager, or Mission Manager: 1) provides the reason why the compliance has not been achieved, e.g. evidence by analysis, technology non-readiness, impact on design; 2) defines the design feature or procedure used to conclude that the non-compliance condition is acceptable; 3) provides the applicable support data as attachment, e.g. drawings, test reports, analysis.
9.	The ESA Study Manager, Project Manager, or Mission Manager reports the recommendation of the Board of the Technical Project Review (if the RFD/RFW has been discussed in the frame of a Technical Project Review).
10.	The ESA Study Manager, Project Manager, or Mission Manager signs the RFD/RFW and submits to the ESA Technical Authority, i.e. the Independent Safety Office (TEC-QI) for further processing.
11.	The Independent Safety Office (TEC-QI) is responsible for the processing of the RFD/RFW, with the technical assistance and expertise of the Directorate in charge of Technical and Quality Management, and the Space Debris Office (HSO-GR) of the Directorate in charge of Operations.
12.	The ESA Technical Authority, i.e. the Independent Safety Office (TEC-QI) indicates if the RFD/RFW is recommended for approval or not approval.
13.	The ESA Programme Director and the ESA Technical and Quality Management Director (D/TEC) sign the RFD/RFW in case of approval.