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DOCUMENT

ESA Space Debris Mitigation Compliance Verification Guidelines

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Update of the requirements from ECSS-U-AS-10C / ISO 24113:2011 to ECSS-U-AS-10C Rev. 1 / ISO 24113:2019		All	All
Update of the rationale, verification methods, and mitigation measures for the requirements		All	All
Update of the References		All	2
Update of data and charts		All	All
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"Should" expressions are replaced with present tense for acceptable practice.		All	All

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Table of contents

Figures

Tables

Introduction

Space Debris Mitigation includes a set of design and operational provisions, which aim at limiting the number of debris in orbit, the probability and effects of on-orbit fragmentation and collision events, and the hazards associated to re-entry, whether expected or planned (re-entry safety).

Spacecraft and launch vehicle orbital stages 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. Spacecraft and launch vehicle orbital stages can also be involved in fragmentation events due to on-orbit break-ups and collisions. Fragmentation debris pose a significant risk for short and long-term survivability of any other operational space object.

High debris density concentrations (clouds) form after on-orbit break-up or collision, and exhibit large changes in the spatial and temporal distribution. For example, in high-inclined LEO orbits, within a few days after the break-up, a debris cloud becomes more uniformly distributed within the orbital plane, and reaches a pseudo-torus distribution. At a later point in time, the debris cloud expands and evolves into a shell distribution.

Re-entering space debris also can 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.

Through progressive steps, ESA has adopted a regulatory framework to ensure Space Debris Mitigation.

In 2004, several European space agencies, including ASI, BNSC (UKSA), CNES, DLR, and ESA agreed on the "European Code of Conduct for Space Debris Mitigation". In 2008, the first ESA Space Debris Mitigation Policy was released. The ESA policy was later updated in 2014 with the ESA/ADMIN/IPOL(2014)2 [RD1], which adopted ECSS-U-AS-10C [RD2] / ISO 24113:2011 [RD3] as standard for Space Debris Mitigation. ESA/ADMIN/IPOL(2014)2 was revised and confirmed in 2018.

In 2017, ESSB-ST-U-004 [RD4] was adopted by ESA as the standard for the re-entry safety requirements.

In 2019 ISO 24113:2019 was published, replacing ISO 24113:2011 with major changes, and followed an by update of ECSS-U-AS-10C Rev. 1, which adopted all the requirements from ISO 24113:2019, with a few clarifications on the interpretation of definitions and requirements.

The ESA regulatory framework for Space Debris Mitigation is in line with the United Nations "Guidelines for the Long-Term Sustainability of Outer Space Activities" (17/07/2018) [RD14].

1 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 [RD1] and its revisions, and the standards ECSS-U-AS-10C Rev. 1 [RD2] / ISO 24113:2019 [RD3]. 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, Engineering and Quality (TEC) Directorate and ESA Operations (OPS) Directorate, including the Space Safety Programme Office (OPS-S), and representatives from the other Programme Directorates.

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

The intended users of this handbook are all ESA projects and its partners, including ESA Directors, ESA Project Managers, Study Managers, Mission Managers, Product Assurance and Safety 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 is also tutorial, since the implementation of the SDM requirements evolves with time.

2 References

[RD1]	ESA/ADMIN/IPOL(2014)2 – Space Debris Mitigation Policy for Agency Projects – ESA Director General's Office, 28/03/2014, 15/04/2019 (add. 1), 08/03/2022 (add. 2).
[RD2]	ECSS-U-AS-10C Rev. 1 – Space Sustainability – Adoption Notice of ISO 24113: Space Systems – Space Debris Mitigation Requirements – 03/12/2019.
[RD3]	ISO 24113:2019 – Space Systems – Space Debris Mitigation Requirements – 07/2019.
[RD4]	ESSB-ST-U-004 Issue 1 – ESA Re-entry Safety Requirements – 04/12/2017.
[RD5]	ECSS-S-ST-00-01C – ECSS System – Glossary of Terms – 01/10/2012.
[RD6]	ECSS-E-ST-10-04C, Rev. 1 – Space Engineering – Space Environment – 30/01/2019.
[RD7]	ECSS-Q-ST-30-02C – Space Product Assurance – Failure Mode Effects (and Criticality) Analysis (FMEA/FMECA) – 06/03/2009.
[RD8]	ECSS-E-ST-32-02C Rev. 1 – Space Engineering – Structural Design and Verification of Pressurized Hardware – 15/11/2008.
[RD9]	ECSS-Q-ST-30C Rev.1 – Dependability – 15/02/2017.
[RD10]	ECSS-Q-HB-30-08A – Components reliability data sources and their use – 14/01/2011
[RD11]	ECSS-U-ST-20C – Space sustainability – Planetary protection – 01/08/2019
[RD12]	ECSS-M-ST-10C Rev. 1 – Space project management – Project planning and implementation – 06/03/2009
[RD13]	ISO 23339:2010 – Space Systems – Unmanned Spacecraft – Estimating the Mass of Remaining Usable Propellant – 01/12/2010.
[RD14]	A/AC.105/C.1/L.366 – Guidelines for the Long-term Sustainability of Outer Space Activities – United Nations, 17/07/2018.
[RD15]	GEN-DB-LOG-00288-OPS-SD – ESA's Annual Space Environment Report – ESA, 27/05/2021. https://www.sdo.esoc.esa.int/environment_report/Space_Environment_Report_latest.pdf
[RD16]	IADC-02-01 Rev. 3 – IADC Space Debris Mitigation Guidelines – IADC, 06/2021
[RD17]	IADC-04-03 Version 7.1 – Protection Manual – IADC, 15/06/2018
[RD18]	ANSI/AIAA S-081B-2018 – Standard: Space Systems – Composite Overwrapped Pressure Vessels – ANSI/AIAA, 01/01/2018.
[RD19]	ANSI/AIAA S-080A-2018 – Standard: Space Systems – Metallic Pressure Vessels, Pressurized Structures, and Pressure Components – ANSI/AIAA, 13/04/2018.
[RD20]	DISCOS, https://discosweb.esoc.esa.int/
[RD21]	SOLMAG, https://sdup.esoc.esa.int/solmag/

3 Terms, definitions and abbreviated terms

3.1 Terms from other documents

a. For the purpose of this handbook, the terms and definitions from ECSS-S-ST-00-01C [RD5] apply, in particular for the following terms:

1. deviation
2. failure
3. ground segment
4. reliability
5. risk
6. single point failure
7. space system
8. verification
9. waiver

b. For the purpose of this handbook, the terms and definitions from ECSS-E-ST-32-02C [RD8] apply, in particular for the following terms:

1. leak-before-burst
2. pressure vessel

c. For the purpose of this handbook, the terms and definitions from ECSS-E-ST-10-04C [RD6] apply, in particular for the following terms:

1. meteoroids

d. For the purpose of this handbook, the terms and definitions from ECSS-Q-ST-30-02C [RD7] apply, in particular for the following terms:

1. failure mode, effects and criticality analysis

e. For the purpose of this handbook, the terms and definitions from ECSS-U-AS-10C [RD2] apply:

1. Earth orbit
2. probability of successful disposal

f. For the purpose of this handbook, the terms and definitions from ESA/ADMIN/IPOL(2014)2 [RD1] apply:

1. approving agent
2. casualty risk
3. disposal
4. disposal phase
5. ESA space system

6. end of life
7. operational phase

   NOTE The term "operation phase" is used in this handbook for "operational phase".

8. orbital lifetime
9. space debris
10. re-entry

g. For the purpose of this handbook, the terms and definitions from ISO 24113:2019 [RD3] 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. passivate
9. protected region
10. spacecraft

h. For the purpose of this handbook, the terms and definitions from ESSB-ST-U-004 [RD4] apply:

1. casualty
2. casualty area
3. controlled re-entry
4. declared re-entry area
5. destructive re-entry
6. re-entry probability
7. safety re-entry area
8. uncontrolled re-entry

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 catastrophic collision

collision which can cause structural break-up of a space system leading to generation of debris

3.1.3 conjunction

close approach between two objects

3.1.4 disposal orbit

final orbit after the end of mission

3.1.5 graveyard orbit

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

3.1.6 interference with Earth orbit

permanent presence or temporary crossing of Earth orbit occurring at any time during the orbital lifetime of a space system

NOTE The term “interference” is used in this handbook for “interference with Earth Orbit”.

3.1.7 mission-related object

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).

3.1.8 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

3.1.9 presence in the LEO or GEO Protected Region after the operation phase

time from the first interference with the LEO or GEO Protected Region after the operation 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 operation phase are termination by atmospheric entry or transfer to a stable disposal orbit.

3.3 Abbreviated terms

For the purpose of this handbook, the abbreviated terms from ECSS-S-ST-00-01C [RD5] and the following apply:

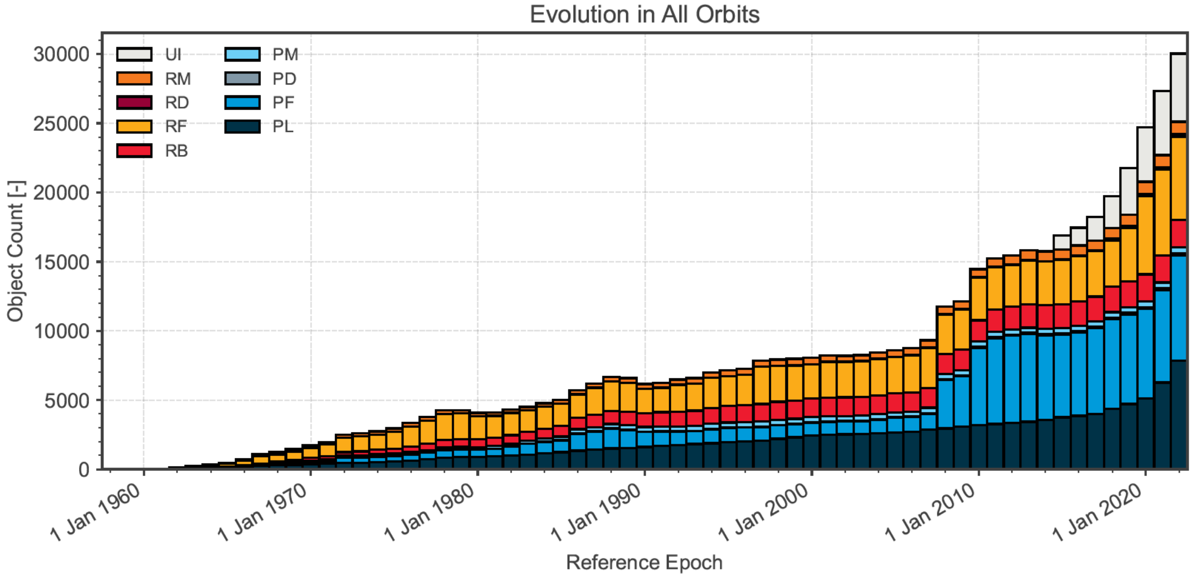
Abbreviation	**Meaning**
BLE	ballistic limit equation
CAM	collision avoidance manoeuvre
CCSDS	consultive committee for space data systems
CDM	conjunction data message
CPO	close proximity operation
CSpOC	combined space operations center

Abbreviation	Meaning
DRA	declared re-entry area
EMR	energy-to-mass ratio
EOL	end of life
EOM	end of mission
FDIR	failure detection, isolation, and recovery
GEO	geostationary Earth orbit
GNC	guidance navigation and control
GTO	geostationary transfer orbit
HEO	highly eccentric orbit
HVI	hypervelocity impact
IGSO	inclined geosynchronous orbits
LEO	low Earth orbit
LPO	libration point orbit
MEO	medium Earth orbit
MMOD	micro meteoroid and orbital debris
MRO	mission-related object
NAVAREA	geographical sea area for navigational warnings
NOTAM	notice to airmen
OMAR	on-orbit manufacturing, assembly and recycling
OOS	on-orbit servicing
SDM	space debris mitigation
SDMP	space debris mitigation plan
SDMR	space debris mitigation report
SEL	Sun-Earth Lagrangian point
SRA	safety re-entry area
SRM	solid rocket motor
SST	space surveillance and tracking
TLE	two-line element

4 Space Debris Mitigation introduction

4.1 Space debris overview

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.



Data from ESA DISCOS database, 6 April 2022: unidentified (UI), rocket mission related object (RM), rocket debris (RD), rocket fragmentation debris (RF), rocket body (RB), payload mission related object (PM), payload debris (PD), payload fragmentation debris (PF), payload (PL)

Figure 4-1: Number of catalogued objects in Earth orbit by object type

According to the ESA DISCOS database [RD20], as of April 2022, 636 break-up events of space systems were recorded, which were attributed to the following causes: 27,62% unknown; 26,7% propulsion system failure; 20,22% anomalous (local degradation/deterioration); 9,10% deliberate; 5,09% aerodynamics (overpressure for interaction with atmospheric drag); 4,32% electrical system failure; 3,86% accidental; 2,16% small impactor (leading to attitude change); 0,93% collisions (between catalogued objects).

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

On 11th January 2007, 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.

On 10th February 2009, the Iridium 33 (an US operational communication satellite, 560 kg) and Cosmos 2251 (a Russian decommissioned communications satellite, 900 kg) collided at 790 km altitude with a relative velocity of 11,6 km/s, which generated more than 2000 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-ups and explosions of spacecraft and launch vehicle orbital stages originate from 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 of reduction of the amount of debris surviving the re-entry of large space structures or 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).

5 ESA Process for Space Debris Mitigation compliance

5.1 ESA Space Debris Mitigation Policy

This Handbook provides the guidelines for ESA projects to comply with the requirements of the ESA Space Debris Mitigation (SDM) Policy [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 and collision with space debris and meteoroids.

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.

The ESA Space Debris Mitigation Policy by making applicable to ESA Projects the standard ECSS-U-AS-10C Rev. 1 [RD2], makes applicable the requirements in section 6 of ISO 24113:2019 [RD3]. The ESA Space Debris Mitigation Policy directly provides the procedural requirements, which are, therefore, not taken from the other sections of ISO 24113:2019 (e.g. section 7 of ISO 24113:2019).

5.2 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 specified in the ESA Space Debris Mitigation (SDM) Policy [RD1].

5.3 Space Debris Mitigation documentation

The Space Debris Mitigation documentation includes:

a. The Space Debris Mitigation Plan (SDMP), which is 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 H.

b. The Space Debris Mitigation Report (SDMR), which is 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 I.

The process for the issue, review, and approval of the SDMP and SDMR is summarized in Table 5-1 according to project review or phase defined in ECSS-M-ST-10C Rev. 1 [RD12].

Table 5-1: Space Debris Mitigation documentation process

Doc.	Review/Phase	Content	ESA Technical Authority Approval/ Review	Ref.
SDMP	Preliminary Requirements Review (PRR)	Preliminary plan for requirements implementation and verification	Review	Annex H
SDMP	System Requirements Review (SRR)	Final plan for requirements implementation and verification	Approval	Annex H
SDMR	Preliminary Design Review (PDR)	Requirements implementation and verification	Review	Annex I
SDMR	Critical Design Review (CDR)	Requirements implementation and verification (update)	Review	Annex I
SDMR	Qualification Acceptance Review (QAR)	Requirements implementation and verification (update)	Review	Annex I
SDMR	Flight Acceptance Review (FAR)	Requirements implementation and verification (update)	Approval	Annex I
SDMR	Flight Readiness Review (FRR)	Requirements implementation and verification (update)	Review	Annex I
SDMR	Prior to mission change	Requirements implementation and verification (update)	Approval	Annex I
SDMR	End of Mission (EOM)	Update of risk analysis	Approval	Annex I
SDMR	Prior to re-entry	Update of risk analysis	Approval	Annex I

6 Space Debris Mitigation requirements verification

The technical requirements, which the ESA Space Debris Mitigation Policy makes applicable to ESA Projects through ECSS-U-AS-10C Rev. 1 [RD2], are in section 6 of ISO 24113:2019 [RD3].

For each requirement, the structure of the sections is summarized in Table 6-1.

Table 6-1: Structure of the sections

Section Number	Section Title	Content
6.Y	Requirement	Text of the requirement (see NOTE)
6.Y.1	Rationale for the Requirement	Background information and justification for the adoption of the requirement
6.Y.2	Methods to Assess Compliance	Guidelines on how to demonstrate project compliance with the requirement
6.Y.3	Mitigation Measures	Possible implementation approaches to comply with the requirement
NOTE: Due to copyrights, the requirement text is found in ISO 24113:2019 [RD3].		

6.1 Requirement 6.1.1.1: Debris release limitation – spacecraft

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.1.1.1
See ISO 24113:2019 for the requirement text.

6.1.1 Rationale for the requirement

The requirement aims at limiting the generation of debris from spacecraft, which represent risk of collision with other objects in orbit and with the spacecraft itself.

The requirement applies to spacecraft to prevent both release of debris directly in bounded Earth orbits and release of debris outside bounded Earth orbits, which can have non-negligible probability to pass or return to Earth orbits.

Debris from spacecraft include, but are not limited to: mission-related objects, e.g. nozzle closures, lens caps, cooler covers, tethers, yo-yo weights and lines, and burst disks, which are larger than 1 mm, and can be intentionally separated or unintentionally released during operations.

For particles from pyrotechnic devices and solid rocket motors, other specific requirements in the ECSS-U-AS-10C Rev. 1 [RD2] / ISO 24113:2019 [RD3] standard apply (see sections 6.4 and 6.5 of this handbook).

For on-orbit release of spacecraft from another spacecraft, e.g. a space transportation platform, see Annex O.

6.1.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to identify any debris released from the spacecraft during normal operations.

b. Analysis, to ensure that the probability of debris returning to Earth orbits, if released in unbounded Earth orbits, or Lagrange points orbits (e.g. around SEL), is negligible.

6.1.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To design the spacecraft with no elements foreseen to be released as part of the nominal mission.

b. To design the spacecraft such that risk of unintentional release of MROs is minimised.

c. To design the spacecraft selecting materials and basic system technologies (e.g. tanks, surface materials, structures) able to be resistant to environmental degradation through time (e.g. due to radiation exposure, atomic oxygen erosion, thermal cycling), excluding when in atmospheric re-entry conditions (e.g. to avoid degrading Velcro leading to a release of MLI patches).

d. To design retention mechanisms or containment structures for deployable appendages, e.g. blocking or deployment mechanisms, thermal protection caps, explosive bolts, etc., such that they do not release fragments in orbit.

e. To design retract mechanisms for extensible appendages, when:

   1. they largely exceed the geometric envelope of the main spacecraft structure
   2. their geometry cannot be tracked by space surveillance facilities
   3. they are no longer necessary

   For example, tethers are retracted after their use.

f. To include in the spacecraft design means to limit uncontrolled build-up of attitude angles rate in case of lost space system attitude control, e.g. through the introduction of passive systems that allow kinetic energy dissipation. More details on how this can be achieved are described in Annex N.

6.2 Requirement 6.1.1.2: Debris release limitation – launch vehicle

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.1.1.2
See ISO 24113:2019 for the requirement text.

6.2.1 Rationale for the requirement

The requirement aims at limiting the number of objects and launch vehicle orbital stages left in orbit. For the launch of multiple payloads (co-passenger spacecraft), the requirement also aims at limiting the number of adapters or other launch mission-related objects, which are intentionally released.

Debris from launch vehicle include, but are not limited to launch mission-related objects, e.g. orbital stages (e.g. upper stages, kick-motor stages), adapters, fairings, connectors, fasteners (e.g. separation bolts, clamp bands, burst disks).

ECSS-U-AS-10C Rev. 1 [RD2] / ISO 24113:2019 [RD3] contains specific requirements for particles from pyrotechnic devices and solid rocket motors (see sections 6.4 and 6.5 of this handbook).

6.2.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to show that the planned number of launch vehicle orbital elements (stages, adapters, and other MROs) is within the allowed quantity.

6.2.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To select a launch vehicle which ensures that no more than the allowed number of elements or debris is accidentally or intentionally released in Earth orbit.

b. To use launch vehicle mission profiles where orbital stages have autonomous capability to perform de-orbit and passivation operations, in compliance with the other applicable requirements in ECSS-U-AS-10C Rev. 1 [RD2] / ISO 24113:2019 [RD3] and re-entry safety requirements.

c. To use launch vehicle mission profiles which ensure the minimum number of elements released in Earth orbit possibly interfering with the LEO and GEO Protected Regions.

6.3 Requirement 6.1.1.3: Debris release limitation – launch vehicle elements on-orbit presence

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.1.1.3
See ISO 24113:2019 for the requirement text.

6.3.1 Rationale for the requirement

The requirement aims at reducing the risk of debris collision by limiting the presence of launch vehicle elements to less than 25 years in the LEO Protected Region and with negligible interference with the GEO Protected Region for at least 100 years. The reference start time for the demonstration of compliance with the requirement is the on-orbit injection time if the space system has no manoeuvre capability, or the end of operation phase if the space system has manoeuvre capability.

The limit of 25 years for the presence of debris in LEO is widely accepted internationally as initially proposed by the Inter-Agency Space Debris Committee (IADC) Guidelines for Space Debris Mitigation [RD16] and adopted by large number of space agencies.

6.3.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to identify any debris released from the launch vehicle during normal operations.

b. Analysis, to demonstrate that the on-orbit presence and propagation of the any debris possibly released from the launch vehicle (e.g. orbital stages, adapters) is limited in the LEO Protected Region to less than 25 years and has negligible interference with the GEO Protected Region for at least 100 years.

The analysis is based on orbit propagation for each launch vehicle element released in orbit and includes:

a. Physical characteristic description, including geometry (i.e. shape and dimensions, e.g. through 2D or 3D drawings or CAD files), mass, and material.

b. Epoch and orbital state vector at the release event.

c. Ejection velocity (Delta-v), if released from a parent body.

d. Trajectory (orbital parameters) propagated for at least 100 years from the release epoch, including expected presence in LEO and GEO Protected Regions.

e. Relevant parameter dispersion, if a statistical analysis can solve the limitation due to uncertainties of a poorly accurate deterministic analysis (Annex M).

f. Re-entry casualty area and casualty risk, if re-entry is foreseen or possible.

In case re-entry is foreseen or possible, compliance with ESSB-ST-U-004 [RD4] is demonstrated.

6.3.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To design launch vehicle elements released in orbit with reduced orbital lifetime.

b. To limit the release of elements by design and operational procedures.

6.4 Requirement 6.1.2.1: Particle limitation – pyrotechnic devices

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.1.2.1
See ISO 24113:2019 for the requirement text.

6.4.1 Rationale for the requirement

The requirement aims at preventing the release of randomly dispersed small particles (> 1 mm) in Earth orbit from any space system. These particles, although small, can damage operational space systems. The requirement focuses on limitation of particles dispersed from pyrotechnic devices, which are largely used in spacecraft and launch vehicles (e.g. cable cutters for solar array deployment, pyrovalves, etc.).

Although the requirement specifically targets pyrotechnic devices, it is also aimed at controlling the dispersion of particles larger than 1 mm which can be released in orbit from other known space system sources or due to degradation processes.

6.4.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review-of-design or Test, to demonstrate that the selected pyrotechnic devices are qualified not to generate particles larger than 1 mm and not to disperse the particles in outer space.

For the compliance verification of a pyrotechnic device, several tests can be performed, which are executed in the relevant flight environment (e.g. especially if the device is installed externally), and under the expected conditions (e.g. worst-case vibration and shock, if relevant to the possible dispersion of particles in space). While avoiding cross-contamination between each test session, the size distribution of the particles collected after the activation of the device is analysed with the objective to check if the 1 mm cut-off particle size criterion is respected. A representative flight environment normally includes at least the pressure difference conditions experienced in space, but not necessarily the potential agglomeration of particles that occur within the involved fluids.

6.4.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To select devices or adopt space system design, which contain particles generated within the structure of the device or the space system.

b. For an internal device, like a pyrotechnic valve in propulsion systems, to implement a filter, or equivalent part downstream the device in order to collect all potential particles larger than 1 mm.

6.5 Requirement 6.1.2.2: Particle limitation – solid rocket motors

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.1.2.2
See ISO 24113:2019 for the requirement text.

6.5.1 Rationale for the requirement

The requirement aims at avoiding release of combustion products (slag) larger than 1 mm from Solid Rocket Motors (SRM) in the LEO and GEO Protected Regions since these particles can damage operational space systems.

It is important not to use solid rocket motors in orbital stages. This is especially relevant if they have by design dead-zones where recirculating gases can concentrate metalized slag, or generate a throat element, or can separate other items (e.g. burst disks), which are eventually released in orbit. The release of particles larger than 1 mm is extremely hazardous at orbit altitudes close to, or higher than, inhabited systems altitude (e.g. ISS).

Although the requirement specifically targets SRMs, it is also aimed at controlling the dispersion of particles larger than 1 mm, which can be released in orbit from other known space system sources or degradation processes. Agglomeration of particles from motors is a relevant phenomenon that can occur and possibly result in generation of debris larger than 1 mm.

6.5.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. 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.

A test procedure to assess the possible generation of particles from a rocket motor can consider:

a. Firing of the motor in a vacuum environment and collection of all the particles from the exhaust with a special net, which has mesh size smaller than the 1 mm (e.g. around 10 times smaller). The test duration also covers the cooling down phase of the motor.

b. Use of a probe collecting samples from different regimes of the motor and at different times. This allows to take into account the erosion of the motor casing and nozzle, and the corresponding impact of the combustion in the boundary layer with subsequently particle generation. At least five motors are used for a sufficient sampling.

c. Assessment of the reaction process and combustion effects of the involved chemical species within the rocket motor in order to check if materials are taking part in the combustion and if reactions are driving material particles to vaporization.

6.5.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. Not to use SRMs for orbital operations, limiting their use to the sub-orbital phase.

b. To use liquid propulsion systems or metal-free propellants as propellant for launch vehicle orbital stages and SRMs.

c. To select materials and basic system technologies (e.g. tanks, surface materials, structures) resistant to on-orbit environmental degradation (e.g. due to radiation exposure, atomic oxygen erosion, thermal cycling).

6.6 Requirement 6.2.1: Intentionally-caused break-up control

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.1
See ISO 24113:2019 for the requirement text.

6.6.1 Rationale for the requirement

The requirement aims at preventing any deliberate generation of space debris in Earth orbit caused by destruction of a space system.

A break-up of a space system can occur before re-entry only provided that the break-up is:

a. controllable, i.e. executable through commands, or occurring at known physical conditions, in a known limited altitude range, and

b. proven to significantly reduce the total re-entry casualty area of the space system, i.e. the sum of the casualty area of the space system and all its fragments is lower than the casualty area of the integral space system, and

c. not generating on-orbit fragments, i.e. the space system and all its fragments re-enter within one orbit without increasing collision risk to other objects.

6.6.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to show that the mission plan does not involve any intentional break-up in orbit.

6.6.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. Not to plan intentional break-ups in orbit.

6.7 Requirement 6.2.2.1: Internally-caused break-up control – probability threshold

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.2.1
See ISO 24113:2019 for the requirement text.

6.7.1 Rationale for the requirement

The requirement aims at reducing the risk of accidental break-up, caused by on-board sources of energy or failure of mechanical parts, to avoid generation and propagation of debris clouds in Earth orbit.

6.7.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Analysis, to demonstrate that the probability of accidental break-up for credible failure modes (excluding impacts with space debris and meteoroids) is less than 10^{-3} for the whole space system until its end of life.

6.7.3 Mitigation measures

Possible mitigation measures to minimise the risk of non-compliance are:

a. To select components and subsystems with low probability of explosion.

b. To design the space system which does not break-up as a consequence of an internal explosion of one of its components (e.g. through use of containment).

6.8 Requirement 6.2.2.2: Internally-caused break-up control – probability computation

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.2.2
See ISO 24113:2019 for the requirement text.

6.8.1 Rationale for the requirement

The requirement aims at supporting the verification of compliance for the requirement ECSS-U-AS-10C, Rev.1 [RD2] / ISO 24113:2019 [RD3] 6.2.2.1 (see section 6.7.1 of this handbook), which aims at reducing the risk of accidental break-up and avoid generation and propagation of debris clouds in Earth orbit.

For externally-caused break-up, e.g. impact with space debris and meteoroids, other specific requirements in the ECSS-U-AS-10C Rev. 1 [RD2] / ISO 24113:2019 [RD3] standard apply (see section 6.16 of this handbook).

6.8.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Analysis, which is performed for each critical component or unit storing energy and is part of a Failure Mode Effects and Criticality Analysis (FMECA) [RD7].

6.8.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. For pressure vessels, to:

   1. verify the design in accordance with the applicable standards (e.g. ECSS-E-ST-32-02C [RD8], ECSS-E-ST-10-04C [RD6], ANSI/AIAA S-081B-2018 [RD18], ANSI/AIAA S-080A-2018 [RD19]).
   2. check that load spectra are within the maximum loads foreseen up to EOL.
   3. perform an analysis to assess thermal effects, environment effects, and effects at system level and adoption of safety design requirements over all mission phases up to EOL.
   4. perform an analysis to demonstrate that propellant dissociation (if present) does not represent a hazard at sub-system level leading to an accidental explosion before EOL.

b. For battery cells, to implement passive propagation resistant design to control thermal runaway propagation preventing:

   1. side wall rupture (e.g. by designing side walls of battery assembly with enough structural safety margin with respect to maximum load from an internal cell burst);
   2. thermal runaway of adjacent cells (e.g. by using adequate cell spacing, or inter-cell passive cooling low density material);
   3. thermal runaway of parallel cells (e.g. by isolating parallel cells);
   4. damages of adjacent cells from ejecta (e.g. by protecting cells to prevent short-circuits caused by electrically conductive ejecta);
   5. flames or sparks, if in non-vacuum environment (e.g. by using arresting screens).

6.9 Requirement 6.2.2.3: Internally-caused break-up control – passivation

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.2.3
See ISO 24113:2019 for the requirement text.

6.9.1 Rationale for the requirement

The objective of the requirement is to prevent the presence in bounded Earth orbits of debris generated by catastrophic release of stored energy after the spacecraft end of life. This can achieved by energy sources "passivation", i.e. "permanent removal of stored energy", or "making the space system safe" by adopting a number of mitigation measures that make the event of generating debris in bounded Earth orbits very unlikely.

Events linked to stored energy sources that can lead to a break-up are:

a. Explosions or bursts of propellant tanks, and pressurized tanks, due to:
   1. Exothermal dissociation of propellant,
   2. Mixture of hypergolic propellants due to leaks,
   3. Pressure build-up of pressurant and propellant due to heating,
   4. Hypervelocity impacts due to penetrating space debris and meteoroids,
   5. Material degradation due to thermal cycling, atomic oxygen, ultraviolet radiation, corrosion, and Stress Corrosion Cracking (SCC), ageing.

b. Explosions or bursts of battery cells, due to:
   1. External short-circuit (leading to thermal runaway),
   2. Internal short-circuit (leading to thermal runaway),
   3. Overcharge (leading to thermal runaway),
   4. Overdischarge (leading to thermal runaway, depending on the cell chemistry and technology),
   5. Overtemperature (leading to thermal runaway),
   6. Overpressure,
   7. Cell degradation, e.g. Stress Corrosion Cracking (SCC), ageing,
   8. Cell manufacturing defects, e.g. dendrite formation, counterfeit,
   9. Hypervelocity impacts (due to penetrating space debris and meteoroids).

c. Explosions or bursts of heat pipes, due to:
   1. Pressure build-up of internal fluids due to heating,
   2. Hypervelocity impacts due to penetrating space debris and meteoroids,
   3. Material degradation due to thermal cycling, atomic oxygen, ultraviolet radiation, corrosion, and Stress Corrosion Cracking (SCC), ageing.

d. Mechanical ruptures of active rotating parts (e.g. reaction and momentum wheels).

When passivation is envisaged, the space system is prepared for the execution of the passivation operations, which foresees dedicated design implementations, commands (e.g. to relays, valves), operational modes for the relevant units, etc..

The robustness and credibility of the design provisions for passivation, including failure tolerance, reliability, qualification status, residual risk assessment, are evaluated in the frame of Design Technical Reviews and by the ESA Technical Authority on the basis of the State-of-the-Art knowhow and technology.

Passivation of the energy sources is always performed when an uncontrolled re-entry is foreseen.

Space systems performing disposal by controlled re-entry at EOM are not needed to implement passivation measures if compliant with the requirements on disposal reliability (ECSS-U-AS-10C Rev. 1 [RD2] / ISO 24113:2019 [RD3]) and re-entry safety (ESSB-ST-U-004 [RD4]).

When requirements on passivation are applied for a space system planning controlled re-entry as its nominal disposal, the requirements are only considered as a fall-back option.

Passivation is considered not completely feasible when a controlled re-entry is in execution since, e.g.:

a. for power passivation, battery depletion needs an amount of time, which is not compatible with power supply requirement and limited duration of the controlled re-entry operations.

b. for propulsion passivation, full tank depletion needs an amount of time, which is not achievable in the short time between the last burn and the re-entry impact.

c. execution of passivation operation needs visibility from ground, which is always possible since the re-entry impact is planned over an Ocean.

After the end of life of a space system, when controlled re-entry is planned, measures for the reduction of the probability of debris generation due to explosions and reduction of environmental impact due to release of toxic substances can be considered, e.g.:

a. depletion of the unused propellant before starting the re-entry manoeuvres, i.e. when starting the re-entry, only the propellant needed for the re-entry operation, including margins, is on-board, while the rest is depleted earlier in advance.

b. keeping thrusting after the last manoeuvre is performed (as long as possible), with also the aim of minimizing the re-entry footprint extension, while still ensuring its correct location.

As best practice for space systems planning controlled re-entry, passivation is also envisaged as a fall-back option in case the planned controlled re-entry cannot be performed. Best practices to reduce the probability of debris generation after the end of life can also be suggested even in the case of no contingency.

6.9.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:

   1. identify all components that store residual energy, which are depleted or made safe during the disposal phase (typical components are listed in Annex F).

   2. identify the implemented measures for depleting or making permanently safe each device storing energy taking into account the environment after the passivation.

b. Analysis or test, to:

1. demonstrate and justify that the adopted design implementation is sufficient to ensure both the capability of execution of the passivation operations at end of mission and the negligible residual risk of debris generation after the execution of the passivation operations (e.g. qualitative or quantitative risk assessments based on proven evidence can be part of the rationale).
2. support the rationale for the acceptance of components that cannot be fully depleted (i.e. by demonstrating negligible explosion probability due to residual energy and effects of hypervelocity impacts).
3. demonstrate, by performing dynamic simulations, that the passivation operations (e.g. venting) do not result in unpredictable attitude or orbit for the space system leading to interference with the LEO/GEO Protected Region, collision with other space objects, or other hazardous conditions (i.e. all possible spurious impulses are controlled).
4. confirm that there are no collateral events preventing successful passivation due to design and operational limitations (e.g. venting lines can be designed to prevent blockage from freezing propellants, etc.).

The residual risk assessment for propulsion (fuel, oxydizer, pressurant) tanks takes into account:

a. Residual pressure in the worst-case thermal condition low enough to result in negligible risk of burst/explosion after passivation.

b. Worst-case analysis based on mathematical models and simulation tools (e.g. EcosimPro, or similar tool) addressing the conditions after the end of mission conditions and taking into account the inaccuracies of the gauging measurement.

c. Maximum amount of energy stored after passivation (i.e. in the maximum remaining amount of fuel, oxidizer, pressurant after their depletion), to check whether this energy is sufficiently low to avoid burst or rupture the tank.

d. Maximum temperature, which can result in decomposition of the propellant and hazardous pressure increase, to check whether the expected conditions after end of mission are within acceptable limit not to generate debris.

e. Shielding capability implemented to protect the tank from hypervelocity impacts.

The residual risk assessment for the battery takes into account:

a. Reliability of the safety features implemented in the battery cells and modules to prevent thermal runaway and explosion.

b. Level of qualification and flight acceptance tests of the battery cells (considering the failure modes and behaviour of the specific battery cell chemistry and technology in reaction to charge, discharge, and environmental conditions).

c. Maximum State of Charge (SoC) of the battery cells after the end of mission and its evolution (minimum SoC implies less risk).

d. Maximum exposure temperature of the battery cells to check that the battery cells can withstand the worst-case thermal on-orbit conditions after end of mission (no attitude control) without exhibiting thermal runaways or explosions.

e. Reliability, life expectancy, failure analysis and worst-case thermal and radiation environment effects on the electronic components used for disconnecting the battery.

f. Estimated Delta-v (kinetic energy) gained by space debris generated in case of battery structural break-up and the likelihood of the debris to interfere in the long-term with the Protected Region, for the case when the spacecraft is outside the Protected Regions (e.g. for spacecraft in graveyard orbit above the GEO Protected Region, in MEO, or in SEL).

6.9.3 Mitigation measures

Possible mitigation measures to minimise risks of generating debris after space system end of life are:

a. For pressure vessels, to vent/depressurize the fluid(s) contained to the minimum residual quantity (pressure) achievable with the State-of-the-Art technology (against possible local accumulation or freezing conditions) and without generating uncontrolled motion of the space system.

b. For bipropellant propulsion systems, to implement inhibits with no Single Point of Failure to prevent hazardous (explosive) mixing, ignition, or chemical decomposition of the hypergolic fluids (e.g. to avoid uncontrolled mixing of fuel and oxidizer).

c. For stored electric energy, discharge the batteries and keep them in a discharged status using design solutions with adequate failure tolerance with respect to radiation, thermal, and ageing conditions. The passivation circuit is typically designed to be single point failure tolerant against inadvertent activation to avoid premature passivation of the space system. The mitigation of the risk of break-up and debris generation can be achieved by combining risk mitigation measures. Some examples are listed below:

    1. disconnection of the battery from the solar array through two fully independent commands (e.g. arm and fire commands) with at least one of the commands set as high priority command activated from the Ground Segment (reversibility of the passivation function, when only one of the two independent commands is accidentally activated, is a provision to avoid accidental execution of passivation, i.e. single point failure tolerance against accidental or spurious commands).

    2. disconnection of the battery from the main bus and connection of the battery to a permanent load by means of latching relay-based circuit.

    3. short-circuiting or disconnecting all solar array sections such as to interrupt further energy transfer to the battery.

    4. the battery State of Charge (SoC) is maintained permanently below the threshold for thermal runaway onset (depending on the cell chemistry and technology) while ensuring by design that the involved electronics stay within operative temperatures limits and remain in stable conditions with respect to the radiation environment. The adoption of this solution implies an accurate analysis that demonstrates that the risk of generating debris due to explosion after end of life is highly unlikely. Orbital environment conditions, time of permanence in orbit after end of life, components reliability are assessed for demonstrating that break-up with debris generation is highly unlikely.

    5. passive thermal protection of the battery to protect the battery from high temperature.

d. For units with rotating parts, e.g. reaction and momentum wheels, to design the unit such that failures do not cause break-up under the worst-case conditions during the presence in orbit.

Examples of design solutions are described in more detail in Annex F.

6.10 Requirement 6.2.2.4: Internally-caused break-up control – passivation for launch vehicle

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.2.4
See ISO 24113:2019 for the requirement text.

6.10.1 Rationale for the requirement

The requirement aims at preventing accidental break-ups of launch vehicle orbital stages in Earth orbit after the completion of their mission operations and post-mission disposal (as in section 6.9.1) by implementing appropriate passivation measures by design in case controlled re-entry is not performed.

6.10.2 Methods to assess compliance

See section 6.9.2

6.10.3 Mitigation measures

See section 6.9.3.

6.11 Requirement 6.2.2.5: Internally-caused break-up control – health monitoring

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.2.5
See ISO 24113:2019 for the requirement text.

6.11.1 Rationale for the requirement

The requirement aims at ensuring that the spacecraft is monitored during its on-orbit operations in order to detect possibly anomalies or trigger events that can result in an accidental break-up, or that can prevent a successful disposal, or both.

Depending on the number of units affected by on-orbit failure, the type of failure and its propagation, the consequences can be:

- Break-up with debris generation, without loss of mission and disposal capability, e.g. explosion or burst of pressure vessel in a payload, which is not functional for disposal;
- Break-up with debris generation, with loss of mission and disposal capability, e.g. due to a catastrophic collision.

Despite the fact that analyses to predict the risk of break-up and risk of disposal failure are performed during the design phases, anomalies, which are unpredictable or at worse conditions than initially assumed, can still occur in orbit resulting in an underestimation of these risks. Therefore, the risk of failures leading to a break-up or an unsuccessful disposal is updated on regular basis, or when relevant events occur, during the operation phase.

6.11.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review of design, to:

1. ensure health monitoring is implemented in the spacecraft through adequate set of sensors and on-board computer functions, with an on-ground response strategy able to keep track of critical space system parameters (e.g. temperature, pressure, absorbed radiation dose at system and unit level). It includes consistency checks to verify the data are correct, while possible corrective actions are defined in case of deviation.

b. Analysis, to:

1. identify, during the design phase, the possible trigger events resulting in anomalies or failures, which implies break-up, or degradation or loss of the disposal capability of the space system.
2. identify all the critical parameters of the spacecraft, extracted from the housekeeping telemetry (HKTM), which can be monitored to detect trigger events, anomalies, or failures.
3. assess, periodically during the operations, the status of the critical parameters to derive or update the worst-case scenarios, probability of failure, trends, or warnings.

Critical pararamters for health monitoring include, but are not limited to:

a. Temperature at local or unit level (e.g. engines, battery cells);

b. Pressure at local or unit level (e.g. engines, tanks, pressure vessels);

c. Absorbed radiation dose (e.g. EEE components regulating power storage units);

d. Duty cycles of units (e.g. battery, EEE components, electromechanical valves, thrusters);

e. Voltage or State of Charge of electrical power storage units (e.g. battery);

f. Performance degradation and wear-outs of units;

g. Attitude and orbital parameters values and their rates (e.g. rotation angles and position and their variations);

h. In-flight configuration changes (if different from the baseline design, e.g. changes in cold/hot redundancies, non-nominal operational modes);

i. Available consumables (e.g. if an anomalous consumption is recorded);

j. Available power (e.g. in the case of solar array and battery degradation).

6.11.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To install on the spacecraft several sensors, preferably redundant and independent, capable to detect possible anomalies in the space system during operations.

b. To perform periodical assessment and reviews of the health of the spacecraft, including operational reviews and trend analysis of critical parameters, frequently enough with respect to the expected reliability of the spacecraft in time and worst-case space environment conditions in which the spacecraft operates.

c. Once a trigger event or an anomaly has occurred, immediately to determine corrective actions (i.e. operational control measures) to allow to minimise the increased risk of on-orbit break-up.

6.12 Requirement 6.2.2.6: Internally-caused break-up control – contingency plan

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.2.6
See ISO 24113:2019 for the requirement text.

6.12.1 Rationale for the requirement

The requirement aims at implementing a contingency plan to respond to a rising risk (slowly or imminent) of on-orbit break-up of the spacecraft. Contingency plans are defined based on best knowledge and lessons learnt and are updated during the operation phase to cope with unpredictable failure scenarios and effects.

6.12.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review of design, to:
   1. ensure that a contingency plan is maintained during the operation phase until end of disposal.

b. Analysis, to:
   1. ensure that worst-case failure scenario have been identified and captured from best engineering practice and available lessons learnt, and responses included in a contingency plan.

6.12.3 Mitigation measures

Possible mitigation measures to minimise the risk of non-compliance are:

a. To maintain and update the contingency plan to take into account the relevant lessons learnt and newly identified worst-case failure scenarios.

6.13 Requirement 6.2.3.1: Externally-caused break-up control – collision avoidance capability for GEO

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.3.1
See ISO 24113:2019 for the requirement text.

6.13.1 Rationale for the requirement

The requirement aims at ensuring that spacecraft operating in the GEO Protected Region have the capability to perform collision avoidance manoeuvres, station-keeping manoeuvres, and disposal manoeuvres.

The GEO Protected Region includes closely confined orbits, which allow spacecraft to appear stationary from Earth ground observers. The orbital slots allocated to individual spacecraft operators in GEO is strictly regulated in order to avoid overlaps (in position and signal transmissions).

Manoeuvre capability allows to:

- maintain an assigned slot
- perform collision avoidance manoeuvres

The consequences of a catastrophic collision in the GEO Protected Region are dramatic since the presence of fragments generated in GEO from a break-up is permanent, or extremely long, and, therefore, representing an irrecoverable risk increase for any other operator.

6.13.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review of design, to:

   1. ensure that the spacecraft is equipped with units and functions enabling the spacecraft to perform collision avoidance and disposal manoeuvres.

b. Analysis, or test, to:

   1. demonstrate that a sufficient amount of propellant is available for performing:

      a) orbital slot maintenance manoeuvres in GEO, if needed;

      b) collision avoidance manoeuvres in GEO;

      c) disposal manoeuvre at end of mission from GEO to graveyard orbit.

   2. demonstrate that a recurring manoeuvring capability is available to ensure separation from possible colliding objects taking into account:

      a) conjunction screenings, which typically result in lead time of up to 14 days;

      b) propulsion system response performance.

6.13.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To establish active data exchange with operators, including sharing of orbit data and manoeuvre plans during the operations.

b. To investigate the collision avoidance capabilities and concepts of operations put in place by the operators of the space systems in the neighbouring slots, such as:

   1. Conjunction Data Messages (CDM) processing methods (CCSDS standards);

   2. Risk escalation and manoeuvre criteria;

   3. Avoidance strategy.

c. To facilitate coordinated collision avoidance operations in the GEO Protected Region.

d. Not to plan operations in the GEO Protected Region for spacecraft which have no capability to perform collision avoidance and disposal manoeuvres, or have low or unknown reliability.

6.14 Requirement 6.2.3.2: Externally-caused break-up control – collision avoidance duties

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.3.2
See ISO 24113:2019 for the requirement text.

6.14.1 Rationale for the requirement

The requirement aims at ensuring that operators limit the risk of collision events by reacting to conjunctions when the probability is above a given threshold. Methods to define a risk threshold and mitigation strategies are presented in Annex B. Active collision risk management allows operators to prevent premature mission termination and avoid events potentially adding several fragments in the space environment.

The requirement specifically addresses spacecraft with recurrent manoeuvre capabilities, operating in orbital regions where actions are taken to prevent long term pollution of the space environment. For spacecraft operating outside the Protected Regions (e.g. HEO), the requirement is intended to be applicable, at least, during the mission phases where the trajectory propagation can interfere with the LEO and GEO Protected Regions (e.g. during periods with LEO or GEO crossings).

For missions disposing the spacecraft, e.g. through controlled orbit lowering, the requirement is applicable as long as the spacecraft is active.

6.14.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review of design, to:

  1. ensure that valid criteria for collision risk management are known and defined, e.g. by the adoption of:

     a) a collision probability threshold above which the operator take corrective actions, e.g. decision-making points, collision avoidance manoeuvres (see Annex B);

     b) a target collision probability threshold being achieved when performing a collision avoidance manoeuvre;

  2. ensure that the spacecraft is capable to perform collision avoidance as response to identified collision risk;

  3. ensure that the spacecraft operation center is capable to actively react in case of collision event warnings;

  4. validate the concept of operations concerning the timeline between alerts and collision avoidance manoeuvre in order to ensure feasibility of the approach (e.g. considering orbit determination strategy, typical turnaround times for exchanges of data with external interfaces, ground station personnel availability);

  5. validate the concept of operations to ensure that appropriate interfaces (e.g. regular updates of trajectory and manoeuvre plan) and data formats have been incorporated.

b. Analysis, to:

1. assess the cumulated collision probability in all orbital regions, especially with respect to Earth orbits, where the spacecraft operate or interfere until disposal (Annex B).
2. determine the minimum amount of resources (propellant mass, Delta-v) necessary to perform the number of collision avoidance manoeuvres expected until disposal, including allocation in case of mission extension (Annex B).
3. demonstrate that the space system has sufficient manoeuvre capabilities until end of life, e.g. propellant availability, pointing requirements.

6.14.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To determine a suitable reaction threshold by following the methodology in Annex B.

b. To allocate an adequate amount of resources (propellant mass, Delta-v) including a margin commensurate to the uncertainty of the collision probability prediction, e.g. using a marging of 2 with respect to the expected number of collisions predicted by analysis.

c. To possibly select an operational orbit where the number of existing space objects is minimised (low risk of collision).

d. In case of missions operating outside the Protected Regions, to assess the feasibility of manoeuvres avoiding potential Protected Region crossings.

e. To re-assess the reaction threshold in case a major break-up event occurs.

6.15 Requirement 6.2.3.3: Externally-caused break-up control – collision risk mitigation

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.3.3
See ISO 24113:2019 for the requirement text.

6.15.1 Rationale for the requirement

The requirement aims at ensuring that spacecraft operators are aware and active in taking actions to reduce the collision risk when potential collision events are identified. Methods to define a risk threshold and mitigation strategies are presented in Annex B. The objective of the requirement is to ensure that the operator has a clear procedure in place when a manoeuvre is necessary.

The requirement is normally applied by spacecraft in their operational orbit, as not performing a Collision Avoidance Manoeuvre (CAM) can endanger the mission itself. In case of conjunctions with other operational spacecraft, the implementation of manoeuvres is usually subject to coordination with the other operators.

6.15.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review of design, to:

1. ensure that the spacecraft operator actively reacts to collision events warnings and is able to perform Collision Avoidance Manoeuvres (CAM), when necessary, until the end of life of the space system.
2. ensure that orbit and manoeuvre data are shared with Space Surveillance and Tracking (SST) provider and other operators.
3. ensure that an update is done on a regular basis, soon after new data is available.

b. Analysis, to ensure procedures are in place to:

1. assess the probability of collision for the nominal trajectory considering the uncertainties in the data received via a Conjunction Data Messages (CDM) from a Space Surveillance and Tracking (SST) provider.
2. perform conjunction screening against ephemeris made publicly available by other operators (ephemeris versus ephemeris screening in-house).
3. assess the possible collision avoidance manoeuvre options, considering specific space system constraints accounting for: time to event, Delta-v, direction of the manoeuvre, operational constraints.
4. assess the probability of collision to verify that it is not below the CAM trigger threshold (or below a trigger threshold lower than the CAM trigger threshold).
5. identify and assess other possible conjunctions in the modified orbit after a CAM planning.
6. re-assess the probability of collision when new SST or orbital data are available.
7. assess that a return manoeuvre after the event does not cause another high collision risk.
8. assess the probability of collision of any planned manoeuvre and modify or cancel the manoeuvre in case a possible high collision risk is detected.

6.15.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To possibly select an operational orbit where the number of existing space objects is minimised (low collision risk).

b. In case the probability of collision is above the defined threshold, perform a CAM, in coordination with other relavant operators, and under the condition that the probability of other conjunction events, resulting from the modified orbit, is below the threshold.

c. To verify that no further conjunctions events arise with a probability above the threshold after a CAM is performed and, if applicable, before the spacecraft plans to return to the original operational orbit.

6.16 Requirement 6.2.3.4: Externally-caused break-up control – vulnerability assessment

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.2.3.4
See ISO 24113:2019 for the requirement text.

6.16.1 Rationale for the requirement

The requirement aims at improving the design of the spacecraft against impacts with space debris and meteoroids in orbit in order to prevent on-orbit break-up and to allow successful disposal.

The selection of an operational orbit is normally defined from the mission objectives. Long-living debris, once left or generated in orbit, worsen the environmental conditions, which ultimately translates in additional efforts for spacecraft developers and operators to implement mitigation measures to reduce space system vulnerability (i.e. protection against impacts).

6.16.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Analysis, to:

   1. assess the probability of break-up due to impacts before end of life (Annex C), including:

      a) definition of mission phases and associated Earth orbits where a considerable amount of time is spent.

      b) definition of one or multiple conditions which can result in the creation of long-lived debris upon impact, considering shock propagation, fragmentation and fracture.

      c) definition of one or multiple conditions which can result in the loss of the mission due to an impact with a debris or an object, which prevents successful disposal.

      d) estimation of the expected number of impacts for each mission phase and until end-of-life.

6.16.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To design the spacecraft with sufficient shielding to protect the critical units (considering also specific accommodation to enhance protection), in order to prevent, debris generation or space system failure in case of collision with untrackable space debris or meteoroids when collision avoidance manoeuvres are not possible.

b. To accommodate the critical units far away from the external panels of the space system structure such as to enhance their protection.

c. To select an operational orbit with lower space debris and meteoroids flux concentration and, hence, reduced probability of impact.

d. To select an operational orbit, where possible generated debris are short-lived.

Possible mitigation measures to minimise risk of impact for other space assets are:

a. To ensure that the space system has cross-sectional area and major dimensions allowing it to be tracked by space surveillance and tracking sensors, e.g.:

   1. for perigee below 2000 km: cross-sectional area greater than a sphere with 0,10 m diameter (i.e. 0,00785 m^2), and characteristic dimension of at least 0,10 m in each major dimension.

   2. for perigee above 2000 km: cross-sectional area greater than a sphere with 0,50 m diameter (i.e. 0,19625 m^2), and characteristic dimension of at least 0,50 m in each major dimension.

b. To implement proven detectability enhancements (e.g. markers, laser retro-reflectors, optical targets, etc.) to facilitate identification from ground tracking and attitude reconstruction, e.g. in case of loss of attitude control or in view of close proximity operations (if planned). More details on how this can be achieved are described in Annex N.

6.17 Requirement 6.3.1.1: Successful disposal assurance – probability threshold

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.1.1
See ISO 24113:2019 for the requirement text.

6.17.1 Rationale for the requirement

The requirement aims at ensuring, by design and operation, a minimum 0,90 probability of performing the disposal of the space system, including disposal manoeuvres and passivation. The objective of the requirement is to minimise the risk for a space system to remain in the LEO or GEO Protected Regions, or generate debris in any Earth orbit, after the end of mission.

The requirement addresses the probability of successful disposal at the time the disposal is executed. The probability is unconditional and is different from a mission success probability requirement, that typically has the nature of a forecast, not facing a failure causing the loss of the mission.

The probability of successful disposal is mainly linked to the reliability of the space system, which is maintained above 0,9 for successful disposal up to the point when the disposal is executed. It is, then, important to react to in-flight anomalies affecting the disposal function. 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 disposal reliability cannot be maintained anymore during the mission.

The assessment of the performance of a space system takes into account the effects of nominal and non-nominal scenarios, and includes statistical analysis, based on adequate dispersions, when a deterministic analysis with poor accuracy is insufficient.

The current approaches for estimating the reliability of a space system through time include:

a. As per design, where nominal design assumptions and failure rates are considered;

b. Updated model, where operational events (e.g. anomalies, failures, change of cold/hot redundancies, duty cycles) are taken into account and more realistic assumptions from on-orbit experience are considered (when they become available).

For the cases of controlled re-entry, the requirement can also contribute to ensure that there is low probability of fragments falling over populated areas, which results in an increase of the expected casualties, although for re-entry the requirements in ESSB-ST-U-004 [RD4] are prevalent to ensure minimum risk to the Earth population and assets.

The "0,90 probability of successful disposal" requirement is the minimum applicable to any individual spacecraft, regardless if belonging or not to a constellation or a series of spacecraft.

Space systems are expected to have reliability higher than 0,90 for the disposal functions, which are actively performed (i.e. de-orbit, and passivation).

Space systems with undetermined reliability are always expected to comply with ECSS-U-AS-10C Rev. 1 [RD2], also in absence of planned disposal operations (i.e through natural orbit decay).

The fastest execution of disposal operations after the end of mission allowed by propulsion capability is the preferred strategy to also limit the residual on-orbit break-up risk.

6.17.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review of design, to:

   1. Identify the system unit and functions that are necessary to allow to perform successful disposal with probability of success higher than 0,90.

b. Analysis, to:

   1. forecast, during the development phase, the probability of successful disposal at the planned time (see Annex G), considering:

      a) uncertainties in the availability of resources, such as Delta-v and propellant for disposal;

      b) inherent reliability at functional level of subsystems and units that are necessary to conduct the disposal, under assumptions validated for the mission profile, according to ECSS-Q-ST-30 [RD11] and ECSS-Q-HB-30-08 [RD10], by means of Reliability Block Diagram (RBD), Worst-Case Analysis (WCA), Fault-Tree Analysis (FTA), Probabilistic Risk Assessment (PRA), and Radiation Hardness Assurance (RHA), when applicable;

      c) assessment of the effects of nominal and non-nominal scenarios, including statistical analysis, based on adequate dispersions, when a deterministic analysis has poor accuracy (e.g. for space systems operating in unbounded Earth orbits, or Lagrange points orbits, e.g. around SEL-1/2, taking into account the probability that the space system can return to bounded Earth orbits due to orbit unstability).

      d) assessment of a space segment monitoring plan to verify that all data used to assess the health and performance of any unit and function of the space system used for the disposal are acquired on-board and can be obtained by the ground segment with the necessary accuracy and frequency. Acquired data allow the detection, in a timely manner, of any performance variation or degradation which require subsequent changes to the space system operation in order to ensure successful disposal (e.g. including possible prevention measures to infant mortality and wear-out of units, and operational remediation procedures when degradations or failures are observed).

   2. update regularly the forecast probability of a successful disposal to take into account the as-flown conditions on the basis of:

a) assessment of the as-flown environmental and operational conditions of the space system in comparison with the specified conditions considered in the initial forecast of the probability of successful disposal. If the as-flown conditions are worse than the specified conditions, the forecast of the probability of successful disposal is updated accounting for the as-flown conditions and the disposal strategy is adapted accordingly. If the as-flown conditions are less demanding than the specified conditions, the forecast of the probability of successful disposal can be updated and possibly used to support an extension of the useful life of the space segment);

b) identification of the root-causes of degradation phenomena in the space system units needed for the disposal;

c) continue monitoring of any unit and function of the space system needed to perform the disposal to determine if their performance satisfies the minimum needs to successfully complete the disposal. If a performance degradation is detected affecting the residual life, the minimum performance level needed for successful disposal is estimated (e.g. operational time, duty cycles, etc., and the disposal strategy is adapted accordingly).

3. in case of an anomaly, occuring during the operations of the space system, identified from telemetry: assess the root-cause, and, based on severity, recurrence, operational constraints, and expert judgement, define the most efficient response plan, e.g. through an Anomaly Review Board (ARB), if relevant.

4. in case of mission extension: re-assess the probability of successful disposal at the time of the extended mission termination (the re-assessment is perfomed during the operations, immediately before the start of the mission extension, and repeated regularly during the mission extension period, based on on-orbit experience and anomaly record).

c. Test, to:

1. validate, during the design phase, possible technological solutions which can enhance successful disposal for the space system, including autonoumous devices, or assistance with a servicer (e.g. Active Debris Removal service), if available or planned.

6.17.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To take into account, in the early phases of the space system development, the requirement on disposal together with an EOL strategy.

b. To identify, in the earliest phases of the space system design and development, the need for health and performance monitoring of the units and functions necessary for the disposal.

c. To re-evaluate the reliability of successful disposal periodically during the life of the space system (e.g. every 6-12 months, and in case of an anomaly) in order to take into account evolution of the reliability of the functions and the availability of the on-board resources such that the EOL strategy can be timely modified to ensure a 0,90 probability of successful disposal.

d. To regularly evaluate the Delta-v needed for disposal manoeuvres (with 3σ accuracy) and assess the availability of propellant. Constantly preserve a minimum allocation of propellant mass for disposal operations including adequate margins to cover the inaccuracy associated with the propellant mass estimation method (see Annex E).

e. 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 disposal go below what has been planned at the beginning of the mission (e.g. due to large use of propellant, degradation of subsystems).

f. To design the system such that no single point failures can result in the loss of the space system and generation of space debris or prevent a successful disposal.

g. To implement a Failure Detection, Isolation and Recovery (FDIR) system to control all known failure modes, which can prevent the space system to perform a successful disposal.

h. To systematically maintain and update a record of on-orbit anomalies, lessons learnt, and response plans in the Space Debris Mitigation Report (SDMR) and in an ESA Anomaly Report, preferably with the broadest ESA missions coverage, e.g. in the ESA Anomaly Report Tracking System (ARTS). Sharing flight telemetry data and events in ESA available databases, e.g. Mission Utility and Support Tools (MUST), also helps to minimise future recurrence or effects of an anomaly, or to enhance early identification of an incoming anomaly.

i. If the space system is not capable to ensure a successful disposal by its own means, alternative disposal methods with assistance of a servicer can be used, if a valid service is available and has been approved for Active Debris Removal operations. The space system can be designed to be prepared for removal by an external servicer, e.g. by implementing mechanical capture interfaces compatible with the servicer, markers to support relative navigation, features to limit uncontrolled build-up of attitude angles rate in case of space system control lost, etc. More details about the approach to prepare a satellite for removal are provided in Annex N.

6.18 Requirement 6.3.1.2: Successful disposal assurance – vulnerability assessment

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.1.2
See ISO 24113:2019 for the requirement text.

6.18.1 Rationale for the requirement

The requirement aims at limiting the likelihood of components of a spacecraft, which are critical to perform the disposal function, to fail upon an impact or as a result thereof.

6.18.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review of design, and analysis, to:

1. Assess the vulnerability of the spacecraft (Annex C), including:

   a) definition of mission phases and associated Earth orbits where a considerable amount of time is spent.

   b) definition of the architectural design of the spacecraft.

   c) identification of the critical components.

d) definition of a failure mode (typically penetration) and assessment of the impact-induced probability of no failure (PNF) per critical component until end of life.

e) estimation of the global PNF with respect to the probability of successful disposal.

6.18.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To introduce additional shielding.

b. To reconsider the accommodation of critical components in the space system, e.g. by accomodating them behind other non-critical components with respect to the preferred impact flux direction.

c. To introduce redundancy for the identified critical components.

d. To select an operational orbit where less impacts are expected.

6.19 Requirement 6.3.1.3: Successful disposal assurance – disposal criteria

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.1.3
See ISO 24113:2019 for the requirement text.

6.19.1 Rationale for the requirement

The requirement aims at ensuring that the space system mission includes a defined disposal plan. The disposal plan is developed first during the design phase, and is systematically re-evaluated during the operation phase. The disposal plan is subject to changes in order to enhance successful disposal or implement workaround solutions in case of unpredicted on-orbit failures. The disposal plan includes pre-defined specific criteria for the initiation of the actual disposal operations.

6.19.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review of design, to:

1. ensure that a disposal plan is developed during the design phase and is consistent with the space system capability, resources, and mission profile.
2. ensure that the disposal plan is included in the flight operation procedures and its implementation is monitored, reviewed and updated as necessary during the operations until disposal is finally executed.

b. Analysis, to:

1. confirm that specific criteria are defined for the initiation of the disposal operations on the basis of the space system conditions and all relevant mission constraints.
2. confirm that the space system, at time of the disposal, is reliable, has sufficient resources with margins to perform a successful disposal and allow permanent clearance of the Earth Protected Regions with probability higher than 0,90, and is in compliance with the re-entry safety requirements (ESSB-ST-U-004 [RD4]), if re-entry is foreseen.

6.19.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To develop a disposal plan, which covers start, duration, and end time of execution, with sufficient allocation of space system and ground system resources, and a well-defined initiation criteria to ensure mitigation of potential violation of the clearance of the Earth Protected Regions and is in compliance with the re-entry safety requirements (ESSB-ST-U-004 [RD4]) in the worst-case scenarios.

6.20 Requirement 6.3.1.4: Successful disposal assurance – health monitoring

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.1.4
See ISO 24113:2019 for the requirement text.

6.20.1 Rationale for the requirement

The requirement aims at ensuring that the spacecraft status is monitored during its on-orbit operations in order to detect possibly anomalies or trigger events that prevent successful disposal.

Despite the fact that analysis to predict the risk of disposal failure is performed during the design phases, anomalies, which are unpredictable or at worse conditions than assumed, can still occur in orbit resulting in underestimation of this risk. Therefore, during the operation phase, the risk of failure leading to unsuccessful disposal is re-evaluated on a regular basis, or when relevant events occur, during the operation phase.

6.20.2 Methods to assess compliance

See section 6.11.2.

6.20.3 Mitigation measures

See section 6.11.3.

6.21 Requirement 6.3.1.5: Successful disposal assurance – contingency plan

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.1.5
See ISO 24113:2019 for the requirement text.

6.21.1 Rationale for the requirement

The requirement aims at implementing a contingency plan to respond to a rising risk (slowly or imminent) of unsuccessful disposal of the spacecraft. Contingency plans are defined based on best knowledge and lessons learnt and are updated during the operations to cope with unpredictable failure scenarios and effects.

6.21.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review of design, to:

   1. ensure that a contingency plan is maintained during the operation phase until the end of disposal.

b. Analysis, to:

   1. ensure that worst-case failure scenario have been identified and captured from best engineering practice and available lessons learnt, and responses included in a contingency plan.

6.21.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To constantly maintain the contingency plan to keep it up to date on the basis of the relevant lessons learnt, as they become available, to be able to cope with newly identified worst-case failure scenarios.

b. To implemenent operational changes in response to anomalies in order to ensure the conditions for a successful disposal of the spacecraft are met with a minimum risk of violating the space debris mitigation and re-entry safety requirements (e.g. to assess the opportunity of anticipating the disposal manoeuvres and the execution of passivation).

6.22 Requirement 6.3.1.6: Successful disposal assurance – mission extension conditions

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.1.6
See ISO 24113:2019 for the requirement text.

6.22.1 Rationale for the requirement

The requirement aims at ensuring that, after the nominal mission duration, the spacecraft mission can only be extended after further assessment of the risk for the extended period, since this can result in future non-compliance with the applicable space debris mitigation and re-entry safety requirements.

Often space system operators, in order to maximize the return of investment and mission objectives for service customers (e.g. Earth observation and science data gathering, telecommunications, etc.), try to extend the duration of the mission to a point where the space system, although apparently still functional, can have already exceeded its qualification limits and its actual status is ill-defined. In the past, space systems were often operated as long as consumables were available, without implementing a disposal plan at a specified time, therefore representing a hazard for space sustainability.

Spacecraft exhibit progressive degradation with time (ageing), especially when the nominal mission duration is exceeded and life-limited items are operating close to or beyond their qualification limits, as normally verified by tests. When the nominal mission is excedeed, failures can occur more frequently at an unpredictable rate (wear-out effects), and compromise the capability to perform a successful disposal.

A mission extension can be authorized only if the probability of successful disposal is accurately re-assessed at the end of the nominal mission. The re-assessment takes into account the actual conditions experienced in orbit in view of the extended mission termination date (e.g. possible failures or loss of redundancies during the current mission, residual reliability margin, different duty cycles, thermal and radiation conditions, or operational modes different than those considered during the design phase, available consumables). The aim of the assessment is to ensure that probability of successful disposal is proven to be still higher than 0,90 at the time of the termination of the extended mission.

Under Research and Development activities, ESA is currently investigating methods to improve reliability prediction beyond the nominal mission and to take into account wear-out effects. Methods to re-asses the reliability during the operation phase, or in view of planning mission extensions, include, pending justification and validation:

a. Diagnostic, health monitoring and Return of Experience (REX), based on data collected on-orbit and elaborated using Bayesian techniques to possibly correct overly optimistic or pessimistic (conservative) predictions;

b. Re-assessment of the validity of qualification activities performed before the operation phase, based on review of the actual operational conditions with respect to possibly overly optimistic or pessimistic (conservative) design assumptions;

c. Prognostic of Remaining Useful Life (RUL), based on physical models, stochastic models, and trend analysis to interpret telemetry data and predict more accurately the onset of wear-out phenomena.

6.22.2 Methods to assess compliance

See section 6.17.2. The indicated verification methods are re-applied with due regards to the prevalent conditions at the end of the nominal mission.

6.22.3 Mitigation measures

See section 6.17.3. The indicated mitigation measures are re-considered with due regards to the prevalent conditions at the end of the nominal mission.

6.23 Requirement 6.3.2.1: GEO clearance – disposal conditions

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.2.1
See ISO 24113:2019 for the requirement text.

6.23.1 Rationale for the requirement

The requirement aims at clearing the GEO Protected Region from launch vehicle orbital stages once they have completed their mission.

The requirement applies as well to launch vehicles orbital stages 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) in order to ensure that their probability of interference is negligible.

6.23.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:

   1. ensure that the planned disposal manoevres secure the space system into a disposal orbit outside the GEO Protected Region, taking into account the space system status and flight parameters, the reliability of its propulsion system, and the availability of propellant mass.

b. Analysis, to:

   1. evaluate the orbital trajectory propagation of the space system for at least 100 years after the completion of the planned disposal maneuvers (see Annex A).

   2. check that the selected disposal approach of the space system has a negligible probability of interference with the GEO Protected Region for at least 100 years (possible metrics for the interference with orbital regions are listed in Annex M).

   3. assess the resulting collision risk of the space system with other objects associated with the selected disposal approach, e.g. an analysis with multiple metrics allows to determine short-term effects (e.g. collision risk) and long-term effects (e.g. object accumulation) associated to the selected approach (see Annex L and Annex M).

   4. assess the robustness of the selected approach of the space system with respect to relevant sources of uncertainty (e.g. disposal epoch, cross-sectional area, other parameters, as listed in Annex M).

In addition, in relevant special cases the compliance verification includes:

1. In case re-entry is foreseen or possible: the compliance with the requirement needs demonstration of compliance as well with the re-entry safety requirements (ESSB-ST-U-004 [RD4]).
2. For operations in MEO: although not yet explicitly defined as Protected Region in ECSS-U-AS-10C Rev. 1 [RD2] / ISO 24113:2019 [RD3], since MEO includes operational regions of relevance for strategical applications (e.g. GNSS constellations), criteria for preservation are applied (see Annex K).
3. For operations in lunar orbits: it is important to assess the consequences of all possible disposal scenarios to ensure compliance with Planetary Protection requirements (ECSS-U-ST-20C [RD11]) (see Annex K).
4. For operations close to unbounded Earth orbits (e.g. injection towards SEL-2), or around SEL: the successful disposal consists in disposal into heliocentric orbits with no revisit closer than 1,5 million km to Earth (or negligible probability to interfere with the Earth Sphere of Influence) within the next 100 years, including demonstration that the probability of successful disposal is higher than 0,90 (see Annex L and Annex M).

6.23.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To lower the apogee below the GEO Protected Region, which avoids later interference with GEO, but without interfering with lower existing or assigned operational orbits (e.g. recent studies suggest apogee lower than 235-680 km from GEO, depending on the inclination and the effective area-to-mass ratio).

b. To select a disposal orbit on which natural perturbations lead to a permanent clearance of, or negligible interference with, the GEO Protected Region and any other operational orbit (in GTO, MEO, HEO) after the end of the operation phase.

c. For operations in MEO, GTO, IGSO, HEO, lunar orbits: see Annex K.

d. For operations close to unbounded Earth orbits (e.g. injection towards SEL-2), or around SEL: see Annex L and Annex M.

6.24 Requirement 6.3.2.2: GEO clearance – disposal execution for continuous presence

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.2.2
See ISO 24113:2019 for the requirement text.

6.24.1 Rationale for the requirement

The requirement provides the criteria to allow clearance of the GEO Protected Region for spacecraft.

The requirement applies the GEO Protected Region clearance conditions, which are derived from IADC recommendations ([RD16]) pointing out disposal above the GEO Protected Region, including:

- clearance altitude 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);
- clearance altitude of (1000 * Cr * A/m) in km, additional to 235 km, which reflects the effect of the solar radiation pressure depending on the physical properties area and mass of the spacecraft, and the solar radiation pressure coefficient (Cr);
- 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.

6.24.2 Methods to assess compliance

See section 6.23.2.

6.24.3 Mitigation measures

See section 6.23.3.

6.25 Requirement 6.3.2.3: GEO clearance – disposal execution for periodical presence

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.2.3
See ISO 24113:2019 for the requirement text.

6.25.1 Rationale for the requirement

The requirement aims at ensuring that spacecraft that periodically reside within the GEO Protected Region have no or negligible interference with the GEO Protected Region after end of life.

For spacecraft with recurrent presence such as IGSO missions, clearance is achieved as defined in 6.24.

For missions with sporadic presence (e.g. HEO, SEL), negligible interference is acceptable as defined in Annex M.

6.25.2 Methods to assess compliance

See section 6.23.2.

6.25.3 Mitigation measures

See section 6.23.3.

6.26 Requirement 6.3.3.1: LEO clearance – disposal conditions

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.3.1
See ISO 24113:2019 for the requirement text.

6.26.1 Rationale for the requirement

The requirement aims at limiting the presence in LEO of space systems that have ended their mission reducing therewith the risk of collision which can generate large quantities of debris and ultimately render some orbital regions completely unusable (Kessler syndrome).

The historical practice of abandoning spacecraft and launch vehicle orbital stages at EOM has allowed about 3 million kg of debris to accumulate in LEO [RD15]. Catastrophic collisions among these objects and secondary collisions (due to fragments of collisions) led to an uncontrolled increase in the object number.

The 25 years limit is derived from the 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 need to limit 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 of space systems in orbit after EOM.

For any orbit both totally included or just 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, 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.

For space systems operating in orbital regions with chaotic behaviour, or in cases where the assessment of the residual lifetime is dependent on predicted parameters, a statistical analysis can be used to verify the compliance.

Space systems which have no capability to perform collision avoidance manoeuvres or are technological demonstrators with unknown reliability (e.g. cubesats), cannot stay in orbit longer than 25 years from the time of the first interference with the LEO Protected Region (orbit injection epoch). These space systems are operated when it is proven that they can re-enter in less than 25 years without the use of disposal operations (i.e. by natural orbital decay).

6.26.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:

1. ensure that the planned disposal manoevres allows the space system to clear the LEO Protected Region, taking into account the space system status and flight parameters, the reliability of its propulsion system, and the availability of propellant mass.
2. for space systems that have no capability of performing collision avoidance or disposal manoevers: identify the space system physical characteristics including geometrical shape, mass and material properties that influence the orbital natural decay.

b. Analysis, to:

1. evaluate the orbital trajectory propagation for at least 100 years (see Annex A) after the first interference with the LEO Protected Region or after completion of the planned disposal maneuvers-if applicable.
2. demonstrate that the space system does not interfere with the LEO Protected Regions on the basis of orbital trajectory propagation after the completion of the planned disposal maneuvers (see Annex A).
3. demonstrate that the probability to interfere with the LEO Protected Region for longer than 25 years after the end of the operation phase (e.g. by using Monte Carlo methods within a reasonable parameter space, or justified assumption and analyses) is negligible (see Annex K and Annex L).
4. for space systems which have no capability to perform collision avoidance manoeuvres: demonstrate that the presence in orbit is limited to the minimum duration compatible with the mission objective and not exceeding 25 years from the on-orbit injection epoch.
5. for space systems equipped with propulsion or AOCS system, which are technology demonstrators, or have unknown or low reliability: demonstrate that the worst-case energy, which can accidentally be added to the space system (i.e max Delta-v per boost), does not result in hazardous orbit altitude change ending up in an unrecoverable violation of the end of mission clearance of the LEO Protected Region, taking into account duty margins in the operational altitude, i.e. such that re-entry is still possible within 25 years from the worst-case highest reachable apogee altitude.
6. assess the resulting collision risk of the space system with other objects associated with the selected disposal approach, e.g. an analysis with multiple metrics allows to determine short-term effects (e.g. collision risk) and long-term effects (e.g. object accumulation) associated to the selected approach (see Annex L and Annex M).

7. assess the robustness of the selected approach of the space system with respect to relevant sources of uncertainty (e.g. disposal epoch, cross-sectional area, other parameters, as listed in Annex M).

In addition, in relevant special cases the compliance verification includes:

a. In case re-entry is foreseen or possible: the compliance with the requirement needs demonstration of compliance as well with the re-entry safety requirements (ESSB-ST-U-004 [RD4]).

b. For operations in MEO: although not yet explicitly defined as Protected Region in ECSS-U-AS-10C Rev. 1 [RD2] / ISO 24113:2019 [RD3], since MEO includes operational regions of relevance for strategical applications (e.g. GNSS constellations), criteria for preservation are applied (see Annex K).

c. For operations in lunar orbits: it is important to assess the consequences of all possible disposal scenarios to ensure compliance with Planetary Protection requirements (ECSS-U-ST-20C [RD11]) (see Annex K).

d. For operations close to unbounded Earth orbits (e.g. injection towards SEL-2), or around SEL: the successful disposal consists in disposal into heliocentric orbits with no revisit closer than 1,5 million km to Earth (or negligible probability to interfere with the Earth Sphere of Influence) within the next 100 years, including demonstration that the probability of successful disposal is higher than 0,90 (see Annex L and Annex M).

6.26.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To select a disposal orbit on which natural perturbations lead to a permanent clearance of the LEO Protected Region within 25 years from the earliest of the two events:

   1. The space system has completed the nominal mission.

   2. The space system is in orbit with no capability to actively manage collision avoidance manoeuvres.

b. For operations in MEO, GTO, IGSO, HEO, lunar orbits: see Annex K.

c. For operations close to unbounded Earth orbits (e.g. injection towards SEL-2), or around SEL: see Annex L and Annex M.

6.27 Requirement 6.3.3.2: LEO clearance – disposal execution

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.3.2
See ISO 24113:2019 for the requirement text.

6.27.1 Rationale for the requirement

The requirement identifies various approaches to limit the presence of space systems in LEO at EOM by lowering their orbit altitude to facilitate re-entry.

In general, the most energy-efficient way to comply is to shorten the orbital lifetime by moving the space system to an orbit from which natural decay can occur within 25 years after the end of the operation phase.

The retrieval of a space system and return to Earth by means of an external servicer, i.e. Active Debris Removal (ADR), can be a viable option in the future. However, adoption as baseline solution for all spacecraft and launch vehicle orbital stages is pending on-orbit demonstration and consolidation of operational costs and constraints.

6.27.2 Methods to assess compliance

See section 6.26.2.

6.27.3 Mitigation measures

See section 6.26.3.

6.28 Requirement 6.3.4.1: Re-entry – safety requirements

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.4.1
See ISO 24113:2019 for the requirement text. • *"ESSB-ST-U-004 – ESA Re-entry Safety Requirements" is the ESA applicable standard.* • *For ESA Space Systems for which the System Requirements Review has already been kicked off at the time of entry into force of ESA/ADMIN/IPOL(2014)2 (28/03/2014), casualty risk minimisation shall be implemented on a best effort basis and documented in the Space Debris Mitigation Report.* • *For ESA Space Systems for which the System Requirements Review has not yet been kicked off at the time of entry into force of ESA/ADMIN/IPOL(2014)2 (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.*

6.28.1 Rationale for the requirement

The requirement identifies the applicable requirements for re-entry safety, in case a re-entry event is planned or there is a non-zero probability of re-entry to Earth.

ESSB-ST-U-004 [RD4] contains re-entry safety requirements for space systems, which are part of ESA space missions (spacecraft, launch vehicle orbital stages, debris).

6.28.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review-of-design, Analysis and Test, to:

    1. verify the compliance with the requirements in ESSB-ST-U-004 [RD4].

6.28.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To ensure compliance with the requirements in ESSB-ST-U-004 [RD4] and applicable national regulation.

6.29 Requirement 6.3.4.2: Re-entry – risks threshold

ECSS-U-AS-10C Rev. 1 / ISO 24113:2019 6.3.4.2
See ISO 24113:2019 for the requirement text.

6.29.1 Rationale for the requirement

The requirement defines the maximum expected risk associated with a re-entry event, which is set by the relevant Approving Agent. ESA as "approving agent" has set the maximum acceptable casualty risk of 10^{-4} for each re-entry event of space systems procured under ESA Programmes, including spacecraft, launch vehicles stages, crewed or robotic vehicles [RD1].

ESSB-ST-U-004 [RD4] contains re-entry safety requirements, including the 10^{-4} re-entry causalty risk threshold.

Re-entry casualty risk is determined by the fragments which are generated from the spacecraft under the effect of aerothermal and mechanical loads during re-entry and survive along the re-entry trajectory until ground impact.

Items from a typical spacecraft that can survive a re-entry event include, but are not limited to:

- Pressure vessels, e.g. tanks in Titanium alloy, Carbon Fiber-Reinforced Polymer (CFRP), or Aluminum (if of large size);
- Reaction wheels, especially if of large size, or some parts thereof (e.g. flywheel and ball bearing units);
- Magnetotorquers, especially if of large size;
- Mechanisms, e.g. Solar Array Drive Mechanisms (SADM), intrument and antenna pointing mechanisms;
- Optical elements, e.g. mirrors and lenses in ceramic-glass material, like silicon carbide (SiC), fused silica, Zerodur (lithium-aluminosilicate glass-ceramic);
- Large antenna parts, e.g. from Synthetic Aperture Radar (SAR) antennas;
- Other parts in stainless steel, Invar (Nickel-Iron alloy, e.g. in optical support elements), Titanium alloy (e.g. in valves, housing boxes, optical support elements, tri/pi-pods and fixings, star trackers), Inconel (Nickel-Chromiun alloy, e.g. in thrusters), Tungsten, Carbon Fiber-Reinforced Polymer (CFRP), Glass Fiber-Reinforced Polymer (GFRP), etc..

6.29.2 Methods to assess compliance

The verification methods used to assess the compliance are:

a. Review of design, to:

1. ensure that the risks associated to re-entry of the space system, either foreseen or planned, are taken into account in the mission procurement, design, operation, and disposal phases.

b. Analysis to:

1. assess the re-entry casualty risk and verify the compliance with the requirements in the ESA standard ESSB-ST-U-004 [RD4]; additional information on the re-entry casualty risk analysis is also provided in Annex D.

c. Test to:

1. support the validity of the assumptions taken for re-entry casualty risk analysis or "design for demise" implementations in the space system.

6.29.3 Mitigation measures

Possible mitigation measures to minimise risks associated to the requirement are:

a. To design the space system such that the demise of units, parts and materials can occur during re-entry, e.g giving preference to materials with low melting temperature (e.g. Aluminium instead of Titanium, stainless steel, Tungsten) or appropriate space system architecture favouring structural demise ("design for demise" implementations).

b. To perform a controlled re-entry over uninhabited areas for all missions for which an uncontrolled re-entry does not ensure compliance with ESSB-ST-U-004 [RD4].

Annex A
Orbit propagation analysis

A.1 Objectives

An orbit propagation analysis is performed to estimate the time spent in orbit after the operation phase with regard to the interference with the Protected Regions for:

a. Spacecraft after the operation phase and disposal maneuvers

b. Launch vehicle orbital stages after the operation phase and disposal maneuvers (if applicable)

c. All MROs (by the time of the release).

The orbit propagation analysis includes:

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 a 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 provided here to cover all relevant aspects.

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

a. Disposal orbit parameters (section A.2.2)

b. Ejection velocity (Delta-v) 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 which Protected Region (i.e. LEO or GEO) is of interest, the propagation of the orbit after the operation phase needs an appropriate or conservative level of the model accuracy and a minimum set of assumptions. The assumptions and accuracy depend on the type of initial orbit, e.g. LEO, MEO, HEO, GTO, GEO, Lagrangian Points, with an ephemeris-based approach considered as more suitable for heliocentric trajectories. 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. It involves 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 includes:

a. Orbital trajectory parameters propagation

b. Time spent in the LEO or GEO Protected Region after the operation phase (see Annex M)

c. Configurations and metrics related to stochastic analyses.

A.2.2 Disposal orbit parameters

The disposal orbit is the orbit that the space system has attained after the end of the operation 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 is 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) is 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 worst-case direction (e.g. acceleration into flight direction) is assumed.

b. An impulsive release maneuver can be assumed.

c. The initial orbit is computed by vector addition of the parent object osculating orbital state with the release velocity vector.

d. If applicable, an appropriate dispersion of the release Delta-v is considered for a stochastical analysis.

e. The resulting osculating state is converted into a single average (over true anomaly) orbital state.

A.2.4 Atmospheric drag

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

$$\vec{F}_{drag} = -\frac{1}{2}\rho A_{drag} C_D V_r \vec{V_r} \qquad \text{[A-1]}$$

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 [RD6]):

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 are avoided as they are not sufficiently accurate.

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 ±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 [RD6]). As a minimum, the following approximations are recommended:

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 is taken into account with appropriate accuracy when involving the following orbits (see also [RD6]). As a minimum, the following approximations are used:

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 sun-synchronous or quasi-sun-synchronous 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 an 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} $Wm^{-2}Hz^{-1}$) and geomagnetic index Ap, i.e. the index to describe fluctuations of the geomagnetic field (range 0-400), is used with the highest possible accuracy and when effective forecast models exist. Atmosphere models compatible with the solar activity proxies are interesting for future propogation, in contrast to atmosphere models that are a posteriori calibrated on variable proxies.

The following approaches can be adopted (in order of fidelity):

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 [RD21].
b. ECSS sample solar cycle:
   It is based on repeated cycles for the solar flux taking from the ECSS solar cycle per [RD6] (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.
c. 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.

d. Equivalent solar flux (only for LEO Sun-synchronous orbits)
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 is derived in with the following conditions:

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

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 [RD6])

Additional approaches for specific cases are:

a. 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 finds the underlying probability density function for the physical process behind each solar cycle.

b. 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.

For the solar cycle method, a DRAMA/OSCAR orbit propagation analysis, using the assessment of the expected presence in orbit computed with, as a minimum: the "ECSS sample solar cycle" and the "Monte Carlo Sampling with at least 5 samples cycles", is performed. Given that the solar activity and geomagnetic index are dependent on the epoch of analysis, the orbit propagation analysis for lifetime assessment is repeated in case the launch or the disposal window extend over a significant time span (e.g. a year), as detailed in Annex M.

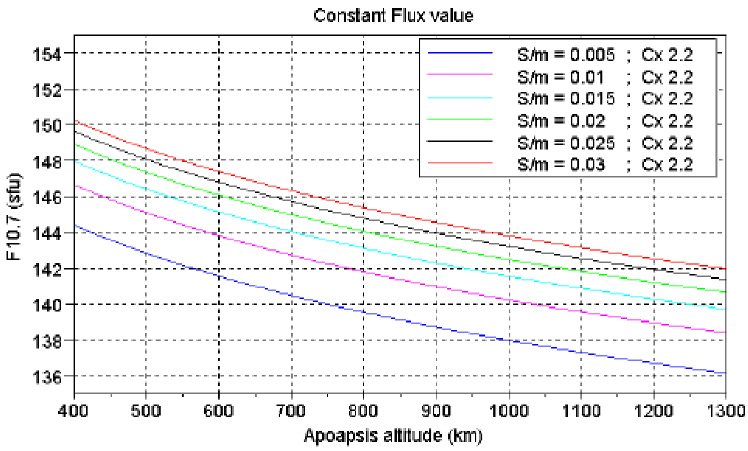


Figure A-1: Equivalent solar flux vs. orbit altitude for different values of the ballistic coefficient (area-to-mass ratio, S/m; drag coefficient, Cx)

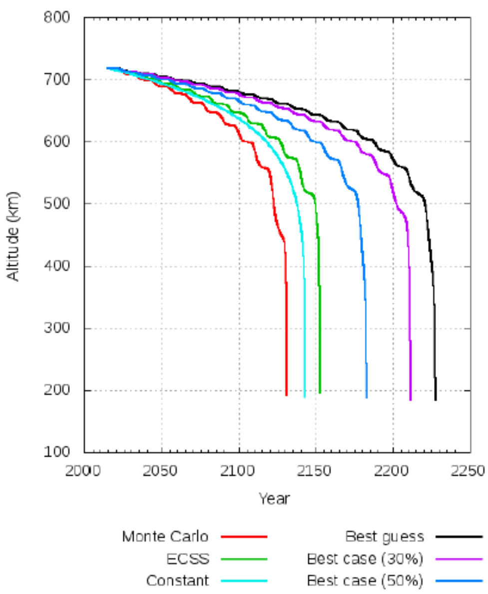


Figure A-2: An example of orbital lifetime assessment for different solar cycle scenarios: Cryosat-2, initial orbit of 710 km x 726 km, 92° inclination, epoch 01/01/2015, cross-section 3,4 m²

A.2.9 Solar radiation pressure

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

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

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)

u unit vector of the Sun-object direction

The solar radiation pressure is significant for orbits with a strong coupling to 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 a mission-related object is a parameter necessary to compute the atmospheric drag and the solar radiation pressure. Nevertheless, these forces do not act necessarily along the same direction, and, therefore, the relevant cross-sectional areas can be different. 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 the cross-sectional area is expected to 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 are identified and used for several numerical propagations in order to cover all possible scenarios without neglecting the worst cases. For example, the following considerations are taken into account:

a. Object geometrical configuration at the beginning of the assessment, i.e. release time of a MRO, end of the operation 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 is 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 or chaotic, the longer-term predictions on the rotation axis are uncertain, and also damping effects are very difficult to quantify during the development phase, it can be often assumed that the object attitude 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 (DSMC).

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

A.2.12 Object mass

The mass (M) of mission-related object at the time or phase of the prediction is 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. ±20% at PRR/SRR, 10% at PDR, and 5% at CDR) is taken into account.

A.2.13 Object ballistic coefficient

The ballistic coefficient (M) is defined as:

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

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 decreases with ageing.

If C_r has not been determined with sufficient accuracy, a conservative value is 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 output frequency

For propagation time and output frequency, it is important to use the following settings:

- Propagation time of at least 200 years, unless re-entry occurs before
- Frequency of the output orbital states of at least 1 per day for all type of orbits (for heliocentric trajectories, lower frequency can be used)

The output frequency is not equivalent to the step size of a numerical integration. The integration step-size needs always to be adapted according to the dynamics of the system. For example, this can be achieved by using a variable step-size integrator with a step-size correction scheme, e.g. the Runge-Kutta-Fehlberg 78.

A.2.16 Result uncertainties distribution

Since there are uncertainties on the physical parameters and assumptions in the models, an error can affect the accuracy of the determination of orbital lifetime and presence in the LEO or GEO Protected Regions, and re-entry casualty risk. Therefore, the final value is considered with an understood distribution and an appropriate error margin on the main estimator. The error margin can be higher than 10%, if the analysis is based on too few simulations or poor or rough assumptions.

However, this error margin does not take into account the numerical error related to the orbit propagation itself, since this is generally controlled outside the uncertainties analysis.

A.2.17 Analysis tool(s)

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 for bounded Earth orbits, which implement a sufficient solution methodology and are endorsed by ESA:

a. DRAMA/OSCAR

Use of other different tools is also possible, pending a priori discussion and agreement of the selected tool 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 NRLMSISE-00 atmosphere model, an 8th order and degree gravity model, lunisolar perturbations, solar radiation pressure, and solar activity predictions (best last update prediction) from SOLMAG [RD21], currently included in the ESA tool DRAMA. 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 useful only for rough assessments.

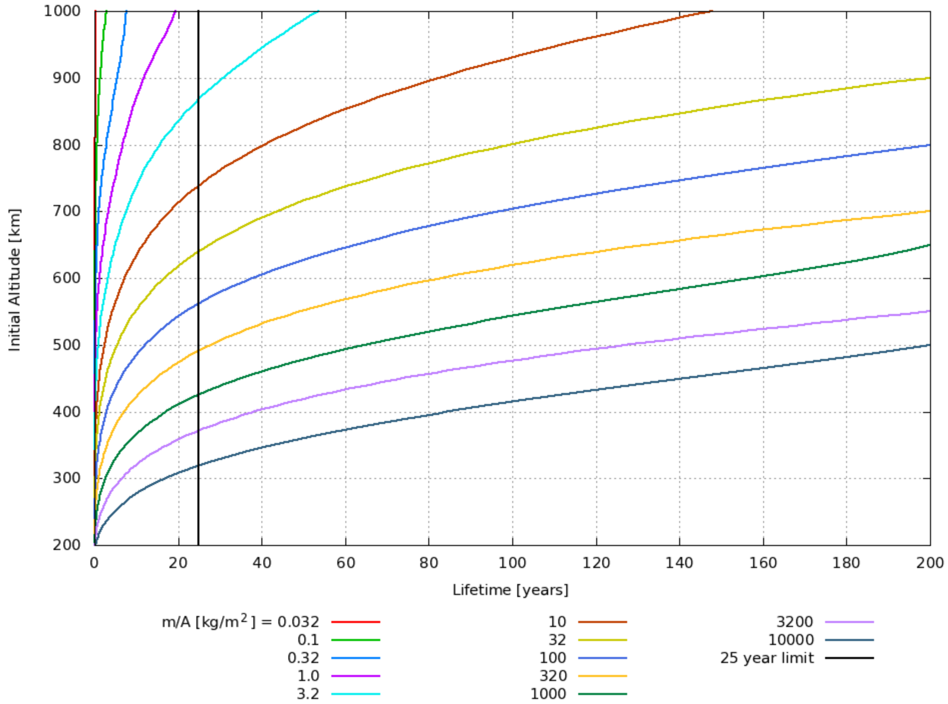


Figure A-3: Orbital lifetime and (initial) geodetic altitude for an object decaying from a circular orbit as function of mass-to-area ratio

Annex B
On-orbit collision risk analysis

B.1 Objectives

An on-orbit collision risk analysis estimates the collision risk and provides the basis for possible mitigation measures (e.g. avoidance maneuvers). An on-orbit collision risk analysis is performed for:

a. Spacecraft

b. Spacecraft tether systems (if applicable)

c. Launch vehicle orbital stages (if applicable).

An on-orbit collision risk analysis includes:

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 probability of catastrophic collision against space objects

e. Determination of the accepted collision probability level above which a collision avoidance manoeuvre is performed

f. Determination of the collision avoidance maneuvers to reduce the probability of catastrophic collision with space objects.

B.2 Methodology

B.2.1 General

A conjunction, or conjunction event, between two objects is a possible intersection between the trajectory of two objects. Information on the trajectory of an object and its related uncertainty is obtained by Space Surveillance and Tracking (SST) services, which, based on their observations of the object, determine a covariance matrix for that object at the orbit determination epoch.

The covariance matrix is usually a 6 x 6 matrix, which provides the covariance information in position and velocity (variance in diagonal elements), in a given reference frame. The covariance matrix can also have larger size, for example, if the drag or the solar radiation pressure components are also included, depending on the orbit determination process of the SST provider.

Based on observations for all tracked objects, which are usually part of a catalogue, a conjunction between two objects can be found by propagating the orbits and checking for close approaches, based on safety ellipsoids (e.g. 25 km x 10 km x 10 km). A collision can occur when the miss distance between two objects, i.e. the relative distance between the conjuncting objects at the Time of Closest Approach (TCA), is smaller than the sum of the maximum radius (including the longest appendages) of the two conjuncting objects.

From the covariances matrices propagated through the predicted Time of Closest Approach, a probability of collision between the two objects is determined, which combines miss distance, uncertainties, geometry and direction of the conjuncting objects. In order to reduce these uncertainties, accurate information about the trajectories of the possible conjuncting objects is needed from reliable Space Surveillance and Tracking systems.

Given that any on-orbit catastrophic collision can be a dramatic event, generating clouds of debris fragments, the criteria for the collision avoidance approach depends on the type of space system and the risk acceptance associated to it. For example, a space system related to human spaceflight uses stricter criteria, e.g. lower collision probability threshold and geometric clearance volumes.

The probability of an accidental break-up due to an impact or collision against space objects is never zero. Collisions between space objects 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.

Two kinds of inert objects constitute a hypervelocity-impact (HVI) risk to space systems:

a. Meteoroids, i.e. cometary or asteroidal fragments.

b. Human-made space debris, including large trackable objects and small untrackable particles.

It is possible to perform Collision Avoidance Manoeuvres (CAMs) only against space objects that are regularly tracked, e.g. with Space Surveillance and Tracking (SST) systems. Against all other objects, only passive protection can avoid critical damage to a space system (Annex C).

Most of the space debris in the space environment currently resides in, or near to, the LEO region occupied by operational spacecraft, in particular in the 700-900 km geodetic altitude shell. The following two points condition risk assessment in LEO:

- Currently, not all debris can reliably be tracked and hence are not avoidable, and
- The location uncertainty of poorly tracked objects can be of the order of kilometres (assuming a Gaussian distribution), which makes an avoidance manoeuvre operationally prohibitive.

As a consequence, a risk threshold in space debris dense orbital regions, such as LEO, is best defined in terms of achievable risk reduction with respect to the unavoidable background population. The same approach to define a risk reduction can also be used for low-density regimes (e.g. HEO).

B.2.2 Collision avoidance maneuvers against tracked objects

Performance of collision avoidance maneuvers is normally a duty of the operator of the space system. It needs a spacecraft propulsion system with an appropriate capability, an adequate propellant mass, and an assessment already in the design phase by the developer of the space system.

The assessment includes the following steps:

a. Definition of the phase (e.g. launch phase, operation phase, disposal phase)

b. Definition of the orbit state vector (as per mission design for the phase under analysis)

c. Definition of the propulsion system able to perform collision avoidance and returning maneuvers (as per system design)

d. Definition of the propellant mass available at the time of the manoeuvre (included in the mass budget)

e. Definition of the time between an event prediction and the event occurrence (e.g. 2 days)

f. Definition of the position and velocity uncertainties

g. Definition by the project of the collision avoidance strategy (e.g. specific risk threshold and target probability reduction when doing a CAM)

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 operation phase duration

j. Estimation of the amount of propellant for collision avoidance and returning maneuvers with sufficient margins.

Each space system operator defines its strategy for collision avoidance according to the applicable policy. The strategy for collision avoidance is defined 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, necessary functions, propellant allocation)

b. Collision detection measures, including analysis performed by the operator, or supplied by accepted providers, e.g. CDM (Conjunction Data Messages), whose data are distributed directly to the operators by CSpOC (Combined Space Operations Centre), or USSPACECOM, or space-track.org, or their future evolution (or other SST providers, if available)

c. Criteria for notification (i.e. probability of collision)

d. Criteria for conducting avoidance maneuvers

e. Strategy to access contact points to plan coordinated avoidance maneuvers, and data exchanging rules.

B.2.3 Initial orbit data

The major orbital parameters for a stochastic analysis of the collision risk during the space system design phase are: semi-major axis, eccentricity, inclination, and argument of perigee.

The right ascension of the ascending node (RAAN) does not, instead, play a major role when no recent break-up event occurred in view of the stochastic nature of the space debris environment and frequency of close encounters (e.g. RAAN can be simply set to 0˚).

Immediately after a recent break-up event, since the debris cloud generated has orbital planes near to the plane(s) of the originating object(s), the RAAN is significant in short-term propagation until the debris cloud is dispersed (from weeks to months). The argument of perigee is relevant for eccentric orbits. The true anomaly is usually not relevant for the analysis.

B.2.4 Epoch and mission duration

The number of objects in the space environment is typically growing over time. Therefore, in order to ensure the results of collision risk analysis are valid and consistent with the actual space environment, the analyses are repeated for epochs ranging from the start of the mission to the end of the mission, including nominal mission duration, and possible mission extensions.

B.2.5 Space system radius and cross-sectional area

The radius and cross-sectional area of the space system used for collision risk analysis are determined consistently with the actual geometry of the space system.

Stochastic collision risk analysis, operational conjunction risk analysis, and collision avoidance process normally consider spherical space objects. A collision occurs when the minimum distance between two objects (i.e. a target and a chaser) is less than the sum of the radii of the two objects. Use of complex shape models of the space system, rather than spherical, can lead to lose of consistency between stochastic collision risk analysis and operational approaches, and is, therefore, normally discouraged.

Different approaches can be selected to determine the radius of the equivalent sphere used for the target object in the analysis, which take into account the overall dimensions of the space system, including solar panels and longest appendices (listed from the most to the least conservative estimators):

1. To define as centre of the sphere the centre of mass (CoM) of the object and seek for the largest distance from the CoM (commonly used for ESA missions).
2. To consider the longest diagonal of the object to compute the radius of the equivalent object sphere.
3. To consider the radius of a circle equivalent to the area of the space system (modelled with the ESA tool DRAMA/CROC), which is exposed to the direction facing the largest space debris and meteoroids flux (derived from the ESA tool MASTER).

B.2.6 Space debris and meteoroid flux and possible conjunction types

A good comprehension of the space debris and meteoroids flux on the selected target orbit is fundamental to improve the accuracy of collision risk analysis and optimize the collision avoidance strategy, as:

- The directionality of the flux drives the typical avoidance manoeuvre approach, e.g. in case the majority of close approaches is nearly head-on (as for near-polar orbits in the populated altitude range near 800 km) a late radial CAM is a typical solution, whereas in case of frequent lateral approach geometries earlier phasing CAMs can be considered.
- The radius of the space system size can be derived from the surface exposed towards the highest flux.

The comprehension of the space debris and meteoroids flux analysis can be done with the ESA MASTER (Meteoroid and Space Debris Terrestrial Environment Reference) model, for dimension down to 1 μm in Earth orbit. Computer models have been used to simulate the generation of objects due to all known debris sources and their orbit evolution over time. Once specified the input orbital parameters, the time interval (mission duration), and the minimum size of the particles (dependent on the altitude of the orbit and the Space Surveillance and Tracking catalogue), MASTER (“target orbit” mode) allows to compute the azimuth and elevation of the flux encountered by a space system, which indicate the direction from where the conjunctions are expected, and their relative velocity with respect to the object (the highest velocity for head-on conjunctions).

B.2.7 Accepted Collision Probabiliy Level (ACPL) and number of Collision Avoidance Manoeuvres planning

Collision probability-based criteria are recommended as trigger for close approach warnings. Distance-based criteria, used in the past as trigger for conjunction warnings, are, instead, not recommended as they ignore important aspects of the approach geometry, orbit covariance and the cross sections of the objects.

Orbit information accuracy is a driver for the conjunction warning rate, which is the statistical rate by which the Accepted Collision Probability Level (ACPL) is exceeded over time. The less accurate the orbit information is, the higher number of conjunction warnings is generated, which can eventually result in a collision, or not. Highly accurate information limits the conjunction warnings to only the acute cases. Orbit information accuracy is also a driver for the residual collision probability. Events having a collision probability smaller than the ACPL are typically more numerous than the ones above the ACPL, and contribute to the residual collision risk, which grows with increasing ACPL. The avoided risk, e.g. by performing CAMs, is the accumulated collision probability of the events above the ACPL.

In order to simulate the confidence level, the covariance can be scaled with a scaling factor. The uncertainties depend on the time between the orbit determination epoch and the epoch of the conjunction (the covariance is generated at the orbit determination epoch and then propagated in the future until the conjunction).

The number of CAMs based on ACPL can be determined with the ESA tool DRAMA/ARES according to the following procedure:

a. Inputs:

1. In "Basic Settings" tab:

   a) Set the functionality to "F2 – Avoidance Schemes Assessment" in "Basic Settings" tab (additional outputs are computed if using the functionalities "F3 – Required Delta-v" or "F4 – Required Propellant").

   b) Set "begin date" corresponding to the epoch of the start of the mission.

   c) Set the orbital parameters (if they are not single averaged Kepler elements, the conversion can be done by clicking on "Import Orbital States").

   d) Set the "Spacecraft radius" as determined in B.2.5.

2. In "Detailed Population Data" tab:

   a) Set the minimum and maximum size for the space debris and meteoroids (the maximum is usually 100 m, the minimum is 0.01 m corresponding to detectable objects, e.g. with radars from CSpOC, or USSPACECOM, or their future evolution).

   b) The "radar equation" filters out from the detected risk the objects which cannot be detected, based on the current performance of the SST providers (i.e. radars or telescopes from CSpOC, or USSPACECOM, or their future evolution).

3. In "Orbit Uncertainties" tab:

   a) Set the "time between the event prediction and the event occurrence" to a typical time span needed in operations between the decision to perform or not a CAM and the time of closest approach (if such information is not available, a value of 1 day can be selected).

b) Set the global scaling factor for position uncertainties to 1 and the uncertainties definition to use the CDM-Data.

c) Set the position uncertainties at epoch to use the catalogue uncertainties.

4. In the "Collision Avoidance Strategy" tab:

a) Set the number of ACPL values to 10, and keep the default values ranging between 10^{-6} and 10^{-2} (at least initially).

b) Set Alfriend & Akella as collision risk algorithm.

b. Outputs:

1. After "Run", assess the following main output plots:

a) "Risk reduction vs. ACPL" shows the risk reduction (or avoided risk) vs. the residual (or ignored) risk, as a function of the ACPL, including the total risk due to trackable objects, and the remaining risk due to the untrackable objects for which CAMs are not possible (Annual Collision Probability for the detected and for the whole population are displayed in the "Summary" tab).

b) "Mean No. of Man. Vs. ACPL" shows the mean number of avoidance manoeuvres as a function of the ACPL. The numeric results used to generate the plots can be seen by clicking on the "Data File" tab under the plots.

2. In order to decide on an ACPL level for the mission, check when the risk reduction fraction (from column "F.Risk.Red" in "data file" tab) reaches 50%, 66%, 90% and higher, and the number of manoeuvres associated. In the selection of the ACPL, a compromise is done between the remaining accepted risk and the number of manoeuvres that the spacecraft can perform, which depends on operational and availability constraints, e.g. amount of propellant on board, service availability during the manoeuvres, manoeuvre planning time, disruption of routine procedures, etc.. The ideal target is to reduce at least 90% of the detected risk. This target is recommended typically for LEO with altitude between 500-900 km, while for other orbits a different collision risk reduction approach can be adopted considering the number of collision avoidance manoeuvres, and acceptable residual collision probability. The lower the ACPL is, the higher the risk reduction is, but also the higher the number and resources for manoeuvres are.

3. Runs are repeated for the epoch of end of mission to verify if the selected ACPL is still valid in terms of the desired level of risk reduction and the maximum number of manoeuvres, or changes are necessary.

4. Runs are repeated for the orbit parameters of each of the phases of the mission while the spacecraft is still active.

c. Sensitivity

Repeating the analysis by varying the "time between the event prediction and the event occurrence" (using double or half the lead-time) and the catalogue accuracy (using a scale factor of 2) is recommended in order to improve the sensitivity of the analysis. The variation with the "time between the event prediction and the event occurrence" can reflect cases with longer, or shorter, processing and preparation time for operations, or availability of the CDMs at a CAM decision point. Shorter lead-times make the covariance to be smaller. The variation of the catalogue accuracy reflects the confidence in the CDMs generation process, and the observation means. Doubling the covariance has a major impact on the results, but provides a worst-case approach to the environment.

Furthermore, in case of a major space object break-up event near the spacecraft orbit altitude, the space environment changes significantly, which can lead to higher manoeuvre rates for a given ACPL. Therefore, the analysis is repeated by taking into account the updated space environment data.

Detailed information on the procedure can be found at:

https://sdup.esoc.esa.int/drama/downloads/documentation/Technical-Note-ARES.pdf

B.2.8 Risk thresholds in non-LEO regions

In case the operational mission orbit crosses, or resides in, other orbital regions than the LEO Protected Region, the crossing of denser regions such as the GEO Protected Region and GNSS region is identified and the risk assessed. For the period of crossing these regions, the following steps are considered:

1. A mission not residing in the LEO Protected Region establishes the cumulative collision risk of passing through the GEO Protected Region or other operationally used regions. The cumulative collision risk covers the normal operations and at least 100 years after end of mission.
2. An ACPL based on a relative risk reduction is investigated and, if significant, applied.

In the GEO Protected Region, significant risk figures can accumulate and care is taken when establishing operational procedures in close presence of other operators. Collision risk due to the operational procedures of other operators are, however, not captured in flux based methods such as DRAMA/ARES.

B.2.9 Analysys tool(s)

The DRAMA/ARES tool is used to determine the expected annual manoeuvre rates with respect to the target orbit and the debris environment and is endorsed by ESA.

Use of other tools is also possible, pending a priori discussion and agreement of the selected tool with ESA.

B.3 Formation Flying (FF), Close Proximity Operations (CPO), On-orbit Servicing (OOS), and On-orbit Manufacturing, Assembly and Recycling (OMAR)

In addition to the collision avoidance approach against catalogued objects, missions including Formation Flying (FF), Close Proximity Operations (CPO), On-orbit Servicing (OOS), or On-orbit Manufacturing, Assembly and Recycling (OMAR), need care to avoid collisions between the involved space systems (e.g. two or more space system elements; or a servicer and a target spacecraft).

The Failure Detection, Isolation and Recovery (FDIR) is the key element that contains the design features to efficiently allow to perform collision avoidance manouvers.

For example, the FDIR implemented in an active spacecraft (e.g. a leading spacecraft in Formation Flying, or a servicer spacecraft) is expected to continuously allow to assess the collision risk (e.g. between two spacecraft in Formation Flying, or between a servicer and a client spacecraft during CPO, OOS and OMAR), and, when the collision risk exceeds the acceptable threshold, the FDIR helps in raising a warning and performing avoidance manouvers, as needed.

For operations requiring capture, docking, or berthing between a servicer and a client spacecraft, the FDIR modes are tailored to the specific case of the client spacecraft, considering if cooperative or uncooperative.

The analysis includes demonstration that possible hazardous contact or collision between the involved space systems, possibly occuring as a result of failures, are not likely to generate debris in orbit in the worst-case scenarios. The probability of collision for the nominal trajectory is assessed considering the measured relative state vector between the space systems involved. The analysis can be based on a quantitative assessment showing that:

a. the probability of occurrence of a collision between the involved space systems (or space system elements) is less than 10^{-4}, or

b. the cumulate probability of break-up of the involved space systems (or space system elements) is less than 10^{-3}, through their end of life.

The following guidelines for the operations are notable:

a. To possibly implement on-board capability to autonoumsly detect potential collision events and perform successful avoidance manoeuvres during the operations.

b. To implement design solution in line with the available ESA Guidelines (e.g. "ESA-TECSYE-TN-022522 - ESA Guidelines on Safe Close Proximity Operations").

c. To perform an autonomous CAM such to guarantee that no collision occur for a defined period of time (e.g. at least 3 days) in case the probability of collision is above the defined threshold.

To ensure that on-board FDIR allows to actively manage collision events and is able to autonomously perform Collision Avoidance Manoeuvres (CAM), when necessary, until the end of life of the space system.

Annex C On-orbit break-up and vulnerability risk analysis

C.1 Objectives

An on-orbit break-up and vulnerability analysis estimates the damage risk against impact with space debris and meteoroids, frequently also called micrometeoroids and orbital debris (MMOD), especially when they are untrackable (in view of their limited size) and still have significant kinetic energy.

Mitigation measures (e.g. shielding) are adopted when the outcome of the on-orbit break-up and vulnerability analysis is unacceptable since collisions avoidance manoeuvres are not possible against untrackable objects.

For example, for a typical spacecraft in LEO, the probability of a mission-terminating impact by a lethal untrackable object is about one order of magnitude higher than the one by the monitored space debris. Typically, untrackable lethal objects are in the size range of mm to cm. The majority of the space debris in this size range resulted from previous break-up events, solid rocket motor firings and impact-induced debris generation events (ejecta).

The outcome of a on-orbit break-up and vulnerability analysis is considered to eventually enhance the spacecraft, or launch vehicle, making it more resilient to the space environment by avoiding hazardous generation of space debris and premature mission termination, which can prevent a successful mission completion and disposal.

An on-orbit vulnerability analysis is performed for:

a. Spacecraft

b. Passive de-orbit devices (if applicable)

c. Launch Vehicle Orbital Stages (if applicable)

An on-orbit break-up and vulnerability analysis includes:

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 space debris and meteoroids

f. Determination of the share of risk that can be mitigated by introducing collision avoidance maneuvers to reduce the probability of catastrophic collision with large objects

g. Determination of the probability of damage or failure due to collisions (vulnerability).

C.2 Methodology

C.2.1 General

The probability of an accidental break-up due to an impact or collision against space objects is never zero. Collisions with space debris or meteoroids can cause:

a. Spacecraft or launch vehicle stage break-up, which beyond a certain impact energy can result in complete fragmentation.

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.

Two kinds of inert objects constitutes a hypervelocity impact (HVI) risk to space systems:

a. Meteoroids, i.e. cometary or asteroidal fragments.

b. Human-made space (orbital) debris, including large trackable objects and small untrackable particles.

C.2.2 Catastrophic collisions threshold

The main parameter to determine whether a collision is catastrophic, i.e. it generates fragments, is the energy-to-mass ratio (EMR), which relates the kinetic impact energy of the projectile to the mass of the space system:

$$EMR = \frac{\frac{1}{2} M_p V_{imp}^2}{M_t} \qquad \text{[C-1]}$$

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)

Although there is no *EMR* cut-off value, which is proven to be a valid threshold for a catastrophic collision $(EMR)_{cc}$ in all collision scenarios, a threshold for a catastrophic collision $(EMR)_{cc}$ is approximately assumed, when no better accurate analysis is available:

$$EMR \geq (EMR)_{cc} = 40\,{}^{\mathrm{J}}\!/_{\mathrm{g}}$$

The $(EMR)_{cc}$ assumption was derived from Tedeschi, W.J., et al., "Determining the effects of space debris impacts on spacecraft structures", Acta Astronautica, vol. 26, no. 7, p. 501-512, 07/1992.

In order to compute the impact velocity (V_{imp}) against a debris, the worst-case is 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:

$$V_{imp} = \sqrt{\frac{2\mu}{r_p} - \frac{\mu}{a}} + \sqrt{\frac{\mu}{r_p}} \qquad \text{[C-2]}$$

where:

μ gravity constant (i.e. the product of the gravitational constant and planet mass; for Earth $\mu = 3{,}98604415 \cdot 10^{14}$ m^{3}s^{-2}, [RD6])

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:

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

In order to convert this threshold mass to a threshold size, a standard projectile with the following characteristics can be used:

a. Area-to-mass ratio: $(A_p/M_p)_{std} = 0{,}01\ m^2/kg$

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 $D_p \geq \sqrt{\frac{8}{\pi}\left(\frac{A_p}{M_p}\right)_{std} \frac{(EMR)_{cc} M_t}{V_{imp}^2}}$ → Catastrophic Collision Analysis

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

The catastrophic collision threshold is assessed implicitly in the flux evaluation using the ESA tool DRAMA/MIDAS.

The definition of threshold for a catastrophic collision is valid only when the space system is a compact object, i.e. the impacting projectile encounters a significant share of the target object mass (e.g. a box-shaped spacecraft), while effects of direct impact on appendages (e.g. solar array wings) are not strictly taken into account in the defintion of the threshold. Therefore, the target object mass for a box-wing spacecraft model can only entail the mass associated with the main body (box) and not the solar array wings. Similar considerations are made for the collision cross-sectional area.

However, detailed studies, based on simulations and tests, demonstrate that the bare criteria based on EMR threshold cannot be always conservative and is generally not sufficient to exhaustively determine whether a specific collision event is catastrophic, i.e. if the collision event causes failure of the space system preventing its disposal, or break-up of the space system with generation of secondary debris, which are hazardous for other space systems, or trigger new collision events with other objects. In fact, for an exhaustive assessment of the effects of a collision event, the evaluation of specific factors like direction of the impact, geometry, number of fragments, size distribution, and velocity distribution, is relevant. Therefore, only detailed analysis or test, can actually prove which are the realistic effects of a specific collision scenario. Detailed analysis can be performed with Finite Element (FE) and Smoothed-Particles Hydrodynamics (SPH) codes, e.g. LS-DYNA (Livermore Software Technology Corporation, USA), and SOPHIA (Fraunhofer EMI, Germany). The analysis can determine the density and dimension of possible secondary debris clouds, which are generated as a consequence of an impact, and further allow to facilitate tracking from SST systems.

C.2.3 Collision cross-sectional area

The collision cross-sectional area (A_{Coll}) of a space system is the envelope of the maximum projected area of the space system and the area of the impacting object (space debris and meteoroids). For complex shaped objects, it can be determined by considering the enveloping ellipsoid area or developing a 3D projection model. As a conservative estimate, considering the circular area with diameter equal to the maximum object size has been used, for example by applying the expression:

$$A_{Coll} = \frac{\pi}{4}\left(D_t + D_p\right)^2 \qquad \text{[C-4]}$$

where:

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

D_p debris or meteoroids diameter

C.2.4 Probability of catastrophic collision

Typically spacecraft or launch vehicle stages with mass smaller than 50 kg can be destroyed or damaged even by small space debris and meteoroids, 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 0,01 m, depending also on the associated impact characteristics.

This assessment allows to understand the likelihood of a catastrophic break-up and mitigation measures to mitigate a substantial share of the risk, such as active collision avoidance maneuvers. The share of objects that can be avoided is typically constrained by the sensitivity limits of the Space Surveillance and Tracking networks, such as the US Space Surveillance Network (SSN). As an example, the sensitivity of radar-based tracking declines with the fourth power of the distance to the tracked object. A typical detection limit for the US SSN is 8 cm diameter at 800 km.

The assessment includes the following steps:

a. Definition of the life cycle phase(s) of the space system (e.g. launch phase, operation phase, disposal phase)

b. Definition of the phase duration (as per mission design for the phases under analysis)

c. Definition of the orbit state vector for nominal and potential orbit evolution (as per mission design for the phases under analysis)

d. Definition of the design parameters of the space system (mass and cross-sectional area)

e. Definition of the threshold size of space debris and meteoroids likely to cause a catastrophic collision (as per criterion in C.2.2)

f. Determination of the impact strength of the material and estimation of the thickness of the layers of material for shielding impacts

g. Use of the space debris and meteoroids flux model, available from the ESA tool MASTER, for the phases under analysis

h. Estimation of the position uncertainty level, if possible

i. Determination of the catastrophic collision probability over the phase duration, e.g. from the annual collision probability ($P_{c,yr}$):

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

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

j. Determination of the share of that risk that can be mitigated by conducting active collision avoidance.

C.2.5 Probability of non-catastrophic impacts

Projectiles with impact energy below the catastrophic collision threshold can still cause considerable damage to a space system and eventually generate new debris. The risk associated to debris generated by non-lethal impacts due to cratering are assessed through the following steps:

a. Definition of the life cycle phase(s) of the spacecraft or launch vehicle stage (e.g. launch phase, operation phase, disposal phase)

b. Definition of the phase duration (as per mission design for the phases under analysis)

c. Definition of the orbit state vector for nominal and potential orbit evolution (as per mission design for the phases under analysis)

d. Definition of the of the design parameters of the space system (mass and cross-sectional area)

e. Use of the space debris and meteoroids flux model, available from the ESA tool MASTER, for the phases under analysis

f. Determination of the associated crater size distribution for the obtained flux

g. Determination of the released debris mass assuming half-sphere cratering.

C.2.6 Probability of damage or failure due to collisions

This assessment provides the vulnerability level of the spacecraft or launch vehicle stage against the impact with space debris or meteoroids.

The assessment includes the following steps:

a. Definition of the life cycle phase(s) of the space system (e.g. launch phase, operation phase, disposal phase).

b. Definition of the phase duration (as per mission design for the phases under analysis).

c. Definition of the orbit state vector for nominal and potential orbit evolution (as per mission design for the phases under analysis).

d. Definition of the of the design parameters of the space system (mass and cross-sectional area).

e. Definition of the impact survivability requirement with minimum Probability of "No Failure" (PNF_{min}) for each critical component, which 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.

f. 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).

g. Determination of the at-risk surface areas for all critical parts, units, or components, which are predominant contributors to space system failure (e.g. the least protected units from space debris and meteoroids impact) considering that:

   1. In the case the critical surface is equally protected by other spacecraft parts, the at-risk area is the total area of the critical surface.

   2. In the case the critical surface is partially 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.

   3. If a unit (e.g. an electronics box) is attached to the inside of the outer wall of the vehicle, the at-risk area is the area of the unit on the side attached to the outer wall. If the unit is attached to the exterior of the outer wall of the vehicle, the at-risk area can be the total area of the unit, excluding the side attached to the outer wall.

   4. The identified at-risk areas is checked to be correct with the orientation of the surface with respect to the spacecraft attitude.

h. 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.

i. Determination of the impact-induced "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).

j. 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).

k. Determination of the expected number of impacts likely to cause damages or failures.

l. Determination of the contribution to the probability of successful disposal due to the PNF combination of all critical components.

The ballistic limit for each surface can be determined through the Ballistic Limit Equations (BLE), which are equations based on experimental data from hypervelocity impacts over surfaces. The BLEs depend on impactor data (density, diameter, velocity), and surface data, including number of layers (e.g. single wall, multiple walls), material properties (density, failure factors), type (e.g. homogeneous, composite, honeycomb), and geometry (thickness, spacing between walls). Information on the use of BLEs can be found in DRAMA/MIDAS tool user manual, or IADC Protection Manual [RD17]. Conservative assumptions are taken for the BLEs parameters when in absence of a validated justification (e.g. test).

The DRAMA/MIDAS user manual is available at:

https://sdup.esoc.esa.int/drama/downloads/documentation/DRAMA-Software-User-Manual.pdf

C.2.7 Appendages and passive de-orbit devices

C.2.7.1 Tethers

Tethers are flexible long and narrow structures, with two dimensions much smaller than the third one, which can be extended from a spacecraft.

The potential to damage operating spacecraft does not depend solely on the tether mass and cross-sectional area. The probability of collision with large objects, space debris, or meteoroids, is assessed with a specific analysis for the tethers using the same methodology in section C.2.6 for the time the tether is deployed in space (i.e. during operation phase and disposal phase).

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

$$A_{Coll.T} = D_{Ti} L \qquad \text{[C-6]}$$

where:

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

L tether length

In case a deployed tether is severed, 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.

C.2.7.2 Sails

A sail is a deployable low-mass structure which significantly increases the cross-sectional area of a space system and can be used to reduce the ballistic coefficient to enhance faster passive de-orbit by exploiting atmospheric drag or solar radiation pressure. The probability of collision with space debris or meteoroids is analysed using the same methodology in C.2.4 and C.2.5 from the time the sail is deployed by the space system.

For the catastrophic collision risk assessment, the cross-sectional area for the space system can be reduced to the cross-sectional area of its primary structure (excluding the sail) only when it is proven that space debris and meteoroid impacts on the sail area do not result in catastrophic break-up, nor create new debris.

Additional structural elements can also be present with a sail, like booms. It can be assumed that space debris or meteoroids impact on booms either penetrate or severe the boom in two pieces. This can have an impact on the effectiveness of the sail to ensure the subsequent orbit descent, which is specifically assessed.

C.2.8 Analysis tool(s)

The 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 (e.g. ESABASE2/Debris by Etamax Space GmbH), pending a priori discussion and agreement of the selected tool with ESA.

Annex D Re-entry casualty risk analysis

D.1 Objectives

A re-entry casualty risk analysis estimates the risk of a space system to cause human injuries or fatalities due to the space system. A re-entry casualty risk analysis is performed for the re-entry of:

a. Spacecraft

b. Launch vehicle orbital stages

c. All MROs

Hazards to human health and Earth environment, human injuries and fatalitites, and damages to assets, which are associated to the re-entry of a space system, can be caused by:

a. Impacting fragments

b. Floating fragments

c. Pressurized or explosive fragments

d. Hazardous chemical substances

e. Radioactive substances

The re-entry casualty risk analyses is documented such as to provide:

a. Description and justification of the methodology of the computation;

b. Description and justification of the model assumptions and uncertainties;

c. Description and justification of the initial or boundary conditions;

d. Determination of the re-entry probability ($P_{re\text{-}entry}$);

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
9. Toxicity (if any)

10. Radiation hazard (if any)

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)

The requirements and documentation for re-entry casualty risk analysis are defined in ESSB-ST-U-004 [RD4].

D.2 Methodology

D.2.1 Re-entry probability

The calculation of the casualty risk to human population takes into account the probability of a re-entry scenario to occur ($P_{re\text{-}entry}$) within a specified timeframe after the end of the operation phase (e.g. at least 100 years). 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 Point (see also Annex L). Any mission-related object falls within one of the following cases:

- $P_{re\text{-}entry} = 1$ for objects with permanent or periodic presence in the LEO Protected Region.
- $P_{re\text{-}entry} = 0$ for objects in non-Earth orbit (e.g. heliocentric orbits), on HEOs, or on orbits around Sun-Earth Lagrange Points, with periodic vicinity to the Earth, if a precise long-term orbit propagation does not indicate a re-entry within the specifed timeframe.
- $0 \leq P_{re\text{-}entry} \leq 1$ 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) can be used to assess accurately $P_{re\text{-}entry}$ within the specified timeframe.

D.2.2 Controlled re-entry

When planning to perform controlled re-entry for a space system, a trade-off of potential strategies is undertaken considering: risk of an uncontrolled re-entry, propulsion system design options, Delta-v for orbit and attitude control, system design requirements, launch mass (relevant also for launch vehicle identification), re-entry execution complexity, available support technology, and reliability.

A controlled re-entry operation sequence normally involves three phases:

1. Clearance of the operational orbit, e.g. by first performing small altitude decrease.
2. Perigee altitude decrease to the minimum controllable altitude by the AOCS system.
3. Final re-entry manoeuvre to target perigee altitude to allow re-entry fragments over a target re-entry area compliant with ESSB-ST-U-004 [RD4].

The controlled re-entry operation sequence is determined by taking into account several factors, including:

- attitude and orbit control modes,
- number and position of the thrusters,
- available and depleted propellant amounts,

- propulsion tank pressurization level (i.e. minimum level needed for thrust performance),
- available power supply from power generators, and energy storages,
- capability of the space system primary and secondary structures (e.g. solar array and appendages) to withstand with maximum forces and torques experienced in orbit until the final atmosphere entry,
- other constraints associated to the space system design and space environment.

A Fault-Tree Analysis (FTA) can map the failure scenarios and operational constraints to identify decision making points at each phase. An example of constraint is the need to switch to a safe mode in case of sensor failure occurring at low altitude.

D.2.2.1 Uncertainties for nominal controlled or off-nominal uncontrolled re-entry

The re-entry casualty risk analysis is 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 prior to enter the nominal case.

The uncertainties for the nominal and off-nominal cases are 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 %]

e. Thrust pitch angle: Gaussian, 3σ (e.g. 2°)

f. Atmospheric density dispersion: Uniform, ±20%

g. Drag coefficient dispersion: Uniform, mean 2,2 ± 0,55 (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, 3σ = 6 km

k. Off-nominal scenarios are identified and considered in case of spacecraft boost failure at re-entry. The following error can 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 D.2.2a. to D.2.2k. are 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.

For example, the following dispersion parameters can be considered for a controlled re-entry of a spacecraft in LEO:

a. Atmospheric density dispersion: Uniform, ±50%

b. Fragments ballistic coefficient: Uniform, ±50%

c. Lift over drag ratio dispersion: Uniform, ±50%

d. Final perigee altitude: ±10 km

e. Argument of perigee: 180°, 225°, 270°

f. Nominal Delta-v inaccuracy: 1% to 5% (depending on the argument of perigee)

g. Thrust pointing error: 1° conical

The thrust pointing error is applied to the last burn as the error on the previous burns are considered known by orbit determination and available through the operation process. The re-entry trajectory initial state vector is at 120 km geodetic altitude and a non-explosive break-up is estimated to occur at lower altitude.

In order to determine the DRA and SRA associated with a controlled re-entry, the location of the fragments under the aforementioned uncertainties are of driving importance. Typically, three fragment types are generally present: short-lived (ballistic coefficient 8 kg/m² with a lift over drag ratio of 0,1), medium-lived, and long-lived (ballistic coefficient 300 kg/m² with a lift over drag ratio of 0,1). These values coupled with a break-up event altitude are suitable to identify the re-entry footprint. For a refined computation of the DRA and SRA, coupling with a fragmentation model based on the actual space segment design is considered.

D.2.3 Initial conditions

The re-entry trajectory, when defined for a deterministic simulation, includes the following:

a. Epoch, initial orbital state vector from the end of the operation phase (beginning of the disposal phase) as per assessment in Annex A.

b. Planned disposal manoeuvres, including epoch, initial orbital state vector, target state vector, boosts magnitude (Delta-v) and direction, manoeuvred 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.

In order to obtain accurate results, which are not biased by limited, or ill-posed, assumptions typical of a deterministic simulation, a stochastic similation is performed. The approach to define the initial conditions for a stochastic simulation, which takes into account trajectory uncertainties according to the type of re-entry and orbit, is summarized in Table D-1.

Table D-1: Approach for initial conditions for stochastic simulations

Re-entry type	Orbit type	Initial Conditions	Uncertainties
Uncontrolled	**Decaying circular orbit**	Altitude: 130 km x 130 km Semi-major axis: 6501 km Eccentricity: ≤ $1 x 10^{-6}$	Argument of perigee with uniform distribution across an orbit
Uncontrolled	**Targeted circular**	Altitude: 120 km x 120 km Semi-major axis: 6491km Eccentricity: ≤ $1 x 10^{-6}$	Epoch with justified distribution (e.g. corrected Gaussian or Beta distribution) True anomaly matching the epoch
Controlled	**Targeted near circular**	Sufficient number of possible state vectors (from mission analysis)	Initial conditions with an uncertainty profile at a suitable re-entry interface (D.2.2.1)
Uncontrolled or controlled	**Highly eccentric or interplanetary**	Sufficient number of possible state vectors (from mission analysis) to identify potential impact zones from the chords of the ground-track	Initial conditions with an uncertainty profile at a suitable re-entry interface (Annex L, Annex M)

D.2.4 Approaches for fragmentation during destructive re-entry

An approtiate fragmentation model is selected in order to assess the risk associated to a destructive re-entry. Possible approaches for fragmentation phenomena occurring during destructive re-entry are:

a. Object-oriented

b. Component-oriented

c. Spacecraft-oriented approach

D.2.4.1 Object-oriented approach

So-called object-oriented re-entry survivability analysis tools (see Figure D-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 modelled assuming to be either randomly tumbling, or otherwise with a fixed attitude, if justified.

d. The ablation method for the modelled components is approximated, e.g. generally based on melting for metals, or charring for CFRPs, from the outside layer-by-layer, hence maintaining their shape type.

e. The trajectory analysis of all fragments considers translational motion only with fixed attitude (three degrees of freedom).

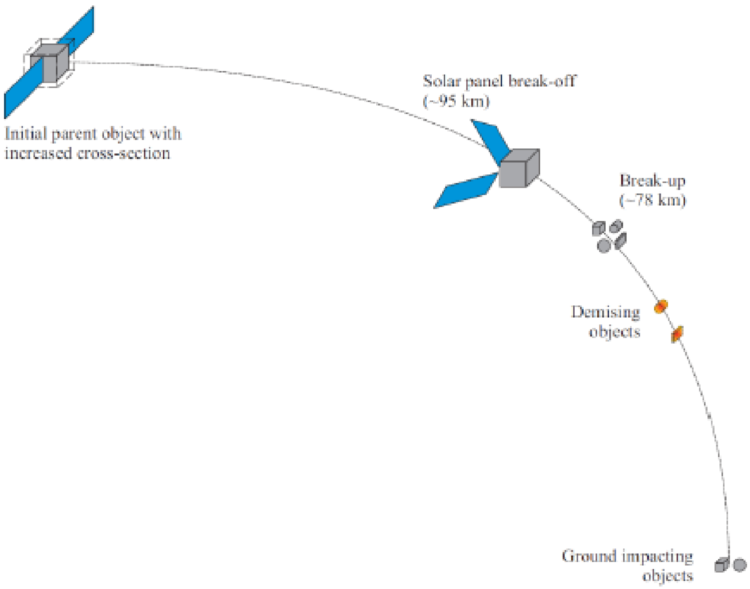


Figure D-1: Object-oriented tool concept

Analysis with an object-oriented tool can be:

a. Deterministic

b. Stochastic

In a deterministic analysis, the model of the space system 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.

In a stochastic analysis, Monte Carlo simulations are performed. The accuracy of the results is improved only if justified assumptions are taken in the range and probability distribution of the relevant variables.

The assumption of the release (separation) point at a fixed altitude can be based on experimental evidence (e.g. test, re-entry observation), otherwise a conventional assumption (78 km altitude) is considered. It is remarked that higher release (separation) altitude assumptions can lead to:

- Lower quantity of re-entry surviving fragments, which can be overly optimistic, and, therefore, not conservative nor suitable without justification when determing the risk for uncontrolled re-entry.
- Larger size of the footprint of re-entry survivig fragments (i.e. larger distance between the toe and heel fragments), which can be conservative when the fragments are actually not demising, and, therefore, suitable when determining the re-entry notification areas for controlled re-entry.

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.; Hyperschall Technologie Göttingen)

b. ASTOS/DARS (ESA; Astos Solutions GmbH)

c. DAS (NASA)

d. ORSAT (NASA)

e. DEBRISK (CNES)

D.2.4.2 Component-oriented approach

The component-oriented approach is conceived as an extension of the object-oriented approach. In order to improve the space system model for re-entry, two types of relationships can be considered between objects:

a. "Connected-to", i.e. two peer objects share a common surface through which heat exchanges are considered until the release (separation) point, occurring when one of the two objects is fully melted (conservative and recommeded fragmentation trigger criterion, as observed for metals), or reaches its melting (transition phase) temperature (if justified).

b. "Included-in", i.e. a parent-child relationship is assumed between two objects, with no heat exchange between the two objects until the release (separation) point, occurring when the parent object is fully melted (conservative and recommeded fragmentation trigger criterion, as observed for metals), or reaches its melting (transition phase) temperature (if justified).

The combination of primitives enables probabilistic based approach, which makes component-oriented based tools suitable for studying the influence of uncertaintites on the re-entry casualty risk and better quantify a confidence level.

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

a. DRAMA/SESAM/SERAM, from the release 3 (ESA; DEIMOS Space S.L.U.; Hyperschall Technologie Göttingen; Belstead Research Limited)

b. SAM (Belstead Research)

D.2.4.3 Spacecraft-oriented approach

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 D-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 can be used as input to probabilitic methods as a calibration point and be useful for "design for demise" studies. 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 (HTG) GmbH) is a tool based on the spacecraft-oriented approach.

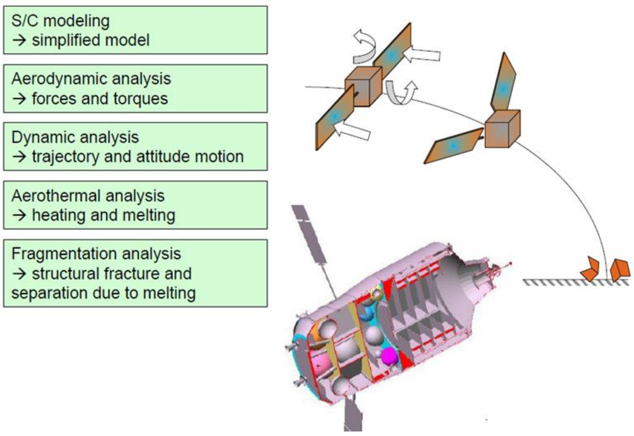


Figure D-2: Spacecraft-oriented tool concept

The re-entry model of the spacecraft –oriented tool is 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 includes:

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. Young's modulus, Poisson's ratio, ultimate tensile stress

d. Justification for any assumption or simplification in the model with respect to the real structure.

D.2.5 Earth population density

Earth population density data is used to assess the casualty expectancy and risk. 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 calculated.

The population density is estimated in order to reflect in the best possible way the situation at the expected epoch of the re-entry, taking into account the population growth trend.

Earth population density data are 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° x 0,25°.

Once the Earth population density is estimated, it is possible to derive latitude dependent population density values $\rho_p(\phi, \Delta, \phi)$, i.e. the density summed up along a latitude band $\Delta\phi$ (e.g. Figure D-3) .

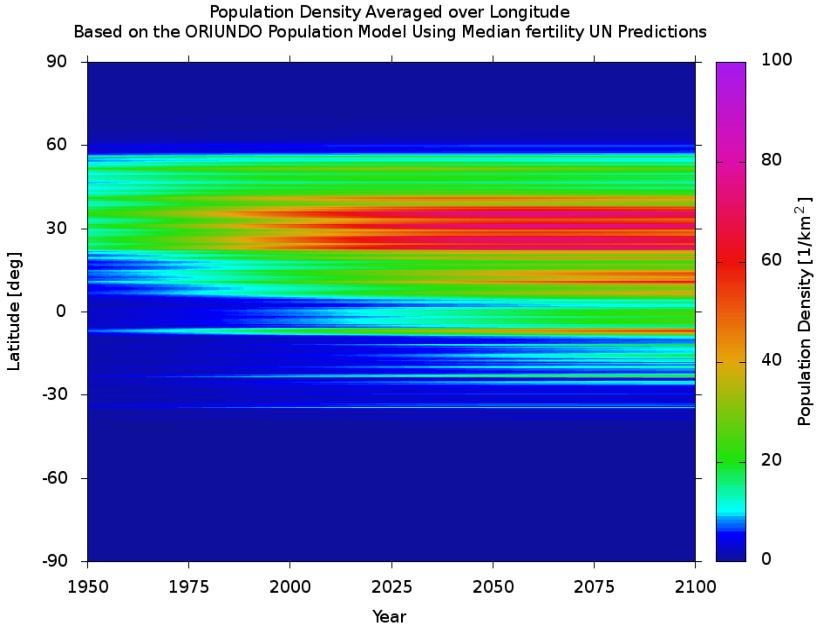


Figure D-3: Earth population density, latitude-dependent, using median UN predictions for the future growth rate (Rev. 2010)

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

a. UN Population data

b. Gridded Population of the World (GPW)

c. DRAMA/SERAM implemented model based on GPW

D.2.6 Ground impact probability for uncontrolled re-entry

For a given fragment, the ground impact probability represents the probability that the fragment impacts in a certain location. The calculation of the ground impact probability depends on the type of re-entry, since uncontrolled re-entry only allows a rough estimation of fragment impact location in terms of latitude range, while controlled re-entry allows to predict more precisely the impact location in terms of latitude and longitude.

D.2.6.1 Ground impact probability (1D)

When the average population density, i.e. mono-dimensional (1D) population density, is used to determine the casualty risk for an uncontrolled re-entry, the ground impact probability is 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 is derived by calculating the weighted sum of longitudinal averaged population density distribution over the whole latitude range. The ground impact probability distribution $P_i(i, \phi, \Delta, \phi, \omega)$ is, then, used as follows to determine the average population density ρ_p:

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

where ρ is the population density along the latitude, ϕ is the latitude, $\Delta\phi$ 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 $P_i(i, \phi, \Delta, \phi, \omega)$ is such that:

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

Depending on the eccentricity of the re-entry orbit, appropriate formulations of the ground impact probability is used (see section D.2.6.2 and D.2.6.3).

D.2.6.2 Ground impact probability for circular re-entry orbits

For re-entry from near circular orbits, an analytical solution is available, which provides an approximation of the impact probability depending on the latitude range. An ESA study provided the impact probability that an uncontrolled re-entry from a near circular orbit of inclination i in the interval $(0, \pi)$ occurs in a latitude band of width $\Delta\phi$, centred at latitude ϕ in the interval $(-\pi/2, \pi/2)$, excluding effects of J2 Earth gravity parameter, is (see Figure D-4):

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

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

NASA provided an alternative equivalent solution, which is valid for $\phi \le i$ and $0 < i < \le \pi/2$:

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

In any case, a 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.

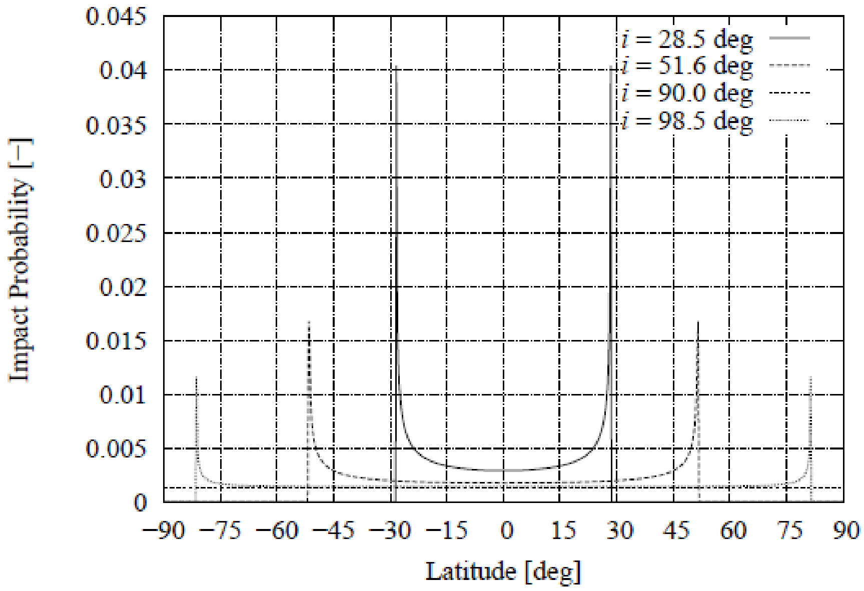


Figure D-4: Ground impact probability as function of latitude (approximation without effects of J2 Earth gravity parameter)

D.2.6.3 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.

Due to the stabilizing effect of Earth gravity induced perturbations, it is possible to estimate the geographic latitude and the geodetic longitude with a given impact probability. The impact probability for the uncontrolled re-entry from highly eccentric orbits can initially be considered as 1D, with uniform longitude distribution and limited latitude range. Closer to the re-entry epoch, the uncertainities on propagation decreases and the longitude can be better estimated, so that the probability is more precise in location and can be then considered as 2D. Figure D-5 shows an example of the re-entry prediction for HEO mission, with visible latitude band and limited longitude, performed 7 year previous the targeted re-entry epoch.

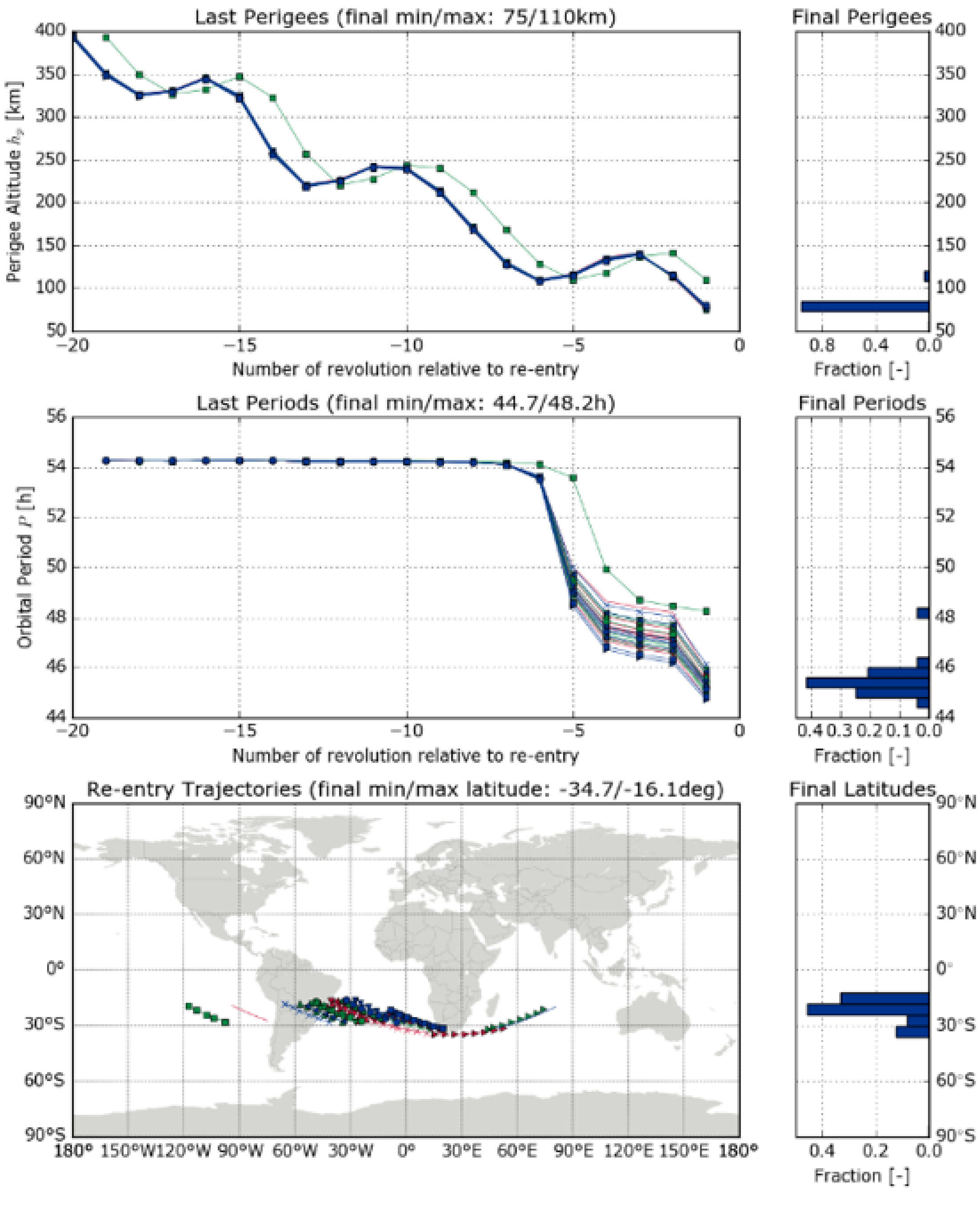


Figure D-5: Example of re-entry analysis for HEO mission with latitude band and delimited longitude

D.2.6.4 Ground impact probability (2D) for controlled re-entry

For controlled re-entry the latitude and longitude of the impact location for each fragment is predicted, i.e. bi-dimensional (2D) impact grid. The impact probability of a given fragment is theoretically 1 at its impact location and 0 in all the other locations, if no error exists on the trajectory. However, for quantitative analyses of the fragments distribution, a stochastic approach (Monte-Carlo) with varying input parameters accounting for the uncertainties on the impact locations prediction, is commonly used. The same consideration applies as well to re-entry events from SEL and interplanetary trajectories.

D.2.7 Re-entry trajectory

The re-entry trajectory is 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 operation phase to ground impact.

b. Ground track of the re-entry trajectory.

D.2.8 Explosion probability assessment

The effect of explosions due to potential residual fuel, oxidizer, and pressurant in the tanks of a re-entering space system, when local temperature increases during the re-entry trajectory, is assessed.

The effect can be relevant especially for space systems performing controlled re-entry, when the tanks are not completely empty as passivation is not performed in orbit. End of mission passivation and minimisation of residuals clearly allow to minimise the effect. If necessary, tanks depletion measures prior to re-entry are considered. A rationale is expected to justify if the risk associated to residual fuel, oxidizer, or pressurant explosions during re-entry whether is considered negligible or already included in the worst-case re-entry casualty risk analysis.

Sensitivity analysis can be performed with the ESA tool DRAMA in order to simulate the effect of spacecraft fragmentation events at different altitudes and the consequent on-ground footprint of the surviving fragments.

An example of an assessment of the effects of explosions during a re-entry was performed for ATV, which is described in detail in "MPA/2005/190/DB - Computational Analysis of ATV Re-entry Flow and Explosion Assessment" (2006). The approach is summarized as follows:

a. The predicted re-entry trajectory is used to set the boundary conditions for a CFD simulation (with the assumptions of ballistic re-entry, and Standard US 76 atmosphere model, with nominal and ±20% density), including:

   1. Pressure
   2. Temperature
   3. Density
   4. Velocity
   5. G-load
   6. Convective heat flux
   7. Radiative heat flux

b. A CFD simulation is performed to determine the flow-field around the spacecraft at different trajectory locations in order to predict the heat-flux distribution over the surface.

c. A fissure is placed on the spacecraft surface where the maximum heat-flux occurs and a CFD simulation is performed to determine the flow-field external and internal to the spacecraft for a break-up altitude (derived from spacecraft fragmentation, due to thermo-mechanical loads, determined with the tool SCARAB) and a higher altitude. The output of the internal and external CFD simulation included:

1. Maximum velocity
2. Maximum temperature
3. Maximum pressure
4. Maximum partial pressure of diatomic and atomic oxygen
5. Maximum heat-flux

An approximation was taken by assuming the trajectory as a succession of steady states (neglecting dynamic aspect of the re-entry and changes in atmospheric conditions along the trajectory). The approximation is valid only if the time scale of the spacecraft filling at the fissure creation by the external gas is negligible in comparison with the time scale of change in the external conditions. An assessment of the time scale of the filling is performed using a set of equations for isentropic flow of perfect gas, assuming the fissure acts as the throat of a Laval nozzle.

d. An explosion analysis is performed, investigating the critical conditions for the ignition of the on-board propellants. The ignition conditions of the chemical reactions can be derived from available analytical formulas, or experimental data, of the minimum auto-ignition pressure depending on composition and temperature.

e. The probability of explosion is computed by coupling the explosion analysis with the flow-field computations, under the assumption of leakage of residual on-board stored propellant, and comparing the magnitude of the partial pressure with the minimum ignition pressure. If the computed partial pressure is higher than the minimum ignition pressure of the propellants, an explosion is likely to occur.

D.2.9 Characteristics of surviving fragments

The physical characteristics of all expected fragment surviving re-entry are 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 D.2.12)

i. Fragment Floating or non-floating capability over water or oceans

j. Toxicity of the fragment.

D.2.10 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), is defined as (see Figure D-6):

$$A_{C,k} = \left[\sqrt{A_{i,k}} + \sqrt{A_h}\right]^2 \qquad \text{[D-6]}$$

where:

$A_{i,k}$ average projected area of the k-th fragment surviving the re-entry (determined as arithmetic mean for non-convex objects, or analytically otherwise)

A_h cross-section of a human, which is conventionally defined equal to 0,36 m^2

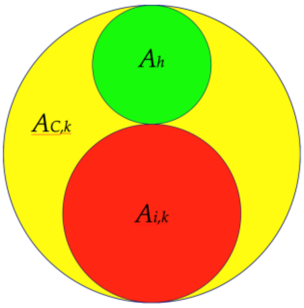


Figure D-6: Casualty area definition

D.2.11 Total casualty area

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

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

D.2.12 Casualty risk

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 is lower than a given value, it is not necessary to project the full risk profile. The Markov's Inequality gives an upper limit for the probability distribution:

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

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

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

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 D.2.6). 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 is 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 is weighted with the resulting casualty risk. A functional Fault Tree can be identified 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 is compared to the requirement of 1:10000 and fulfilled for any disposal strategy (controlled or uncontrolled re-entry).

The re-entry casualty risk is computed in the practice through the casualty expectation approximating the casualty probability. The re-entry casualty risk is computed as follows, depending on controlled or uncontrolled re-entry case:

a. The re-entry casualty risk 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 $P_i(i, \phi, \Delta\phi)$ or $P_i(i, \phi, \Delta\phi, \omega)$ depending on the orbit eccentricity (section D.2.7), which is a function of the orbit inclination i, the latitude step size $\Delta\phi$, and the argument of perigee at the epoch of atmospheric capture ω:

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

b. The re-entry casualty risk 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):

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

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

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

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

d. The re-entry casualty risk for a space system, which is not nominally planned 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 risk for an uncontrolled re-entry and the re-entry probability ($P_{re-entry}$):

$$E_{C,\mathrm{prob},re-entry} = E_{C,unc} P_{re-entry} = A_C \rho_v(i, \varphi, \Delta\varphi) P_{re-entry} \quad \text{[D-13]}$$

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

$$E_{C,comb} = \mathrm{E}_{C,nom} R_{nom} + \sum_{r=1}^{Z} E_{C,non-nom,k}\, P_{non-nom,k} \quad \text{[D-14]}$$

where $E_{c,nom}$ is the casualty risk for the nominal controlled re-entry, R_{nom} is the reliability to perform the nominal controlled re-entry, $P_{non-nom,,k}$ is the probability to have the r-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), $E_{C,non-nom,r}$ is the casualty risk associated to the r-th non-nominal case, and Z is the number of non-nominal re-entry scenarios.

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

D.2.13 Declared Re-entry Area (DRA) and Safety Re-entry Area (SRA)

The Declared Re-entry Area (DRA) and the Safety Re-entry Area (SRA) are computed following several simulation runs (Monte Carlo), which are based on the dispersions of the relevant variables to cover all uncertainties of the model (see section D.2.2), where the amount of runs yield stable confidence intervals (see Figure D-7):

a. The Declared Re-entry Area (DRA) delimits the area where the debris are enclosed with a probability of 99% given the delivery accuracy.

b. The Safety Re-entry Area (SRA) delimits the area where the debris are enclosed with a probability of 99,999% given the delivery accuracy.

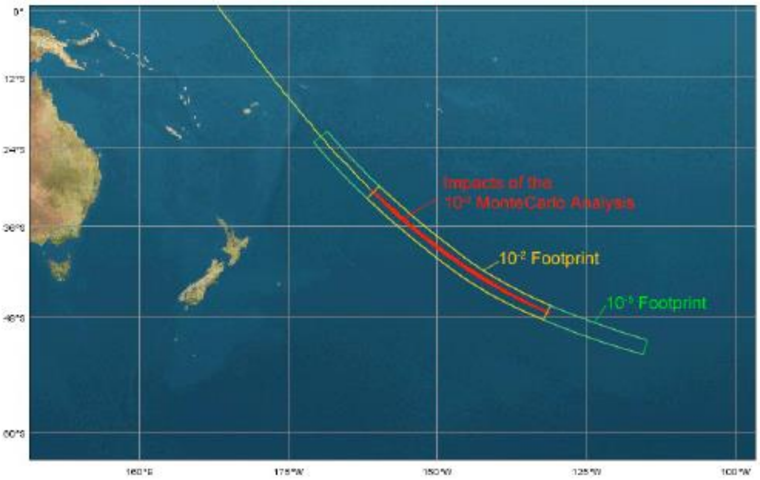


Figure D-7: Example of DRA (10^{-2} Footprint) and SRA (10^{-5} Footprint)

The definition of the DRA and SRA, which to the first order include the footprint estimation, is based on the probability of having potentially surviving fragments impacting in a specific zone, but the confidence level attached to these values is not given. A confidence level of 90 or 95% is practically used, given that classical stochastic (Monte-Carlo) simulations contain millions of samples.

Methodologies making use of surrogate models, or parametric sampling, of the input distributions can be used to quantify the extremes of the impact location, which are treated as a distribution at the noted confidence level requiring a lower number of samples than in a classical Monte-Carlo simulation. The uncertainty distributions, which are associated to a re-entry event, are driving the results, and, therefore, it is fundamental that their assumptions are correctly justified.

D.2.14 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 D-8).

This can be used in conjunction with a population density model based on the Gridded Population of the World (GPW) v4 for the year 2020, 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 D-9).

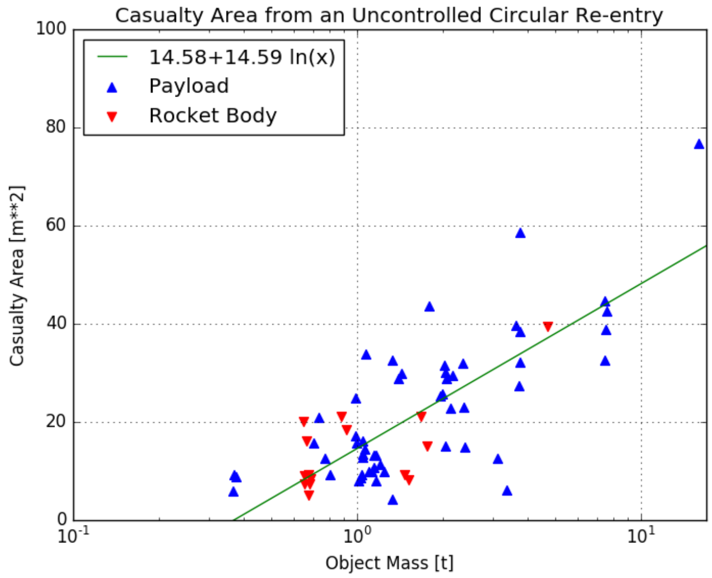


Figure D-8: Fit of historical re-entry assessment for the casualty area as a function object mass

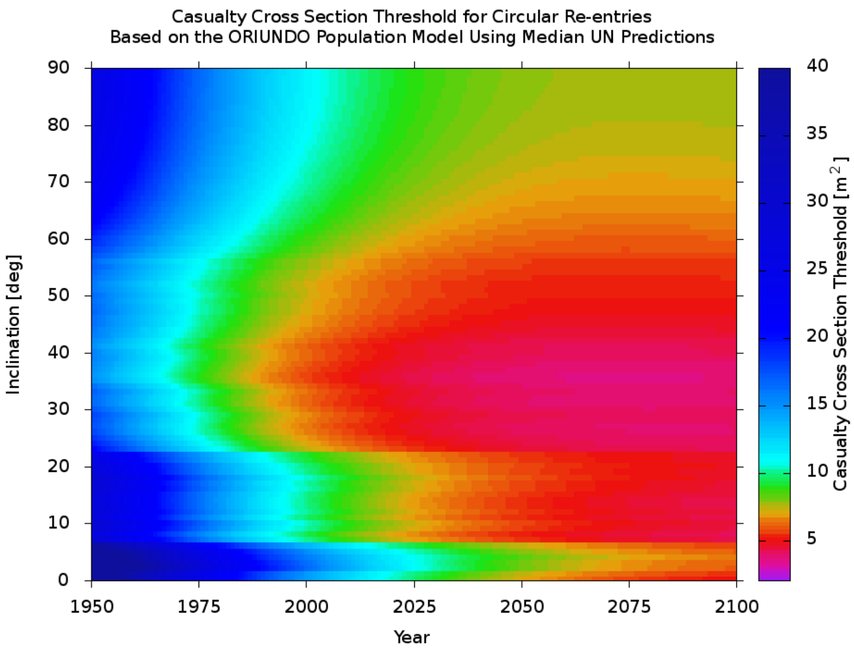


Figure D-9: Casualty cross-section threshold for a 10^{-4} casualty risk and uncontrolled re-entries from circular orbits, using median UN predictions for the population growth (Rev. 2010)

D.2.15 Analysis tool(s)

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, pending a priori discussion and agreement of the selected tool with ESA.

D.2.16 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. 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 to be taken at mission SRR on whether or not a controlled re-entry is part of the mission. If the decision is to plan for an uncontrolled re-entry, "design for demise" measures can be used to mitigate the re-entry casualty risk, which are, then, implemented and verified at PDR.

Accordingly, a multi-step approach is used (see Table D-2).

Table D-2: Re-entry casualty risk analysis process

<table>
<tr><th>Review / Phase</th><th colspan="3">Action</th></tr>
<tr><td rowspan="2">Phase 0</td><td colspan="3">1. First assessment of $P_{re\text{-}entry}$ for the mission orbit (D.2.1).
2. First assessment of E_C for an uncontrolled re-entry (two options):
a. Rough order of magnitude assessment of $E_C = f(inclination,\ Epoch,\ dry\ mass)$ (D.2.14) (for circularised LEO re-entry orbits).
b. First assessment of E_C (D.2.12) using likely re-entry conditions (D.2.3) and a first space system model using object-oriented tools (for all orbits) (D.2.4).</td></tr>
<tr><td colspan="3">Result: First indication of $P_{re\text{-}entry} \cdot E_C$ (controlled re-entry to be considered if $P_{re\text{-}entry} \cdot E_C > 10^{-4}$)</td></tr>
<tr><td rowspan="4">PRR
Phase A</td><td colspan="3">1. Refined assessment of $P_{re\text{-}entry}$ 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 (D.2.4)</td></tr>
<tr><td colspan="3">Result: Preliminary decision on the re-entry approach (controlled or uncontrolled)</td></tr>
<tr><td colspan="2">Uncontrolled re-entry</td><td>Controlled re-entry</td></tr>
<tr><td>Circular re-entry (D.2.6.2)</td><td>Eccentric Re-entry (D.2.6.3)</td><td>First sizing of debris fall-out footprint (D.2.13)</td></tr>
<tr><td rowspan="4">SRR
Phase B</td><td colspan="3">1. Final assessment of $P_{re\text{-}entry}$ for the mission orbit (D.2.2).
2. Final assessment of a space system model in object-oriented tools, including uncertainty quantification (D.2.4.3)
3. Establishment of a model in a spacecraft-oriented tools (for confirmation) (D.2.5), determination of explosion likelihood and consideration of these effects (D.2.9), and conclusion on passivation measures.</td></tr>
<tr><td colspan="2">Uncontrolled re-entry</td><td>Controlled re-entry</td></tr>
<tr><td>Circular re-entry
(spacecraft-oriented tool)</td><td>Eccentric Re-entry
(spacecraft-oriented tool)</td><td>1. Preliminary assessment of SRA and DRA.
2. Preliminary assessment of reliability figures and failure modes.
3. First assessment of $E_{C,comb}$ (D.2.14).</td></tr>
<tr><td colspan="3">Result: Final decision on the re-entry approach (controlled or uncontrolled)</td></tr>
</table>

<table>
<tr><th>Review / Phase</th><th colspan="5">Action</th></tr>
<tr><td rowspan="3">PDR
Phase B</td><td colspan="5">1. Refinement of the model for the spacecraft-oriented tool</td></tr>
<tr><td colspan="2">Uncontrolled re-entry</td><td colspan="3">Controlled re-entry</td></tr>
<tr><td>Circular re-entry
If $P_{re\text{-}entry} \cdot E_C > 10^{-4}$, mitigation options:
1. Implementation and verification of “design for demise” measures;
2. Passivation to prevent explosive break-ups.</td><td>Eccentric Re-entry
If $P_{re\text{-}entry} \cdot E_C > 10^{-4}$, 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.</td><td>$E_{C,unc,fail}$
Mitigation options:
1. Improvement of system reliability;
2. Implementation and verification of “design for demise” measures;
3. Passivation to prevent explosive break-ups.</td><td>$E_{C,nom}R_{nom}$
Mitigation options:
1. Lower re-entry perigee;
2. Alternate target area;
3. Passivation to prevent explosive break-ups.</td><td>$E_{C,off\text{-}nom}P_{off\text{-}nom}$
Mitigation options:
1. Improvement of system reliability;
2. Passivation to prevent explosive break-ups.</td></tr>
<tr><td rowspan="2">CDR
Phase C</td><td colspan="2">Uncontrolled re-entry</td><td colspan="3">Controlled re-entry</td></tr>
<tr><td colspan="2">As before</td><td colspan="3">As before</td></tr>
<tr><td rowspan="2">FAR
Phase D</td><td colspan="2">Uncontrolled re-entry</td><td colspan="3">Controlled re-entry</td></tr>
<tr><td colspan="2">As before</td><td colspan="3">As before</td></tr>
<tr><td rowspan="2">Mission Change</td><td colspan="2">Uncontrolled re-entry</td><td colspan="3">Controlled re-entry</td></tr>
<tr><td>Circular re-entry
1. Verify that changes to re-entry epoch (on-ground population growth) do not lead to a violation the requirement.</td><td>Eccentric re-entry
1. Verify that changes to re-entry epoch do not lead to re-entry latitude with higher population densities.</td><td>$E_{C,unc,fail}$
1. Verify that P_f has not reached critical levels.</td><td>$E_{C,nom}R_{nom}$
1. Verify that R_{nom} has not reached critical levels.</td><td>$E_{C,off\text{-}nom}P_{off\text{-}nom}$
1. Verify the $P_{off\text{-}nom}$ has not reached critical levels.</td></tr>
</table>

<table>
<tr><th>Review / Phase</th><th colspan="4">Action</th></tr>
<tr><td rowspan="2">EOL</td><td>Uncontrolled re-entry</td><td colspan="3">Controlled re-entry</td></tr>
<tr><td>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.</td><td>Uncontrolled re-entry
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.</td><td>Controlled re-entry
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.</td><td>Degraded Controlled re-entry
1. Same actions as for controlled re-entry.</td></tr>
</table>

Annex E
Propellant gauging methods

Possible methods to gauge the propellant mass for any phase during the mission are summarized in Table E-1, which is derived 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 ISO 23339:2010 [RD13].

Table E-1: Examples of estimation methods

Method	Measurement Principle	Advantages	Disadvantages
pVT	Measurement of the tank temperature and pressure, calculation of the tank ullage volume, and thereby the remaining propellant mass by applying the gas law	No additional equipment and low cost; frequently used on spacecraft	Decreasing accuracy towards EOL; low accuracy with conventional pressure transducer
Thermal Knocking	Heating of the propellant tank and measurement of its thermal response related to the propellant load	No additional equipment	Low accuracy; high calibration efforts; long operational gauging times
Gas Injection	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; propulsion system modifications needed; high accuracy pressure transducer needed; high calibration effort; high costs
Liquid Levelling	Measurement of liquid level in tanks when propellants are settled due to spin acceleration (e.g. spin-stabilized spacecraft), or during thruster operations	Simple system; Very high or improvable accuracy	Limited to spin-stabilized spacecraft, or during thruster firings with minimum thrust level and duration

Method	Measurement Principle	Advantages	Disadvantages
Bookkeeping	Calculation and recording of propellant consumption and manoeuvre data (e.g. pulse duration, pulse mode, thruster temperature) during each manoeuvre, with support of on-ground calibration test data of individual thrusters	Simple system with no additional equipment needed; frequently used on spacecraft	In-flight calibration needed; operational effort on ground
Flometer	Integration of mass flow rate measurements during operations	High accuracy sensor	Sensor still to be developed; low accuracy for pulse firings; sensitivity to gravity vector, thermal environment, and small gas bubbles formation

Annex F
Passivation methods

Table F-1 summarizes, although not exhaustively for all cases, passivation measures that can be used for the most common components storing energy.

Table F-1: Passivation measures

Subsystem	Unit	Passivation Measures	Critical Issues and Remarks
GNC	Attitude Control Sensors and Actuator	1. Disconnection from power supply sources	• A dedicated GNC mode can be implied.
GNC	Cold Gas Thruster	1. Depletion of gas supply source	• A dedicated GNC mode can be implied.
GNC	Control Moment Gyro	1. Disconnection from power supply sources 2. De-spin/stop rotating parts	• Mobile parts can lead to mechanical ruptures due to fatigue. • A dedicated GNC mode can be implied.
GNC	Reaction or Momentum Wheel	1. Disconnection from power supply sources 2. De-spin/stop rotating parts	• Mobile parts can lead to mechanical ruptures due to fatigue. • A dedicated GNC mode can be implied.
Mechanism	Any rotating or movable part	1. Fix and block the relative movements	• Mobile parts can lead to mechanical ruptures due to fatigue.
Mechanism	Electro-explosive device	1. De-activation, if not useful any more 2. Disconnection from power supply sources	
Mechanism	Pyrotechnic device	1. De-activation, if not useful any more 2. Disconnection from power supply sources	
Power	Battery	1. Self-protection 2. Discharge 3. Disconnection from any charging source (e.g. solar array)	• Discharging and keeping the battery in a permanently discharged status is the best approach. • Disconnection from the solar array can be sufficient since it leads to a complete battery discharge. Battery discharge can initially occur via the power bus loads, and, then, via the leakage current of control electronics connected to the battery, or with a permanent electrical drain to prevent recharging. • The preferred passivation device is robust enough to cope with ageing and the harsh environment at EOL (e.g. loss of temperature control, radiations) to avoid losing passivation after some time. • When not possible to eliminate all energy or disconnect the batteries, a risk assessment is performed to demonstrate that the design/operational solution ensures that the likelihhod of debris generation is very low.

Subsystem	Unit	Passivation Measures	Critical Issues and Remarks
			• Small batteries (e.g. for cubesats) can be protected in containers to prevent generation of debris in case of failure. An assessment is performed to demonstate that the energy potentially released by the small battery is not sufficient to generate a space system break-up. • The assessment of the risk of debris generation due to catastrophic battery failure takes into account: 1. Available information on the battery cell procurement to check if they come from well-reputed supplier. 2. Available information on qualification and lot acceptance tests, or abuse tests (e.g. in extreme charge and thermal conditions expected in the space environment), performed on the cells (e.g. possible certifications or test campaigns). 3. Assessment of the safety protection devices of the battery cells, e.g. Current Interrupt Devices (CID), Positive Temperature Coefficient thermistors (PTC), circuit breakers, vent valves, leak-before-burst design, etc.. 4. Adequacy of measures foreseen by design for the end of mission passivation to withstand the thermal/radiation environment for the entire on-orbit duration (e.g. in LEO, 25 yrs before re-entry). 5. Assessment of the worst-case (max) residual energy stored after end of life and check if it is likely to create a hazard (e.g. only few cells for small spacecraft at low altitude cannot be considered a critical hazard). 6. Assessment of the worst-case (max) temperature of the battery cells after end of life and check if the battery cells can withstand it (i.e. not resulting in risk of thermal runaway). 7. Determination of the criticality of the worst-case scenarios in terms of debris generation effects with respect to the type of orbit. For example, debris generated at 300 km do not have the same impact on long-term sustainability of debris generated at 800 km.
Power	Fuel Cell	1. Self-protection 2. Discharge 3. Disconnection from any charging source (e.g. solar array) 4. Depressurization of the cells (if needed)	• See Battery.
Power	Power Conditioning and Distribution Unit (PCDU)	1. Isolate power storages from power generators 2. Switch-off all possible circuits	• The PCDU can include a function to short-circuit the spacecraft bus to derive battery discharge.
Power	Solar Array	1. Disconnection from power bus or batteries 2. Short-circuit	• See Battery.

Subsystem	Unit	Passivation Measures	Critical Issues and Remarks
Propulsion	Pipeline	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	• Hazards from venting include: uncontrolled accelerations, attitude, or orbit changes, increase of the likelihood of collision with other objects, fragmentation, mixing of fuel and oxydizer, blockage due to freezing propellants or venting fluids. • Residual propellant can be scavenged without generating solid particles greater than 10 μm. • The risk of explosion of pipelines, which are not connected to pressure vessels with high stored energy, can be demonstrated to be minor if high design safety factors and low volume are involved.
Propulsion	Pressurant Tank	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	• Hazards from venting and depressurization include: uncontrolled accelerations, attitude, or orbit changes, increase of the likelihood of collision with other objects, fragmentation, structure embrittlement. • Risk of explosions due to over-pressure or over-temperature can be mitigated by using relief valve mechanisms. • In case residual gas cannot be drained from pressure vessels, safe conditions include: no burst in case of penetrating impacts (demonstration via HVI tests and analysis); vessel design and thermal protection able to inhibit pressure build-up (e.g. relief valve mechanisms). • Hazards from venting, depletion burn, and depressurization include: uncontrolled accelerations, attitude, or orbit changes, increase of the likelihood of collision with other objects, fragmentation, structure embrittlement, spin-up of the vehicle, inadvertent mixing of vented hypergolic propellants.
Propulsion	Propellant Tank	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	• Leak-before-burst tank designs, although beneficial, are not sufficient to prevent explosions in all scenarios (depressurization is still needed). • Depressurization of pressure vessels with pressure-relief mechanisms is not an issue 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 do not to 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	1. Switch-off the telemetry transmitter with monitoring RF signal	
Thermal Control	Heat Pipe	1. Demonstration of low probability of rupture	

The validity and justifiability of a passivation measure (i.e. by design or operational control measure) for each unit storing energy can be estimated through a risk assessment. Risk assessment is used to evaluate qualitatively or quantitatively the severity and probability of debris generation associated to the possible failure, inefficiency, or limitations of the passivation measure. The risk assessment can make use of a criticality ranking matrix (as in ECSS-Q-ST-30-02C, clause 5.3), or a qualitative hazard risk matrix as in Table F-2. In absence or insufficience of passivation measures, on-orbit break-up with debris release is considered a catastrophic hazard event as it involves pollution of the space environment and increase of probability of debris collision with other uninhabited or inhabited space assets and debris. The hazard event is particularly relevant if it occurs directly in Earth orbit, or leads debris to interfere with Earth orbits.

Table F-2: Hazard risk matrix (example)

Hazard severity	Hazard likelihood				
	Very low	Low	Moderate	High	Very high
Catastrophic					
Critical					
Major (severe)					
Moderate					
Minor (negligible)					

Legend		
Acceptable with no actions	Acceptable with rationale	Not acceptable – Design improvement or RFW

The hazard likelihood (P) can be defined, qualitatively or quantitatively, as, e.g.:

1. Very low, e.g.: event estremely remote to happen; $P \leq 10^{-5}$;
2. Low, e.g.: event not expected to happen; $10^{-5} < P \leq 10^{-4}$;
3. Moderate, e.g.: event not likely to happen; $10^{-4} < P \leq 10^{-3}$;
4. High, e.g.: event likely to happen; $10^{-3} < P \leq 5{\cdot}10^{-3}$;
5. Very high, e.g.: event very likely to happen; $P > 5{\cdot}10^{-3}$.

The hazard severity can be defined as, e.g.:

1. Minor (negligible), e.g.: no or minor damages to the space system; no debris release in orbit;
2. Moderate, e.g.: moderate damages to the space system; no debris release in orbit;
3. Major (severe), e.g.: significant damages to the space system involving performance degradation of the hazard control function implementation (e.g. degradation of passivation function); possible debris released in orbit not interfering with Earth orbits, planetary bodies, uninhabited or inhabited space assets;
4. Critical, e.g.: significant damages to the space system involving loss of the hazard control function implementation (e.g. loss of passivation function); possible debris released in orbit not interfering with Earth orbits, planetary bodies, uninhabited or inhabited space assets;
5. Catastrophic, e.g.: destructive damages to the space system involving debris release in orbit, whose trajectory evolution interfere with Earth orbits, can cause collision with uninhabited or inhabited space assets, or can result in violation of a planetary body surface (when planetary protection requirements are applicable).

The risk assessment compilation (severity and likelihood) for a hazard with respect to an adopted passivation measure takes into account:

- ESA Technical Authority for Space Debris Mitigation recommendations
- Subject Matter Experts judgements

- Project assessments (e.g. assessment of the probability of explosion/burst given the space environment conditions and design qualification data)
- ESA Alerts applicable to the perimeter of the hazard
- State-of-the-Art knowhow available in ESA

Annex G
Disposal reliability

G.1 Objectives

This Annex describes guidelines for the assessment of the probability to successfully perform the disposal at the end of life of a space system, further on referred also as “disposal reliability”.

G.2 Methodology

G.2.1 General

A successful disposal of a space system can be assured by performing the following assessment of the probability of successful disposal:

a. During the development phase, to ensure system and operation plan compliance with design-to requirements.

b. During the operation phase, to monitor and maintain compliance with the defined disposal reliability requirements.

The principal contributor to the probability of successful disposal is the reliability of the on-board systems to perform successfully the functionality for disposal with proven performance level. Other significant aspects for successful disposal include:

a. Control of break-up due to internal explosion or burst.

b. Control of break-up due to collision with other space systems, debris and meteoroids.

c. Availability of sufficient consumables, mostly propellant, on board for disposal operations.

d. Capability of the operator to monitor the space system health status and plan and execute correctly all disposal maneuvers.

Margins and uncertainties are 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).

G.2.2 Disposal reliability assessment during the development phase

The probability of successful disposal is the unconditional probability that the space system is capable to complete the disposal, which depends on the time of execution and termination of the disposal and the space system elements (units and functions) involved.

The probability of successful disposal involves as main contributors the reliability of:

a. All the elements of the space system which are used to successfully perform the disposal with functionality at a proven performance level, i.e. probability of the “must-work” items to successfully complete the disposal.

b. All the elements of the space system which are not used to successfully perform the disposal, but can prevent nominal performance of the “must-work” in case of failure, i.e. “must-not-work” items which can prevent “must-work” items to successfully perform the disposal, e.g. in case of propagation of short-circuits, generation of hot spots, or excessive electromagnetic interference.

The model for the assessment of the probability of successful disposal is a self-standing probability model and not simply a sub-set of a mission reliability model. Therefore, when performing the assessment of the probability of successful disposal, possible waivers affecting the reliability of “must-work” and “must-not-work” flight hardware affecting the disposal capability are relevant.

For successful disposal, it is important also to assess and control hazards from events, which can prevent successful disposal, including:

a. Internal explosions, e.g. depending on the quality and ageing of components and units, and identified in FMECA or specific assessments.

b. Collisions with other space objects likely to cause break-up or irrecoverable failure (Annex C).

It is normally desiderable that the space system design follows best practice design rules, e.g. being compliant with the applicable ECSS standards with respect to tank pressurization, thermal design propellant vapour segregation, battery charge/discharge control, space debris and meteoroids protection, such as to ensure that the contribution to the break-up probability can be negligible. However, reliability predictions for the design of space systems can be affected by limitation in the models and data availability (e.g. obsolete data sets, poor data on in-space behaviour of new developments, use of Commercial Off-the-Shelf (COTS) items from non direct space business). In order to investigate and overcome the current limitations in reliability assessment methods, ESA is leading development activities to define more realistic reliability prediction models.

G.2.3 Disposal reliability on-orbit assessment

The assessment of the actual disposal reliability of the space system is performed during the mission in order to monitor and maintain compliance with the defined disposal reliability requirement, because:

a. Reliability predictions, performed during the development phase, cannot cover systematic faults that were not detected prior to the launch and can evolve into system failures once activated under the actual on-orbit operational and environmental conditions. 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 disposal unreliability.

b. A space system can experience a random failure on equipment for disposal operations during its mission. Loss of redundancy on items used to perform disposal operations is reflected in an update of the disposal reliability estimation.

c. The as-designed disposal reliability prediction from the development phase is typically accounting for worst-case environmental and operational conditions. In practice the actual conditions experienced by the space segment on-orbit differ from those assumed during development. While environmental conditions are typically less stressing on-orbit than assumed for the development phase reliability model, operational conditions can be more demanding. Examples are an increase in usage of demand-based equipment (e.g. valve cycles) or operating an equipment in warm redundancy rather than the assumed cold redundant scheme. Where on-orbit conditions are more demanding than originally assumed, the disposal reliability prediction is updated to account for it. Less demanding on-orbit conditions than assumed during the development phase do not mandatorily need be considered in an update to the disposal reliability prediction because disregarding them simply leads to a conservative estimate. However, they can be considered and provide a valuable contribution to the rationale for a potential extension of the space segments operation beyond its nominal lifetime.

d. Monitoring the performance of Life Limited Items (LLI) during the operation phase is important 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.

e. Monitoring the health of a space system is important to identify unanticipated degradation faster than expected. This can be either an early loss of redundancy or degradation in performance of equipment needed for disposal operations. In such a case the disposal reliability is adversely affected and is 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 G-1 for a model accounting for random failure).

f. For the re-assessment of the disposal 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.

g. In addition to controlling the risk in case of in orbit anomalies, confirming the good health of space system disposal functions in orbit can allow to extend a mission beyond its nominal life. In order to evaluate the possibility of a mission extension, it is relevant to determine if the planned extension still allows to consider units to be operated in a domain where random failure behaviour or wear-out phenomena prevail, implying an increase in the failure rate.

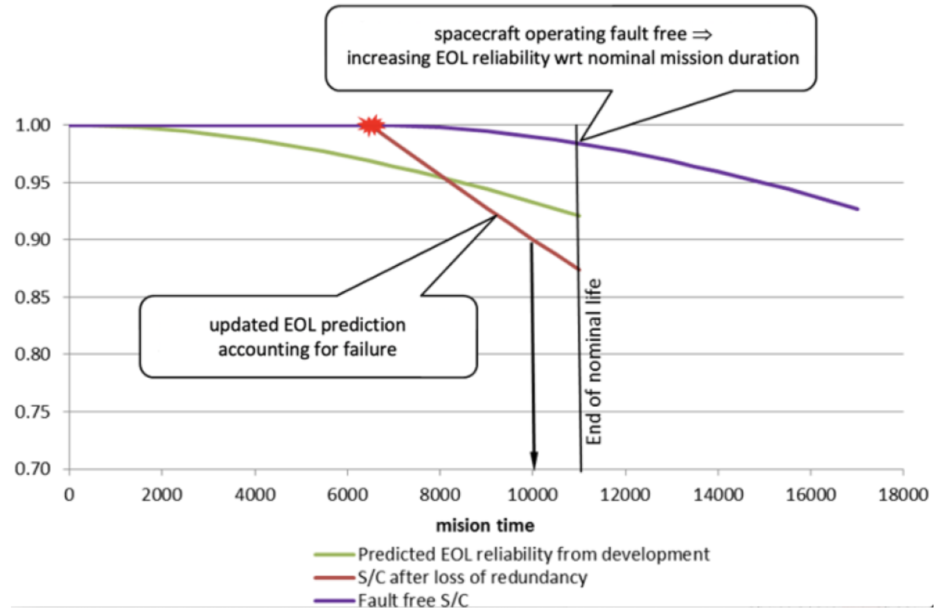


Figure G-1: Principle of disposal reliability updating along the space system life

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

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

During the qualification activities for a space system or unit, assumptions are taken, which are also used to determine the reliability. However, some of these assumptions can be re-considered, based on later on-orbit experience, in order to investigate the possibility to derive more realistic reliability predictions with respect to the design phase. For example:

- The successful result of a qualification test of a process (e.g. soldering) can be re-assessed using the Coffin-Manson Law, based on actual temperatures and duty cycles, to derive a longer lifetime than the nominal one.
- The qualification test of a product (e.g. a mechanism), if it has duly demonstrated a safety margin higher than expected, can be considered to exploit additional margin and derive longer lifetime.

Space system heath monitoring, including Life Limited Items (LLI) performance monitoring, require an assessment of the on-orbit performance of the space system. Performance monitoring is not only relevant for planning extension of the useful life of a space system, but also for identifying early performance degradation that can cause loss of the space system before its nominal end of life, which eventually lead to space debris if not accounted in an update to the disposal strategy.

ESA is leading studies on how performance monitoring and forecasts can be performed for different types of units. Table G-1 provides an overview of possible approaches for performance monitoring and forecast, which, however, is not exhaustive as the possible approaches are not fully supported yet by a validated end-to-end performance monitoring and forecast programme.

Some approaches have been investigated with respect to their capability to improve the assessment of the probability of successful disposal, including:

- Health monitoring:
  It is currently the most common practice, which is based on evaluating the current status of the spacecraft units from the data available on-board and transmitted to the ground segment. From this data useful information can be derived on the status and degradation of the spacecraft units, which can be used to update and improve the accuracy of the disposal reliability model.
- Return of Experience (REX) and Bayesian techniques:
  More realistic failure models or failure rates are obtained by exploiting information on on-orbit experience (e.g. failures, duty cycles), in order to improve the accuracy of reliability predictions made during the development phase. The Bayesian approach is based on updating unit failure rates by accounting for on-orbit observations. The Bayesian approach requires care about the collection and use of data since even small differences in design, usage and environmental conditions can invalidate a Bayesian model update.
- Model based prognostic:
  Performance, or degradation of units, is predicted through engineering models based on their operating conditions. This allows to derive cumulative effects, the estimation of the remaining useful life (RUL) of units, or trends of performance parameters, e.g. to optimize operating conditions by reducing stress and preserving remaining useful lifetime.
- Prognostic based on data trends:
  The approach is based on the statistical exploitation of the observed performance degradation with the objective to replace the current simple exponential failure model not accounting for degradation by a more realistic model. Various methods are around, e.g. linear failure rate evolution, lognormal law, adjustment of parameters of the Weibull distribution, to reflect the degradation behaviour of a monitored unit. The approach needs the collection of a large amount of valid data.
- Risk assessment:
  A risk assessment is performed to address the effect of failure combinations under the condition of a preceding failure on a different unit, subsystem or functional chain having occurred (while classical FMEA/FMECA analyse only the effect of single failures at a time). The assessment can be done in different forms, by a matrix type of analysis or by fault tree analysis for the loss of disposal capability as top event.

Table G-1: Overview of possible spacecraft unit assessment approaches for disposal during the operation phase

Unit(s)	Health monitoring	REX and Bayesian techniques	Model based prognostic	Prognostic based on data trends	Risk assessment	Conclusions and recommendations
Battery	Useful for behaviour and performance check. Estimation not always possible or accurate.		Very useful approach for future performance and RUL prediction. Currently the most commonly used approach.	Useful for improving health monitoring and investigation of anomalies. Currently poorly practically used since the needed amount of data.	Useful to derive the impact of performance degradation or failure propagation.	**Model based prognostic** together with **health monitoring** are currently used. **Prognostic based on data trends** and **Risk assessment** are recommended.
Solar array	Useful for behaviour and performance check, or defining actions in case of low/negative power margins.		Used to predict performance degradation and array size. Major drawback related to lack of data and	Allow to detect anomalies and plan preventive actions when failures are not yet severe.	Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** and **Model based prognostic** are complementary. **Prognostic based on data trends** is very promising.

Unit(s)	Health monitoring	REX and Bayesian techniques	Model based prognostic	Prognostic based on data trends	Risk assessment	Conclusions and recommendations
	Currently the most commonly used approach.		confidence interval.			**Risk assessment** is recommended.
Solar Array Drive Mechanism (SADM)	Requires suitable telemetry data, currently not easily available. Currently the most commonly used approach.				Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is mostly used. There is a general need of availability of larger and more accurate data, and additional monitoring. **Risk assessment** is recommended.
Chemical propulsion thrusters	Useful for behaviour and performance check (via direct or indirect telemetry data), or defining actions in case of anomalies or rapid degradations. Currently the most commonly used approach.		Useful for performance prediction. Accuracy and validity questionable in when operating conditions are different from qualification test conditions.	Useful for improving health monitoring and investigation of anomalies. Currently poorly practically used since the needed amount of data.	Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is currently used. **Prognostic based on data trends** and **Risk assessment** are recommended.
Electrical propulsion thrusters (Hall effect thrusters)	Useful for behaviour and performance check (via direct or indirect telemetry data), or defining actions in case of anomalies or rapid degradations. Currently the most commonly used approach.		Useful for performance prediction. Accuracy and validity questionable in when operating conditions are different from qualification test conditions.	Useful for improving health monitoring and investigation of anomalies. Currently poorly practically used since the needed amount of data.	Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is currently used. **Prognostic based on data trends** and **Risk assessment** are recommended.
Other propulsion units	Requires suitable telemetry data, currently not easily available. Currently the most commonly used approach.	Useful to better evaluate the reliability and lifetime with a large number of flight units. Limitation in keeping coherent design of units.			Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is mostly used. **REX** is recommended for technology heritage**.** There is a general need of availability of larger and more accurate data, and additional monitoring. **Risk assessment** is recommended.

Unit(s)	Health monitoring	REX and Bayesian techniques	Model based prognostic	Prognostic based on data trends	Risk assessment	Conclusions and recommendations
Reaction wheels	Useful for behaviour and performance check or defining actions in case of anomalies or rapid degradations. Currently the most commonly used approach.		Possible use of unit Supplier mathematical models for performance assessment (however, not a clear/complete knowledge on accuracy and validity of the models).	Useful for improving health monitoring and investigation of anomalies. Currently poorly practically used since the needed amount of data.	Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is mostly used. **Prognostic based on data trends** and **Risk assessment** are recommended.
Magneto torquers	Useful for behaviour and performance check. Currently the most commonly used approach.					**Health monitoring** is mostly used. The degradation is typically low and not critical for planning disposal or mission extension.
Magnetometers	Useful for behaviour and performance check. Currently the most commonly used approach.					**Health monitoring** is mostly used. The degradation is typically low and not critical for planning disposal or mission extension.
Sun sensors	Useful for behaviour and performance check. Currently the most commonly used approach.		Useful for behaviour and performance check.			**Health monitoring** is mostly used. **Model based prognostic** is used for degradation prediction and complementary to health monitoring but not critical for planning disposal or mission extension.
Star Tracker	Useful for behaviour and performance check and improving the accuracy of the reliability models with real operating temperatures (not the ones computed at CDR). Currently the most commonly used approach.			Useful for improving health monitoring and investigation of anomalies. Currently poorly practically used since the needed amount of data.	Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is mostly used. **Prognostic based on data trends** is very promising. **Risk assessment** is recommended.

Unit(s)	Health monitoring	REX and Bayesian techniques	Model based prognostic	Prognostic based on data trends	Risk assessment	Conclusions and recommendations
Gyroscopes	Useful for behaviour and performance check and improving the accuracy of the reliability models with real operating temperatures (not the ones computed at CDR). Currently the most commonly used approach.			Useful for improving health monitoring and investigation of anomalies. Currently poorly practically used since the needed amount of data.	Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is mostly used. **Prognostic based on data trends** is very promising. **Risk assessment** is recommended.
GNSS	Useful for behaviour and performance check and improving the accuracy of the reliability models with real operating temperatures (not the ones computed at CDR). Currently the most commonly used approach.			Useful for improving health monitoring and investigation of anomalies. Currently poorly practically used since the needed amount of data.	Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is mostly used. **Prognostic based on data trends** is very promising. **Risk assessment** is recommended.
Earth sensors	Currently this Useful for behaviour and performance check. Currently the most commonly used approach.				Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is mostly used. **Risk assessment** is recommended.
Thermal control	Useful for behaviour and performance check and improving the accuracy of the reliability models with real operating temperatures (not the ones computed at CDR). Currently the most commonly used approach.			Useful for improving health monitoring and investigation of anomalies. Currently poorly practically used since the needed amount of data.	Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is mostly used. **Prognostic based on data trends** is very promising. **Risk assessment** is recommended. Thermal analysis models and assumptions can benefit from data exploitation.
Rotary actuators mechanisms	Requires suitable telemetry data, currently not easily available.	Evaluated and useful only for the same design on the same orbit. Hence	Physics of failure is good for new applications, but it is		Useful to derive the impact of performance degradation or	**Health monitoring** is mostly used. A combined approach

Unit(s)	Health monitoring	REX and Bayesian techniques	Model based prognostic	Prognostic based on data trends	Risk assessment	Conclusions and recommendations
	Currently the most commonly used approach.	very good for constellation at some point, but not good at all for single missions.	important to focus on dominant failure modes in order to limit the complexity. It might be difficult to validate the model because of the lack of data		failure propagation.	incorporating prognostic models, random failures on design and space debris and meteoroids risk, and Bayesian updates can improve the failure predictions. However, limited data and dependency on design and application similarity make REX and model-based prognostic are difficult to use.
Other electronics units	Useful for behaviour and performance check or defining actions in case of anomalies or rapid degradations. Currently the most commonly used approach.			Useful for improving health monitoring and investigation of anomalies. Currently poorly practically used since the needed amount of data.	Useful to derive the impact of performance degradation or failure propagation.	**Health monitoring** is mostly used. **Prognostic based on data trends** is very promising. **Risk assessment** is recommended.

Annex H Space Debris Mitigation Plan (SDMP) document description

H.1 Scope

This document, called Space Debris Mitigation Plan (SDMP), provides the plan for the implementation and verification of the Space Debris Mitigation requirements. The document is issued at Preliminary Requirements Review (PRR) for approval at the System Requirements Review (SRR) in order to define a stable baseline for the development and operations of the space system.

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 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, crewed, 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
    a) Identification of the methodology to assess on-orbit collision risk
    b) Identification of strategies to prevent on-orbit collisions through avoidance maneuvers
5. Space system on-orbit break-up and vulnerability risk
    a) Identification of the methodology to assess on-orbit break-up and vulnerability risk
    b) Identification of strategies to prevent on-orbit impact with space debris and meteoroids resulting in break-up, or unsuccessful disposal of the space system

6. 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 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) Preliminary analysis to determine the probability of successful disposal (based on available information)
    e) Tether systems disposal plan (if applicable)
7. Space system passivation
    a) List of the components storing energy (e.g. Annex F)
    b) List of the components, which are passivated at the end of the operation phase
    c) Description of the design measures and operations implemented for the passivation of the components
    d) List of the component, which cannot be passivated at the end of the operation 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
8. Space system re-entry
    a) Preliminary re-entry casualty risk analysis of the spacecraft and any other MRO likely to re-enter (Annex D), 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 D)
9. Space system hazardous materials
    a) Identification of hazardous materials on-board (e.g. toxic, radioactive)
10. Contingency plan

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 H-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 H-1: Example of preliminary compliance and verification matrix

Req. Id. (see Note 1)	**Requirement short-title** (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	Debris release limitation – spacecraft					
6.1.1.2	Debris release limitation – launch vehicle					
6.1.1.3	Debris release limitation – launch vehicle elements on-orbit presence					
6.1.2.1	Particles limitation – pyrotechnic devices					
6.1.2.2	Particles limitation – solid rocket motors					
6.2.1	Intentionally-caused break-up control					
6.2.2.1	Internally-caused break-up control – probability threshold					
6.2.2.2	Internally-caused break-up control – probability computation					
6.2.2.3	Internally-caused break-up control – passivation					
6.2.2.4	Internally-caused break-up control – passivation for launch vehicle					
6.2.2.5	Internally-caused break-up control – health monitoring					
6.2.2.6	Internally-caused break-up control – contingency plan					
6.2.3.1	Externally-caused break-up control – collision avoidance capability for GEO					
6.2.3.2	Externally-caused break-up control – collision avoidance duties					
6.2.3.3	Externally-caused break-up control – collision risk mitigation					
6.2.3.4	Externally-caused break-up control – vulnerability assessment					
6.3.1.1	Successful disposal assurance – probability threshold					
6.3.1.2	Successful disposal assurance – vulnerability assessment					
6.3.1.3	Successful disposal assurance – disposal criteria					
6.3.1.4	Successful disposal assurance – health monitoring					
6.3.1.5	Successful disposal assurance – contingency plan					
6.3.1.6	Successful disposal assurance – mission extension conditions					
6.3.2.1	GEO clearance – disposal conditions					
6.3.2.2	GEO clearance – disposal execution for continuous presence					
6.3.2.3	GEO clearance – disposal execution for periodical presence					
6.3.3.1	LEO clearance – disposal conditions					
6.3.3.2	LEO clearance – disposal execution					

Req. Id. (see Note 1)	**Requirement short-title** (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	Re-entry – safety requirements					
6.3.4.2	Re-entry – risks threshold					

1. Identification of the requirement ECSS-U-AS-10C Rev. 1 / ISO 24113:2019.
2. Short title 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 I Space Debris Mitigation Report (SDMR) document description

I.1 Scope

This document, called Space Debris Mitigation Report (SDMR), provides 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).

I.2 Table of contents

a. Introduction

b. Scope

c. References

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   2. Reference documents

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   1. Terms and definitions
   2. Abbreviated terms

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   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:
      1) Object type
      2) Object dry mass and fuel or fluids mass (if foreseen)
      3) Object materials
      4) Object dimensions
      5) 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) Probability of accidental break-up analysis and monitoring:
        1) Space system break-up risk analysis
        2) Space system health monitoring (parameters and approach for control of break-up risk)
        3) Anomaly report (status during the operations, if relevant to break-up risk)
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:
        1) Type of particles, e.g. SRM slags, pyrotechnic particles
        2) Size, mass and density (ranges) of the particles
        3) Conditions and orbit where released
4. Space system on-orbit collision risk
    a) Assessment of the collision risk for the spacecraft (Annex B), including:
        1) Analysis to determine the expected number of collisions against trackable objects
        2) Review of design to ensure that adequate resources (Delta-v, propellant mass) are allocated for collision avoidance manoeuvres
    b) Consolidated measures to manage collision avoidance maneuvers
5. Space system on-orbit break-up and vulnerability risk
    a) Assessment of the on-orbit break-up and vulnerability risk for the spacecraft (Annex C), including:
        1) Analysis to determine the probability of catastrophic collisions with objects or debris over the launch, operation, and disposal phases
        2) Analysis to determine the probability of damage or failure due to collisions over the launch, operation, and disposal phases
    b) Assessment of the on-orbit break-up and vulnerability risk for tether systems (if applicable), including:
        1) Analysis to determine the probability of catastrophic collisions with objects or debris over the launch, operation, and disposal phases
        2) Analysis to determine the probability of damage or failure due to collisions over the launch, operation, and disposal phases

6. 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 to accomplish any disposal manoeuvre

   c) Plan of the maneuvers to accomplish the disposal phase, including engine type, thrust level, maneuvre or coasting 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:

      1) Object physical and geometrical parameters and related dispersion margins:
         - Cross-sectional area
         - Drag coefficient
         - Mass
         - Ballistic coefficient
         - Solar radiation pressure reflectivity coefficient
      2) Model assumptions:
         - Atmospheric density, if relevant
         - Earth gravitational attraction
         - Lunisolar attraction
         - Solar activity (solar flux and geomagnetic index)
      3) Initial conditions:
         - Orbital parameters and epoch at the end of the operation phase
         - Orbital parameters and epoch after the disposal maneuvers
      4) Tool and methodology used
      5) Justification for the used methodology and assumptions
      6) Estimation of the presence in the LEO or GEO Protected Regions

   e) Probability of successful disposal analysis and monitoring (Annex G):

      1) Space system reliability analysis and other relevant analysis (until completion of disposal)
      2) Space system health monitoring (parameters and approach for control of risk of degradation of the disposal operations)
      3) Anomaly report (status during the operations, if relevant to risk of degradation of the disposal operations)

   f) Tether systems disposal status (if applicable)

7. Space system passivation

For each component storing energy (e.g. Annex F), 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

8. 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 D), including:

1) 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)

2) 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

3) Initial conditions
   - Orbital parameters and epoch at re-entry, i.e. last orbit before fragmentation events
   - Attitude at re-entry

4) Tool and methodology used

5) Justification for the used methodology and assumptions

6) 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 risk
- 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

9. Space system hazardous materials

a) Summary of the hazardous materials contained on the space system, including:

1) Chemical and commercial name of the material
2) Description of how material is hazard to humans (e.g. explosive, carcinogen, toxic, radioactive)
3) Material state (gas/liquid/solid/powder) and mass/volume and pressure at launch
4) Material state (gas/liquid/solid/powder) and mass/volume and pressure at the operation phase
5) Material state (gas/liquid/solid/powder) and mass/volume and pressure at EOM/end of passivation
6) Material state (gas/liquid/solid/powder) and mass/volume and pressure at released in the atmosphere at re-entry
7) Material state (gas/liquid/solid/powder) and mass/volume and pressure expected to survive at re-entry

10. Contingeny plan

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 I-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 I-1: Example of compliance and verification matrix

Req. Id. (see Note 1)	**Requirement short title** (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	Debris release limitation – spacecraft					
6.1.1.2	Debris release limitation – launch vehicle					
6.1.1.3	Debris release limitation – launch vehicle elements on-orbit presence					
6.1.2.1	Particles limitation – pyrotechnic devices					
6.1.2.2	Particles limitation – solid rocket motors					
6.2.1	Intentionally-caused break-up control					
6.2.2.1	Internally-caused break-up control – probability threshold					
6.2.2.2	Internally-caused break-up control – probability computation					
6.2.2.3	Internally-caused break-up control – passivation					
6.2.2.4	Internally-caused break-up control – passivation for launch vehicle					
6.2.2.5	Internally-caused break-up control – health monitoring					
6.2.2.6	Internally-caused break-up control – contingency plan					
6.2.3.1	Externally-caused break-up control – collision avoidance capability for GEO					
6.2.3.2	Externally-caused break-up control – collision avoidance duties					
6.2.3.3	Externally-caused break-up control – collision risk mitigation					
6.2.3.4	Externally-caused break-up control – vulnerability assessment					
6.3.1.1	Successful disposal assurance – probability threshold					
6.3.1.2	Successful disposal assurance – vulnerability assessment					
6.3.1.3	Successful disposal assurance – disposal criteria					
6.3.1.4	Successful disposal assurance – health monitoring					
6.3.1.5	Successful disposal assurance – contingency plan					
6.3.1.6	Successful disposal assurance – mission extension conditions					
6.3.2.1	GEO clearance – disposal conditions					
6.3.2.2	GEO clearance – disposal execution for continuous presence					
6.3.2.3	GEO clearance – disposal execution for periodical presence					
6.3.3.1	LEO clearance – disposal conditions					
6.3.3.2	LEO clearance – disposal execution					

Req. Id. (see Note 1)	**Requirement short title** (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)
6.3.4.1	Re-entry – safety requirements					
6.3.4.2	Re-entry – risks threshold					

1. Identification of the requirement ECSS-U-AS-10C Rev. 1 / ISO 24113:2019.
2. Short title of the requirement.
3. Status of the 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. Justification of the requirement applicability or not applicability and rationale for the statement of compliance.
6. Reference to any documentation that demonstrates compliance with the requirement (e.g. section of the SDMR, reports, analysis, RFD/RFWs).
7. Close-out status: open or closed.

Annex J
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 on 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 is 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 (OPS-SD) 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 (RFW)	Page 1 of X
1. RFD/RFW Number:	**XXX - Issue Y**	**2. Date:** **dd/mm/yyyy**

3. Requested Type:

☐ Deviation (RFD) ☐ Waiver (RFW)

4. Title of RFD/RFW:

5. ESA Project Submittal:	Directorate:

6. Applicable Requirement:

ECSS-U-AS-10C Rev. 1 - 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 ☐ Approval ☐ Not Approval

ESA Technical Authority - Head of Independent Safety Office (TEC-QI)

Signature: Date: xx/xx/xxxx

13. Approval

ESA Programme Director (D/XXX)	**ESA TEC Director (D/TEC))**
Signature:	Signature:
Date: xx/xx/xxxx	Date: xx/xx/xxxx

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

Table J-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. 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).*
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 Rev. 1.
7.	The ESA Study Manager, Project Manager, or Mission Manager provides a technical description of the non-compliance. If necessary, detailed description can 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 Technology, Engineering and Quality, and the Space Debris Office (OPS-SD) 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 Technology, Engineering and Quality Director (D/TEC) sign the RFD/RFW in case of approval.

Annex K Disposal from bounded Earth orbits outside the Protected Regions and from lunar orbits

Space systems can operate, partially or completely, in bounded Earth orbits other than the LEO and GEO Protected Regions, including orbits which are in between the LEO or GEO Protect Regions, or, which, at a certain time after the nominal operation and disposal phase, can interfere with the LEO and GEO Protected Regions, or MEO operational regions, e.g. due to natural forces (e.g. third-body perturbation, solar radiation pressure, Earth oblateness gravitational effects).

These orbits include, e.g.:

a. Medium Earth orbits (MEO), which have perigee altitude and apogee altitude between the LEO Protected Region and the GEO Protected Region (e.g. GNSS constellations orbits, which usually have semi-major axis below 42164 km and orbital period below 1 day)

b. Geostationary transfer orbits (GTO), or other Earth orbits, which have perigee or apogee interfering with the LEO or GEO Protected Region

c. Inclined geosynchronous orbits (IGSO), which have perigee below the GEO Protected Region and apogee above the GEO Protected Region (e.g. Tundra or Quasi-Zenith orbits, which usually have semi-major axis equal to 42164 km and orbital period of 1 day)

d. Highly eccentric orbits (HEO), which have perigee below the GEO Protected Region and apogee above the GEO Protected Region

e. Lunar orbits, which are considered part of the Earth-Moon system

Disposal in MEO is allowed, except for space systems operating in LEO which can only de-orbit and re-enter, therefore requiring implementation of all the measures to prevent debris generation, including end of mission passivation.

However, MEO includes operational regions, which, although not explicitly defined in ECSS-U-AS-10C Rev. 1 [RD2] / ISO 24113:2019 [RD3] as Protected Regions, are of relevant importance as hosting operational GNSS constellations, and, therefore, are avoided when selecting graveyard orbits. As of today, there is no technical definition of MEO operational regions, therefore, these regions are defined using the best of available knowledge.

MEO operational regions include GNSS constellations, but are not limited to:

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

Recent studies suggest considering ±50 km semi-major axis range from the nominal semi-major axis for typical GNSS operational region.

In order to optimize the resources availability and system capability for disposal, and collision risk reduction, different options to select a target disposal orbit are possible.

Disposal options from MEO, IGSO, HEO, and GTO include:

a. Stable orbit (low eccentricity variation) outside the LEO and GEO Protected Regions (e.g. by means of manoeuvres in MEO);

b. Unstable orbit (high eccentricity variation) to LEO and re-entry (under the effect of natural orbit perturbations);

c. Direct de-orbit to LEO (by means of manoeuvres) and re-entry (under the effect of atmospheric drag, or by means of manoeuvres);

d. Re-orbit above the GEO Protected Region (by means of manoeuvres).

Unstable orbits can be selected by manoeuvring the space system to an orbit with determined eccentricity, argument of perigee, right ascension of ascending node and inclination at a specified epoch. Unstable orbits and direct de-orbit from MEO can result in an increase of the collision risk as these orbits can likely interfere (or evolve in future interference) with LEO or MEO populated regions. Unstable orbits and direct de-orbits can also possibly lead to re-entry, which ensures permanent clearance, but requires compliance with the re-entry safety requirements (ESSB-ST-U-004 [RD4]). Therefore, a trade-off is performed to preventively assess the consequences associated to the selected disposal option for each space system and in order to eventually identify the least risky approach.

Direct de-orbit from MEO, although typically hard to meet because of demanding resources, can be performed, e.g. using low thrust electrical propulsion systems, provided that the disposal duration is not long enough to create increase of the collision risk until re-entry (e.g. residual cumulative collision risk through time not higher than the one associated to the operational orbit in MEO).

Lunar orbits are considered part of the Earth-Moon system, therefore, the same requirements to prevent debris in Earth orbits are applicable. Disposal options from lunar orbits, which can naturally evolve into interference with LEO and GEO Protected Regions, or MEO operational orbits, require specific control in order to comply with the Space Debris Mitigation requirements (ECSS-U-AS-10C Rev. 1 [RD2]) and not to interfere with MEO GNSS constellations. Disposal from lunar orbits can also result into collapse on the Moon, which, if planned or possible, requires compliance with the Planetary Protection requirements (ECSS-U-ST-20C [RD11]). Additional requirements can be applicable to protect possible human spaceflight assets. Similar general remarks can apply to orbits around Earth-Moon Lagrange points (EMLs).

For a space system operating in MEO, GTO, IGSO, HEO, or lunar orbits, re-entry, although usually not foreseen in the baseline, is possible, e.g. in the cases when the space system:

a. is separated from the launch vehicle and released in an orbit with perigee in LEO, and failure in orbit raising system prevent the space system to reach the nominal orbit (i.e. the probability of re-entry is dependent from the platform reliability at the early phase of the mission).

b. is on an unstable orbit (e.g. as result of wrong positioning in orbit, or if foreseen by a disposal plan), where natural forces can drive lowering the perigee altitude down to LEO where atmospheric drag effects become relevant.

c. is intentionally manoeuvred to lower the perigee altitude down to LEO where atmospheric drag effects become relevant.

The verification of compliance is based on:

a. Analysis, to:

1. assess that the selected disposal approach has a negligible probability of interference with the LEO and GEO Protected Regions, and the MEO operational regions for at least 100 years (possible metrics for the interference with orbital regions are listed in Annex M).
2. assess that presence in the LEO Protected Region is limited to less than 25 years since the first interference event after the disposal, if interference with LEO is possible.
3. assess the resulting collision risk with other objects associated with the selected disposal approach, e.g. an analysis with multiple metrics can allow to determine short-term effects (e.g. collision risk) and long-term effects (e.g. object accumulation) of the selected approach.
4. assess the re-entry casualty risk to ensure the compliance with the re-entry safety requirements (ESSB-ST-U-004 [RD4]), if re-entry is planned or possible.
5. assess the consequences of all the possible disposal scenarios to ensure compliance with Planetary Protection requirements (ECSS-U-ST-20C [RD11]), if relevant (e.g. lunar orbits).
6. assess the robustness of the selected approach with respect to relevant sources of uncertainties (e.g. disposal epoch, cross-sectional area, other parameters, as listed in Annex M).

The tool used to perform analysis of the collision risk outside Earth orbits can be different, depending on the mission and the orbit. Currently there is no fully dedicated tool for such analysis, while customized codes, or worst-case assumptions, can be used. A first order assessment of the collision risk can be performed by analysing the flux experienced by a mission on the planned trajectory, e.g. using the ESA tool MASTER, if the orbital region is covered by the tool. Data on the specific population of on-orbit objects is relevant in order to assess accurately the collision risk to other objects.

Possible mitigation measures to minimise the risk of non-compliance are:

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 the probability of interference is negligible (for at least 100 years).

b. To de-orbit the space system to a lower altitude with sufficient clearance from the MEO operational regions and, as well as the LEO and GEO Protected Region, such that the probability of interference is negligible (for at least 100 years).

c. To ensure a minimum clearance from MEO operational regions, including ESA GNSS constellations (i.e. Galileo) and other known GNSS constellations, of at least ±300 km in semi-major axis with eccentricity below 0,001 (TBC) (orbit circularization) for at least 100 years. Optimal solutions are feasible for closer semi-major axis to the MEO operational regions with suitable orientation with respect to Moon ad Sun.

d. To ensure that the selected disposal orbit does not interfere nor result in non-negligible collision probability with other space systems previously disposed (e.g. spacecraft and launch vehicle orbital stages) for at least 100 years.

e. To manoeuvre 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 (e.g. less than 5 years) and in compliance with ESSB-ST-U-004 [RD4].

f. 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 operation phase.

Annex L
Disposal from unbounded Earth orbits and from Sun-Earth Lagrange points orbits

Space systems can operate in unbounded Earth orbits, or Sun-Earth Lagrange points (SELs) orbits, which can interfere with the LEO and GEO Protected Regions, or MEO operational regions, at a certain time after the nominal operation and disposal phases, e.g. due to natural forces (e.g. third-body perturbation, solar radiation pressure).

These orbits include, e.g.:

a. Unbounded Earth orbits, which have Earth at a focal point

b. Sun-Earth Lagrange points (SEL) orbits, which involves trajectories around the SEL points

The orbits of space systems around SEL 1, 2 and 3 (collinear libration points) are naturally unstable since natural forces (perturbations) can lead the space system to return to Earth orbits, or re-enter. In these orbits, the Space Debris Mitigation requirements are fully applicable in order to avoid that debris are generated or can return to Earth orbits.

The orbits of space systems around SEL 4 and 5 (triangular libration points) are naturally stable since natural forces (perturbations) do not lead the space system to return to Earth orbits, or re-enter. In these orbits, the Space Debris Mitigation requirements, although not mandatory, are recommended in order to avoid debris generation in orbital regions of possible interest (e.g. for science).

Successful disposal from SEL orbits implies disposal into heliocentric orbits with no revisit closer than 1,5 million km to Earth (or negligible probability to interfere with the Earth Sphere of Influence considering the space system area-to-mass ratio distribution) within the next 100 years.

Trajectory analyses typically show that if disposal manoeuvres to place a space system from SEL-1,2,3 orbit into a heliocentric orbit are successfully executed with an adequate Delta-v during given dates, the probability of remaining into heliocentric orbit can be 1,0 (for at least 100 years).

If the space system uses a finite Delta-v for disposal (e.g. 10 m/s), the residual probability to return to Earth orbits can be zero if the manoeuvre is performed during a determined limited number of useful dates per year. The useful dates are usually concentrated in a couple of windows per year. An increase in the Delta-v for disposal, implies an increase of the duration of the windows with useful dates. Therefore, there can be several combinations of Delta-v allocated for disposal and date of execution of the disposal manoeuvres, which can guarantee no return to Earth orbits, providing that the space system is capable to perform the manoeuvres in full (i.e. with a reliability higher than 0,90 as normally for disposal functions). In fact, the probability of return to Earth orbits is related to the energy level of the orbit, which can be changed with a manoeuvre, whose efficiency in reaching energy level allowing to avoid return to Earth orbits depends on the epoch when the manoeuvre is executed. In case the disposal manoeuvres from SEL-1,2,3 are unsuccessful, or are not performed, there is a distinct probability to return to Earth orbits, which can be conservatively assumed to be 0,50.

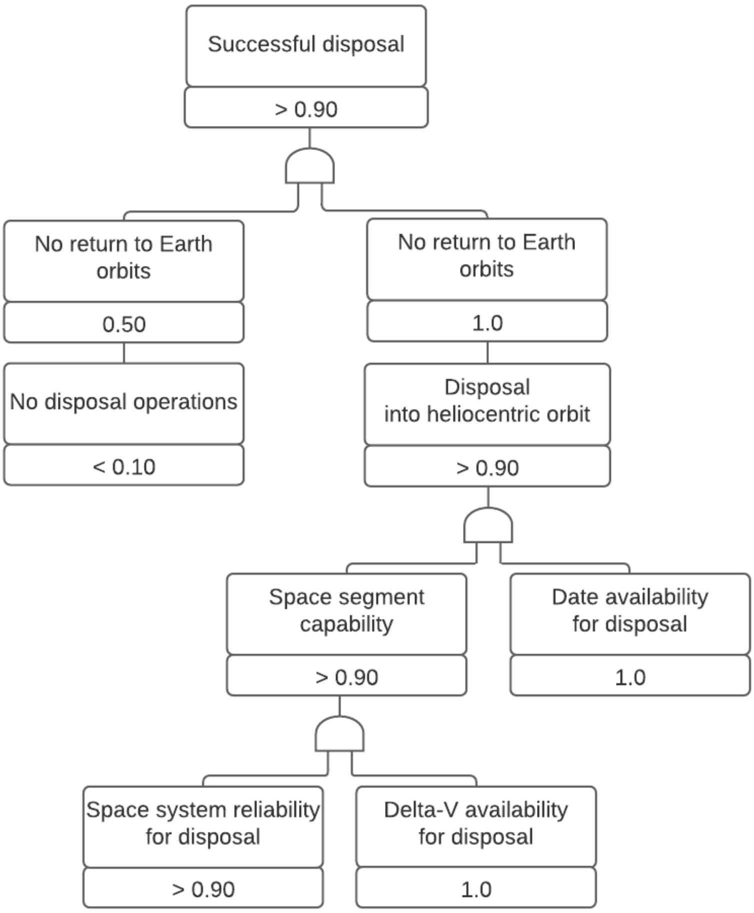


Figure L-1: Example diagram for probability of successful disposal for a SEL mission

The verification of compliance is based on:

a. Analysis, to:

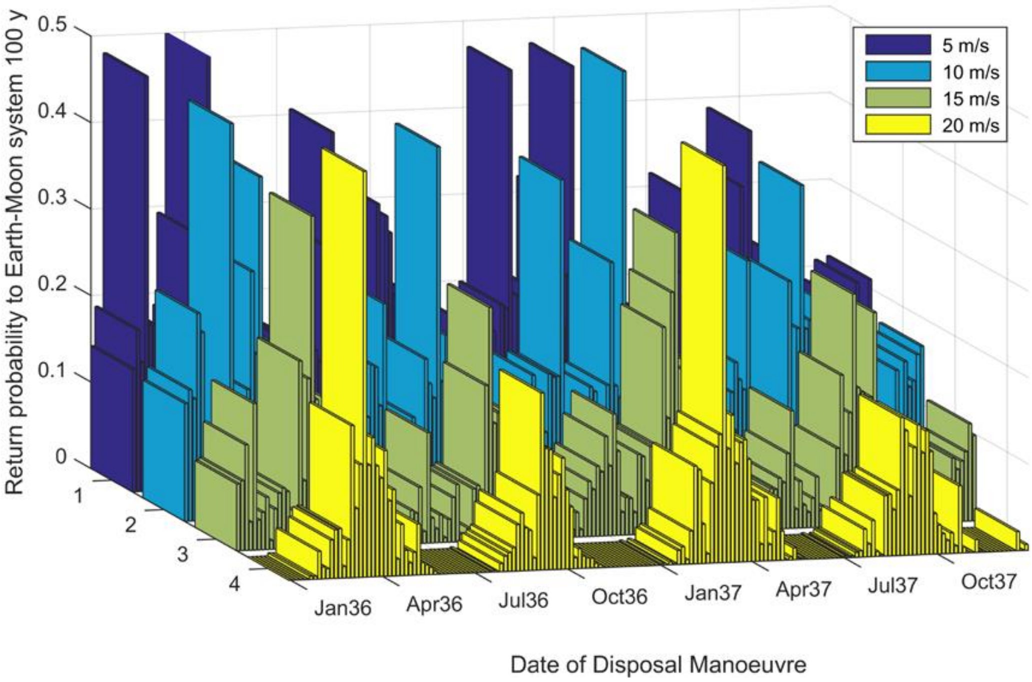
1. assess the probability of successful disposal, i.e. disposal into heliocentric orbits with no revisit closer than 1,5 million km to Earth (or negligible probability to interfere with the Earth Sphere of Influence) within the next 100 years, in order to demonstrate that the probability of successful disposal is higher than 0,90 (see Annex M), considering:
    a) orbit propagation with stochastic simulations (e.g. Monte Carlo)
    b) trajectory uncertainties
    c) space system reliability at time of the disposal execution
    d) space system area-to-mass ratio distribution
2. determine the minimum resources allocation (propellant mass, Delta-v) and time availability to allow successful disposal, i.e. disposal into heliocentric orbits with no revisit closer than 1,5 million km to Earth (or negligible probability to interfere with the Earth Sphere of Influence) within the next 100 years (see Annex M).
3. assess the re-entry casualty risk to ensure the compliance with the re-entry safety requirements (ESSB-ST-U-004 [RD4]), if re-entry is planned or possible.

Possible measures to minimise risk of non-compliance are:

a. To dispose the space system into heliocentric orbits with no revisit closer than 1,5 million km to Earth within (or negligible probability to interfere with the Earth Sphere of Influence considering the space system area-to-mass ratio distribution) the next 100 years.

b. To maximize the resources (propellant mass, Delta-v) allocated for the disposal manoeuvres to a stable orbit with no return to bounded Earth orbits.

c. To optimize the Delta-v budget allocation from lessons learnt in the operations of similar missions in order to possibly increase the allocation for disposal manoevres, e.g. operations of past SEL-2 missions shown a surplus of unused Delta-v (with respect to typical design allocation) for launch dispersion correction to ensure 3-σ trajectory accuracy (where a Delta-v of about 5 m/s is tipically used), and for station keeping (where a Delta-v of about 1 m/s per year is typically used).



Example of probability to return to Earth orbit for a generic SEL-2 mission as a function of date of the disposal manoeuvres execution and allocated Delta-v for disposal

Figure L-2: Example probability of return to Earth orbit for a generic SEL-2 mission

Orbits around SEL-4/5, distant about 150 million km from Earth, are stable compared to SEL-1/2/3.

Although SEL-4/5 orbits are not yet considered as Protected Regions from Space Debris Mitigation point of view, preservation of SEL-4/5 orbital regions is considered reasonable in view of their possible future use and scientific interest (e.g. for space weather monitoring through mission requiring lower Delta-v for orbit injection compared to SEL-1/2/3 with an orbit amplitude below 2 million km around SEL-4/5).

Although no direct operations have been performed as of today and risk of collision is, therefore, low, predictive analysis of orbit evolution can be performed to minimise interference with future space systems in view of the stable nature of the orbital region, which can lead to potential future volume concentration.

"Nice to have" recommendations for Space Debris Mitigation are under consideration for SEL-4/5, including:

a. To minimise the presence in SEL-4/5 orbits of space systems by performing disposal manoeuvres, considering that any manoeuvre can be beneficial. A Delta-v allocation of about 42 m/s can allow the space system to move to a horseshoe orbit (where the object slowly oscillates between the two triangular SEL points, passing through the region around the SEL-3 point) in order to slowly diverge from SEL-5 (or SEL-4) towards SEL-4 (or SEL-5), which can serve as a graveyard orbit for the space system. Smaller Delta-v can leave the space system in a tadpole orbit (where the distance of the object from the Earth does not reach the SEL-3 point).

b. To implement passivation measures at end of mission in order to minimise debris generation as excessive Delta-v from a break-up (e.g. order of magnitude of 1000 m/s) can lead debris to return to Earth orbits from SEL-4/5 in about 200 years.

Annex M
Orbit interference assessment

When assessing the compliance with space debris requirements, for mission with uncertain trajectory propagation, it is relevant to quantify the probability of interference with the Protected Regions in order to determine whether the probability is within an acceptable threshold. In fact, in several conditions, deterministic analysis of a single trajectory cannot be sufficiently representative to characterise correctly the compliance with the Protect Region clearance requirements, e.g. when the trajectory of a space system is depending on specified launch vehicle trajectory dispersions, or has strong dependence on the orbital dynamics, parameters (e.g. solar cycle and geomagnetic activity), and time of execution of operations. For orbital regimes where chaotic behaviour can emerge, the robustness of the assessment to the variation of mission parameters is relevant to determine the solution domain. When deterministic approach is insufficient, a stochastic approach (e.g. Monte Carlo simulations) can provide relevant information about the dependencies of parameters.

This Annex provides indications on how to treat uncertainties through a stochastic approach.

The approach can be based on two aspects:

a. Identification of a suitable metric for interference;

b. Evaluation of the metric considering the relevant dependencies on the mission parameters.

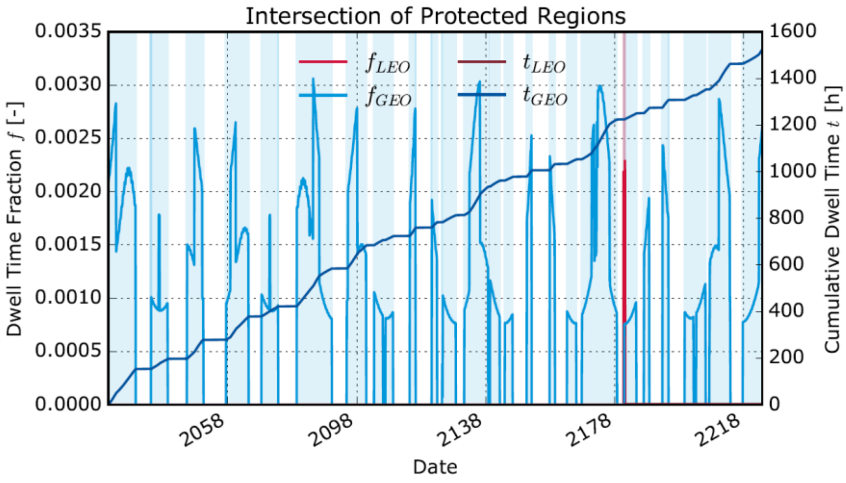
M.1 Metrics for stochastic analysis

The definition of interference is intended primarily as an estimator on whether the trajectory of the analysed space system systematically and significantly crosses a Protected Region during its orbital lifetime. Despite the objective of Space Debris Mitigation is to have no interference with the Protected Region, in some cases, e.g. Libration Point Orbits (LPO) or interplanetary orbits, the occurrence of the interference is an event, which has a probability depending on the boundary conditions (e.g. a specific launch date), with, or without, a limited duration and periodicity. Possible metrics can be used to quantify the risk of interference with the Protected Regions, e.g.:

a. Cumulated residence time in the Protected Region

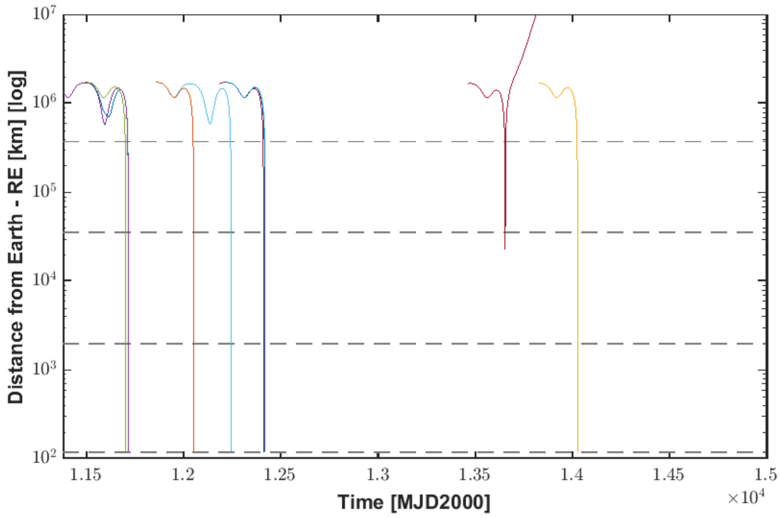
b. Cumulated collision probability

The computation of the cumulated collision probability for the analysed object can be performed considering the flux, or concentration of objects, encountered along the trajectory, e.g. using the ESA tools MASTER and DRAMA/ARES/MIDAS. However, when analysing orbital regions with low object concentration (e.g. MEO), or along non-circular orbits (e.g. HEO), the collision risk assessment can be relatively low compared to orbital regions with higher object concentration (e.g. LEO). For example, a disposal from HEO through LEO can result in an increase of the cumulated collision risk although with an attempt to avoid free drift involving multiple GEO crossings. Therefore, cumulated collision probability evaluation is considered only with respect to the collision probability associated to the original orbital region.



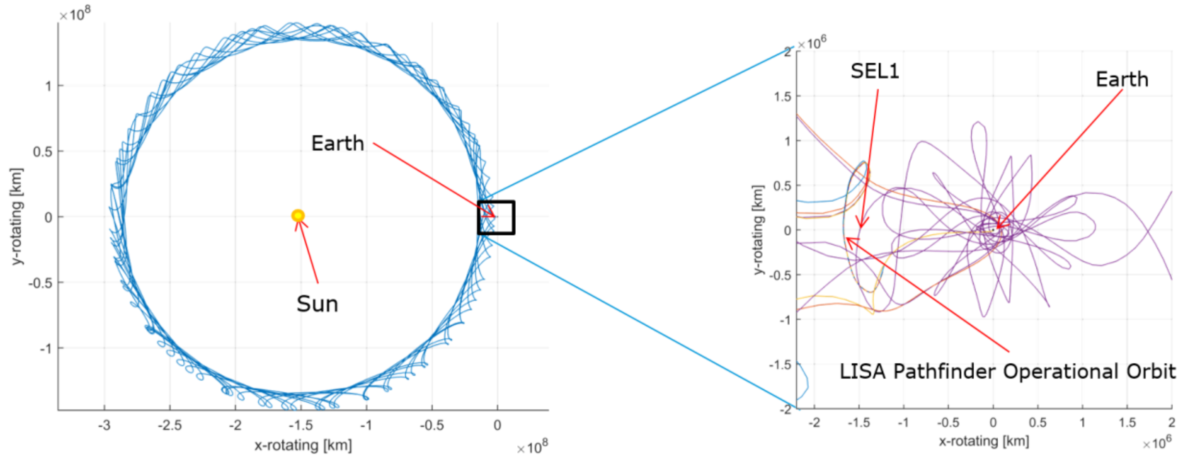
Example of 200 year propagation and crossing of the LEO and GEO Protected Regions in terms of time fraction per orbit and cumulative time (ESA XMM spacecraft operating in HEO)

Figure M-1: Example of long-term propagation and crossing of the LEO and GEO Protected Regions



The horizontal lines indicate (top to bottom) the Earth's Sphere of Influence at 1,5 million km, the GEO altitude, the LEO boundary at 2000 km, and the atmosphere interface at 120 km

Figure M-2: Analysis of return trajectories from a LPO



Lisa Pathfinder trajectories: on the left the heliocentric trajectories are shown after departure from SEL-1, on the right potential trajectories in the Earth-Moon System are shown (the orbits have a very high apogee altitude and thus very long orbital periods; the trajectories do not remain in the Earth-Moon system for prolonged periods of time and interference with the GEO / LEO Protected Regions is short during the actual orbits and overall)

Figure M-3: Lisa Pathfinder trajectories

M.2 Distribution functions and convergence criteria for stochastic analysis

In several conditions, the analysis of a single trajectory cannot be relevant to correctly characterize the compliance with the space debris mitigation requirements. For example, in the cases where there is a strong time-dependence of the dynamics (e.g. return probability from Libration Point Orbits), several epochs are taken into consideration. In addition, for orbital regimes where a chaotic behavior can emerge (e.g. disposal from the MEO region), the robustness of the assessment to the variation of mission parameters can be considered. Similarly, also in case of dependence on predicted parameters (e.g. solar and geomagnetic activity for the analysis of lifetime estimation for LEO crossing objects), a statistical approach is recommended.

When Monte Carlo simulations are performed, the minimum number of runs is set to ensure, with a minimum confidence level, that the solution, which is not known *a priori*, is convergent as the number of runs further increases. Relevant parameters taken under consideration include:

a. Solar cycle and geomagnetic activity, e.g. with sensitivity as for orbit propagation analysis (Annex A)

b. Cross-sectional area, e.g. with distribution between the minimum and maximum value, accounting for relevant space system configurations (e.g. solar array fully deployed, partially deployed, not deployed) for orbital regions where chaotic behaviour can occur; for simpler cases, testing the extreme cases (i.e. minimum and maximum values) is enough to define the envelope of evolution

c. Epoch of launch, injection, separation, or relevant manoeuvre, e.g. with uniform distribution across time interval or at discrete times, accounting for launch delays, extended windows, and the 11-year solar cycle, i.e. using the levels of solar and geomagnetic activity corresponding to the different analysis epochs

d. Separation dispersion, e.g. sampling of the covariance matrix depending on the launch vehicle trajectory

e. Manoeuvre dispersion, e.g. sampling of the covariance matrix depending on the thruster, or sampling around the nominal thrust direction with perturbations in magnitude and direction

f. Initial state dispersion, e.g. sampling the knowledge dispersion matrix of the initial state due to orbit determination uncertainties or random failures

In cases the problem can be formulated with a binary criterion (e.g. re-entry/no re-entry, crossing/no-crossing), a single Monte Carlo run can be considered as a binomial process with only two possible outcomes, generally labelled as "success" and "failure". The output of the Monte Carlo simulation is considered as a binomial variable $X \sim B(n, p)$, with n number of trials and p success probability for each trial. Applying an approximation to a normal distribution, the mean value is $\mu = np$, the variance is $\sigma = np(1-p)$, and the confidence interval (Wald's confidence level) is:

$$\hat{p} - z_{1-\alpha/2}\sqrt{\frac{1}{n}\hat{p}(1-\hat{p})} \leq p \leq \hat{p} + z_{1-\alpha/2}\sqrt{\frac{1}{n}\hat{p}(1-\hat{p})} \qquad \text{[M-1]}$$

where $\hat{p}$ is the probability of success estimated from the statistical sample, $z_{1-\alpha/2}$ is the $\left(1-\frac{\alpha}{2}\right)$ quantile of a standard normal distribution, $\alpha = 1 - c$ is the error quantile, with c confidence level.

When the probability p tends to 1 or 0, a more adequate estimation of the confidence level (Wilson's confidence level) is used, considering that, for an observed value $\hat{p}$, there are two values of the mean p of a normal distributed variable that can put $\hat{p}$ at the limits of a confidence interval about p:

$$\frac{\hat{p} + \frac{z^2{}_{\alpha/2}}{2n} - z_{\alpha/2}\sqrt{\frac{\hat{p}(1-\hat{p})}{n} + \frac{z^2{}_{\alpha/2}}{4n^2}}}{1 + \frac{z^2{}_{\alpha/2}}{n}} \leq p \leq \frac{\hat{p} + \frac{z^2{}_{\alpha/2}}{2n} + z_{\alpha/2}\sqrt{\frac{\hat{p}(1-\hat{p})}{n} + \frac{z^2{}_{\alpha/2}}{4n^2}}}{1 + \frac{z^2{}_{\alpha/2}}{n}} \qquad \text{[M-2]}$$

Equation [M-2] is not suitable for a low number of simulations ($n < 30$) since it an approximation for the discrete binominal distribution. For higher number of simulations ($n > 30$), an approximation for continuity can be used, (e.g. Yale's correction, which is demonstrated to be conservative for $n > 50$):

$$\hat{p} - \left(\frac{1}{2n} + z_{\alpha/2}\sqrt{\frac{\hat{p}(1-\hat{p})}{n}}\right) \leq p \leq \hat{p} + \left(\frac{1}{2n} + z_{\alpha/2}\sqrt{\frac{\hat{p}(1-\hat{p})}{n}}\right) \qquad \text{[M-3]}$$

In the case where an explicit target for p exists, the expression of Wilson's confidence interval can be used, for example, to determine the minimum number of runs with no *failures* ($\hat{p} = 0$) to ensure that p is below the defined threshold, with the selected confidence interval. The approach is currently implemented in the ESA DRAMA tool.

M.3 Bootstrapping

A bootstrapping technique can be used to estimate the properties of an estimator (e.g. mean value, median, variance) by measuring those properties when sampling with replacement from an approximating distribution. For example, in the case a re-entry casualty risk analysis associated to the failure of a spacecraft in SEL, the initial population is generated by propagating the spacecraft trajectory assuming a random failure time. Only a subset of these trajectories results in a re-entry, which represent the subset of interest. For each of these trajectories, the relevant parameter, e.g. casualty cross-section area, is computed and the resulting collection of values represent the empirical distribution. This distribution is then re-sampled (with a selected sample size) multiple times (running a sample estimator on each re-sampled set of observations).

M.4 References

More in-depth information on the subject of this Annex is available in the following references:

- F. Letizia, S. Lemmens, B. Bastida Virgili, H. Krag, Application of a debris index for global evaluation of mitigation strategies, Acta Astronautica, Vol. 161, pp. 348–362, 2019. doi: 10.1016/j.actaastro.2019.05.003
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Annex N Spacecraft Design for Removal

N.1 Objectives

This Annex describes guidelines for the preparation of removal of a spacecraft by an external servicer spacecraft in order to enhance compliance of disposal of the spacecraft with the space debris mitigation and re-entry safety requirements. The removal operation is frequently referred as "Active Debris Removal" (ADR). The set of specific design features, which allow spacecraft readiness for being removed by a servicer spacecraft is referred as "Design for Removal".

N.2 Methodology

An ADR operation normally involves two entities:

a. A servicer spacecraft, i.e. a spacecraft having the function of removing another spacecraft (i.e. the client spacecraft), from its orbit after its end of mission, either in a cooperative or uncooperative scenario.

b. A client spacecraft, i.e. a spacecraft designed and prepared for being removed by another spacecraft (i.e. the servicer spacecraft).

"Design for Removal" can facilitate the operation of removal of a spacecraft, as potential issues associated to removal operations (including close proximity and rendezvous operations) are preventively tackled. "Design for removal" can enhance reduction of complexity and increase of efficiency for a removal operation. For a client spacecraft in LEO, removal operations normally imply capture, de-orbit, and safe re-entry of the client spacecraft with the assistance of a servicer spacecraft.

"Design for removal" can be actuated in:

- a "cooperative scenario", i.e. when the client spacecraft is still operational (i.e. the client spacecraft is able to control its attitude and maintain its orbit), or
- an "uncooperative scenario', i.e. when the client spacecraft is non-operational (i.e. the client spacecraft lost its control).

"Design for Removal" needs the following main functions:

- Tracking and attitude reconstruction on ground, in order to have accurate knowledge of the target attitude and orbit of the client spacecraft before starting its removal operation.
- Acquisition of suitable attitude, in order to ensure compatibility of the client spacecraft with the operational envelope of the servicer spacecraft.
- Relative navigation support, in order to minimise or avoid any risk for the close proximity and rendezvous operations through precise synchronisation of the servicer spacecraft with the client spacecraft and simplification of the capture.
- Mechanical capture interface, in order to allow the capture of the client spacecraft by the servicer spacecraft and the transfer of the mechanical loads between them during the removal operations.

Currrent State-of-the-Art technology and feasibility studies suggest some guidelines on how to implement "Design for Removal" function for different scenarios, which are grouped in:

- Generic risk mitigation guidelines
- Specific risk mitigation guidelines for cooperative scenarios
- Specific risk mitigation guidelines for uncooperative scenarios

N.2.1 Generic risk mitigation guidelines

Some generic guidelines are considered regardless of the scenario:

a. To know if the client spacecraft is in condition to be serviced, the client spacecraft is able to determine and share with the servicer spacecraft its status (e.g. structural integrity), orbital parameters and attitude parameters with level of accuracy defined by the servicer spacecraft.

b. To support the rendezvous of the servicer spacecraft with the client spacecraft, the client spacecraft is equipped with relative navigation aids (e.g. markers), which allow precise and accurate measure of the client spacecraft attitude facilitating synchronisation and alignment of the servicer spacecraft with the client spacecraft. Different relative navigation aids can be necessary, which:

   1. are able to support unbiased reconstruction of the client spacecraft attitude by the servicer spacecraft from a safe distance before the synchronisation and alignment manoeuvres start.

   2. are normally installed on the panel including the mechanical interface for capture, to support the precise determination of the relative motion and misalignments (once the synchronisation and alignment have been initiated) for proper use of the capture mechanism and minimum collision risk.

   3. are robust to different and changing illumination conditions.

   4. cover different rendezvous sensors, if possible, e.g. cameras in different wavelengths, lidars, etc., in order to be compatible with different sensor suits of the servicer spacecraft.

c. To allow for the capture, the client spacecraft includes a mechanical interface compatible with the capture mechanism of the servicer spacecraft. The mechanical capture interface for capture:

   1. supports the mechanical loads to damp residual motion or to correct misalignements between client and servicer spacecraft.

   2. is designed to allow for capture before contact, to avoid that the client spacecraft escapes after capture.

d. To rigidize the capture and allow the transfer of the de-orbit manoeuvre loads between the servicer and the client spacecraft, if necessary, the client spacecraft provides access to a stiff interface, e.g. a launch adapter ring. The loads:

   1. depend on the servicer spacecraft design.

   2. are expected to be higher for a controlled re-entry, very likely requiring a stiff interface to manage the transfer of loads.

e. To minimise errors and misalignments during rendezvous and capture, the client spacecraft ensures the correct mounting and precise alignement of the mechanical interface for capture and the relative navigation aids, and is able to share the related information according to the requirememts from the servicer spacecraft.

f. To prevent the interference of other hardware with capture operations, the client spacecraft has no equipment or appendage, in any of its operational positions, that blocks the access to the mechanical interface for capture or actuation of the rigidisation hardware.

g. To support the rendezvous and capture, the client spacecraft has all the relative navigation aids with enough clearence to be seen by the servicer spacecraft rendezvous sensor suit.

h. To prevent the release of debris during the de-orbit manoeuvres, the client spacecraft is designed to support the de-orbit manouvre accelerations and loads.

N.2.2 Specific risk mitigation guidelines for cooperative scenarios

Some specific guidelines are considered for cooperative scenarios:

a. To simplify the rendezvous and capture operations, the client spacecraft is capable of acquiring and maintaining a suitable attitude for the rendezvous and capture by the servicer spacecraft.

b. To avoid the risk of the client spacecraft reacting against the capture, during the rendezvous and capture operations the client spacecraft is put in a operational mode that is robust to the close proximity and capture by the servicer spacecraft.

c. To prevent the interference with the post-capture operations, once the servicer spacecraft has confirmed the capture, the client spacecraft shuts down possibly perfoming passivation (if safe).

N.2.3 Specific risk mitigation guidelines for uncooperative scenarios

Unpredictable angular rates are a major driver for the feasibility and risk of the removal service in an uncooperative scenario.

Some specific guidelines are considered for uncooperative scenarios:

a. To prevent the increase and minimise the magnitude of its tumbling motion, the client spacecraft can implement de-tumbling methods to dissipate kinetic energy once it becomes uncooperative, e.g. passive magnetic detumbling in LEO orbits. The de-tumbling methods:

    1. can be fully passive or automatically activated in case of an unrecoverable failure of the client spacecraft and work without the need of any resource provided by the client spacecraft.

    2. do not interfere with the client spacecraft system during the operation phase.

    3. allow the reduction of the angular rates below the maximum magnitudes defined by the servicer spacecraft.

b. To support the tracking determination of its orbital parameters and attitude reconstruction on ground, once in an uncooperative state, the client spacecraft implements passive features that allow its identification (geometry) from ground and attitude reconstruction independently of its pose and attitude, e.g. laser retro reflectors (LRRs).

c. To minimise or avoid the risk of the rendezvous and synchronisation of the servicer spacecraft and client spacecraft in a uncooperative scenario (client spacecraft attitude not controlled), the relative navigation aids are able to support the determination of the client spacecraft attitude by the servicer independently of the client spacecraft attitude.

N.3 Detection, identification, and trackability

An on-orbit collision risk analysis implying collision avoidance manoeuvres can only be performed for objects included in the so-called Space Surveillance and Tracking (SST) catalogues, i.e. when the data on the orbital state are available in a timely manner and are actionable quality. A pre-condition for an object (space debris, active spacecraft, or launch vehicle orbital stage) to be included in an SST catalogue is that the object is detectable and trackable.

Detection is the property of a space surveillance system to statistically detect an object passing through the field of view of a sensor.

Tracking is the ability to use space surveillance data to unambiguously monitor the orbital evolution of an identified object, accounting for uncertainties that in turn are used for the collision avoidance processes.

In addition, a third ability of the surveillance network is to engage in identification, i.e. the ability to use space surveillance data to identify uniquely correlated objects and ideally confirm the source.

As part of a collision avoidance strategy, an SST catalogue and service is a necessary input. It is common practice to assess the impact of the accuracy and timeliness of the service on the strategy. In some cases, such as physically small objects or objects on trajectories reaching large distance from the space surveillance sensors, a dedicated analysis is performed to ensure that objects are detectable and trackable by the identified catalogue provider. This can generally be achieved by, if necessary, adding orbit tracking enhancement or attitude determination aids such as laser retro-reflectors or other devices tuned to the identified catalogue provider. Additional identification provisions, across different catalogue providers, ensures efficient cooperation between operators in case of close approach operations.

Annex O
Space transportation platforms

A space transportation platform is a space system, which although is not part of a launch vehicle development, is put in orbit by a launch vehicle with the function to embark, perform repositioning manoeuvres, and finally release in orbit other space systems, usually smaller spacecraft provided by a different customers, i.e. hosted complement spacecraft.

A space transportation platform is a spacecraft, e.g. an on-orbit motorized dispenser, which is, therefore, subject to comply to the space debris mitigation and re-entry safety requirements, including the following:

a. The compliance with all the space debris mitigation requirements (ECSS-U-AS-10C Rev. 1 [RD2] / ISO 24113:2019 [RD3]) is verified both for the space transportation platform and all the hosted complement spacecraft.

b. The space transportation platform has the propulsion capability to ensure clearance of the Earth Protected Regions.

c. The release of the hosted complement spacecraft is executed in a way such as to avoid re-contact or collision between each released hosted complement spacecraft and the space transportation platform or the other released hosted complement spacecraft.

d. On-orbit functional testing and monitoring of each hosted complement spacecraft before the release is performed in order to avoid release of non-functional spacecraft (space debris).

e. The compliance with on-orbit break-up risk threshold (10^{-3}) is verified as aggregate risk for the space transportation platform in both scenarios:

   1. including all the undeployed hosted complement spacecraft (deployment failure) by considering the break-up risk of the space transportation platform and the hosted complement spacecraft.

   2. excluding the hosted complement spacecraft (deployment success) by considering the break-up risk of the space transportation platform only.

f. In case re-entry is foreseen, the compliance with ESSB-ST-U-004 [RD4], in particular with the re-entry casualty risk threshold requirement (10^{-4}), is verified as aggregate risk for the space transportation platform in both scenarios according to their probability of occurrence:

   1. including all the undeployed hosted complement spacecraft (deployment failure),

   2. excluding the hosted complement spacecraft (deployment success).

g. In case the hosted complement spacecraft are not known at time of the development of the space transportation platform (e.g. if provided by a customer different than the developer), worst-case reference mission scenarios are considered for the preliminary compliance verification, e.g. based on release altitude range, orbital lifetime, energy stored (in batteries, propulsion tanks, pressure vessels, etc.), casualty risk contribution of the expected hosted complement spacecraft.