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ESA SPACE DEBRIS MITIGATION COMPLIANCE VERIFICATION GUIDELINES

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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 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 hundreds of catalogued objects, including spacecraft, launch vehicle orbital stages, and fragments re-enter uncontrolled the Earth atmosphere. A few tens of these objects are large and heavy enough to partially survive a destructive atmospheric re-entry.

Through progressive steps, ESA has adopted a regulatory framework to ensure Space Debris Mitigation, which is currently founded on the policy ESA/ADMIN/IPOL(2023)1 [RD01] and the standards ESSB-ST-U-007 [RD02] and ESSB-ST-U-004 [RD03].

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 [RD04], which adopted ECSS-U-AS-10 [RD05] / ISO 24113:2011 as standard for Space Debris Mitigation. ESA/ADMIN/IPOL(2014) [RD04] was revised and confirmed in 2018.

In 2017, ESSB-ST-U-004 [RD03] was adopted by ESA as the standard for the re-entry safety requirements.

In 2019, ISO 24113:2019 [RD06] was published, replacing ISO 24113:2011 with major changes, and followed an by update of ECSS-U-AS-10 [RD05], which adopted all the requirements from ISO 24113:2019 [RD06], with a few clarifications on the interpretation of definitions and requirements.

In 2022, ESA introduced the Zero Debris approach, with the goal to significantly limit the production of debris in Earth and Lunar orbits by 2030 for all future missions. As part of this effort, in 2023 ESA updated both its policy ESA/ADMIN/IPOL(2023)1 [RD02] and the technical requirements applicable to its developments and operations with the adoption of ESSB-ST-U-007 [RD02]. ESSB-ST-U-007 [RD02] is the current ESA Space Debris Mitigation requirements baseline, to which the present handbook refers.

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) [RD07].

ESSB-ST-U-007 [RD02] foresees requirements applicability based on the mission risk scenarios. Risk scenarios are defined on the basis of elements such as:

a. The casualty risk associated to an uncontrolled re-entry of the spacecraft.

b. The operation in the protected regions.

c. The duration of the natural orbit decay from the operational orbit.

d. The cumulative collision probability of the spacecraft with space debris objects larger than 1 cm, once the spacecraft is no longer able to perform manoeuvres.

e. The type of space system (i.e. single spacecraft, constellation, large constellation).

Figure 1-1 provides a schematic representation of this approach for the case of a mission composed by a single spacecraft, showing how, depending on parameters above, some requirements become applicable and have different thresholds. More details on this approach and the selection of the metrics for risk characterisation can be found in ESSB-ST-U-007 [RD02].

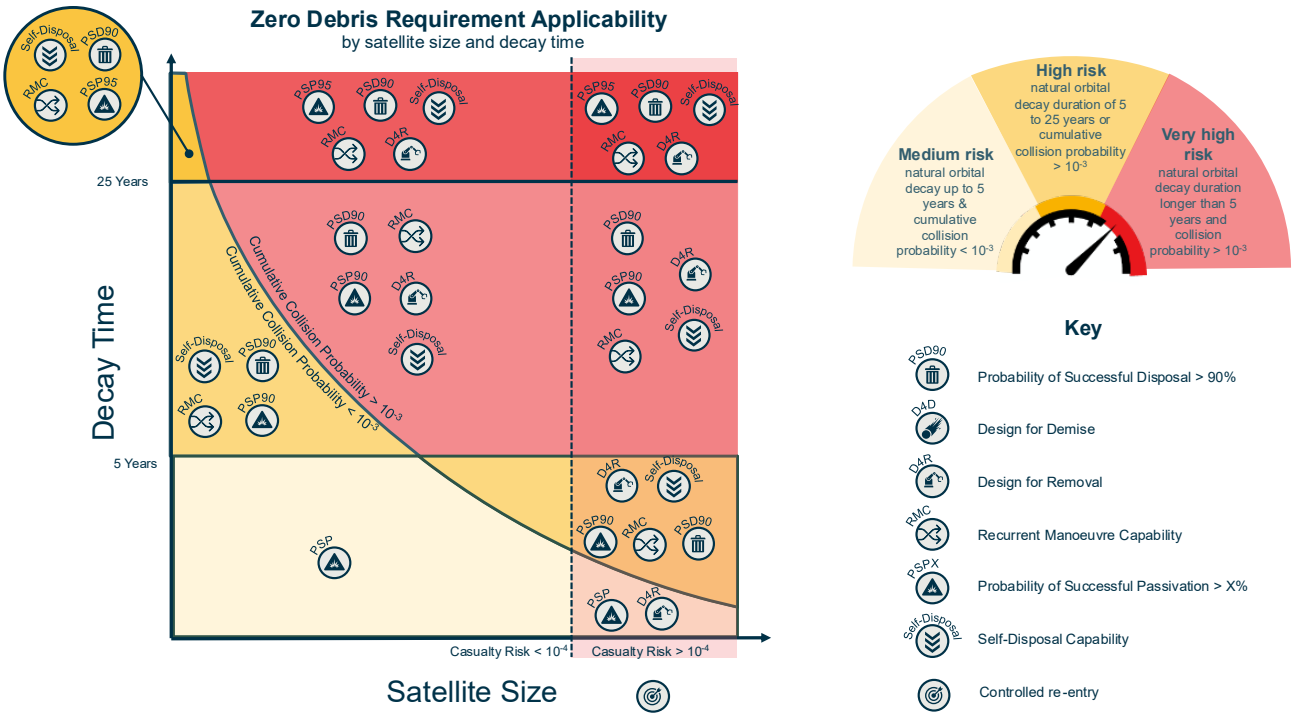

Figure 1-1: Example of requirement applicability based on mission risk scenarios for a single spacecraft mission

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(2023)1 [RD01] and the standard ESSB-ST-U-007 [RD02].

This handbook 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.

The intended users of this handbook are all ESA Projects stakeholders, including ESA Directors, 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 of space systems with respect to the implementation of the ESA SDM requirements.

This handbook is also tutorial, since the implementation of the SDM requirements evolves with time.

The content of this handbook is focused on the implementation and verification of the technical requirements from ESSB-ST-U-007 [RD02], while the ESA principles and process for Space Debris Mitigation, including applicability, roles and responsibilities, are established in ESA/ADMIN/IPOL(2023)2 [RD01], and the documentation requirements, including Space Debris Mitigation Plan (SDMP) and Space Debris Mitigation Report (SDMR), are specified in ESSB-ST-U-007 [RD02].

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3
Terms and abbreviated terms

3.1 Terms defined in other documents

a. For the purpose of this handbook, the terms and definitions from ECSS-S-ST-00-01 [RD08] apply, in particular for the following terms:

1. **deviation**
2. **element**
3. **equipment**
4. **failure**
5. **ground segment**
6. **launch vehicle**
7. **part**
8. **reliability**
9. **risk**
10. **segment**
11. **single point failure**
12. **space debris**
13. **space system**
14. **verification**
15. **waiver**

b. For the purpose of this handbook, the terms and definitions from ESSB-ST-U-007 [RD02] apply, in particular for the following terms:

1. **acceptable collision probability**
2. **approving agent**
3. **break-up**
4. **casualty risk**
5. **close proximity operations**
6. **collision avoidance**
7. **conjunction**
8. **constellation**
9. **controlled re-entry**
10. **cumulative collision probability**
11. **demise**
12. **design for demise**
13. **disposal**
14. **Earth orbit**

NOTE The definition includes unbound Keplerian orbit with Earth at a focal point. However, it is accepted that the requirements can be waived for space objects in an unbounded Earth orbit if, for at least 100 years after the space objects enter the unbounded Earth orbit:

- the assessed risk of the space objects interference with the LEO and GEO protected regions, or
- the assessed risk of the space objects re-entry is less or equal to the corresponding threshold set by the approving agent.

15. **end of life**
16. **end of mission**
17. **ephemeris**
18. **external removal service**
19. **formation flying**
20. **free drift**
21. **geostationary Earth orbit**
22. **graveyard orbit**
23. **inhabitable**
24. **large constellation**
25. **launch vehicle orbital stage**
26. **lunar orbit**
27. **natural orbital decay**
28. **near Earth orbit**
29. **normal operations**
30. **orbit lifetime**
31. **passivate**
32. **probability of successful disposal**
33. **probability of successful passivation**
34. **prognostics**
35. **protected regions**
36. **re-entry**
37. **recurrent manoeuvre capability**
38. **space object**
39. **space surveillance segment**
40. **space traffic coordination**
41. **spacecraft**
42. **uncontrolled re-entry**

c. For the purpose of this handbook, the terms and definitions from ECSS-E-ST-32-02 [RD09] apply, in particular for the following terms:

1. **leak-before-burst**
2. **pressure vessel**

d. For the purpose of this handbook, the terms and definitions from ECSS-E-ST-10-04 [RD010] apply, in particular for the following terms:
   1. **meteoroids**

e. For the purpose of this handbook, the terms and definitions from ECSS-Q-ST-30-02 [RD011] apply, in particular for the following terms:
   1. **failure mode, effects and criticality analysis**

f. For the purpose of this handbook, the terms and definitions from ESSB-ST-U-004 [RD03] apply:
   1. **casualty**
   2. **casualty area**
   3. **declared re-entry area**
   4. **destructive re-entry**
   5. **re-entry probability**
   6. **safety re-entry area**

3.2 Terms specific to the present document

3.2.1 area-to-mass ratio

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

NOTE The area-to-mass ratio can differ over a mission lifetime. The dry mass is usually considered for disposal considerations.

3.2.2 catastrophic collision

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

3.2.3 disposal orbit

final orbit after the end of mission

3.2.4 interference with Earth orbits

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

NOTE 1 The term “interference” is used in this handbook for “interference with Earth Orbits”.

NOTE 2 Permanent presence or temporary crossing of Earth orbits are defined in ESSB-ST-U-007 [RD02].

3.2.5 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).

3.2.6 operational phase

period of time during which a space system performs planned tasks and functions prior to its disposal

3.3 Abbreviated terms

The following abbreviations are defined and used within this document:

Abbreviation	Meaning
ACPL	accepted collision probability level
AOCS	attitude and orbit control system
ARES	assessment of risk event statistics
BLE	ballistic limit equation
CAM	collision avoidance manoeuvre
CCD	coupled charged device
CCSDS	consultive committee for space data systems
CDM	conjunction data message
CDR	critical design review
CFD	computational fluid dynamics
CFRP	carbon fiber-reinforced plastic
COG	centre of gravity
CoM	centre of mass
CONOPS	concept of operations
COPV	composite overwrapped pressure vessel
CPO	close proximity operation
CRP	cathode reference potential
Delta-v	delta velocity
DoD	depth of discharge
DoF	degrees of freedom
DRA	declared re-entry area
DRAMA	debris risk assessment and mitigation analysis
DSMC	direct simulation Monte Carlo
ECSS	European Cooperation for Space Standardization
FDIR	failure detection, isolation, and recovery
EOL	end of life
EPS	electrical power system
FF	formation flying
FMEA	failure modes and effects analysis
FMECA	failure modes, effects, and criticality analysis

Abbreviation	Meaning
FTA	fault tree analysis
GEO	geostationary Earth orbit
GNC	guidance navigation and control
GNSS	global navigation satellite system
GODOT	general orbit determination and optimisation toolkit
GTO	geostationary transfer orbit
HEO	highly eccentric orbit
HET	hall effect thruster
HVI	hypervelocity impact
LED	light-emitting diode
LEO	low earth orbit
LRR	laser retroreflectors
MEO	medium earth orbit
MLI	multi-layer insulation
MMOD	micro meteoroid and orbital debris
NAVAREA	geographical sea area for navigational warnings
NDM	navigation data message
NOM	nominal
NOTAM	notice to airmen
OEM	orbit ephemeris message
OSCAR	orbital spacecraft active removal
PCDU	power conditioning and distribution unit
PRR	preliminary requirements review
PVT	pressure volume temperature
RAAN	right ascension of the ascending node
RAMS	reliability, availability, maintainability, and safety
REX	return of experience
RFW	request for waiver
RUL	remaining useful life
SADM	solar array drive mechanism
SARA	(re-entry) survival and risk analysis
SDM	space debris mitigation
SDMP	space debris mitigation plan
SEL	Sun-Earth Lagrange point
SRA	safety re-entry area
SRR	system requirements review

Abbreviation	Meaning
SST	space surveillance and tracking
TLE	two-line element
TM	telemetry
WCA	worst case analysis

3.4 Conventions

For the purpose of this handbook, the conventions introduced in ESSB-ST-U-007 [RD02] apply, in particular for the following terms:

a. Launch vehicle orbital element.

b. GEO protected region.

c. LEO protected region.

d. Orbital regimes.

In particular, the latter states that

- The wording "**operating in**" applies to spacecraft or launch vehicle orbital stage operating continuously or periodically within a certain orbital region before end of mission: this refers to missions that are functional while being in a certain orbital region; for a spacecraft to be considered as *operating* in a region, it is not required that its orbit is fully within the given region at a given moment in time. For example, a launcher in a GTO is considered to be *operating* in the LEO protected region. Similarly, a spacecraft functional in a HEO such that the perigee altitude is below 2000 km is considered to be *operating* in the LEO protected region.
- The wording "**crossing**" applies to any space object crossing a certain orbital region at any point during the orbital lifetime. For example, a non-functional spacecraft in HEO with a perigee reaching altitudes below 2000 km is considered to be *crossing* the LEO protected region. Interplanetary missions performing a fly-be of the Earth are considered to be *crossing* Earth orbit.
- The wording "**injected into**" applies to any space object when launched or released in a certain orbital region.

4 Space debris mitigation requirements

4.1 Overview

4.1.1 Requirement 5.1: ECSS-U-AS-10

Rationale for the Requirement

The ESA Space Debris Mitigation requirements are built on top of a European baseline from the standard ECSS-U-AS-10 [RD012].

4.2 Space debris release restriction

4.2.1 Requirement 5.2.1.a: Space debris release avoidance

Rationale for the Requirement

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

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, or analysis, to ensure not to release any debris from the spacecraft during normal operations into Earth orbit.

Mitigation Measures

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

a. To design the spacecraft with no elements releasable 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 by selecting materials and technologies (e.g. tanks, surface materials, structures, adhesives, fasteners, MLI) resistant to space environmental degradation over time (e.g. avoiding the release of space debris larger than 1 mm, due to radiation exposure, atomic oxygen erosion, thermal cycling, but excluding those due to atmospheric re-entry and impacts with space debris and meteoroids).

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

e. To design mechanisms to retract extensible appendages (e.g. tethers), 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.

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

4.2.2 Requirement 5.2.1.b: Launch vehicle space debris restriction

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.

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.

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 intentionally released in Earth orbit.

b. To use orbital stages with the autonomous capability to perform de-orbit and passivation operations, in compliance with the other applicable requirements in ESSB-ST-U-007 [RD02] and re-entry safety requirements ESSB-ST-U-004 [RD03].

4.2.3 Requirements 5.2.2.a-c: Space debris from pyrotechnic, solid or hybrid propellant rocket motors, or resulting from environment-induced degradation in GEO

Rationale for the Requirement

See Requirement 5.2.1.a: Space debris release avoidance in 4.2.1.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, analysis, or test, to ensure that pyrotechnics, solid or hybrid propellant rocket motors, or environment induced degradation of adhesive and hook and loop fasteners in GEO, do not lead to generation and release of debris larger than 1 mm into Earth orbit.

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.

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 consider 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.

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

The verification of environment-induced debris release in the GEO protected region is an extension of the material ageing test practices established in the scope of mission success, e.g. accelerated testing in radiation facilities, UV (Ultraviolet) exposure, thermal cycling, etc. The focus is on elements such as adhesives and fasteners that are known to be susceptible to degradation and can lead to large objects being released. For said elements, the test procedure can include the mission design lifetime and be extended up till representing 50 years. In addition, the degradation of the material itself poses a risk for the generation of space debris, which can be documented as part of the testing (e.g. the observation of paint flakes, MLI patches, etc., during the testing), i.e. during the test it is relevant to ensure that no debris particles greater than 1 mm are released due to the long-term exposure.

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 technologies for adhesives and hook and loop fasteners resistant to on-orbit environmental degradation (e.g. due to radiation exposure, thermal cycling).

4.3 Avoiding break-ups in Earth orbit

4.3.1 Requirement 5.3.1a: Intentional break-up

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. Design for demise measures leading to a fragmentation during atmospheric re-entry are not considered intentional break-ups.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

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

4.3.2 Requirement 5.3.2.1.a / ECSS-U-AS-10 7.2.1.1: Accidental break-up probability threshold

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.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to include:
   1. FMECA, in accordance with ECSS-Q-ST-30-02 [RD011], to identify all components storing energy which can lead to an accidental break-up scenario and assess their failure modes and likelihood. The following steps are applied:
      (a) Identify all failure modes that can contribute to the accidental break-up (e.g. pressure vessels, batteries, tanks).
      (b) Extract the likelihood for the failure modes during the whole mission duration (in correlation with existing reliability predictions).
      (c) Compute the total likelihood by cumulation of all the single failure modes (extended to the mission duration).
   2. Fault-Tree Analysis (FTA), when the likelihood of a combination of failures is not negligible, to compute the accidental break-up probability. The following steps are applied:
      (a) Consider as top event “accidental break-up" and identify sub-events and failure modes that can lead to the top event.
      (b) Attribute to each sub-event its likelihood consistently with reliability prediction and FMECA, using one of the methods from ECSS-Q-ST-30 [RD013].
      (c) Compute the total likelihood (extended to the mission duration).

The FMECA considers one failure at a time. Since accidental break-up can be the result of a combination of failures (with significant likelihood), the information extracted from the FMECA is integrated in the FTA, analysing all the possible combinations of causes leading to the undesired accidental break-up.

The risk of burst due to residuals after propulsion passivation is addressed as well in the FTA.

The accidental break-up probability is calculated until end of life.

Mitigation Measures

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

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

b. To design the space system which does not release debris in orbit as a consequence of an internal explosion of one of its components (e.g. through use of containment).

4.3.3 Requirement 5.3.2.1.a / ECSS-U-AS-10, 7.2.1.2: Accidental break-up probability computation

Rationale for the Requirement

The requirement aims at supporting the verification of compliance for the requirement 5.3.2.1 (Requirement 5.3.2.1.a / ECSS-U-AS-10 7.2.1.1: Accidental break-up probability threshold in 4.3.2 and subsequent requirements), to reduce the risk of accidental break-up from design and manufacturing to avoid generation and propagation of debris clouds in Earth orbit.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Methods to Assess Compliance for ECSS-U-AS-10, 7.2.2.1 [RD012] (probability threshold).

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-32-02 [RD09], ECSS-E-ST-10-04 [RD010], ANSI/AIAA S-081B-2018 [RD014], ANSI/AIAA S-080A-2018 [RD015]);
   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).

4.3.4 Requirement 5.3.2.1.a / ECSS-U-AS-10, 7.2.1.3: Health monitoring

Rationale for the Requirement

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

Even though analyses to predict the risk of break-up are performed during the design phases, anomalies, which are unpredictable, or worse conditions than initially assumed, can still occur in orbit resulting in an underestimation of this risk. Therefore, the risk of failures leading to a break-up is updated on regular basis, or when relevant events occur, during the operation phase.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, or analysis, to ensure that health monitoring means are implemented for all the failures leading to the feared events identified in the FMECA and FTA (adequate set of sensors and on-board computer functions, consistency checks). Refer to the recommendations for the requirements 5.4.1.2 (Requirement 5.4.1.2.a / ECSS-U-AS-10, , 7.3.1.3: Disposal criteria in 4.4.3 and the subsequent requirement) for details on the verification methods for heath monitoring. Critical parameters for health monitoring include, but are not limited to:
   1. Temperature at local or unit level (e.g. engines, battery cells);
   2. Pressure at local or unit level (e.g. engines, tanks, pressure vessels);
   3. Absorbed radiation dose (e.g. EEE components regulating power storage units);

4. Duty cycles of units (e.g. battery, EEE components, electromechanical valves, thrusters);
5. Voltage or State of Charge of electrical power storage units (e.g. battery);
6. Performance degradation and wear out of units;
7. Attitude and orbital parameters values and their rates (e.g. rotation angles and position and their variations);
8. In-flight configuration changes (if different from the baseline design, e.g. changes in cold/hot redundancies, non-nominal operational modes);
9. Available consumables (e.g. if anomalous consumption);
10. Available power (e.g. if solar array and battery degradation).

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 the failures leading to the feared events identified in the FMECA and FTA during operations (refer to the ECSS-U-AS-10, 7.2.2.1 [RD012]).

b. To perform periodical assessment of the health of the spacecraft, including trend analysis of critical parameters.

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

4.3.5 Requirement 5.3.2.1.a / ECSS-U-AS-10, 7.2.1.4: Contingency plan

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.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to ensure that worst-case failure scenarios have been identified and captured together with available lessons learnt and applicable responses (e.g. isolation of faulty components).

Mitigation Measures

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

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

4.3.6 Requirement 5.3.2.2.a: Passivation capability

Rationale for the Requirement

The requirement aims at implementing passivation capabilities for spacecraft and launch vehicle orbital stage in Earth orbit, regardless of the selected re-entry strategy. Available passivation capabilities are important against the risk of break-up and debris generation in case a controlled re-entry cannot be successfully performed.

Examples of possible break-up events and causes from elements storing energy 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).

Passivation involves dedicated design implementations, commands (e.g. to relays, valves), operational modes for the relevant units. 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 based on the State-of-the-Art knowhow and technology.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to ensure that passivation capabilities are implemented for all relevant sources of energy.

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).

Existing tests on the selected components, which are publicly available in literature or available to the spacecraft developer, can be used as supporting evidence to the analysis work.

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 or 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).

Mitigation Measures

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

a. To check that the implementation of passivation capabilities is in line with Annex E.

b. For pressure vessels, to vent or 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.

c. 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).

d. 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.

e. 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.

4.3.7 Requirement 5.3.2.2.b: Passivation condition

Rationale for the Requirement

The requirement aims at executing the passivation operation before end of life, with the only exception of a controlled re-entry. A controlled re-entry is typically not compatible with a spacecraft or launch vehicle full passivation operation timeline. Passivation, although helpful for ground safety and environmental impact reduction due to release of toxic substances, can be not completely feasible when a controlled re-entry is executed since, for example:

a. For power passivation, battery depletion needs an amount of time, which is not compatible with power supply 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 not always possible since the re-entry impact is planned over an Ocean.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to ensure that passivation can be executed before end of life.

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.3.2.2.a: Passivation capability in 4.3.6.

4.3.8 Requirement 5.3.2.2.c: Probability of successful passivation

Rationale for the Requirement

The requirement aims at guarantying, on top of the availability of passivation capabilities, a minimum probability of successful passivation. The minimum probability of successful passivation for spacecraft or launch vehicles is 0,90, while a more demanding threshold of 0,95 is requested for high-risk scenarios as defined in Section 4.1.3 in ESSB-ST-U-007 [RD02], i.e. LEO with natural orbital decay longer than 25 years, and GEO.

The probability of successful disposal includes the probability of successful passivation.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Methods to Assess Compliance for Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1.

In the case that a controlled re-entry is the nominal disposal strategy, the probability of successful passivation is not meant to be demonstrated in contingency scenarios (e.g. after major failure of the propulsion system that prevents the execution of the controlled re-entry), but rather it is important to show that in nominal conditions (e.g. considering the expected mission duration), the critical components for the passivation implementation are expected to function with a high success rate.

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1.

4.3.9 Requirement 5.3.2.2.d: Passivation implementation

Rationale for the Requirement

The requirement aims at prioritising two implementation actions for passivation. A safe level of passivation is reached when any remaining source of energy cannot cause an accidental break-up.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to ensure that the passivation implementation for all the energy sources and the approach chosen, do not trigger on-orbit break-up risk.

b. Analysis, to provide evidence that, within the boundary conditions of the scenario at time of, and after, the passivation execution, the passivation implementation and sequence allow to completely and irreversibly deplete the on-board energy storages preventing future loadings from the energy sources (Requirement 5.3.2.2.d: Passivation implementation, bullet 1), or demonstrate the final level of residual energy sources is safe (Requirement 5.3.2.2.d: Passivation implementation, bullet 2).

Mitigation Measures

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

a. To implement recommendations provided in Annex E.

4.3.10 Requirement 5.3.3.1.a: Collision risk assessment during design

Rationale for the Requirement

The requirement aims at informing about the likelihood of collisions leading to mission termination (impacts with objects larger than 1 cm) and generation of new space debris (impacts with objects larger than 1 mm).

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to assess the probability of catastrophic collisions leading to complete break-up due to space debris or meteoroid impacts, including:
    1. Definition of the mission phase(s) of the space system (e.g. launch phase, operation phase, disposal phase);
    2. Definition of the trajectory, free drift trajectories after orbit injection, end of mission, and disposal, and during normal operations, until re-entry or up to 100 years;
    3. Definition of the catastrophic collision threshold of space debris and meteoroids leading to a complete break-up (as per criterion in C.2.3.1);

4. Evaluation of the space debris and meteoroids flux for the phases under analysis, e.g. from the ESA tool MASTER;
5. Determination of the catastrophic collision risk, based on the number of impacting particles that exceed the catastrophic collision threshold (as per criterion in C.2.3.1);
6. Determination of the share of the risk that can be mitigated by conducting active collision avoidance.

b. Analysis, to assess the probability of non-catastrophic collisions leading to partial break-up due to space debris or meteoroid impacts, including:
1. Definition of the mission phase(s) of the system (e.g. launch phase, operation phase, disposal phase);
2. Definition of the trajectory, free drift trajectories after orbit injection, end of mission, and disposal, and during normal operations, until re-entry or up to 100 years;
3. Definition of the appropriate cratering equation(s) and relevant failure mode (e.g. release of number of particles larger than 1 cm, total mass release);
4. Evaluation of the space debris and meteoroids flux for the phases under analysis, e.g. from the ESA tool MASTER;
5. Determination of the associated crater size distribution for non-catastrophic collisions;
6. Determination of the risk to encounter defined failure mode for non-catastrophic collisions;
7. Determination of the share of the risk as function of impactor size, velocity and directionality, which can be used to collect a database of spacecraft sensitivities to non-catastrophic collisions and to possibly mitigate risk by design.

In the verification of the requirement, the computation of the impact rates for the two size thresholds, and in the four trajectory conditions can be considered sufficient at a first stage and the additional steps above can be pursued if specific concerns arise from the analysis of the impact rates as represented in Figure 4-1. A more detailed vulnerability assessment is specifically requested to demonstrate the probability of successful disposal as explained in Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1.

Mitigation Measures

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

a. To design the spacecraft with sufficient shielding to protect relevant units, in order to prevent debris generation in case of collision with untrackable space debris or meteoroids.

b. To accommodate relevant 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.

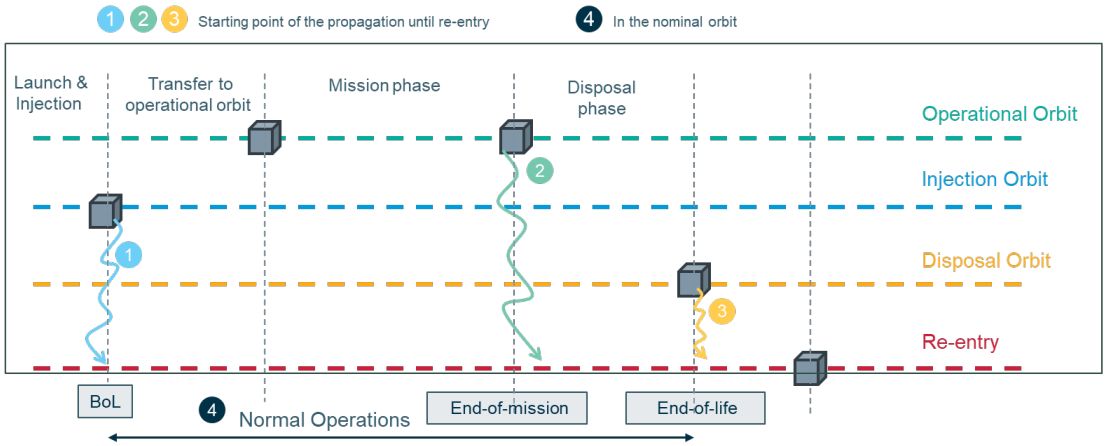

Figure 4-1: Representation of the simulation scenarios for the verification of the requirement 5.3.3.1.a

4.3.11 Requirement 5.3.3.2.a: Cumulative collision probability in case of object release

Rationale for the Requirement

The requirement aims at controlling the collision risk during on-orbit release of a spacecraft from a launch vehicle or another spacecraft. During the launch and orbit insertion phase, as well as during operations involving the release of spacecraft or elements, collision risk arises from the proximity. A challenging aspect is that typically no orbit information is available before the release event and larger uncertainties can be involved.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:

1. Identify the responsibility for the assessment of the cumulative collision probability for the release event (e.g. the launch service provider, all involved parties);
2. Identify, during design, roles and responsibilities, e.g. if the responsibility for launch collision avoidance is shifted from a launch service provider (typically covering a few hours after launch) to an operator for releasing spacecraft;
3. Have a coordinated release sequence such as to minimise mutual collision risk;
4. Ensure that the launch service provider obtains all the information related to orbital elements for the release, potential parking orbit, powered flight portions and forwards it to the responsible space surveillance segment (to mitigate potential colocation and potentially conflicting information during close approaches);
5. Share the release sequence as widely (preferably publicly) as possible;
6. Ensure that collision risk with active spacecraft can be addressed in the early mission phase.

b. Analysis, to:

1. Verify, considering dispersion of possible trajectories, that the cumulative collision probability for each possible combination involving a releasing object (i.e. launch vehicle orbital stage, or spacecraft releasing other spacecraft) and a released object remains below 10^{-6} (the cumulative collision probability is computed for a 3-day time after the release

based on the assumption that involved spacecraft can attain manoeuvrability in that period);

2. Identify, in early mission design phase, a plausible (and likely) release scenario and demonstrate that the means to conduct such an analysis (as indicated in b.1 above) exist.

Mitigation Measures

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

a. To release objects such that mutual collision risks are minimised, e.g. release on slightly different orbits.

b. To manoeuvre the releasing spacecraft or launch vehicle orbital stage as quickly as possible away from the released objects.

c. To attain full manoeuvring capability for the involved spacecraft as quickly as possible.

4.3.12 Requirement 5.3.3.2.b: Launch collision avoidance with inhabitable space objects

Rationale for the Requirement

The requirement aims at defining a collision probability threshold to protect inhabitable space objects, which can be in the vicinity of a launch vehicle. It is assumed that the orbits of inhabitable space objects are generally known before the launch.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to check the responsibility for the assessment of the cumulative collision probability against inhabitable space objects (i.e. the launch service provider).

b. Analysis, to:

1. Plan the launch at a date and time where potential close approaches are avoided;
2. Verify, considering dispersion of possible trajectories, that the cumulative collision probability between the launch vehicle orbital stage and each inhabitable space object remains below 10^{-6} for a 3-day period after launch.

Mitigation Measures

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

a. To select the launch window such as to minimise collision risk with inhabitable space objects.

b. To dispose the launch vehicle orbital stage as quickly as possible.

c. To perform a dedicated conjunction screening as soon as the first orbit information for the launch vehicle orbital stage becomes available.

4.3.13 Requirement 5.3.3.2.c: Recurrent manoeuvre capability

Rationale for the Requirement

The requirement aims at defining the criteria to have recurring manoeuvring capability for a spacecraft to conduct collision avoidance manoeuvres. The requirement applies both to LEO and GEO Protected Regions, and specific mission concepts where a spacecraft can experience close approaches. Manoeuvre capability allows maintaining assigned slots and mitigate potential collision risk in an entire region, like GEO and within constellations. The consequences of a collision can be long-lived fragments that can represent an irrecoverable risk increase for other current and future operators.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to ensure that the spacecraft is equipped with units and functions enabling to perform collision avoidance and disposal manoeuvres.

b. Analysis, or test, to:

   1. Demonstrate that enough resources are available to perform:
      (a) Orbital maintenance manoeuvres, if needed.
      (b) Collision avoidance manoeuvres.
      (c) Disposal manoeuvre at end of mission.

   as requested by Requirement 5.3.3.2.d: Collision avoidance operational impact in 4.3.14 and Requirement 5.3.3.3.d: Collision avoidance capability assessment in 4.3.20.

   2. Demonstrate that a recurring manoeuvring capability is available to ensure separation from possible colliding objects, considering:
      (a) The timeliness requirements defined in Requirement 5.3.3.3.h: CAM first-time capability in 4.3.24 and Requirement 5.3.3.3.j: Collision avoidance operations timeliness in 4.3.26 .
      (d) The risk reduction defined in Requirement 5.3.3.3.a: Acceptable collision probability threshold in 4.3.17, Requirement 5.3.3.3.b: Acceptable collision probability threshold in congested regions in 4.3.18 and Requirement 5.3.3.3.i: CAM effect in 4.3.25.
   3. Assess the remaining orbital lifetime of the operational or disposal orbit, according to the verification Requirements 6.2.f-g: Probabilistic assessment of the orbital lifetime in 5.1.5;
   4. Assess the cumulative collision probability with space objects larger than 1 cm until end of life, assuming no manoeuvring capability, following the approach in Annex Section C.3.

The population of objects larger than 1 cm is used even if usually objects smaller than 10 cm are not tracked and, therefore, they are not the subject of collision avoidance manoeuvres. The cumulative collision probability with objects larger than 1 cm is used as a proxy of the risk level of a mission.

Mitigation Measures

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

a. To plan the mission outside the domains the requirement is referring to, if the intended system has no (or low, unknown, unreliable) capability to perform collision avoidance and disposal manoeuvres.

4.3.14 Requirement 5.3.3.2.d: Collision avoidance operational impact

Rationale for the Requirement

The requirement aims at assessing the cumulative impact on the design and operation of a spacecraft due to collision avoidance. This requirement is complemented with the Requirement 5.3.3.2.e: Expected number of conjunctions in 4.3.15, which covers the criteria for the calculation of the expected number of collision events during the spacecraft lifetime.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to prove, during the design phase, that the impact of responding to collision warnings is assured.

The analysis can have better accuracy if it is performed by the spacecraft developer in cooperation with the selected operator as the impact of collision events partly depend on the operational context and available resources. Alternatively, it is important that the assumptions taken by the spacecraft developer are clearly communicated to the selected operator to match with the available, or needed, facilities.

The impact of a single "generic" collision warning event can be assessed and scaled by the results from Requirement 5.3.3.2.e: Expected number of conjunctions in 4.3.15. Typical topics, which are included in the analysis to respond to a collision warning, are:

a. Operational effort:
    1. Processing of conjunction information and updates and translation into required mitigation activities;
    2. Coordination with other operators in case of a conjunction event with active spacecraft;
    3. Design, planning and implementation of mitigation activities (i.e. collision avoidance manoeuvre);
    4. Validation of mitigation activities with respect to safety, mission compatibility, and collision risk reduction effectiveness;
    5. Monitoring of execution of mitigation activities, including reporting to other parties.

The response timeliness of these activities is important, e.g. if the response capability is limited to working hours only, or 24/7 response capability is needed.

a. Ground system availability and resources:
    1. Communications links with the spacecraft to support uplink, monitoring and execution of collision avoidance activities;
    2. Tools and ground systems to allow the processing of collision warning and their production and execution.

b. Space system availability and resources:
    1. Impact on consumables (e.g. propellant) or life-limited items (e.g. thrusters) to perform a collision avoidance manoeuvre;
    2. Impact on system availability due to the mitigation activities, either related to the spacecraft routine engineering activities, or to the payload ability to execute its mission;
    3. Impact of mitigation activities on orbit maintenance and the subsequent impact on the ability of the mission to achieve its goals.

Mitigation Measures

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

a. On-board autonomy to allow the spacecraft to manoeuvre in response to provided CDMs, considering the resulting ability to predict, monitor, communicate and validate collision avoidance.

b. On-board autonomy to simplify the execution of a ground-commanded mitigation activity and therefore reduce the required operational effort.

c. Automation of ground systems to allow many of the ground activities to be executed autonomously, reducing the operational burden, provided that the system has demonstrated robust validation.

d. To design on-board propulsion capabilities to minimise the impact of mitigation activities, e.g.:
   1. Payload with minimum impact in case of collision mitigation activities;
   2. Chemical propulsion to allow delivery of Delta-v for collision avoidance at any point in the orbit without multiple burns spread over many orbits;
   3. Electric propulsion with high enough thrust level to be compatible with the required timeliness of collision avoidance operations as defined in Requirement 5.3.3.3.j: Collision avoidance operations timeliness in 4.3.26.

The mitigation measures for the requirement are varied and outcome of trade-off between operational complexity and on-board complexity.

In addition, it can be beneficial to update the assessment during the development phase and during operations to reflect changes in the environment. The Requirement 5.4.1.2.f: Probability of successful disposal re-assessment occurrences in 4.4.9 provides some reference milestones for the re-computation of the expected probability of successful disposal that can be adapted also for the assessment of the collision avoidance procedures. ESA makes available different resources to perform such updates:

- ESA's fragmentation frontend [RD017] provides an overview of historical fragmentation and the expected effect on the collision probability at different altitudes and at different epochs (including future ones).
- ESA's DISCOSweb [RD018] contains information on tracked objects (Objects table), on the operational orbits of spacecraft (Destination Orbits table), and on constellations (Constellations table) that can be used to monitor the change in space population in the orbital ranges of interest.

Revising the collision avoidance strategy along the mission development can provide a better picture of the resources to be dedicated to operations, not only in terms of Delta-v spent (usually small), but mostly in terms of coordination and monitoring effort [RD019].

4.3.15 Requirement 5.3.3.2.e: Expected number of conjunctions

Rationale for the Requirement

The requirement aims at assessing the impact of collision avoidance measures on the concept of operations (i.e. in terms of resources and payload availability) and the burden imposed on the operations of other spacecrafts. Orbital regions with high debris density, but also repeating conjunctions typically create a high burden in terms of collision avoidance operations. In the latter, especially if active coordination with one (or several) operators is needed. Studying the orbital neighbourhood in advance can lead to significant optimisation gains.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to:
   1. Identify the individual mission phases, including the nominal mission phase and spanning until the re-entry or at most 100 years (the variability of the 11-year solar cycle is not accounted, while a working assumption with mean, or high, or low, solar and geomagnetic activity over the analysis span is sufficient);
   2. Estimate for the identified analysis span and the different mission phases the expected number of conjunction events for the designed mission above a collision probability level of 10^{-4} and 10^{-6}, based on either the latest MASTER reference population or the latest population forecasts (see Annex B);
   3. Estimate, for spacecraft operated in LEO, for the identified analysis span and the different mission phases the expected number of conjunction events triggered for other spacecraft operating within the orbital neighbourhood, identifying the share of active vs. operational vs. debris objects in the orbital neighbourhood and applying that scaling to the assessed overall manoeuvre rate (a more detailed explanation can be found in the ARES Technical Note [RD020]);
   4. Identify also the share of events that can be associated with inhabitable space objects.

Mitigation Measures

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

a. To review the obtained statistics about operators in the orbital neighbourhood and optimise mission profile for systematic encounters, such as close coordination with constellation operators.

b. To optimise the intended orbit for the nominal mission phase.

c. To optimise the intended disposal orbit.

4.3.16 Requirement 5.3.3.2.f: Expected number of conjunctions for constellations

Rationale for the Requirement

The requirement aims at assessing the aggregate impact of collision avoidance measures for a constellation, e.g. to estimate the effort in terms of operations and coordination with other operators and highlight potential systematic interactions, including the contribution coming after end of life.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to:
   1. Compute the aggregated collision probability with trackable objects for each spacecraft of the constellation;
   2. Compute the aggregated number of estimated manoeuvres according to the methodology applied in Requirement 5.3.3.2.e: Expected number of conjunctions, in 4.3.15 which is repeated for all the spacecraft of the constellation.

For both analyses, the numbers obtained for each individual spacecraft are split into the contribution when the spacecraft is functional and when it is not. The first contribution is summed across all the

constellation members to obtain the assessment during normal operations, whereas the second contribution is summed across all the constellation members to obtain the assessment after disposal.

As the spacecraft of the constellation can operate at very different epochs, the space debris environment at the initial and final epochs can be used for the whole constellation. In case the final epoch is beyond the available space debris environment predictable data, the latest available data can be used.

The assessment is meant to cover both the exposure of the constellation to the space debris environment and the burden to other operators. Conjunctions within the constellation do not need to be considered for the verification of this requirement as they are addressed by the Requirement 5.3.3.3.c: Acceptable collision probability within a constellation in 4.3.19.

The aggregation of the results is performed considering all the planned spacecraft in the constellation, over its entire life cycle.

4.3.17 Requirement 5.3.3.3.a: Acceptable collision probability threshold

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. The absolute upper risk threshold of 10^{-4} per event is chosen to ensure a sufficiently low collision risk throughout a mission lifetime, irrespective of its orbit. This value is widely accepted in the sector as a sustainable balance between safety and mission impact [RD089]. In case of conjunctions with other operational spacecraft, the implementation of manoeuvres is subject to coordination with the other operators, as defined in Requirement 5.3.3.3.o: CAM coordination in 4.3.29.

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 is shared with Space Surveillance and Tracking (SST) provider and other operators and updates are done on a regular basis (see also Requirements 5.3.3.3.k-m: Collision avoidance procedure information in 4.3.27 and Requirement 5.3.3.5.f: Ephemerides frequency in 4.3.41).

b. Analysis, to:
   1. Assess that the CAMs adhere to the defined threshold and their effect on the mission design and operations (a suggested procedure is defined in Annex B.2 and Requirement 6.2.c: Space object population for collision avoidance planning in 5.1.2, including further guidelines on the definition of the spacecraft radius and the population to be used for the assessment). For missions not crossing into LEO, metrics in addition to the collision risk probability can be considered as the density of space debris can be low. For example, many GEO operator adopt a geometric distance criterion in their orbital slot, which implies collision of propagability below 10^{-6}. Further examples are provided in [RD021];
   2. 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 (see also Requirement 5.3.3.3.e: Collision probability

computation during operation in 4.3.21 and Requirement 5.3.3.5.i: CCSDS format in 4.3.44);

3. Perform conjunction screening against ephemerides made publicly available by other operators (ephemerides versus ephemerides screening in-house);
4. 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;
5. Identify and assess other possible conjunctions in the modified orbit after a CAM planning (see also Requirement 5.3.3.3.f: Collision avoidance temporal effectiveness in 4.3.22);
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 generate new 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.

Mitigation Measures

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

a. To select an operational orbit where the number of existing space objects is minimised (low collision risk), when still compatible with the mission objectives.

b. To perform a CAM, in case the probability of collision is above the defined threshold, in coordination with other relevant operators (see also Requirement 5.3.3.3.o: CAM coordination in 4.3.29), and under the condition that the probability of other conjunction events, resulting from the modified orbit, is below the threshold (see also Requirement 5.3.3.3.f: Collision avoidance temporal effectiveness in 4.3.22).

4.3.18 Requirement 5.3.3.3.b: Acceptable collision probability threshold in congested regions

Rationale for the Requirement

The requirement aims at constraining the aggregated collision risk in congested orbital regions by setting a more stringent reaction threshold then in Requirement 5.3.3.3.a: Acceptable collision probability threshold in 4.3.17. A target risk reduction is introduced to reflect the influence of varying collision geometry and uncertainty characteristics in different orbital regimes.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to identify the mission parameters affecting the estimation of collision probability, including the intended orbit, representative hard-body radius and orbit uncertainties.

b. Analysis, to:

1. Obtain the spatial density for the orbital region during the nominal mission phase to determine the applicability of the requirement. The spatial density graph from ESA's Annual Space Environment Report, Figure 2.9 in [RD016], can be used, or a more detailed assessment can be performed using the MASTER model, e.g. to consider also the dependence on the inclination. For eccentric orbits, the space debris density along the orbit can vary significantly and the average density generally accounts for the time spend in the different density regions while averaging;

2. Estimate the accepted collision probability level which results in 90 % risk reduction (the suggested procedure is described in [RD022]).

The verification methods for the requirement are in addition to Requirement 5.3.3.3.a: Acceptable collision probability threshold in 4.3.17.

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.3.3.3.a: Acceptable collision probability threshold in 4.3.17.

4.3.19 Requirement 5.3.3.3.c: Acceptable collision probability within a constellation

Rationale for the Requirement

The requirement aims at quantifying and limiting the risk of intra-constellation collisions, considering potential failures and different mission phases. A possible break-up within a constellation can result in high severity event.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to ensure satisfactory knowledge of the spacecraft states, including their covariance, also in line with the requirement Requirements 5.3.3.3.n: Trajectory catalogue for constellations in 4.3.28.

b. Analysis, to assess the collision probability for each spacecraft, with the following considerations:
   1. The assessment is performed following the same instructions as in Requirement 5.3.3.3.e: Collision probability computation during operation in 4.3.21, i.e. using probabilistic methods and modelling uncertainties in the position and velocity of the spacecraft (see 5.3.3.3.e for full specifications);
   2. As the requirement refers to normal operations, the nominal planned constellation size is considered (e.g. once deployment is complete) and the nominal orbital states. The assessment is performed considering probabilistically the presence of not functional spacecraft, in line with the estimated probability of successful disposal reported in 5.4.1.1 (Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1 and the subsequent requirements);
   3. The contribution from active-vs-active conjunctions can be reduced if an estimation of the probability of successful implementation of collision avoidance manoeuvres is available from the analysis in Requirement 5.3.3.3.d: Collision avoidance capability assessment in 4.3.20.

While this requirement is for operation, it has as well design implications related to the selection of suitable AOCS units and the targeted level of probability of successful disposal, which are assessed during the early design phase (e.g. through a constellation simulator).

During normal operations, the cumulative collision probability of each spacecraft is monitored as part of the routine constellation management. In case of systematic violation of the defined threshold, the approving agent is informed.

Mitigation Measures

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

a. To use GNSS receivers or similar approaches to obtain a good accuracy on each spacecraft state and reduce the associated covariance.

b. To ensure a high level of post mission disposal to limit the chance of collisions within the constellation.

c. To consider the implementation of a preliminary manoeuvre to move the spacecraft from the operational orbit before the initiation of the rest of the disposal strategy.

4.3.20 Requirement 5.3.3.3.d: Collision avoidance capability assessment

Rationale for the Requirement

The requirement aims at assuring the capabilities of the entire system, including both ground and space segment, to perform recurrent manoeuvres for collision avoidance and disposal, and the resulting impact on system design, resources, and operation.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to confirm that the system can perform recurrent manoeuvres for collision avoidance and disposal, including:
    1. Resources allocation and design impact, by considering and extending the analysis performed for Requirement 5.3.3.2.d: Collision avoidance operational impact in 4.3.14 on availability of space and ground segment resources, e.g. considering on-board propellant budget, life-limited items, operational personnel, system resources on-ground, management of feedback from the supplier;
    2. Response timeliness, by considering and extending the analysis performed for Requirement 5.3.3.3.j: Collision avoidance operations timeliness in 4.3.26, including:
        (a) Timeliness (including working hours) of key personnel involved in the ground and space surveillance segments;
        (b) Timeliness of access to any critical systems (particularly ground stations for uplink of collision avoidance activities);
        (c) Duration for the space segment to complete execution of commanded collision mitigation activities.
    3. Data accuracy and quality, by considering and extending the analysis performed for Requirement 5.3.3.5.c: State vector quantification frequency in 4.3.38 and Requirement 5.3.3.5.d: Position accuracy in 4.3.39, including the impact on accuracy of propagation of chaser position caused by the tracking frequency of a given object. For common space surveillance segments serving multiple missions, a generic assessment of their capabilities is usually already available.
    4. System failure probability, by combining the assessment of failure probability of subsystems and units (based on their reliability data) to show the robustness to failures against the ability to perform collision avoidance for the entire system, i.e. space segment and ground segment, and any potential system or interaction between systems.

The analysis is performed prior to entry into service of the system and re-assessed any time there is a deviation. Deviations can be planned, or result from a failure, due to internal or external cause, which can change the capability or performance of the system.

Mitigation Measures

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

a. For resources allocation and design impact:
   1. To refer to the mitigation measures performed for Requirement 5.3.3.2.d: Collision avoidance operational impact in 4.3.14;
   2. To ensure sufficient margins during the design phase;
   3. To consider novel methods such as automation or modification of operations concept, if validated, in case resource allocation is not sufficient.

b. For response timeliness:
   1. To refer to the mitigation measures performed for Requirement 5.3.3.3.j: Collision avoidance operations timeliness in 4.3.26**;**
   2. To ensure constant coverage by personnel;
   3. To consider automation of systems, if validated, which can result in meeting timeliness even without increase of personnel availability;
   4. To design all systems to simplify the process to ensure that a high level of automation can be achieved.

c. For data accuracy and quality:
   1. To check if already handled in the frame of the qualification of the selected space surveillance segment, which can exempt further analysis from the spacecraft developer and operator;
   2. To agree with the selected space surveillance segment on tailored performances in case of specific needs from the mission.

d. For system failure probability:
   1. To consider redundancy of systems, ideally with automatic failover in case of an issue, to minimise the need for active human intervention.

4.3.21 Requirement 5.3.3.3.e: Collision probability computation during operation

Rationale for the Requirement

The requirement aims at determining collision avoidance actions through probabilistic assessments, which are more accurate than distance-based methods, since they capture the involved uncertainties.

Methods to Assess Compliance

The verification methods used to assess compliance are:

a. Review-of-design, to:
   1. Ensure that a selected collision probability method is valid in the envisaged operational regime. Annex N of [RD089] provides an overview of different methods available for the computation of the collision probability;
   2. Ensure that the major contributing factors and their uncertainties are considered by the method in relation to the orbital region, including:

(a) the uncertainties in the position (in all cases) and velocity (particularly in case of low-velocity encounters, e.g. in GEO and other high-altitude regions) of the involved space objects.

(b) the relative distance between the trajectories during the conjunction.

(c) the dimensions of the space objects when known, and

(d) space weather forecasts (most relevant when orbits are crossing the LEO region).

4.3.22 Requirement 5.3.3.3.f: Collision avoidance temporal effectiveness

Rationale for the Requirement

The requirement aims at minimising the risk of the execution of a CAM inducing further probable collisions, which later need additional CAMs and, therefore, increase the overall collision risk with respect to the existing scenario before the CAM execution. Furthermore, the introduced uncertainties and changes in predictions can affect the manoeuvre planning of other operators.

Methods to Assess Compliance

The verification methods used to assess compliance are:

a. Review-of-design, to ensure that a planned CAM can be executed with sufficient accuracy.

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

1. Screening potential manoeuvres against an object catalogue for possible new conjunctions;
2. Planning of subsequent manoeuvres in case additional high-risk conjunctions are identified;
3. Evaluating potential manoeuvres based on different factors such as risk, chaser type, uncertainties of subsequently generated conjunctions.

Mitigation Measures

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

a. To share ephemerides including planned manoeuvres, as outlined in Requirements 5.3.3.3.k-m: Collision avoidance procedure information in 4.3.27, to avoid changes in predictions of other operators.

4.3.23 Requirement 5.3.3.3.g: Ephemerides first-time availability

Rationale for the Requirement

The requirement aims at incentivising a quick identification after insertion, e.g. by the space surveillance segment (see also Requirement 5.3.3.5.e: On-orbit identification in 4.3.40) or combined space, ground and surveillance segments, which is a pre-requisite to provide timely ephemerides for improved conjunction assessments to other operators.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:

1. Identify whether available (or preferred) launch opportunities imply a multi-payload or rather a single spacecraft launch. In the latter case, identification usually happens quickly

and unambiguously, whereas in the former scenario more scrutiny is needed to avoid potential misidentification;

2. Establish the means to exchange ephemerides with a space surveillance segment or other operators as soon as possible after orbit injection.

Mitigation Measures

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

a. To optimise design and introduce functionality to enhance trackability and identification (see also Requirement 5.3.3.5.e: On-orbit identification in 4.3.40).

b. To coordinate, if applicable, with other operators sharing the same launch and identify potential for misidentification between spacecraft and potential to mitigate this.

c. To adopt own orbit determination capabilities to reduce delays coming from relying on third-party tracking.

4.3.24 Requirement 5.3.3.3.h: CAM first-time capability

Rationale for the Requirement

The requirement aims at addressing the collision risk during early operations in a timely manner.

Methods to Assess Compliance

The verification methods to assess compliance are:

a. Review-of-design, to:

1. Identify the chain of tasks (and related timings) during LEOP needed to enable collision avoidance capabilities, both on the space and on the ground segment;
2. Ensure that the spacecraft sub-systems required to execute a CAM are operational within 2 days (e.g. early commissioning of the propulsion system);
3. Ensure the availability of orbital data to allow screening for potential collisions (see also Requirement 5.3.3.3.g: Ephemerides first-time availability in 4.3.23).

In case of a launch constraining an initial operational capability (e.g. a rideshare launch), the requirement objective can be met, if the launch service provider guarantees the release of the spacecraft on an orbit which has been prior assessed not resulting in collision risk above the threshold until the spacecraft has its own capability to perform CAMs.

To verify the applicability of this requirement, with regards to the natural orbit decay, refer to the procedure described in Requirement 5.4.2.3.a: LEO protected region clearance – objects operating in LEO in 4.3.21.

4.3.25 Requirement 5.3.3.3.i: CAM effect

Rationale for the Requirement

The requirement aims at performing effective collision avoidance manoeuvres by targeting risk reduction by two orders of magnitude below the defined collision probability threshold. The value includes margins due to the limit of orbit accuracy and trajectory forecasting. A threshold based on the risk reduction, instead of radial separation, is more general for different orbital regimes.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design (for the operator), to:
   1. Ensure that valid criteria are known and defined concerning the target collision probability threshold achieved when performing a collision avoidance manoeuvre;
   2. Validate the concept of operations concerning the timeline between alerts and collision avoidance manoeuvre implementation, also in consideration of Requirement 5.3.3.3.j: Collision avoidance operations timeliness in 4.3.26 (i.e. need to implement a collision avoidance manoeuvre in less than 12 hours).

b. Analysis (for the spacecraft developer), to determine the minimum amount of resources (propellant mass, Delta-v) necessary to perform the number of collision avoidance manoeuvres expected until disposal, considering the reaction threshold from the Requirement 5.3.3.3.a: Acceptable collision probability threshold in 4.3.17 (and Requirement 5.3.3.3.b: Acceptable collision probability threshold in congested regions in 4.3.18, if applicable) and a manoeuvre size compatible with the reduction of the collision probability by two orders of magnitude.

The DRAMA/ARES tool can be used for such analysis as it is aligned with the Verification and Validation requirements 6.2.a and 6.2.c on space debris population models and offers the option of defining a collision avoidance strategy based on risk reduction **[**RD022**]**.

4.3.26 Requirement 5.3.3.3.j: Collision avoidance operations timeliness

Rationale for the Requirement

The requirement aims at addressing the timeliness for collision risk as a crucial factor. It is relevant from the perspective of coordination needs when two active spacecraft from different operators are involved. Moreover, inherent uncertainties in the contributing factors leading to a conjunction warning can result in sudden changes that can require action on short notice. Various design decisions can result in constraints impeding the timeliness with respect to short-notice reaction capabilities, such as platform limitations, available ground system support.

Methods to Assess Compliance

The verification methods to assess compliance are

a. Review-of-design, to:
   1. Guarantee that, in nominal conditions, an assessment is performed within 4 hours after a warning has been received (in LEO), which involves at least an automated assessment with respect to the defined avoidance action decision criteria;
   2. Assess design drivers impacting the timeliness of the implementation of collision avoidance manoeuvres in nominal conditions. Such design drivers can be influenced both by internal factors and external factors or third-party providers (e.g. the update cycles of space surveillance segments for requested screenings);
   3. Identify potential and required trade-offs to reduce reaction times, including options for automation and externalisation of tasks;
   4. Demonstrate that the mission targets operations with at least 12 hours (including on-call periods) of coverage to react to alerts;

5. Define procedures to communicate the inability to manoeuvre (e.g. also in case of no propulsion capabilities) or to react to a warning (e.g. in case of anomalies or too short lead time).

b. Analysis, to:

1. Guarantee that the selected propulsion system is able to implement a collision avoidance manoeuvre with the desired effect (e.g. reduction of the collision probability by two orders of magnitude as in Requirement 5.3.3.3.i: CAM effect in 4.3.25) within 12 hours, under a range of conditions (e.g. considering the presence of eclipse periods);
2. Estimate the number of missed collision avoidance manoeuvre considering the actual availability, process duration (including the time for the manoeuvre execution), and the performance of the space surveillance segment (in terms of late high risk event notification).

Mitigation Measures

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

a. To actively communicate spacecraft status to other operators.

b. To actively communicate procedure and timeliness aspects thereof.

c. To share ephemerides with other operators in line with the approach to information distribution in Requirements 5.3.3.3.k-m: Collision avoidance procedure information in 4.3.27.

d. To increase system automation.

e. To lower involved cost by pooling resources, offloading or involving a third-party in providing such a service.

4.3.27 Requirements 5.3.3.3.k-m: Collision avoidance procedure information

Rationale for the Requirement

The requirements aim at ensuring that in the case of a conjunction an operator has knowledge of any potential action from the other object, such that an avoidance manoeuvre can be planned without the risk of unknown trajectory changes from the other object, which can invalidate the avoidance plan.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:

1. Ensure sharing of ephemerides that include all possible scenarios, such as nominal orbit, free drift trajectory, and planned manoeuvres (5.3.3.3l);
2. Ensure the inclusion in the procedure of the description of the used space surveillance segment and associated interfaces;
3. Ensure the inclusion in the procedure of the collision avoidance strategy to approach individual events including mission-specific timelines;
4. Ensure the implementation of a workflow to generate ephemerides in a standard format (e.g. OEM/NDM), preferably in the J2000 reference frame (see also Requirement 5.3.3.3.i: CAM effect in 4.3.25);

5. Ensure the implementation of data distribution through services such as space-track, Space Data Association, a publicly accessible website, or similar services, which allow access by other operators.

Specifically for point a.3 above on the documentation of the collision avoidance strategy, two scenarios are possible, depending on the mission complexity and the mission phase:

1. In case there is no possibility that the spacecraft can manoeuvre in response to a collision alert or for other operational reasons (either due to lack of capability by design, failure or due to end of life), then the documentation of the collision strategy consists in clearly communicating this information, so that, for the purposes of collision avoidance coordination, the object can be treated as a ballistic object;
2. In case there is possibility that the spacecraft can manoeuvre, either in response to the collision warning or for other operational reasons, then the documentation of the collision strategy includes high-level information on the planning and decision-making timeline of the response activities, the timeliness possible for interaction between operators, the level of automation of the system.

The distribution of information is intended here to happen between actors involved in a conjunction event and the open publication of information is not required to meet the requirements. For enhanced transparency, an operator can consider sharing ephemerides and supplementary information (such as CONOPS, collision avoidance procedures, current spacecraft status) on a dedicated publicly accessible website since no single service provider typically pools such information and distribute it to all spacecraft operators.

4.3.28 Requirements 5.3.3.3.n: Trajectory catalogue for constellations

Rationale for the Requirement

The requirement aims at facilitating the management of the collision risk for spacecraft in a constellation. A constellation operator performing collision avoidance activities relies on sufficiently good knowledge of the trajectory of their spacecraft to assess and plan any collision avoidance activity safely and to be able to assess the risk of collision in a timely manner.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:
   1. Ensure the adoption by the operator of a reliable, timely and validated way to perform precise orbit determination;
   2. Ensure the use of the data from the point above to maintain the information on the spacecraft orbital states that can be used as input to the collision risk assessment;
   3. Ensure the ability to assess the collision risk and mitigation measures for any conjunctions involving the spacecraft in the constellation in an accurate and timely manner.

4.3.29 Requirement 5.3.3.3.o: CAM coordination

Rationale for the Requirement

The requirement aims at defining preliminary coordination criteria in case of conjunctions between active spacecraft.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:
   1. Ensure that the operator provides points of contact and time availability, in case of coordination needed with other operators;
   2. Ensure that the ground segment can process and produce ephemerides (see also Requirement 5.3.3.3.i: CAM effect in 4.3.25) in standard formats and covariance to enable operator-vs-operator screening;
   3. Check that the operational procedures define suitable timelines to coordinate with the other operator and a decision point on when an action is defined in case of no response;
   4. Check that the operational procedures reflect the order of priority defined in the requirement.

4.3.30 Requirement 5.3.3.4.a: CPO/FF: probability of unintentional contact threshold

Rationale for the Requirement

The requirement aims at defining the probability threshold to limit the risk of unintentional contact between all space objects involved in close proximity operations.

Unintentional contact is defined as contact between two objects in an unintended way, i.e. not through nominal docking or berthing contact as planned in the operations, such that the contact:

a. Is made outside the set geometrical envelope of contact surfaces,

b. Is made outside the set profile of velocities and accelerations, both translational and rotational.

c. Generates debris.

d. Degrades the manoeuvrability or functionality of the involved spacecraft, through both direct effects on structures or functionality, and derived effects from electrostatic discharge and plume impingement.

The requirement is complemented by requirements listed below:

a. Requirement 5.3.3.4.b: CPO/FF: probability of unintentional contact assessment in 4.3.31.

b. Requirement 5.3.3.4.c: CPO/FF: operational procedure against unintentional contact in 4.3.32Requirement 5.3.3.4.c: CPO/FF: operational procedure against unintentional contact.

c. Requirement 5.3.3.4.d: CPO/FF: resources for contingency/recovery against unintentional contact in 4.3.33

d. Requirement 5.3.3.4.e: CPO/FF: CAM temporal effectiveness against unintentional contact in 4.3.34

f. Requirement 5.3.3.4.f: CPO/FF: relative navigation information distribution, in 4.3.35.

In case of close proximity operations for removal, as per Section 5.4.1.3 (Requirement 5.4.1.3.a: Preparation for removal: LEO in 4.4.12and the subsequent requirements), if the serviced spacecraft

follows the directives in Requirement 5.4.1.3.f: Preparation for removal: cooperative in 4.4.17 to inhibit actions during docking, or capture, the serviced spacecraft can be considered passive and the current Section does not apply to it.

If the formation flying between spacecraft is such that loss of control of a spacecraft does not pose a risk of unintended contact with the other spacecraft of the formation flying with a probability above 10^{-4} during the 7 days after the loss of control, the requirement verification can follow the procedures for collision avoidance operations as with other space objects.

Finally, it is important to mention that the notion of catastrophic collision used in space debris modelling is not directly related to the definition of severity categories in the framework of safety assessments (e.g. as defined in [RD08]). In general, when safety assessments are performed (as typically in the case of close proximity operations), debris generation events in orbit can be considered to be critical, major, or negligible (e.g. as defined in [RD08]), except for extraordinary cases related to the breakup of or affecting inhabitable spacecraft. Established metrics and threshold for the mapping of debris generating event into specific safety severity levels are currently not available, but different approaches are under investigation. As a result, project teams can perform their assessment and evaluations based on considerations such the number of objects potentially generated by the event, the duration of their permanence in orbit, the potential interaction with space debris and active spacecraft. It is expected that such a preliminary version of such assessment is performed by SRR and documented in the Space Debris Mitigation Plan and consolidated by PDR in the Space Debris Mitigation Report.

Methods to Assess Compliance

The verification methods to assess compliance are:

a. Review-of-design, to:
    1. Verify that the operations are without risk of unintentional contact under nominal conditions, i.e. the nominal/reference close proximity trajectories are designed such that unintentional contact does not happen, and therefore their contribution to the unintentional contact risk threshold is negligible;
    2. Ensure health monitoring is implemented in the spacecraft through adequate set of sensors and on-board computer functions to detect possible anomalies during the operations that lead to a violation of the probability of unintentional contact, as per 5.4.1.2 in 4.4.3 (Requirement 5.4.1.2.a / ECSS-U-AS-10, , 7.3.1.3: Disposal criteria and the subsequent requirements);
    3. Ensure that the spacecraft is equipped with units and functions enabling to perform contingency operations, as per Requirement 5.3.3.4.c: CPO/FF: operational procedure against unintentional contact in 4.3.32, Requirement 5.3.3.4.d: CPO/FF: resources for contingency/recovery against unintentional contact in 4.3.33, and Requirement 5.3.3.4.e: CPO/FF: CAM temporal effectiveness against unintentional contact in 4.3.34.
    4. Ensure that the level of on-board autonomy permits emergency collision avoidance according to points a.2 and a.3 above and considering the level of ground intervention required. This is verified by analysis as per points b.2 and b.3 hereunder;
    5. Ensure that enough observability and adequate timing for data processing and commanding is granted in case of ground intervention where needed. This is verified by analysis as per points b.2 and b.3 hereunder.

b. Analysis, to:
    1. Prove the nominal CONOPS CPO operations contain no unintentional contact.

2. Quantify the probability of unintentional contact under all feared events and demonstrates that it is below the required threshold, taking into account the conditions in Requirement 5.3.3.4.b: CPO/FF: probability of unintentional contact assessment in 4.3.31. .
3. Demonstrate quantitatively that mitigation measures implemented to lower the risk of collision, such as collision avoidance manoeuvres, as per Requirement 5.3.3.4.c: CPO/FF: operational procedure against unintentional contact in 4.3.32 , Requirement 5.3.3.4.d: CPO/FF: resources for contingency/recovery against unintentional contact in 4.3.33, Requirement 5.3.3.4.e: CPO/FF: CAM temporal effectiveness against unintentional contact in 4.3.34, Requirement 5.3.3.4.f: CPO/FF: relative navigation information distribution in 4.3.35, also result in compliance with the 10^{-4} threshold.
4. Demonstrate that the sequence of events during operations does not create, at any time, a risk of unintended contact larger than the 10^{-4} threshold.

The verification of compliance is performed prior to entry into service of the system and re-assessed any time there is a deviation. Deviations can be planned, or result from a failure, due to internal or external cause, which can change the capability or performance of the system.

Mitigation Measures

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

a. To take the guidelines in [RD023] as part of the design philosophy, in particular the design of passively safe trajectories and a nominal close proximity operations concept of operations with nominal set of trajectories warranting no unintentional contact during operations. This is information expected to be contained in the CONOPS.

b. Mitigation Measures for requirements:
   1. Requirement 5.3.3.4.c: CPO/FF: operational procedure against unintentional contact ; in 4.3.32
   2. Requirement 5.3.3.4.d: CPO/FF: resources for contingency/recovery against unintentional contact in 4.3.33;
   3. Requirement 5.3.3.4.e: CPO/FF: CAM temporal effectiveness against unintentional contact in 4.3.34 and,
   4. Requirement 5.3.3.4.f: CPO/FF: relative navigation information distribution in 4.3.35.

4.3.31 Requirement 5.3.3.4.b: CPO/FF: probability of unintentional contact assessment

Rationale for the Requirement

The requirement aims at supporting the calculation of the probability of unintentional contact between approaching space objects to take into account the uncertainties in the nominal operations, and all failures that can arise during the mission. This quantification is used to verify that the threshold for the probability of unintentional contact is not violated in Requirement 5.3.3.4.a: CPO/FF: probability of unintentional contact threshold in 4.3.30.

During close proximity operations, there are usually only two space objects involved. In this case, the approaching spacecraft is interpreted as the active spacecraft performing the approach manoeuvres. However, during formation flying missions, the number of space objects can likely be higher, and therefore the probability of collision of a space object is calculated with respect to all the spacecraft actively participating to the formation.

Methods to Assess Compliance

The verification methods to assess compliance are:

a. Review-of-design, to:
   1. Verify that all systems, functional chains, and units of all active spacecraft involved in the close proximity operations are identified and:
      (a) are input to the FTA analysis (refer to [RD024] as a guideline) for the calculation of the probability of unintentional contact between the space objects.
      (b) have equipment performance uncertainties from the various systems estimated, and are input to the analyses in point b hereunder.
   2. Verify that the space environment non-negligible perturbations that influence the behaviour of the involved spacecraft have been identified, and are input to the analyses in point b hereunder;
   3. Verify that the analyses and simulations performed to quantify the probability of unintentional contact contain all the active space objects involved in the close proximity operations, and cover:
      (a) nominal scenarios, including performance and environment uncertainties.
      (b) failure scenarios leading to the "unintentional contact" feared event, including combinations of failures.
   4. Verify that, during operations, when the probability of collision is updated, and in particular in the case of mission extension including further close proximity operations, the probability of collision is re-calculated considering the updated RAMS analysis of the involved systems and functional chains, as per Requirement 5.4.1.2.d: Disposal critical function and equipment parameters update in 4.4.7, which also considers wear out modelling.

b. Analysis, to assess with simulations the probability of unintentional contact by:
   1. Using a Fault Tree Analysis (with a top event "unintentional contact" and identifying all the sub-events and failure modes that can lead to that top event, and their corresponding probability of occurrence) and the simulation results from the scenarios assessed in points a.1, a.2 and a.3;
   2. Simulating, stochastically (e.g. Monte Carlo analysis) or deterministically (e.g. worst-case analysis), the nominal/reference trajectories during close proximity operations in the concept of operations and the guidance functions, considering:
      (a) different approach directions during motion synchronisation phase.
      (b) nominal/reference trajectories with dispersion of parameters, taking into account exogenous and endogenous disturbances on the spacecraft, expected uncertainties in the spacecraft performance, and measurement uncertainties from on board or ground during the close proximity operations.
   3. Simulating the abort, cancel and collision avoidance trajectories in support to the verification that there is no collision risk upon execution, as per Requirement 5.3.3.4.a: CPO/FF: probability of unintentional contact threshold in 4.3.30 , Requirement 5.3.3.4.c: CPO/FF: operational procedure against unintentional contact in 4.3.32, Requirement 5.3.3.4.e: CPO/FF: CAM temporal effectiveness against unintentional contact in 4.3.34 and Requirement 5.3.3.4.f: CPO/FF: relative navigation information distribution in 4.3.35.

c. Test, to complement the analyses, to:

1. Verify that mechanism actuations in very close proximity, and in particular during the capture, docking and separation phases, do not cause unintentional contact between space objects;
2. Verify that the functional chains and units for relative navigation between the spacecraft have been identified correctly in the simulations and that they satisfy the set performance parameters during close proximity operations to ensure no unintentional contact can happen under nominal conditions.

Mitigation Measures

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

a. To take the guidelines in [RD023] as part of the design philosophy, in particular the design of passively safe trajectories and of a nominal close proximity operations concept of operations that has a nominal set of trajectories warranting no unintentional contact during operations.

b. To consider the SAVOIR working group guidance regards to the FDIR engineering process in both design and operational phases (**[**RD025] and [RD026]). Section 5.4.1.2 (Requirement 5.4.1.2.a / ECSS-U-AS-10, , 7.3.1.3: Disposal criteria in 4.4.3 and the subsequent requirements) provides further details on the health monitoring processes.

c. To identify the performance of the equipment involved in the close proximity operations (i.e. GNC, propulsion, avionics) as input to the (Monte Carlo) simulation campaigns.

d. During the design phase, to perform the relevant analyses on the functional chains and the equipment involved in the close proximity operations (i.e. GNC, propulsion, avionics) therein, in particular FMEA/FMECA and FTA (see 5.3.2.1.a. (Requirement 5.3.2.1.a / ECSS-U-AS-10 7.2.1.1: Accidental break-up probability threshold in 4.3.2 and the subsequent requirements), **[**RD025] and [RD011]) to define the feared events that lead to the violation of Requirement 5.3.3.4.a: CPO/FF: probability of unintentional contact threshold in 4.3.30 and design contingency operations (i.e. CAM manoeuvres) to lower this risk, as per Requirement 5.3.3.4.e: CPO/FF: CAM temporal effectiveness against unintentional contact in 4.3.34.

e. To define a corridor where the CPO can be performed safely and a keep out zone around the target, subsequently perform Monte Carlo simulations of the baseline CPO operations to verify that the hardware and software in the involved chains have the performance to stay within that corridor, and determine the maximum degradation of the equipment allowed before the space object exits the corridor as input to the FDIR system.

The volumes of the corridor and keep out zone are based on the properties of the target and on the performance of the navigation units of the chaser. In formation flying, the volumes are based on the performance of the individual spacecraft. Outside the corridor the performance is considered too degraded to warrant Requirement 5.3.3.4.a: CPO/FF: probability of unintentional contact threshold in in 4.3.30 and contingency operations (i.e. CAM and subsequent operations) are triggered.

4.3.32 Requirement 5.3.3.4.c: CPO/FF: operational procedure against unintentional contact

Rationale for the Requirement

The requirement aims at ensuring that the operational procedures are defined to include the monitoring of functions and performance of the space objects involved in close proximity operations and their thresholds during operations, and that they describe, for each identified credible contingency scenario, the contingency actions (e.g. planned collision avoidance strategies) and recovery actions (i.e. to restore on-orbit safety conditions), and their respective functions. The operational procedure ensures that the probability of unintentional contact between space objects during close proximity operations is kept under the threshold determined in requirement in Requirement 5.3.3.4.a: CPO/FF: probability of unintentional contact threshold in 4.3.30.

The requirement is applicable to each operator actively involved in the close proximity operations, i.e. excluding dead spacecraft (space debris) being captured.

Methods to Assess Compliance

The verification methods to assess compliance are:

a. Review-of-design, to:
    1. Ensure that combined observability, adequate timing for data processing, and commandability is granted for the execution of nominal and contingency operations (including both the availability of relevant data from the space objects and their timing availability via the relevant space and ground interfaces);
    2. Verify that the criteria for notification of alerts by the space segment FDIR function due to the violation of the probability of unintentional contact between space objects during close proximity operations are known and defined in the operational procedures, including required timeliness of these notifications;
    3. Identify and assign the roles of the space segment and the ground segment regarding the monitoring, contingency and recovery operations to mitigate the probability of unintentional contact through full coverage between the ground segment procedures and the autonomy design of the space segment;
    4. Validate the strategy in the concept of operations concerning the timeline between alerts and collision avoidance manoeuvre implementation. The strategy includes high-level information on the planning and decision-making timeline of the response activities, the timeliness for interaction between operators, the level of automation of the system, and the definition of a decision point for actions in case of absence of response from one of the operators;
    5. Verify the implementation of flight operation procedures to monitor and handle foreseeable failure modes, which can prevent the space system to perform safe close proximity operations;
    6. Verify that there are ground systems, ground system functions and resources to respond to unintentional contact warnings, or triggers;
    7. Verify that the responsibilities of each operator involved in the close proximity operations are exhaustively and unambiguously defined, and without overlaps, to avoid gaps or conflicts in response to contingency scenarios;
    8. Verify the completeness of the procedure, which rules the used ground segment and associated interfaces, for each of the operators involved in the close proximity operations, e.g. the contingency and recovery operations contain the correct communication

procedures between the operators involved, taking into account timeliness of response requirements;

9. Verify the completeness of the recovery procedures and the coverage of the allocation of recovery functions between ground and space segments;
10. Verify that both space and ground segments have the required functions to perform recovery procedures.

b. Analysis, to:

1. Ensure visibility and correct identification of the necessary data for monitoring the collision risk by simulating the combined dynamics of the involved space objects, using pre-flight models;
2. Demonstrate that the sequence of events during nominal and contingency operations does not create, at any time, a risk of collision larger than the 10 -4 threshold.

c. Tests, to:

1. Verify that the ground and space segments are compatible with contingency monitoring and handling. In particular, the verification of communication protocols between ground and space segment, command formatting and correct reaction times are tested;
2. Verify that the sequence of events for nominal and contingency cases is executable within the imposed timing constraints.

Mitigation Measures

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

a. To ensure that the requirement is included in the operational procedures described in [RD027]and [RD028].

b. To validate the simulation environment for anomaly investigation (including ground-based anomalies) and contingency scenario validation of the close proximity operations.

c. To consider the SAVOIR working group guidance with regards to the FDIR engineering process in both design and operational phases ([RD025] and [RD026]).

Details on the methodology to perform the assessment can be found in:

- ECSS-E-ST-70 [RD027], in particular for nominal and contingency procedures
- ECSS-E-ST-70-11 [RD028], and [RD025], in particular for FDIR and recovery procedures.

ECSS-E-ST-70 [RD027] (clause 5.3.2) covers the preparation of a mission operations plan (MOP) that includes operations procedures covering nominal and contingencies operations for both the space segment and ground segment. This mission operations plan therefore includes any procedures related to CAM strategy and collision risk management for close proximity operations.

4.3.33 Requirement 5.3.3.4.d: CPO/FF: resources for contingency/recovery against unintentional contact

Rationale for the Requirement

The requirement aims at ensuring that, in case of non-nominal scenarios, the spacecraft is able to mitigate the contingency arising to prevent the risk of collision in order to fulfil Requirement 5.3.3.4.a: CPO/FF: probability of unintentional contact threshold in 4.3.30.

The requirement specifies the capability to perform contingency operations, either autonomously or via ground command, and recovery operations after contingency to continue to warrant an acceptable probability for risk of unintentional contact.

In case of two active space objects, usually one is involved in the operations, i.e. the servicing vehicle, around the other one that takes a passive role, i.e. the target. It is common to assign the responsibility of the contingency and recovery capability in close proximity operations to the servicing vehicle. In case of formation flying individual capability is expected.

Methods to Assess Compliance

The verification methods to assess compliance are:

a. Review-of-design, to:
   1. Verify that the FDIR engineering has identified and implemented on-board the conditions that can trigger contingency and relayed to ground for ground intervention, including:
      (a) combined autonomous recovery functions implemented in the concerned space objects consistent and compatible in terms of actions for collision anomaly and final ("safe") mode reached by the space object(s).
      (b) observability granted for all parameters necessary for ground intervention.
   2. Verify that contingency operations have been identified and planned for each contingency scenario, including the design of collision avoidance manoeuvres and corresponding trajectories, including:
      (a) operational contingency procedures, as per ECSS-E-ST-70 [RD027].
      (b) all resource budgets (TM/TC, power, propulsion) consistent with all the expected (nominal and contingency) close proximity operations scenarios, considering also the combined geometry and interactions of the objects (e.g. RF interferences, albedo effect for attitude sensors, shadowing).
   3. Verify that the functional chains and units are identified and implemented in the system design to perform contingency operations at all times during close proximity operations, in case of a:
      (a) contingency scenario that does not require a collision avoidance manoeuvre (e.g. equipment failure at low level).
      (b) collision avoidance manoeuvre, including:
         (1) the on-board functional chains and units required to perform the collision avoidance manoeuvre
         (2) the functional chains and units required to inform ground of the need for the triggering of contingency operations, both on ground and on board
         (3) the functional chains and units required on ground to trigger a ground-based contingency operations (i.e. collision avoidance manoeuvre)
         (4) the propulsion system capable to perform the collision avoidance manoeuvres with duly time and propellant

(5) transition to a safe type of mode for the spacecraft to await from ground analysis and receive recovery instructions

4. Verify that the resources exist on ground to analyse the cause of the contingency and react to it, including:
   (a) team composition and training, and certification adequate for all roles in the operations concept for (nominal and) contingency scenario.
   (a) all hardware and software systems in place and all interfaces defined and compatible with the operations execution.
5. Verify that the resources exist on ground and on board to perform recovery operations via ground command, including:
   (a) reconfigurability on board, as well as the capability to send commands from ground to this purpose.
   (b) reconfigurability of the spacecraft (e.g. change to redundant chain or unit) as part of the recovery operations to eliminate the cause of contingency.

b. Analysis, to:
   1. Verify via simulation that the resulting trajectories during contingency operations lead to no collision;
   2. Verify via simulation that the functional chains in the spacecraft are able to perform the contingency and recovery operations;
   3. Verify via simulation that the FDIR is able to trigger the contingency operations and to transition the spacecraft in a safe state to await for recovery via ground command.

c. Tests, to:
   1. Verify the above-mentioned analyses via testing with the involver hardware and software combined;
   2. Verify via simulations the end-to-end execution of contingency scenarios including the operations teams involved;
   3. Verify compatibility and performance end-to-end of the system ground to space segment including ground-to-space segment in a system validation test campaign.

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.3.3.4.c: CPO/FF: operational procedure against unintentional contact in .

4.3.34 Requirement 5.3.3.4.e: CPO/FF: CAM temporal effectiveness against unintentional contact

Rationale for the Requirement

The requirement aims at ensuring that, in case issues arise during operations, there are coordinated manoeuvres designed to handle the contingencies and operational procedures to correctly implement them such that that the probability of unintentional contact is set back under the threshold set in Requirement 5.3.3.4.a: CPO/FF: probability of unintentional contact threshold in 4.3.30. A manoeuvre is effective if it reduces the probability of unintentional contact below 10^{-4} for at least 7 days after the manoeuvre (i.e. the spacecraft is on an orbit that warrants no collisions for 7 days) to allow for failure investigation and recovery operations planning and execution.

Methods to Assess Compliance

The verification methods to assess compliance are:

a. Review-of-design, to:

1. Verify that the relevant analyses in the design of the on-board segment (at both subsystem and system level) identifies those cases that lead to a probability of unintentional contact between the space objects involved to be above the set threshold that require a coordinated manoeuvre. The RAMS (FMEA/FMECA, FTA) methodology is found in [RD024], [RD011].
2. Verify that for each of the identified cases in point a.1, the coordinated manoeuvre consists of:
   (a) a collision avoidance manoeuvre (CAM) designed to send the spacecraft with an opening rate (i.e. increasing distance to other spacecraft) warranting no immediate collision during the CAM, and to leave the spacecraft in an orbit that does not cause collisions for 7 days. This includes the design of the CAM strategy:
      (1) with assignation of the correct manoeuvres to each involved space segment spacecraft
      (2) in complex scenarios such as motion synchronisation around the target.
   (b) the spacecraft being set to a state that warrant no collision for 7 days after the CAM.
3. Verify that the coordinated manoeuvre can be autonomously triggered by the space segment, including:
   (a) CAM computation implemented on board the spacecraft.
   (b) FDIR design containing the fault detection mechanisms to detect those failures and parameter thresholds that have been identified to lead to the violation of the threshold and capable of initiating the CAM and subsequent state changes either autonomously or via ground-trigger (depending on the criticality and timeliness requirement).
   (c) the spacecraft resources to perform the CAM, as per Requirement 5.3.3.4.d: CPO/FF: resources for contingency/recovery against unintentional contact in 4.3.33.
4. Verify that the spacecraft is capable of communicating with the ground segment the need to trigger a CAM and the confirmation of its execution, together with other relevant information for ground for contingency and recovery measures;
5. Verify that the ground segment has:
   (a) the resources to (re)design CAMs and to upload them to the space segment.
   (b) the operational procedures to perform the coordinated manoeuvres in coordination with all involved operators as part of the contingency operations, as per Requirement 5.3.3.4.c: CPO/FF: operational procedure against unintentional contact in 4.3.32 and Requirement 5.3.3.4.d: CPO/FF: resources for contingency/recovery against unintentional contact .

b. Analysis, to verify with simulations that:

1. The coordinated manoeuvres from point a.2 and a.5 result in the probability of unintentional contact below 10^{-4} for 7 days after the manoeuvre, using the same simulation methodology as in Requirement 5.3.3.4.b: CPO/FF: probability of unintentional contact assessment;
2. The FDIR performs as expected in ground and space segments, complemented by tests as per c.3. here below.

c. Test, to:

1. Confirm the analyses in b. with the relevant hardware in the loop;
2. Verify the correct implementation of telecommand and telemetry during the coordinated manoeuvres;
3. Ensure that all FDIR is tested (in accordance with recommendations provided in [RD025]), in particular those critical monitors that are implemented on the FDIR on-board.

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.3.3.4.c: CPO/FF: operational procedure against unintentional contact in 4.3.32.

b. To identify and analyse failures, when occurred in the systems of the involved spacecraft, to detect those that can increase the risk of collision above the acceptable threshold such that mitigation measures, both at design level and at operational level.

c. To implement only the monitors that have been tested on ground in order to avoid modifying the FDIR once in orbit.

d. For close proximity operations involving motion synchronisation, to simulate the CAM manoeuvre in all directions to check that the CAM manoeuvre is performed in the correct direction to avoid unintentional contact during the coordinated manoeuvre.

4.3.35 Requirement 5.3.3.4.f: CPO/FF: relative navigation information distribution

Rationale for the Requirement

The requirement aims at ensuring that the data required by the spacecraft navigation system to perform relative navigation during close proximity operations is complete before flight and updated during flight.

Methods to Assess Compliance

a. Review-of-design, to:
   1. Identify the requirements for relative navigation to achieve the necessary accuracy, which are derived together with the information needs from the servicer, i.e. the data required by the relative navigation sensors:
      (a) geometry, structure, and material properties of the object (including predicted or estimated degradation at time of arrival).
      (b) information on present navigation targets on the client (including predicted or estimated degradation at time of arrival).
      (c) capture operations and obstructions during approach in the field of view of the navigation sensors.
      (d) information on the attitude and orbital (control) state of the target.
   2. Identify key points in the mission operations where this information is checked for updates, typically before and during relative navigation operations;
   3. Verify that the ground segment has the capability to perform data updates during the mission and that ground-to-space data exchange protocols support timely and accurate distribution of updated information.

4. Ensure that the Interface Control Document (ICD) specifies all this information and Analysis, to verify with simulations that the relative navigation functions and units on the involved spacecraft meet their required performance during close proximity operations, based on the information provided by the client's operator through the ICD.

b. Test, to:

1. Confirm the findings from the analysis in Requirement 5.3.3.4.b: CPO/FF: probability of unintentional contact assessment in 4.3.31 using the relevant hardware-in-the-loop;
2. Verify the correct execution of the update capability on the ground, as well as the implementation of telecommands and telemetry;
3. Conduct end-to-end system tests to verify the feasibility and the response of the spacecraft and ground segments to updates in the navigation data.

Mitigation Measures:

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

a. To maintain an ICD between operators with the properties of the target required for the relative navigation and GNC functions, which is updated whenever the target(s) in orbit change state (e.g. planned or unplanned end-of operations, tumbling states, deterioration of the geometry).

b. To verify through analysis and test that the servicer GNC units performance is compatible with the properties of the target (information on the control engineering requirements regarding the need for accuracy can be found in [RD029][RD028]) and continuously monitor the system performance over time to assess the impact of potential degradation of the servicer navigation performance (both and hardware and software level) and the target's properties on the navigation accuracy, which during close proximity operations is a mission-critical function.

c. To ensure capability to perform uplinks with telemetry in the event that an update necessitates sending new information to the space segment, which implies reconfigurability of the space segment, particularly the Guidance, Navigation, and Control (GNC) system, telemetry package configuration, and the ground segment's ability to compile information in the correct format for telemetry (details on the methodology to define the monitoring and control data to be delivered can be found in [RD030]).

4.3.36 Requirement 5.3.3.5.a: Trackability

Rationale for the Requirement

The requirement aims at ensuring that, by design, the spacecraft and launch vehicle trajectory is estimated and predicted by space surveillance segments. This means that the sensor systems are able not only to detect the spacecraft and launch vehicle, but also to repeatedly re-observe (i.e. track) it to estimate the trajectory. The ability to track the target and estimate a trajectory depends on the frequency, quality, and distribution of measurements.

The estimated trajectory of tracked spacecraft and launch vehicles allows:

- Screening for new conjunctions against catalogued debris;
- Computing the collision probability using the estimated uncertainty

Methods to Assess Compliance

a. Review-of-design, to:

1. Identify system units that increase the trackability, e.g.:

(a) large reflective surfaces, active transmission, passive brightness, radar cross-section enhancement.

(b) corner cubes or retroreflectors to enable tracking the spacecraft using satellite laser ranging stations.

(c) retroreflector payloads (as a reference, for retroreflector payloads for spacecraft at 20000 km altitude it is important to have a minimum effective cross-section of 100 million square meters, while for spacecraft at lower or higher orbits the value is scaled to compensate for the increase or decrease in signal strength determined by the object distance to the power of 4 [RD031]).

2. Identify system units that support the identification of spacecraft;
3. Identify the space surveillance segment capabilities;
4. Collaborate with the space surveillance segment and share ephemerides and manoeuvre prediction (see Requirement 5.3.3.5.f: Ephemerides frequency in 4.3.41).

b. Analysis, to:

1. Estimate the reflective properties of the spacecraft using:

(a) the geometric cross-sectional area as the first approximation. The cross-sectional area can be derived from using 3D model of the object, e.g. using DRAMA CROC [RD032].

(b) an optical brightness model as described in Requirement 5.6.a: Visual brightness assessment in 4.6.1**.**

(c) a complex radar cross-section model derived from numerical electromagnetic simulation codes.

2. Estimate the trackability by the space surveillance segment by:

(a) comparing the estimated geometric size, radar cross-section, and optical brightness with a trackability curve or performance table, e.g. following the approach in [RD020]. The performance curve can be provided, with demonstration, by the space surveillance segment or derived from the catalogue. For objects tracked by the US SSN, literature with performance values is available, as represented in Figure 5-1, and can be used to demonstrate that no further analysis is necessary.

(b) simulating the surveillance segment and quantifying the observation opportunities, reporting the number of potential observations per day. It is important to select a time frame for the simulation long enough to identify possible observation gaps, e.g. one week is considered reasonable to identify the re-observation frequency for spacecraft in LEO orbits. Spacecraft and launch vehicles that are observed at least once per day are considered trackable.

Mitigation Measures

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

a. To assess the trackability early during the space system development to identify possible trackability enhancement needs.

b. To identify possible space surveillance segments early to understand detection and trackability limits.

4.3.37 Requirement 5.3.3.5.b: Space surveillance segment

Rationale for the Requirement

The requirement aims at ensuring the availability of space surveillance data for identifying potential collision events throughout the object lifetime and supporting the cataloguing of the spacecraft.

Methods to Assess Compliance

a. Review design, to:
   1. Ensure that a contractual agreement with possible space surveillance data providers is in place;
   2. Ensure that the agreement includes sharing trajectory and manoeuvre data and other information supporting the space surveillance operations;
   3. Validate operational ground segment interfaces with space surveillance segment to share this information;
   4. Provide point of contact information to the space surveillance segment such that it is available to other operators;
   5. When designing a constellation, or a mission performing close proximity operations, determine if a more tailored conjunction screening process than for a single object mission is warranted and can be provided by the space surveillance segment.

Mitigation Measures

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

a. To identify one or multiple space surveillance segments and analyse the suitability of the agreements early in the system development.

4.3.38 Requirement 5.3.3.5.c: State vector quantification frequency

Rationale for the Requirement

The requirement aims at enhancing the assessment of the uncertainty of the spacecraft orbit, which is fundamental for the determination of the collision probability and the avoidance measures. Typically, spacecraft operators (and cooperating ground and space surveillance segments) better assess the uncertainty than third-parties space surveillance service providers. Potential conjunctions between active spacecraft can, therefore, be better assessed if such information is established and shared.

Methods to Assess Compliance

a. Review-of-design, to:
   1. Be able to generate standardised exchange files via the Orbit Data Message (ODM);
   2. Ensure that the state covariance at orbit determination time is regularly estimated;
      (a) from the orbit determination process using data from ground, space, or space surveillance segment (the covariance at the orbit determination time can be considered constant, e.g. if consistent GNSS data is available, or variable depending on data availability).
      (b) from historic data, e.g. using orbit comparisons as in [RD033].
   3. ensure that the state covariance is forecasted according to Annex Section B.3.2 and models uncertainties introduced by imperfect force models and manoeuvres.

b. Analysis, to:

1. Review capabilities of operational process to estimate and predict realistic covariance;
2. Assess orbit determination covariance based on on-board GNSS using a consider-covariance analysis as in Requirement 5.3.3.5.d: Position accuracy in 4.3.39;
3. Review the engaged third-party space surveillance service provider capabilities.

Mitigation Measures

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

a. To identify orbital state and uncertainty sources early in the development, e.g. publicly available TLE typically do not provide covariance information.

b. During the operational phase, to assess quality of position and velocity accuracy in regular time intervals, e.g. by comparing the forecasted covariance with an estimated a posteriori state.

4.3.39 Requirement 5.3.3.5.d: Position accuracy

Rationale for the Requirement

The requirement aims at having position accuracy to enable accurate prediction of conjunction events, i.e. the adequate assessment of the collision probability to avoid missing critical events or issuing unnecessary false alarms. Accurate orbital information allows to reduce the overall number of required avoidance manoeuvres.

In high-velocity encounters, only the position uncertainty is considered relevant for the collision probability calculation. The accuracy thresholds are defined for the orbit determination interval, i.e. the time span covering all observations used in the orbit determination process.

In practice, the propagated uncertainty at the time of close approach is relevant for the collision probability assessment. However, the propagated position uncertainty depends on the initial estimated accuracy, but also on dynamical model inaccuracies (e.g. predicted solar and geomagnetic activity and impact on atmosphere), which differ between altitudes. It is, therefore, considered more practical to define a threshold on the orbit determination accuracy.

Methods to Assess Compliance

a. Review-of-design, to:

1. Identify system units that increase the trackability (see Requirement 5.3.3.5.a: Trackability in 4.3.36) (more frequent and well distributed observations can generally improve the orbit determination accuracy);
2. Identify the space surveillance segment accuracy capabilities;
3. Collaborate with the space surveillance segment and share ephemeris and manoeuvre prediction (see Requirement 5.3.3.5.f: Ephemerides frequency in 4.3.41).

b. Analysis, to:

1. Estimate the reflective properties of the spacecraft as in Requirement 5.6.a: Visual brightness assessment in 4.6.1;
2. Estimate the trackability by the space surveillance segment by:
   (a) determining the expected covariance from the orbit and the estimated geometric size, radar cross-section, and optical brightness with an interpolation or look-up table, e.g. following the approach in [RD020]. The performance curve can be

provided, with demonstration, by the space surveillance segment (preferred option) or derived from historic estimated covariances in the catalogue [RD034][RD035]:

(1) Covariance statistics for the performance assessment can be aggregated from historic conjunction data messages. The last available data per conjunction event is assumed to represent the covariance closest to the orbit determination epoch. The distribution of conjunction events along the orbit is assumed to cover the orbit determination interval

(2) Synthetic covariances are in some cases available and can be used as a reference [RD036][RD090].

(b) For surveillance providers different from US SSN, for which literature data exists, simulating the surveillance segment and quantifying the position accuracies.

Mitigation Measures

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

a. To assess the trackability early during the space system development to identify possible trackability enhancement needs.

b. To identify possible space surveillance segments early to understand accuracy limits.

4.3.40 Requirement 5.3.3.5.e: On-orbit identification

Rationale for the Requirement

The requirement aims at enhancing early identification to support the space surveillance segment in the cataloguing process and maintain a consistent trajectory of the spacecraft or launch vehicle.

Methods to Assess Compliance

a. Review-of-design, to:

1. Identify system units that support the identification of spacecraft;
2. Identify the space surveillance segment capabilities, e.g. laser ranging sensors capable of identifying reflectors with special signatures;
3. Collaborate with the space surveillance segment and share predicted launch trajectory and early operations/manoeuvring plans. Inform surveillance segment early about possible mislabelling in the catalogue (e.g. using the TLE);
4. Review launch sequence, in case of rideshare, to avoid uncoordinated release of spacecraft and cause mislabelling in the catalogue;
5. Identify ground segment capabilities to share early orbital information derived from telemetry and on-board GNSS data.

Mitigation Measures

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

a. To identify possible identification components needs early during the space system development, e.g. for multiple spacecraft release scenarios where it is more difficult to identify each individual spacecraft than in a single spacecraft release. Possible identification components are retroreflectors, e.g. reflecting in a specific spectrum or polarising, or LEDs with a unique sequence.

b. To identify possible space surveillance segments early to understand identification limits.

4.3.41 Requirement 5.3.3.5.f: Ephemerides frequency

Rationale for the Requirement

The requirement aims at enhancing the procurement from a space surveillance segment of ephemerides related to potentially conjuncting space objects. It is only possible for active spacecraft to demand an assessment of the position and velocity accuracy from the operator. For all other objects (including space debris), the space surveillance segment is responsible to determine the state, predict the ephemerides and uncertainties relevant to characterise conjunction events. State and accuracy information is provided in the result of a screening process as conjunction messages, e.g. in CCSDS CDM format.

Methods to Assess Compliance

a. Review-of-design, to:
    1. Ensure collaboration with space surveillance service provider, e.g. through a bilateral agreement, and ensuring daily updates are received and on-demand provided ephemerides of own satellite are screened with low turn-around time;
    2. Ensure provision of at least one daily updated ephemeris file to the space surveillance service provider or make it available by other means to other operators (in high-drag environments, consider providing more than one update per day);
    3. Review the third-party space surveillance service provider proposal, or service level agreement, to be able to:
        (a) (re)process on daily basis the ephemerides of other objects, for spacecraft in high-drag environment, considering even higher frequency screenings.
        (b) generate ephemerides and conjunction warnings in agreed format.

Mitigation Measures

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

a. To identify and possibly get in contact with potential space surveillance service provider(s), already during the design phase, to clarify the accuracy and timeliness of the provided data.

4.3.42 Requirement 5.3.3.5.g: Anomaly notification

Rationale for the Requirement

The requirement aims at facilitating the space surveillance segment to act as a central information hub for operators to inquire about the manoeuvrability status of other spacecraft operators. It is crucial to inform other operators via that channel as quickly as possible about anomalies so that, during potential encounters with active spacecraft, the other parties know soon that they have to act, if the situation required.

Methods to Assess Compliance

a. Review-of-design, to:
    1. Foresee an anomaly resolution board meeting as soon as possible after an anomaly occurred, to assess if the manoeuvrability status is affected;
    2. Ensure that space surveillance segment interfaces exist to describe and set spacecraft status, e.g. operators can register and set the manoeuvrability status of their spacecraft in the operator panel on www.space-track.org.

Mitigation Measures

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

a. To create a procedure that allows to assess the manoeuvrability status (ACTIVE or INACTIVE) of a spacecraft given its potential failure modes. For instance, a failure of a thruster can be directly affecting manoeuvrability, whereas other failures can inhibit the capability to address conjunction events in very different ways.

4.3.43 Requirement 5.3.3.5.h: Ephemerides forecast

Rationale for the Requirement

The requirement aims at improving the predictability of possible conjunctions. The orbit of an active spacecraft is typically determined more accurately using on-board data (e.g. from GNSS receivers), than via tracking by a space surveillance segment due to higher frequency and accuracy of measurements. Forecasted ephemerides for active spacecraft can include planned manoeuvres only known a priori by the operator. Mutually sharing those plans among operators improves the knowledge during close encounters of active spacecraft.

A space surveillance segment needs to identify the spacecraft by correlating new tracking data with catalogued states. A spacecraft is less accurately tracked if it manoeuvred in between observation passes. Sharing ephemerides files with the space surveillance segment ensures that the space surveillance segment keeps custody of all objects.

Methods to Assess Compliance

a. Review-of-design, to:
    1. Ensure capability to determine spacecraft orbit, e.g. on-board GNSS receiver, RF or laser ranging during ground station passes, or via space surveillance segment;
    2. Ensure capability to perform orbit determination, e.g. smoothing data using a non-linear least-squares or Kalman filter. The outcome of the orbit determination is a state and covariance at an orbit determination time, e.g. the time of last available observation or execution time;
    3. Be able to generate forecast ephemerides and covariance using an adequate orbit propagation method. The prediction of the covariance is described in Annex Section B.3.2 using a linear transformation and considering uncertain parameters such a manoeuvre performance or atmospheric density;
    4. Be able to analyse manoeuvre performance during operations, e.g. comparing expected manoeuvre magnitude with calibrated observed orbit change. This allows better describing and predicting the uncertainty introduced by future manoeuvres.

Mitigation Measures

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

a. To consider the orbit determination capability early in the design phase, assess accuracy, and potential changes to the spacecraft design (e.g. retroreflectors or GNSS receiver).

4.3.44 Requirement 5.3.3.5.i: CCSDS format

Rationale for the Requirement

The requirement aims at using the CCSDS formats, which includes standardized exchange formats for various scenarios to improve space safety. CCSDS formats are well adopted by the international community and, therefore, able to minimise misunderstandings in time-critical operations and exchanges between operators.

Methods to Assess Compliance

a. Review-of-design, to:
   1. Be able to generate and process ephemerides via the Orbit Data Messages (ODM);
   2. Be able to generate and process conjunction information via the Conjunction Data Message (CDM);
   3. Be able to generate and process re-entry information via the Re-entry Data Messages (RDM).

4.4 Disposal

4.4.1 Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft

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 the reliability of the disposal manoeuvres and the probability of occurrence of catastrophic impacts of space debris or meteoroids preventing disposal. 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 disposal phase includes all the actions performed by a spacecraft or launch vehicle orbital stage to permanently reduce its chance of accidental break-up and to achieve its needed long-term clearance of the protected regions. The probability of successful disposal also includes the passivation function to the reliability contribution, when performed (see Requirement 5.3.2.2.a: Passivation capability in 4.3.6 and subsequent requirements).

The probability of successful disposal is linked to the reliability of the items and functions used for disposal and to the catastrophic impact probability, which depends on multiple factors such as the size, design and materials of the space system, as well as orbit space debris density.

As an example of allocation, if the probability of catastrophic impacts is 2 % and the probability of successful disposal requirement is 90 %, then the minimum reliability of the disposal items and functions needed for compliance is 91,84 % (if the contribution of the availability of resources and the effect of radiation are negligible).

The assessment considers 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.

In case the orbit is confirmed at a late time during the design phase, for the contribution not related to the reliability of the disposal chain, the worst-case orbit and worst-case conditions are considered in accordance with the cases identified in the vulnerability analysis.

The fastest execution of disposal operations after the end of mission allowed by the propulsion capability is the preferred strategy, as it limits the residual on-orbit break-up risk. Space systems in absence of planned disposal operations (i.e. through natural orbit decay) are also expected to comply with the current requirement.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:
   1. Select reliable equipment for the disposal and passivation functions, to ensure the probability of success threshold;
   2. To have shielding, or protected accommodation (e.g. far from the external panels) for the critical units to avoid catastrophic impacts (refer to mitigation measures for Requirement 5.3.3.1.a: Collision risk assessment during design in 4.3.10).

b. Analysis, to compute, during the development phase, the probability of successful disposal at the planned time, considering:
   1. The reliability of the equipment in charge of the disposal function at the time of the disposal. The reliability analysis is computed by means of a Reliability Block Diagram (RBD), Fault-Tree Analysis (FTA), or an equivalent methodology, according to ECSS-Q-ST-30 [RD013]. The logic is the following:
      (a) applicable equipment: identify the list of all the equipment in the disposal functional chain and whose failure can prevent the successful disposal (directly or through failure propagation).
      (b) functional logic: build the Reliability Block Diagram (or equivalent) for the disposal function.
      (c) mission duration: define the applicable timeline for the disposal. Note that equipment used throughout the whole mission can have a higher impact on the disposal probability.
      (d) probability calculation: use the failure rate data for the different equipment to compute the disposal probability, extracted from one of the applicable methodologies from ECSS-Q-HB-30-08 [RD037] or reliability.SPACE [RD038] (listed below in order of preference):
         (1) supplier's data
         (2) physics of Failure (FIDES [RD039])
         (3) reliability data handbooks (MIL-HDBK-217F [RD040], NPRD-95 [RD041])
         (4) in-flight data (only applicable if the amount of data is sufficient and well justified)
         (5) similarity (by extrapolating the unknown failure rate of a component from a known one, if both are using a similar technology. It is important to apply the new mission environment and justify all assumptions).
   2. The availability of the resources needed to perform the disposal manoeuvre (if any);
   3. The probability of collisions with space debris or meteoroids preventing the successful disposal (refer to Annex C):
      (a) in absence of orbit details for the disposal phase (e.g. in an early phase), consider worst-case orbit and worst-case conditions identified in the vulnerability analysis;
      (b) consider only catastrophic impacts for the disposal function.

4. The probability of failures due to radiation preventing the successful disposal:
   (a) radiation failures with a permanent effect on disposal probability are taken into consideration throughout the whole mission duration.
   (b) as for non-permanent failures during the disposal phase, the probability of occurrence of one of those events potentially impacting is included.

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

Different ways of demonstrating a required level of reliability at a given confidence level exist, as described in [RD038] or [RD042], for instance. Failure rates are provided at 60 % confidence level in the commonly used handbooks, as described in ECSS-Q-HB-30-08 [RD037]. The confidence level can be up to 90 % or 95 % in some specific cases (such as for one-shoot critical elements, among others).

In case of mission extension, the reliability prediction is re-assessed (Requirement 5.4.1.2.e: Probability of successful disposal re-assessment in 4.4.8 and the subsequent requirements).

In case the mission present little or no information on failure rate data for its components, the reliability prediction is based on the available information from the supplier, i.e.:

a. If datasheet and parts lists are available, the reliability calculation is performed with FIDES [RD039], MIL-HDBK-217F [RD040]. Some handbooks differentiate between two methodologies: Parts Stress (detailed information on stresses – temperature, current) and Parts Count (number of parts, quality level, application environment). If little information is available, the latter can be used.

b. If no information on the equipment internal composition and functional scheme is available, reconstruction of the parts list via reverse engineering and Part Count methods are used. Alternatively, life tests, in-flight data, or work with similarity with other designs are investigated (only applicable on a sufficiently large number of data and proper justification).

Mitigation Measures

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

a. To consider, 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 equipment involved in the disposal functional chain and make sure the failure rate data is available.

c. To 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).

d. 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).

e. To avoid single point failures in the design which can prevent a successful disposal.

f. 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.

g. To introduce additional shielding.

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

4.4.2 Requirement 5.4.1.1.b: Disposal reliability for large constellations

Rationale for the Requirement

The requirement aims at providing a minimum allocation of 0,95 to the reliability contribution in the disposal probability of spacecraft in a large constellation in near Earth Orbit. The probability of failure of the equipment involved in the disposal functional chain of each spacecraft is, therefore, quantitatively limited.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to select reliable equipment for the disposal and passivation functions, to ensure a probability of success of 95 %.

b. Analysis, to compute, during the development phase, the contribution from the spacecraft reliability to the probability of successful disposal at the planned time, considering the reliability of the equipment for the disposal function at the time of the disposal. For further details, refer to the methods to assess compliance for the Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1.

Mitigation Measures

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

a. Mitigation Measures for the requirement the Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1.

4.4.3 Requirement 5.4.1.2.a / ECSS-U-AS-10, , 7.3.1.3: Disposal criteria

Rationale for the Requirement

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

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to implement flight operation procedures or a Failure Detection, Isolation and Recovery (FDIR) strategy to control all known failure modes, which can prevent the space system to perform a successful disposal.

b. Analysis, to ensure that a disposal plan is developed during the design phase and is consistent with the space system capability, resources, and mission profile, including that:

1. Specific criteria are defined for the initiation of the disposal operations based on the space system conditions and all relevant mission constraints;
2. 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;
3. The space system, at time of the disposal, is confirmed compliant with the Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1, and Requirement 5.4.1.1.b: Disposal reliability for large constellations in 4.4.2 (if applicable), and the re-entry safety requirements (ESSB-ST-U-004 [RD03]) in worst-case scenarios, if re-entry is foreseen.

Mitigation Measures

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

a. To identify in the disposal plan the worst-case scenarios and the well-defined initiation criteria and to allocate, from the early design stages, sufficient space system and ground system resources.

4.4.4 Requirement 5.4.1.2.a / ECSS-U-AS-10, 7.3.1.5: Contingency plan

Rationale for the Requirement

The requirement aims at implementing a contingency plan to respond to a rising risk of unsuccessful disposal of a 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.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review of design, to ensure that the space segment monitoring can support the evaluation of the ability of the satellite to successfully perform disposal activities.

b. Analysis, to ensure that worst-case failure scenario have been identified and captured from best engineering practice and available lessons learnt, and responses are included in a contingency plan, which is maintained during the operation phase until the end of disposal. The contingency plan can include actions such as (not exhaustive):

    1. The re-evaluation of the conditions of a controlled re-entry in case of mis-performance of the planned manoeuvre(s).
    2. The change in the propulsion branch or in the firing strategy for the disposal burns in case of anomalies in the propulsion system.
    3. The re-evaluation of the expected performance at end-of-life considering observed degradation or updated information of components' performance.
    4. Adaptation of the orbit determination strategy to meet the required positional accuracy for the disposal execution.

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 based on the relevant lessons learnt, as they become available, to be able to cope with newly identified worst-case failure scenarios.

b. To implement operational changes in response to anomalies to ensure the conditions for a successful disposal of the spacecraft are met (e.g. to assess the opportunity of anticipating the disposal manoeuvres and the execution of passivation).

c. To implement a space segment monitoring plan to ensure that the health and performance data of any unit and function of the space system used for the disposal is 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 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).

4.4.5 Requirement 5.4.1.2.b: Failure prognosis

Rationale for the Requirement

The requirement aims at anticipating the occurrence of failures, which can prevent the successful disposal. During the operational phase of a spacecraft, in order to improve the reliability models used during the design phase, the implementation of prognostic methods to provide updated reliability data is considered. Anticipating possible failures and wear out trends can be achieved by exploiting the information provided by degradation models since they allow quantitatively or statistically prediction at any time T0 of the future status (prognostic) of the units and estimate their Remaining Useful Lifetime. RUL is the difference between the current time and the failure predicted time, once a threshold is defined (see Figure 4-2).

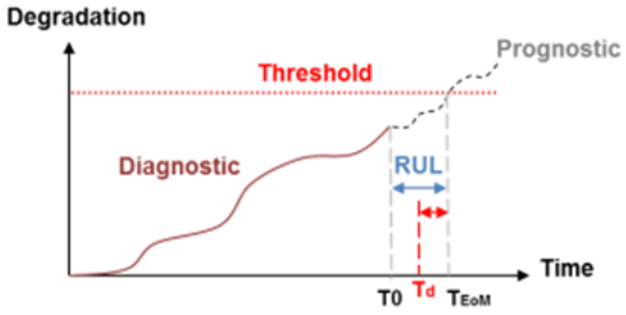

Figure 4-2: Prognostics and remaining useful lifetime

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to implement the failure prognostic methods. The outputs of a prognostic method are an estimated time to failure, which is also referred to as remaining useful life, or remnant life, and the associated confidence limit.

Possible prognostic approaches to assess compliance to this requirement are (see also Annex G, Approach 3: Prognostics):

a. Stochastic models, such as linear evolution of failure rate with time, lognormal law and "mortality" model [RD043] or Weibull laws.

b. Model-based, usually derived from the analysis of the Physic of Failure of the unit or data of ground tests.

c. Data trend monitoring (data-based approach).

To determine the remaining useful life of a component, it is important to know:

a. If the component experiences wear out phenomena and what is its current degraded state.

b. Which failure mode has initiated the degradation and what are the natural factors influencing its degradation.

c. How severe is the degradation (i.e. where the component is on the degradation curve) and what can be the impact at higher level.

d. How quickly is degradation expected to progress from its current state up to functional/physical failure.

e. What novel events can change (e.g. accelerate, retard) the expected degradation behaviour such as additional failures or spacecraft reconfiguration, respectively.

f. How many other aspects specifically linked to the prognostic method (e.g. the type of model, measurement noise, quality and quantity of data) affect the estimate of RUL.

More details on prognostic approaches can be found in [RD044]. The health monitoring or prognostics methods adopted by a mission can differ from those mentioned here, and, therefore, they are assessed by the customer and safety authority prior approval. Annex G provides information on the approach trade-off, degradation phenomena, impact on mission extension and EOL disposal, and examples of application. The other approaches complementing the latter are detailed in the Methods to Assess Compliance for Requirement 5.4.1.2.c: Constellation lessons learned, failures, and anomalies record in 4.4.6.

Mitigation Measures

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

a. To implement prognostic methods for anticipating possible failures and wear out trends.

b. If negative margins are observed in orbit (e.g. for available power, remaining propellant mass, unit performance), to consider different units configuration, or redundancy schemes.

c. If a negative trend on the spacecraft reliability is noted, to shorten the mission lifetime and to anticipate the disposal phase.

4.4.6 Requirement 5.4.1.2.c: Constellation lessons learned, failures, and anomalies record

Rationale for the Requirement

This requirement aims at enforcing operators and developers of constellations to plan in advance for storage of in-flight data, as well as collecting all detected anomalies. The requirement aims at requesting spacecraft developers and operators to make use of the on-board data to understand the overall health of the spacecraft and to implement necessary corrective actions.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Methods to Assess Compliance for Requirement 5.4.1.2.b: Failure prognosis in 4.4.5 and Requirement 5.4.1.2.d: Disposal critical function and equipment parameters update in 4.4.7 (refer also to Annex G).

Mitigation Measures

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

a. To perform periodical assessment and review of the health of the spacecraft, including analysis of critical parameters and application of one of the verification methods.

4.4.7 Requirement 5.4.1.2.d: Disposal critical function and equipment parameters update

Rationale for the Requirement

The requirement aims at allowing the update of reliability data during the lifetime of a spacecraft to improve the reliability assessment performed during the design phase through the implementation of prognostic methods.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to encompass the following methods, together with the Approach 3 (Requirement 5.4.1.2.b: Failure prognosis in 4.4.5, refer also to Annex G):
    1. Approach 0: Reliability as per CDR design, by modifying the redundancy scheme after the occurrence of failures, which lead to the loss of a redundancy path;
    2. Approach 1: Current (in-orbit), by assessing the spacecraft reliability considering the same failure rates "as per design", but exploiting additional information, i.e. new or different operational modes and units real operational environment, e.g. duty cycle, operational modes or cold/hot redundancy, which can differ from those considered in the CDR reliability model;
    3. Approach 2: Diagnosis and Return of Experience (REX):
        (a) health monitoring or diagnosis of the on-orbit status (e.g. in-orbit temperature) of the spacecraft units to update the reliability model and improve its accuracy. In-orbit data can be analysed to derive the degradation and the actual margins with respect to the design of spacecraft units, especially those needed for the mission extension or disposal;

(b) Return of Experience (REX), used to:

(1) compute the failure rate of one unit (e.g. using Chi-Square distributions). Note that a large number of cumulated hours are needed to obtain a lower failure rate than the one assessed with reliability standards

(2) update and better evaluate the unit failure rate by exploiting test and in-orbit data (e.g. using Bayesian techniques)

(3) better understand the anomalies and failures encountered by the unit and its statistical impact on previous mission (including extensions, if any) and on the disposal success.

Mitigation Measures

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

a. To perform periodical assessment and review of the health of the spacecraft, including the analysis of critical parameters and the application of one of the methods described above.

b. To update regularly the forecast probability of a successful disposal to take into account the as-flown conditions.

c. To identify the root-causes of degradation phenomena in the space system units.

d. To continue monitoring 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) and the disposal strategy is adapted accordingly.

4.4.8 Requirement 5.4.1.2.e: Probability of successful disposal re-assessment

Rationale for the Requirement

The requirement aims at ensuring that the spacecraft developer and operator work together in identifying the updated reliability in flight, considering wear out data and based on return of experience or any other failure prognostics methods which have been accepted by the approved authority.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to compute a more accurate reliability of the disposal elements, based on the described methods under Requirement 5.4.1.2.b: Failure prognosis in 4.4.5 and Requirement 5.4.1.2.d: Disposal critical function and equipment parameters update in 4.4.7, and on observed on-orbit events, which are jointly analysed through collaboration between the spacecraft developer and operator.

b. Other methods, if accepted by the approving agent.

Mitigation Measures

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

a. To not plan mission extensions if updated reliability figures (including wear out data and in-flight collected telemetry) based on return of experience, or any other failure prognostics methods, are not available.

b. To shorten the mission lifetime and to anticipate the disposal phase if failures are observed in orbit with a potential irreversible impact on the probability of successful disposal.

4.4.9 Requirement 5.4.1.2.f: Probability of successful disposal re-assessment occurrences

Rationale for the Requirement

The requirement aims at ensuring that the spacecraft operator, with possible support from the spacecraft developer, re-compute the probability of successful disposal, specifically the reliability of the disposal elements, in the case of one of the events mentioned in the requirement occur. As a minimum, the spacecraft operator is requested to recompute the reliability of disposal and provide the probability of successful disposal at 50 % of the nominal lifetime of the mission, unless agreed differently with the approving agent.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to determine the updated reliability of disposal, based on observed anomalies in orbit and considering wear out, as described in requirements Requirement 5.4.1.2.b: Failure prognosis in 4.4.5 and Requirement 5.4.1.2.d: Disposal critical function and equipment parameters update , in 4.4.7 specifically Approach 1 and 2 (and Approach 3, if deemed possible).

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.4.1.2.e: Probability of successful disposal re-assessment in 4.4.8.

b. In case of mission extension, to re-assess the probability of successful disposal at the time of the extended mission termination (the re-assessment is performed 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).

4.4.10 Requirement 5.4.1.2.g: In-flight health status assessment

Rationale for the Requirement

The requirement aims at ensuring that the spacecraft condition is monitored during the on-orbit operations to detect failures resulting in an unsuccessful disposal. Operational procedures need to be implemented to respond to a risk impacting the disposal phase to cope with unpredictable failure scenarios and effects. This facilitates the planning and execution of safe life extension in compliance with the Space Debris Mitigation needs.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to ensure that the worst-case failure scenarios have been identified and captured together with the responses from the spacecraft operator based on available on-orbit health status such to update the probability of successful disposal (refer to Sections Requirement 5.4.1.2.d: Disposal critical function and equipment parameters update in 4.4.7 and Requirement 5.4.1.2.f: Probability of successful disposal re-assessment occurrences in 4.4.9).

Mitigation Measures

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

a. To install several sensors, preferably redundant and independent, capable to detect the failures leading to the feared events identified for the probability of successful disposal.

b. To perform periodical assessment and review of the health of the spacecraft, including trend analysis of critical parameters.

c. To implement corrective actions, once a trigger event or an anomaly has occurred, i.e. operational control measures to allow minimising the increased risk of unsuccessful disposal (according to the operational procedures).

4.4.11 Requirement 5.4.1.2.h: Probability of successful disposal re-assessment information

Rationale for the Requirement

The requirement aims at ensuring that the probability of successful disposal is correctly re-assessed during the operational lifetime, considering the updates based on on-orbit data, and the information is exhaustively shared with the approving agent. The updates allow to anticipate the identification of possible failures during the operation in order not to compromise the capability for successful disposal at end of mission, and to facilitate the planning and execution of safe life extension in compliance with the Space Debris Mitigation needs.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to confirm that:

   1. The probability of successful disposal considers the methodologies implemented for the Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1, together with one or more of the approaches included in Requirement 5.4.1.2.b: Failure prognosis in 4.4.5 and Requirement 5.4.1.2.d: Disposal critical function and equipment parameters update in4.4.7.
   2. The adopted criteria for the mission continuation, termination or extension are in line with the 5.4.1.2.a.(Requirement 5.4.1.2.a / ECSS-U-AS-10, , 7.3.1.3: Disposal criteria in 4.4.3 and Requirement 5.4.1.2.a / ECSS-U-AS-10, 7.3.1.5: Contingency plan in 4.4.4);
   3. The update of the probability of successful disposal based on on-orbit data follows the approach for Requirement 5.4.1.2.f: Probability of successful disposal re-assessment occurrences in 4.4.9 and Requirement 5.4.1.2.g: In-flight health status assessment in 4.4.10;
   4. The number of collision avoidance manoeuvres foreseen up to the end of life and the respective Delta-v allocation are re-assessed in accordance with Annex B and Annex C.

Mitigation Measures

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

a. To not plan mission extensions if the updated probability of successful disposal figure based on return of experience, or any other on-orbit data, is not available.

b. To shorten the mission lifetime and to anticipate the disposal phase if failures are observed in orbit with a potential irreversible impact on the probability of successful disposal.

c. To perform periodical assessment and review of the health of the spacecraft, including trend analysis of critical parameters.

d. To constantly maintain the contingency plan and to keep it up to date based on the relevant lessons learnt, to be able to cope with newly identified worst-case failure scenarios.

e. To implement operational changes in response to anomalies to ensure the conditions for a successful disposal of the spacecraft are met (e.g. to assess the opportunity of anticipating the disposal manoeuvres and the execution of passivation).

4.4.12 Requirement 5.4.1.3.a: Preparation for removal: LEO conditions

Rationale for the Requirement

The requirement aims at implementing "preparation for removal" for spacecraft in specified mission risk scenarios in LEO.

In general, "preparation for removal" ("design for removal") is the set of specific design features, which allow a client (target) spacecraft readiness for being removed by an external servicer (chaser), and cover aspects such as capture interfaces, support to relative navigation, attitude reconstruction from ground, detumbling. "Preparation of removal" can enhance the chances of successful disposal of the spacecraft to comply with the space debris mitigation and re-entry safety requirements. "Preparation for removal" can facilitate the operation of removal of a spacecraft if potential issues associated to removal operations (including close proximity and rendezvous operations) are preventively tackled. "Preparation for removal" can, therefore, enhance reduction of complexity and increase of efficiency for a removal operation.

"Preparation for removal" aims at:

- De-risking the possibility of the removal of space objects in high-risk scenario by an external servicer, in case it is unable to perform its own disposal directly.
- Defining a minimum set of design implementations that can allow the safe clearance of the space object by an authorised and reliable servicer space object.
- Mitigating the risk of accidental events leading to the generation of space debris during the execution of removal operations.

For a client (target) spacecraft in LEO, removal operations normally imply capture, detumbling, de-orbit, and safe re-entry of the client spacecraft with the assistance of a servicer spacecraft.

Removal can occur 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), but is unable to perform end of life functions with respect to removal from orbit.
- An "uncooperative scenario", i.e. when the client spacecraft is non-operational (i.e. the client spacecraft lost its control) and tumbling.

Example of cooperative and uncooperative mission scenario are reported in [RD045]. In a cooperative scenario the spacecraft is able to acquire and maintain a stable attitude (i.e. using the Safe Mode) assuring that the AOCS does not react against the capture (i.e. interfering with close proximity operations, activation of the thrusters, appendages obstructing the Field of View (FoV) of relative navigation supports).

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:
   1. Identify the applicability of "preparation for removal" measures;
   2. Ensure suitable implementations of "preparation for removal" measures.

b. Analysis, to:
   1. Assess the expected number of casualties, the natural orbit decay and the cumulative collision probability following the methodologies described respectively Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2 and Requirement 5.4.2.3.a: LEO protected region clearance – objects operating in LEO in 4.4.21;
   2. Ensure the correct implementation of "preparation for removal" measures.

c. Test, to verify "preparation for removal" measures.

Please note that the analysis at point b.1 on the expected casualty risk on ground is performed considering the case of uncontrolled re-entry (even in the case where a controlled re-entry is planned); similarly, the lifetime and cumulative collision probability assessment are performed considering the free drift from the operational orbit.

Mitigation Measures

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

a. To design a spacecraft compliant to expected number of casualties lower than 10^{-4} for uncontrolled re-entry following "design for demise" guidelines specified in [RD088].

b. To operate in an orbit with either natural decay below 5 years or cumulative collision probability with space debris larger than 1 cm lower than 10^{-3} in free drift from operational orbit.

4.4.13 Requirement 5.4.1.3.b: Preparation for removal: GEO conditions

Rationale for the Requirement

The requirement aims at defining that a spacecraft operating in the GEO protected region is always prepared for removal in view of the poor natural orbit clearance effects in GEO.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to ensure suitable implementations of "preparation for removal" measures.

b. Analysis, to ensure the correct design of "preparation for removal" measures.

c. Test, to verify "preparation for removal" measures.

4.4.14 Requirement 5.4.1.3.c: Preparation for removal: features

Rationale for the Requirement

The requirement aims at specifying the high-level features needed for a spacecraft to be prepared for removal both in cooperative and uncooperative scenario in terms of mechanical interfaces and relative navigation supports. The compliance is met if compatibility with at least one possible removal service interface is demonstrated.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:
   1. Ensure suitable mechanical interfaces and support for relative navigation solutions;
   2. Prevent the interference of other hardware with potential capture operations;
   3. Ensure enough clearance for the relative navigation aids.

b. Analysis, to:
   1. Derive thermo-mechanical load environment for interface during capture, detumbling, removal and re-entry phases;
   2. Verify stiffness and strength for the interface at spacecraft structure and passive interface level;
   3. Verify the location on the spacecraft and visibility of relative navigation supports is robust to different and changing illumination conditions;
   4. Assess the compatibility with different rendezvous sensors, if possible.

c. Test to:
   1. Verify the mechanical interface by static tests at structural level and dynamic tests at equipment level (i.e. sine, random and shock);
   2. Verify visibility of relative navigation supports.

The strength to mechanical loads is derived from the loads exchanged during capture, detumbling, removal and re-entry phases. The re-entry strategy (i.e. uncontrolled or controlled) is a key design driver of the mechanical interface, and in particular:

- Metallic parts (handles) mounted on the spacecraft structure to allow its capture with a robotic gripper before rigidisation of the compound spacecraft and removal vehicle are example of mechanical capture interface in case of uncontrolled re-entry.
- Launch Adapter Ring is an example of interface in case of controlled re-entry due to higher load environment with respect to uncontrolled re-entry case.

Good practices for the interface design are the minimisation of the risk of escape after capture and the minimum perturbation of the approach dynamics. [RD045] provides set of requirements and technological solutions for mechanical interfaces and mechanical loads definition for capture, detumbling and removal for LEO spacecraft both in cooperative and uncooperative scenario.

For what concerns relative navigation, 2D and 3D markers are considered examples of supports for far and near range navigation in terms of attitude, distance and velocity estimation by the removal vehicle. Compatibility with different rendezvous sensors (i.e. cameras, radar and lidar) for the relative navigation supports is considered a good practice for design.

[RD045] provides set of requirements and technological qualified solutions for support to relative navigation for LEO spacecraft.

4.4.15 Requirement 5.4.1.3.d: Preparation for removal: attitude evolution

Rationale for the Requirement

The requirement aims at de-risking the possibility of the removal of space objects. Assessment of the long-term evolution of the spacecraft attitude in free drift in its operational orbit allows proper design of the removal operation.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to assess the long-term evolution of spacecraft attitude, starting from a nominal condition expected in case of failure in orbit:

   1. For Phase 0 and A: As moments of inertia, centre of mass, and outer shape of a space object are generally not yet known accurately, analytical attitude models based on spin-averaged and orbit-averaged torques, or analyses based on empirical data from space surveillance segments, on similar objects similar, can be used for estimation of the attitude evolution;

   2. For Phase B, C and D: Two cases can generally occur for a space object with constrained (well-known) moments of inertia, centre of mass, and outer shape, i.e. aerodynamically optimised (e.g. Very Low Earth Orbit missions), or simply shaped (e.g. spheroids, or elongated cylinders such a launch vehicle stages) can have accurate analytic attitude motions models and continue to use them. Many space objects, however, including classical spacecraft, are optimised for a long-term attitude motion and numerical simulations are important to assess the spacecraft attitude evolution. Tools such as iOTA (In-Orbit Tumbling Analysis, available for ESA member states upon request) allows to perform short (days), medium (months) and long term (year) propagation of the orbit and attitude motion (6-DoF) of spacecraft in Earth orbit, starting from the centre of mass and moments of inertia of the spacecraft.

b. Analysis, either analytically or numerically, to estimate the likelihood of entering the attitude modes identified under point a. hereabove. It is known that chaotic motion can exists (e.g. on eccentric orbit or when multiple torques have similar orders of magnitude), however the analysis aims to identify and constrain probable long-term motion categories.

4.4.16 Requirement 5.4.1.3.e: Preparation for removal: LEO uncooperative features

Rationale for the Requirement

The requirement aims at specifying the features needed, in addition to Requirement 5.4.1.3.c: Preparation for removal: features in 4.4.14, for a spacecraft in LEO protected region to be prepared for removal in an uncooperative scenario. The magnitude of the spacecraft angular rate is a major design driver for chaser GNC, propulsion and telecommunications subsystems design, and it is important that it is kept as low as possible. However, once the spacecraft is uncontrolled, the angular rates cannot be easily controlled, and the damping of the tumbling motion rates is essential to make the removal feasible.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:

1. Evaluate the inclusion of supports attitude reconstruction on ground, such as Laser Retroreflectors (LRR) corner cubes. [RD045] provides a valid reference for the requirement definition and for technological qualified solutions for attitude reconstruction from ground based on LRR corner cubes;
2. Ensure the limitation and damping of spacecraft angular rates through measures such as (not exhaustive);
3. Have angular rates damping systems of device such as Passive Magnetic Detumbling System through short-circuiting Satellite Magnetic Torquers;
4. Have Safe Mode designed to minimise the solar radiation pressure torque by orientation of appendages;
5. Have passivation of propulsion system not increasing the spacecraft angular rates (e.g. zero-torque venting pipe outlet).

b. Analysis, to:

1. Assess the performance (i.e. accuracy in the angular rate vector magnitude and direction) and the process for attitude reconstruction from ground considering available sensors. Passive optical telescopes, lasers or synthetic radars are example of sensors capable of observing cooperative targets with retroreflectors or non-cooperative targets. Depending on the sensor, products such as light curves, laser ranges, RCS evolution are obtained, which can be sued to reconstruct the trajectory and the attitude of the object. iOTA (In-Orbit Tumbling Analysis, available for ESA member states upon request) is a valid tool for reconstruction of attitude, requiring as input the Spacecraft dynamics properties (CoM, CoG dry mass and MoI), the initial attitude state and the surface material properties (diffuse reflection and absorption factors);
2. Characterise the angular rate evolution over time and estimate the time requested to reach the target value of 1 deg/s.

Section 2.2.3 of [RD045] provides further information and requirements for the AOCS.

4.4.17 Requirement 5.4.1.3.f: Preparation for removal: cooperative features

Rationale for the Requirement

The requirement aims at specifying the features needed to facilitate the removal in a cooperative scenario. The compliance is met if compatibility with at least one possible removal service interface is demonstrated, with [RD045] provided as a possible design reference.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:

1. Ensure that the safe mode attitude and configuration is compatible with the cooperative capture and removal from a servicer. Examples of recommended attitudes can be found in Section 2.1.3 of [RD045];
2. Ensure that the AOCS does not react against the capture and the use of AOCS actuators does not impact on the servicer close proximity operations (e.g. by perturbing the relative navigation sensors);

3. Ensure that stable angular rates can be achieved. Example target values for the angular rates can be found in Section 2.1.3 of [RD045];
4. Assess the feasibility and safety of passivation after confirmed capture.

b. Test, to verity the functions of the system modes.

4.4.18 Requirement 5.4.1.3.g: Preparation for removal: documentation

Rationale for the Requirement

The requirement aims at documenting the implemented "design for removal" features, including geometrical and material properties, to facilitate the design and operation of a servicer removal mission.

Methods to Assess Compliance

The verification methods used to assess the compliance is:

a. Review-of-design, to ensure information is properly documented about "preparation for removal" functions and interfaces for spacecraft or launch vehicle orbital stage involved in close proximity and removal operations. Sections 2.1.7 and 2.2.7 of [RD045] are a valid reference for the minimum documentation needed the removal service providers.

4.4.19 Requirement 5.4.2.1.a: General Earth orbit clearance

Rationale for the Requirement

The requirement aims at defining disposal strategies for spacecraft and launch vehicle orbital elements operating continuously or periodically in Earth orbit by minimizing the interference with other space objects.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to:
   1. Assess the feasibility of the possible disposal options, e.g. manoeuvred de-orbit, disposal towards unstable trajectories (i.e. high eccentricity variation) resulting into Earth re-entry, disposal towards stable graveyard orbits (i.e. low eccentricity variation);
   2. Define the disposal strategy consistently with the space system proven capability, the reliability of its propulsion system, the allocated propellant mass (in case of disposal manoeuvres), and the reliability of the disposal systems (e.g. passive de-orbit with sails, tethers, and other non-propulsive devices);
   3. Evaluate the orbital trajectory propagation of the space system for at least 100 years after the completion of the planned disposal (Annex A) in order to demonstrate:
      (a) LEO Protected Region clearance compliance (Requirement 5.4.2.3.a: LEO protected region clearance – objects operating in LEO in 4.4.21 and the subsequent requirements), when operating or crossing LEO.
      (b) the selected graveyard orbit with negligible interference (e.g. considering the dwell time) with the GEO Protected Region and with the orbits of known constellations (e.g. GNSS) identified in the Requirement 6.3.a: List of constellations and inhabitable space objects in 5.2.1 for at least 100 years.

(c) the selected graveyard orbit, or disposal trajectory, with cumulative collision probability with space objects larger than 1 cm below 10^{-3} for up to 100 years after the end of life applying the methodology described in Annex C.3.

4. Validate the selected disposal approach with respect to relevant sources of uncertainty (e.g. disposal epoch, cross-sectional area, other parameters, as listed in Annex H);

5. Demonstrate that the disposal strategy is compliant in worst-case scenarios, e.g. for space systems equipped with propulsion or AOCS system, which are technology demonstrators, or have unknown or low reliability, when energy can accidentally be added to the space system to cause possible deviation from compliance.

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

a. In case re-entry is foreseen or possible (e.g. in degraded cases): the compliance with the requirement needs demonstration of compliance with the re-entry safety requirements (ESSB-ST-U-004 [RD03])).

b. For space objects in MEO, studies suggest ensuring a minimum clearance from GNSS operational constellations of at least ±300 km in semi-major axis with eccentricity below 0,001 (TBC) (orbit circularisation) 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.

c. For operations close to unbounded Earth orbits 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 (Annex G).

d. For disposal into heliocentric orbits, it is important to assess the consequences of all possible disposal scenarios to ensure compliance with Planetary Protection requirements (ECSS-U-ST-20 [RD046]).

e. Additional elements for specific scenarios such as interplanetary missions are provided in Annex G.

Mitigation Measures

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

a. To perform, for launch vehicles, a direct delivery to GEO, to lower the apogee below the GEO Protected Region (e.g. apogee lower than 235-680 km from GEO) to avoid later interference with GEO [RD047].

b. 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.

4.4.20 Requirement 5.4.2.2.a: GEO protected region clearance

Rationale for the Requirement

The requirement aims at providing the criteria for the GEO Protected Region clearance for spacecraft placed in a graveyard orbit. The clearance criteria are derived from IADC recommendations [RD048] with the intention to avoid permanent or periodic interference with the GEO protected regions under the effect of perturbation forces (i.e. lunisolar, geopotential, and solar radiation pressure). More details on the different contributions can be found at [RD047].

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to ensure that the planned disposal manoeuvres secure the space system into a disposal orbit outside the GEO Protected Region, considering 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 manoeuvres (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 (e.g. considering the dwell time in the protected region);
3. 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 H).

4.4.21 Requirement 5.4.2.3.a: LEO protected region clearance – objects operating in LEO

Rationale for the Requirement

The requirement aims at identifying the most sustainable disposal status (orbit), which limits the risk of collision, especially if generating large number of long-lived debris and ultimately rendering some orbital regions within LEO completely unusable. In order to achieve this, it is of importance to both limit the time left in orbit when a space object does not have the capability to manoeuvre and additionally to limit the cumulative collision probability after end of life.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:

1. Ensure that the planned disposal manoeuvres allow the space system to clear the LEO Protected Region, considering the space system status and flight parameters, the reliability of its propulsion system, and the availability of propellant mass;
2. For space systems with passive de-orbiting systems (i.e. non-propulsive systems): identify the space system physical characteristics (including appendage, if influencing the cross-sectional area for the drag) including geometrical shape, long-term expected attitude motion, mass and material properties as well as reliability of the de-orbiting system;
3. For space systems that have no capability of performing disposal manoeuvres: identify the space system physical characteristics (including appendage, if influencing the cross-sectional area for the drag) including geometrical shape, long-term expected attitude motion, mass and material properties that influence the orbital natural decay;
4. Identify appendages that contribute to the cross-sectional area calculation for the collision risk assessment, i.e. extensions to the main structure such as solar panels or dish or radar antennas that shatter upon impact but not structures that tear only without creating debris upon impact (e.g. wire grids or foils).

b. Analysis, to:

1. Demonstrate that the space system has a limited orbital lifetime within the LEO Protected Region based on orbital trajectory propagation considering the orbital and attitude state after the completion of the planned disposal that is compatible with the overall 5-year limit

(see Annex A and Annex H for recommendations regarding the orbital propagation and stochastic analysis respectively);

2. Demonstrate that the space system in case of no recurrent manoeuvre capability, has a limited orbital lifetime within the LEO Protected Region during its normal operations based on orbital trajectory propagation considering the orbital and attitude state after injection that is compatible with the overall 5-year limit (see Annex A and Annex H for recommendations regarding the orbital propagation and stochastic analysis respectively);
3. For space systems which have no capability to perform any manoeuvre: demonstrate that the presence in orbit is limited to the minimum duration compatible with the mission objective and not exceeding 5 years from the on-orbit injection epoch;
4. Demonstrate that the cumulative collision risk with space objects later than 1cm is below 10^{-3} based on orbital trajectory propagation considering the orbital and attitude state after end of life, the geometrical shape and relevant appendages, and accounting for the space debris environment model requirements in Section 5 (see Annex C for recommendations regarding the risk estimation methodology);
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 or in an increase of the accepted cumulative collision risk with space objects larger than 1 cm, taking into account duty margins in the operational altitude, i.e. such that re-entry is still possible within 5 years and the cumulative collision probability until re-entry remains below 10^{-3} from the worst-case highest reachable apogee altitude;
6. 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 H).

In addition, 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 [RD03]).

Mitigation Measures

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

a. To select a disposal or injection orbit on which natural perturbations lead to a permanent clearance of the LEO Protected Region within 5 years and a cumulative probability of collision with objects larger than 1 cm below 10^{-3} 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. To limit the cross-sectional area that can create space debris upon impact with space objects larger than 1 cm after the end of life.

4.4.22 Requirement 5.4.2.3.b: LEO protected region clearance – objects crossing LEO

Rationale for the Requirement

The requirement aims at setting risk-limiting criteria for space systems entering the LEO Protected Region because of natural effects. Any abandoned space system leads to an increase of the collision risk. For certain orbits, such as highly eccentric orbits or Lagrange point orbits, the risk can over time affect the LEO Protected Region. In addition, disposal from the MEO or GEO Protected Region can include eccentricity growth strategies that inevitably also enter the LEO Protected region.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:
   1. Ensure that the planned disposal manoeuvres allow the space system to clear the LEO Protected Region in no longer than 100 years after end of life (from the orbit of the nominal mission) and within 25 years from the first LEO intersection (after the nominal mission), considering the space system status and flight parameters, the reliability of its propulsion system, and the availability of propellant mass;
   2. Consider also Methods to Assess Compliance for Requirement 5.4.2.3.a: LEO protected region clearance – objects operating in LEO in 4.4.21 (review-of-design).

b. Analysis, to:
   1. Demonstrate that the space system has limited interference within the LEO Protected Region based on orbital trajectory propagation considering the orbital and attitude state after the completion of the planned disposal that is compatible with the overall 100 year limit (see Annex A and Annex H for recommendations regarding the orbital propagation and stochastic analysis respectively);
   2. Demonstrate that the space system in case of no recurrent manoeuvre capability, has limited interference within the LEO Protected Region during its normal operations based on orbital trajectory propagation considering the orbital and attitude state after injection that is compatible with the overall 25-year limit (see Annex A and Annex H for recommendations regarding the orbital propagation and stochastic analysis respectively);
   3. Demonstrate, for space systems which have no capability to perform any manoeuvre, that the presence in orbit is limited to the minimum duration compatible with the mission objective and not exceeding 25 years from the epoch of first interference with the LEO Protected Region;
   4. Demonstrate that the cumulative collision risk with space objects later than 1 cm is below 10^{-3} based on orbital trajectory propagation considering the orbital and attitude state after end of life, the geometrical shape and relevant appendages, and accounting for the space debris environment model requirements in Section 5 (see Annex C for recommendations regarding the risk estimation methodology);
   5. Demonstrate, for space systems equipped with propulsion or AOCS system, which are technology demonstrators, or have unknown or low reliability, 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 or in an increase of the accepted cumulative collision risk with space objects larger than 1 cm, taking into account duty margins in the operational altitude, i.e. such that the total orbit lifetime after end of life does not exceed 100 years, the cumulative collision probability until re-entry remains

below 10^{-3} and the orbit lifetime from the epoch of first intersection with the LEO protected region does not exceed 25 years from the worst-case highest reachable apogee altitude;

6. 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 H).

In addition, 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 [RD03]).

Mitigation Measures

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

a. To select a disposal or injection orbit on which natural perturbations lead to a permanent clearance of the LEO Protected Region within 100 years after end of life and 25 years after first intersection with the LEO Protected Region, a cumulative probability of collision with objects larger than 1 cm below 10^{-3} until re-entry.

b. To refer to the guidelines in Annex G for operations outside the protected regions.

4.4.23 Requirement 5.4.2.3.c: LEO protected region clearance – large constellations

Rationale for the Requirement

The requirement aims at providing a more stringent prescription than in Requirement 5.4.2.3.a: LEO protected region clearance – objects operating in LEO in 4.4.21 for the disposal orbit of spacecraft of large constellation in LEO due to the large number of spacecrafts involved. The apogee altitude limit (375 km) is set to limit the interference with other operators, including inhabited space systems.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design to

1. Ensure that the disposal manoeuvres, if planned, allow the space system to reach the desired apogee altitude, considering the space system status and flight parameters, the reliability of its propulsion system, and the availability of propellant mass;
2. Ensure reliable system functionality down to altitudes to 375 km;
3. Ensure that the disposal strategy is robust considering the likelihood and effects of worst-case scenarios in term of energy that can accidentally be added to the operational space system, which can modify its orbit.

Mitigation Measures

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

a. To perform an initial manoeuvre to first move the spacecraft outside the constellation operational orbit before performing the remaining manoeuvres to reach the desired apogee altitude.

4.4.24 Requirement 5.4.2.4.a: Insertion orbit for constellations – no crossing other constellations

Rationale for the Requirement

The requirement aims at limiting the compounded collision risk associated with infant mortality of spacecraft of constellations, defining orbit insertion criteria to facilitate testing of operational capability in a lower risk scenario.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to demonstrate that the selected initial insertion orbit does not cross the orbits of known constellations that operate at a fixed operational altitude, as identified in Requirement 6.3.a: List of constellations and inhabitable space objects in 5.2.1.

b. The analysis is expected to be updated until the approval of the Space Debris Mitigation Plan, or equivalent. However, it is considered good practice to update the assessment during later phases.

Mitigation Measures

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

a. To confirm that the disposal plan is still valid at time of the disposal execution in view of the latest status of the space environment.

4.4.25 Requirement 5.4.2.4.b: Insertion orbit for constellations – cumulative collision probability threshold

Rationale for the Requirement

See Rationale for the Requirement 5.4.2.4.a: Insertion orbit for constellations – no crossing other constellations in 4.4.24.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to assess the cumulative collision probability, considering the free drift scenario from the initial insertion state, described in Annex C.3, in line with validation and verification requirements: Requirements 6.2.a and 6.2.b: Space debris and meteoroid models in 5.1.1 and 6.2.e (Requirements 6.2.f-g: Probabilistic assessment of the orbital lifetime in 5.1.5).

The approving agent can decide to relax this requirement in case a low risk of dead-on-arrival is expected given the adoption of quality standards (e.g. ECSS-Q-ST-10 [RD049], ECSS-Q-ST-20 [RD050]).

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.4.2.4.a: Insertion orbit for constellations – no crossing other constellations in 4.4.24.

4.4.26 Requirement 5.4.2.4.c: Insertion orbit for large constellations – natural decay threshold

Rationale for the Requirement

See Rationale for the Requirement 5.4.2.4.a: Insertion orbit for constellations – no crossing other constellations in 4.4.24.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to assess the orbit lifetime, considering the free drift scenario from the initial insertion state, described in Annex A, in line with validation and verification requirements 6.2.f and 6.2.g (Requirements 6.2.f-g: Probabilistic assessment of the orbital lifetime **in 5.1.5**).

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.4.2.4.a: Insertion orbit for constellations – no crossing other constellations in 5.1.5.

4.5 Re-entry

4.5.1 Requirement 5.5.a: ESSB-ST-U-004 applicability

Rationale for the Requirement

The requirement aims at applying the ESA Re-entry Safety Requirements [RD03] for the design and operations of any ESA inhabitable space system, i.e. procured by ESA or operated under ESA responsibility.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to check that ESA Re-entry Safety [RD03] are made applicable.

b. Analysis, to assess the casualty risk compliance (see Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2).

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2, and Requirement 5.5.d: Re-entry casualty risk threshold for large constellations in 4.5.4.

4.5.2 Requirement 5.5.b: Re-entry casualty risk threshold

Rationale for the Requirement

The requirement aims at limiting the risk of causing casualties per re-entry event by adopting the ESA risk threshold and mitigation measures. The requirement complements the ESA Re-entry Safety Requirement [RD03].

The expected number of casualties per re-entry is the number of people who are predicted to be killed or seriously injured by the re-entry of a space object, which corresponds to an approximation of the re-entry casualty risk. Therefore, the nomenclature "casualty risk" is conventionally used with similar meaning of "expected number of casualties per re-entry".

Furthermore, this requirement:

a. Specifies that the application of proven "design for demise" is always considered as a mitigation measure for any destructive re-entry, either controlled or uncontrolled, and it is the preferable option as it can likely lead to the lightest and smallest possible impacting elements, therefore, the lowest risk.

b. Clarifies that the assessment of the expected number of casualties per re-entry of a spacecraft, or launch vehicle orbital stage, take also into account the possible parts released on-orbit before its re-entry (e.g. kick-stage, a payload adapter), even if they re-enter independently (as in [RD051]).

c. Clarifies that the assessment of the expected number of casualties per re-entry of a spacecraft, or launch vehicle orbital stage, composed of combined or stacked space objects (e.g. docked servicing spacecraft, space tugs, or spacecraft assembled in orbit) is computed for the combined or stacked space object (as in [RD051]).

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to ensure that the risks associated to re-entry of the space system, either foreseen or planned, are identified and mitigated through the mission procurement, design, operation, and disposal phases, according to clause 6 (Re-entry Safety Verification Requirements) of the ESA Re-entry Safety Requirements [RD03].

b. Analysis, to assess the expected number of casualties, according to clause 6 (Re-entry Safety Verification Requirements) of the ESA Re-entry Safety [RD03] and the guidelines provided in Annex D.

c. Analysis, in the case of controlled re-entry, to assess the severity of a failed re-entry strategy (e.g. resulting in a uncontrolled re-entry) and the corresponding casualty risk on ground to obtain the combined casualty risk as detailed in Annex D.

d. Test, to support the validity of the assumptions taken for the assessment of the expected number of casualties and "design for demise" implementations in the space system, following the guidelines provided in DIVE [RD088]. For example, testing is particularly relevant for the so called "design for containment" methods, given that solely simulation tools (e.g. DRAMA) are insufficient to demonstrate the validity of the design approach.

Mitigation Measures

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

a. To implement design for demise strategies by designing 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, or Tungsten) or appropriate space system architecture favouring structural demise. The application of "design for demise" can reduce the expected number of casualties, resulting in an effective way to keep low cost and complexity, e.g. lighter structural mass, lower fuel load, and lower operational cost in case of uncontrolled re-entry of a demisable space system. Guidelines for the implementation are provided in DIVE [RD088]. Different studies are currently on-going to assess the impact of

re-entries on the atmosphere, but the results are not mature enough to prescribe specific recommendations in terms of material selection.

b. To perform a controlled re-entry over a designated areas in compliance with the ESSB-ST-U-004 [RD03]. For additional guidance check the guidelines provided in Annex D.

c. Other design or operation practices can be implemented only if authorised by the approving agent and provided that their effectiveness in satisfying the requirement without creating further safety hazards is demonstrated by analysis or test.

4.5.3 Requirement 5.5.c: Re-entry casualty risk – probabilistic assessment

Rationale for the Requirement

The requirement aims at assessing the expected number of casualties per re-entry of a spacecraft, or launch vehicle orbital stage, including elements thereof, which is affected by several uncertainties. Therefore, it is important to use a probabilistic approach (such as Monte-Carlo) to account for them. The requirement complements the ESA Re-entry Safety Requirements [RD03].

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:
   1. Identify the relevant sources of uncertainties for the re-entry scenario. Additional information on the minimum level of uncertainties to be considered is provided in Table D-1: Initial conditions and uncertainties depending on re-entry type. Further guidelines are available in Section 4.2.4 of DIVE [RD088] and Annex D;
   2. Ensure the definition of representative distributions for the variables related to the relevant uncertainties.

b. Analysis, to:
   1. Assess the robustness of the selected approach with respect to the relevant sources of uncertainty (e.g. Re-entry epoch, re-entry trajectories, other parameters), adopting the following approach:
      (a) For missions up to Phase A, the probabilistic assessment can be performed by varying the anomaly along the orbit and the atmosphere density. When ESA's DRAMA tool is used for the analysis, such options can be directly reached from the Graphical User Interface (GUI).
      (b) For missions in later phases of development where the compliance to the Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2 is achieved by relying on the adoption of "design for demise" solutions, a deeper probabilistic analysis is needed as elaborated in the point b2, DIVE [RD088] and [RD086]. Some example scripts to perform such analysis with ESA's DRAMA tool are available at [RD060].
      (c) For missions using SCARAB (or other approved spacecraft-oriented tool) is understood that a probabilistic assessment in the sense of a large Monte Carlo analysis can be too demanding and it is not considered needed. In this case, the requirement can be fulfilled through a parametric analysis (i.e. a few datapoints) with a variation of the initial states (e.g. changing the initial anomaly) and of the atmosphere density.

(d) For small satellites (e.g. CubeSats), with a computed null casualty probability in the nominal scenario, the request for the probabilistic assessment can be relaxed.

2. Demonstrate the demise of an element, equipment, or partial/full component of the re-entering object. In this case, the minimum number of simulations is driven by the acceptable confidence level. Usually, this value is between 90 % and 95 %. Further guidelines in Annex D and Annex H;
3. Evaluate the expected number of casualties. In most cases, the evaluation of the median is sufficient to validate at system level. In case of multi-modal distribution of the results, with at least one mode above the threshold, it is important to take into account the mitigation measures for the specific mode and repeat the probabilistic assessment;
4. Re-assess the probabilistic approach for the expected number of casualties periodically to account for the variability of the relevant uncertainties at different mission phases, as defined in Table D-2: Characteristics of the different re-entry modelling approaches.

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2, and Requirement 5.5.d: Re-entry casualty risk threshold for large constellations in 4.5.4.

4.5.4 Requirement 5.5.d: Re-entry casualty risk threshold for large constellations

Rationale for the Requirement

The requirement aims at controlling the aggregate risk associated to the re-entry of spacecraft from a large constellation. Large number of spacecraft adopting the similar design and mission profiles, especially if operating in LEO, can lead to an equivalent large number of re-entries, therefore, an increasing aggregate expected number of casualties in the long-term, if safer design measures are not taken. The requirement complements the ESA Re-entry Safety Requirements [RD03].

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Methods to Assess Compliance for Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2 .

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2 .

b. To design the spacecraft, including platform, appendages and payloads, as the most demisable, lightest and smallest possible such that the casualty area is the closest to zero.

c. To plan the launch date of successive spacecraft from the constellation only provided that the number of casualties resulted from previously executed re-entries of similar spacecraft from the constellation, is still confirmed to be below the 10^{-6} threshold per spacecraft, otherwise to introduce mitigation measures.

4.6 Dark and quiet skies

A set of requirements was introduced to address the emerging topic of dark and quiet skies. The purpose of these requirements is to assess the optical and radio frequency interference of the space objects with ground and on-orbit (LEO) astronomy and protect radio astronomy observations and radio quiet zones.

Acceptable threshold values for spacecraft brightness have been derived from the Vera Rubin Observatory by simulating trails and crosstalk artifacts and correct them below noise level with post-processing algorithms [RD052]. They were reported to UN COPUOS in the "Dark and Quiet Skies for Science and Society – Report and recommendations" [RD053]. The report recommends that the apparent magnitude m_{Sat} of a spacecraft is limited by

$$m_{Sat} \geq 7.0 + 2.5 \cdot \log\left(\frac{h_{Sat}}{550\ \text{km}}\right)$$

where the apparent magnitude describes the brightness of the spacecraft in the visible band (Johnson V bandpass) of the spectrum and h the orbital height. The limits in **[RD052]** are originally derived for multiple colour bands for constellation on circular orbits at 550 km. The equation rescales the threshold to any altitude. The velocity in the sky and frequency of bright passes of the satellite is not considered in the recommendation.

The apparent magnitude can be computed from the irradiance ratio of the spacecraft with respect to the solar illumination irradiance E_{Sun} and the apparent magnitude of the Sun m_{Sun} at Earth distance

$$m_{Sat} = m_{Sun} - 2.5 \cdot \log\left(\frac{E_{Sat}}{E_{Sun}}\right)$$

The irradiance of the spacecraft is a function of the solar illumination, the distance d to the observing instrument, any extinction, and the reflectance function $f_r(\vec{l}, \vec{v})$

$$E_{Sat} = E_{Sun}\ \frac{f_r(\vec{l}, \vec{v})}{d^2}$$

The vector $\vec{l}$ describes the direction of the incoming light from the Sun and $\vec{v}$ the direction from the spacecraft to the observing sensor. The function describes the overall effect of individual surfaces and materials, self-shadowing and specular as well as diffuse reflection in the analysed visual bandwidth.

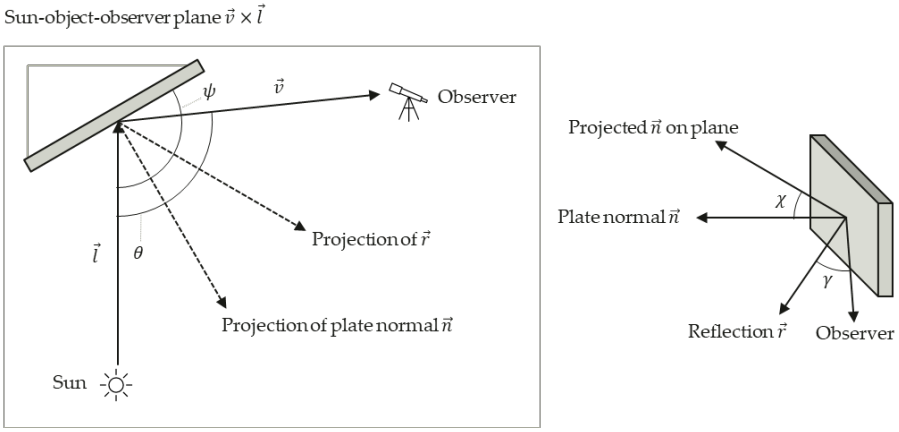

Figure 4-3: Observation geometry for a flat plate.

The reflectivity of the individual surfaces is described with a bi-directional reflectance distribution function (BRDF). In case of a flat plate, the overall reflectivity $f_r(\vec{l},\vec{v})$ is derived from the surface point BRDF, the cross-section area A and the observation geometry. The vector $\vec{v}$ describes the viewing direction from the object to the observer. The Sun direction is described with $\vec{l}$. The geometry is illustrated in Figure 4-3. The function can be decomposed in a diffuse and specular term, that is $f_r = f_d + f_s$. An analytical expression for f_d for diffuse Lambertian scattering from a flat panel with the normal vector $\vec{n}$ is

$$f_d(\vec{l},\vec{v}) = \rho_d \cdot A \cdot (\vec{n} \cdot \vec{l}) \cdot (\vec{n} \cdot \vec{v})$$

where ρ_d is diffuse reflectance coefficient. The specular term is computed according to Phong reflection model

$$f_s(\vec{l},\vec{v}) = \rho_s \cdot A \cdot (\vec{r} \cdot \vec{v})^{\alpha} (\vec{n} \cdot \vec{v})$$

with the reflection direction

$$\vec{r} = 2(\vec{n} \cdot \vec{l}) \cdot \vec{n} - \vec{l}$$

and the specular reflection coefficient ρ_s and the shininess constant α.

The equations can be reduced to three dimensions by defining the angles as in Figure 4-3 to visualise the impact of the geometry on the brightness, where the diffuse term is $f_d(\theta,\psi,\chi)$ and $f_s(\theta,\psi,\chi)$ the specular one. The transformation is described with the following vectors:

$$\vec{l} = \begin{pmatrix} 0 \\ 1 \\ 0 \end{pmatrix}, \qquad \vec{v} = \begin{pmatrix} -\sin\theta \\ \cos\theta \\ 0 \end{pmatrix}, \text{ and } \vec{n} = \begin{pmatrix} \cos\psi\cos\chi \\ \sin\psi\cos\chi \\ \sin\chi \end{pmatrix}.$$

The variable θ is the so-called phase angle between the viewing and the Sun direction, ψ describes the orientation of the plate with respect to the Sun in the Sun-object-observer plane, and χ is the title angle of the plate.

Diffusive and specular reflection for surfaces is described using physical or empirical models, e.g. ideal Lambertian diffuse as described in the previous examples, or empirical Phong reflection models, ABg scattering, or binomial scattering [RD055]. These models typically require more parameters to describe the reflective properties. Empirical functions can be derived in the laboratory for individual materials, subcomponents, or the whole spacecraft. They often represent spacecraft materials such as solar panels and MLI better than simplified models.

4.6.1 Requirement 5.6.a: Visual brightness assessment

Rationale for the Requirement

The requirement aims at limiting the visual brightness, measured as apparent magnitude, of the spacecraft to reduce the impact on ground and space-based optical astronomy.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:
   1. Identify system units that can cause specular reflections, strong scattering, or can increase the brightness;
   2. Identify materials with strong scattering or specular reflectivity properties.

b. Analysis, to:

1. Predict, during the mission design phase, the apparent magnitude at the planned altitude by:

(a) assessing, in early mission design phase (phase 0 to B), the brightness assuming a combination of diffuse and specular reflection for primitive shapes (e.g. sphere and flat plate as described above). To assess the maximum brightness (lowest apparent magnitude) the function f_r is maximised, e.g. at phase angle $\theta = 0°$ for the Lambertian sphere, and the distance minimised, i.e. using the orbital height $d = h$. Analytical expressions for a combination of diffuse flat plates in an Earth-pointing or Sun-pointing scenario are provided in [RD054].

(b) alternatively, rescaling measured magnitude values of similar spacecraft to planned altitude using the formula $m_B - m_A = 5 \log \frac{d_A}{d_B}$, where the apparent magnitude m_A of the spacecraft A at distance d_A is rescaled to distance d_B.

(c) using, in later phases (from phase B to C), the full 3D geometrical models, which describes the overall system using subcomponents with different material properties and considering self-shadowing. The geometrical models can be used to assess the apparent brightness for different observation geometries. To quantify the brightness independently on the location on the orbit and the observer position, a look-up table / map for $f_r(\vec{l}, \vec{v})$ is created. This allows identifying conditions with larger brightness values and conditions that cause glints and strong reflections. The apparent magnitude is then computed as described above. It is important to select a step size (e.g. in degrees) for the look-up table / map computation small enough to allow for the identification of bright flares and to sample illumination and observation angles evenly.

(d) determining the average apparent magnitude and its variation (e.g. standard deviation, percentile, or histogram) from the map considering different operational phases/modes:

(1) Operational Phase: Earth-fix / inertial pointing mode, Slow rotation mode, Fast rotation

(2) Post operations phase: Random tumbling mode, dynamically converged stable.

The distance and observation geometry are computed for all possible observer latitudes to derive the statistics.

(e) quantifying flares and glints occurrence (e.g. a function defining upper limit of amplitude vs. frequency) depending on phase/mode in b.1.d).

2. Regularly update the brightness estimate by considering:

(a) measured reflectance and scattering or full BDRF of materials, subcomponents, small scale system mock-ups, or during critical design phase the whole system if facilities are available (e.g. for nano-, cube- and small satellites). Specular and scattering properties of materials can be measured using spectrophotometry and provides simplified scattering data at discrete angles and wavelengths.

c. Test, to validate that the BDRF or simplified material models is done through ground-based measurements of in-orbit apparent magnitude as function of the distance and spacecraft orientation with respect to the Sun and observer.

Mitigation Measures

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

a. To consider in early phase of the space system development, the requirement on brightness characterisation.

b. To identify, in early phases the system parameters and subcomponents that drive the overall brightness of the system.

c. To develop a brightness model along the development of the spacecraft, with increasing representativity from conceptual to critical design.

d. To regularly observe the system from ground or collect information from surveillance system or exchange with the astronomy community (upon agreement), to update the brightness and reflectivity maps accounting for material degradation (material property changes) during the life of the spacecraft and after disposal.

4.6.2 Requirement 5.6.b: Visual brightness reduction for constellations

Rationale for the Requirement

See Rationale for the Requirement 5.6.a: Visual brightness assessment .in 4.6.1.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Methods to Assess Compliance for Requirement 5.6.a: Visual brightness assessment in .4.6.1

b. Analysis, to assess (following the analysis and tests in Requirement 5.6.a: Visual brightness assessment in 4.5.2) the aggregated effect of the spacecraft constellation, i.e. number of spacecraft simultaneously in the sky, or frequency of passes for different latitudes.

Mitigation Measures

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

a. To design spacecraft with reduced brightness.

b. To limit flares and strong reflections by operational mode.

c. To validate, during the design phase, possible technological solutions that can reduce the brightness of the system, including changes to the operational altitude, attitude mode, subcomponents, coating materials.

4.6.3 Requirement 5.6.c: Radio Astronomy protection

Rationale for the Requirement

The requirement aims at controlling spacecraft carrying active RF instruments from interfering with observations, or even damaging radio telescope receivers or instruments, due to deliberate and undeliberate transmission. Radio quiet zones are defined by national governments and the ITU to avoid any radio interference caused by direct transmission, antenna sidelobes, and leakage in protected bands.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:
   1. Avoid any unwanted emissions of a spacecraft falling into the frequency bands of Radio Astronomy and ensure them within levels provided in Section 5.5.1.2 of ECSS-E-ST-50-05-REV.2 [RD056].

b. Analysis, to:
   1. Assess interfaces and procedures to inform radio telescopes whenever the spacecraft passes the area [ITU-R RA.2259-1][RD057];
   2. Assess the aggregated effect of spacecraft constellation.

Mitigation Measures

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

a. To investigate, as earlier during the design phase, if special operational modes in the CONOPS can have impact on radio quiet zones.

b. To identify and limit possible undeliberate RF transmission or leakage during the spacecraft design phase.

4.6.4 Requirement 5.6.d: Information distribution for astronomy protection

Rationale for the Requirement

The requirement aims at facilitating the sharing of information, including the current and planned orbit, attitude profile and any other relevant spacecraft data, with the scientific communities to allow mitigate deteriorating effects on astronomy. Based on this information, mitigation can be indeed done afterward through postprocessing methods, e.g. removing artefacts introduced by the spacecraft, or before the passes, e.g. during planning to avoid the appearance of the spacecraft in the field-of-view.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to:
   1. Ensure that the ground segment of the spacecraft can provide accurate orbital and attitude data, with sufficient precision to assess the effect on the instrument. It is important that standard formats are used for the distribution where possible, e.g. CCSDS APM for attitude profiles and CCSDS ODM for orbital data;
   2. Ensure that the CONOPS contains the option to switch-off high-power active RF instruments (or point away, thereby considering also transmission through side-lobes) when passing over recognised radio quiet zones;
   3. Ensure that the ground can identify and report whenever the spacecraft passes over a radio telescope;
   4. Identify and collect surface material optical properties in support of brightness analyses;
   5. Ensure that a brightness map is available from simulations or measurements as described in Requirement 5.6.a: Visual brightness assessment in 4.6.1.

b. Analysis, or test, to:

1. Demonstrate capability of performing attitude manoeuvres over the entire mission to assure commanded pointing;
2. Determine the antenna pattern or an alternative quantity to predict the signal strength on ground including the impact of sidelobes.

Mitigation Measures

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

a. To consider, early in the design phase, the provision of the orbital and attitude data.

4.7 Lunar orbits

4.7.1 Requirement 5.7.1.a: Debris release avoidance in lunar orbit

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.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, or analysis, to ensure not to release any debris from the spacecraft during normal operations into Lunar orbit.

4.7.2 Requirement 5.7.1.b: Intentional break-up in lunar orbit

Rationale for the Requirement

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

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.

Mitigation Measures

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

a. Mitigation Measures for 5.3.2.1a (Requirement 5.3.2.1.a / ECSS-U-AS-10 7.2.1.1: Accidental break-up probability threshold in 4.3.2 and the subsequent requirements).

4.7.3 Requirement 5.7.1.c: On-orbit break-risk threshold in lunar orbit

Rationale for the Requirement

The requirement aims at minimizing the creation of debris in lunar orbit by applying the lessons learned from Earth orbits while reducing the risk of accidental break-ups, which are caused by on-board sources of energy or failure of mechanical parts.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to calculate the accidental break-up similarly as for Requirement 5.3.2.1.a / ECSS-U-AS-10 7.2.1.1: Accidental break-up probability threshold in 4.3.2, also considering the contribution from external factors as in Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1, taking into account the different dynamics of the lunar orbit and the related space debris scenario.

Mitigation Measures

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

a. Mitigation measures for Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1.

b. To perform periodical assessment and review of the health of the spacecraft, including trend analysis of critical parameters.

4.7.4 Requirement 5.7.1.d: On-orbit break-risk assessment in lunar orbit

Rationale for the Requirement

The requirement aims at determining the accidental break-up probability quantitatively by considering all known failure modes for the release of stored energy, including those from external sources such as impacts with space debris and meteoroids. The requirement specifically refers to the lunar environment, for which it is important to consider specific models and information.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Methods to assess compliance for Requirement 5.7.1.c: On-orbit break-risk threshold in lunar orbit in 4.7.3, additionally including the contribution of the external sources (Annex C), and, when orbit details are not available yet, considering the cases with worst-case orbit and worst-case conditions identified in the vulnerability analysis.

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.7.1.c: On-orbit break-risk threshold in lunar orbit in 4.7.3.

4.7.5 Requirement 5.7.2.a: Ephemerides determination for lunar orbit

Rationale for the Requirement

The requirement aims at ensuring that, for mission in lunar orbits, it is possible to produce ephemerides that can be used for monitoring and coordination with other operators.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Methods to Assess Compliance for Requirement 5.3.3.5.h: Ephemerides forecast in 4.3.43.

4.7.6 Requirement 5.7.2.b: Ephemerides information distribution for lunar orbit

Rationale for the Requirement

The requirement aims at ensuring that an operator in Lunar orbits provide relevant information and means for coordination with other operators, similarly to equivalent requirement for Earth's orbit.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Methods to Assess Compliance for 5.3.3.3.k (Requirements 5.3.3.3.k-m: Collision avoidance procedure information in).

4.7.7 Requirement 5.7.3.a: Probability of successful disposal in lunar orbit

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. The objective of the requirement is to minimise the risk for a space system to remain in hazardous state in the lunar orbit, or generate debris, after the end of mission.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Methods to assess compliance for Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1, while considering the different space debris and meteoroids scenario provided in lunar orbit and the disposal (heliocentric orbit, lunar impact, Earth re-entry, or Lunar graveyard orbit, as per Requirement 5.7.3.e:).

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.4.1.1.a: Probability of successful disposal for single spacecraft in 4.4.1.

4.7.8 Requirement 5.7.3.b: Disposal from lunar orbits

Rationale for the Requirement

The requirement aims at limiting the pollution of the lunar orbits and associated long-term risk given the increasing presence of spacecraft or launch vehicle orbital elements orbiting the Moon and the absence of a Moon atmosphere that can naturally cause the decay of these objects. Disposal criteria help to reduce this risk. The disposal into heliocentric orbit is the preferred one whenever achievable, while regarding disposal resulting in lunar impact, Earth re-entry, or a Lunar (graveyard) orbit there is no order of preference since this is still an area of open research [RD058].

Methods to Assess Compliance

The verification methods used to assess the compliance are similar to those for the Requirement 5.4.2.1.a: General Earth orbit clearance in 4.4.19 with the exception of those referring to Earth orbits.

Available software tools to assess compliance are:

- GODOT, with the recommended force model for lunar orbit propagation as described in Annex A.2.18.
- CUDAjectory, for the computation of the probability of lunar impact and Earth re-entry through Monte Carlo analysis accounting for the relevant dispersion and uncertainties (Annex H) [RD059].

More information on the tools is available in Annex A.2.18.

Mitigation Measures

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

a. To design the mission such that heliocentric disposal is feasible through a careful operation orbit selection and spacecraft design capable of heliocentric disposal.

b. For Lunar impacts, to avoid areas of historic and scientific value [RD060][RD061]. This can be assessed with the approach described in Requirement 5.7.3.c: in 4.7.9.

c. For Earth re-entry, to apply "design for demise" approaches as described in the Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2.

4.7.9 Requirement 5.7.3.c: Trajectory propagation in lunar orbit

Rationale for the Requirement

The requirement aims at assessing the effect of the disposal orbit evolution and its final status, given that merely disposing the spacecraft or launch vehicle orbital element in lunar orbit is not sufficient to ensure the successful long-term orbit clearance due to the high relative magnitude of perturbing forces. In the highly perturbed region of the Moon, long-term propagation and impact probability assessments are performed as:

a. In case of trajectories leading to Earth orbits, there is a non-negligible risk of impacting on the Earth surface, therefore, the occurrence rate and locations are determined as fundamental inputs towards determining the casualty risk.

b. In case of heliocentric or lunar trajectories, there is a non-negligible risk of impacting on the Moon surface, therefore, the occurrence rate and locations are determined as fundamental inputs towards assessing potential consequences for space heritage sites and operational assets on the Lunar surface.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, with an orbit propagation software for long-term propagation of Monte Carlo samples for probability studies [RD059]. Annex H elaborates on the parameters to vary in such Monte Carlo studies.

Mitigation Measures

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

a. Mitigation Measures for Requirement 5.7.3.b: in 4.7.8.

4.7.10 Requirement 5.7.3.d: Probability of Earth re-entry from lunar orbits

Rationale for the Requirement

The requirement aims at permitting exemption from performing the impact area evaluation for spacecraft and launch vehicle operating in lunar orbit, but with extremely low probability of Earth re-entry such that ESSB-ST-U-004 [RD03] is not violated. In case the free drift trajectories of spacecraft or launch vehicle disposed from Lunar orbit result in possible return and impact on Earth, as described in Requirement 5.7.3.c: in 4.7.9, the probability of Earth re-entry is used to assess the re-entry casualty risk (also referred as "expected number of casualties"). However, if the order of magnitude of the probability of Earth re-entry is equal to or less than the critical threshold for the re-entry casualty risk, as defined in Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2, then also the latter is expected to be compliant with the critical threshold, and therefore, the impact area evaluation is not necessary.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Review-of-design, to ensure that the order of magnitude for the probability of Earth re-entry is equal to or less than the critical threshold for the expected number of casualties, defined in Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2.

Mitigation Measures

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

a. To design the disposal from Lunar orbits such that the probability of Earth re-entry is minimised.

4.7.11 Requirement 5.7.3.e: Lunar graveyard orbit

Rationale for the Requirement

The requirement aims at identifying monitorable and stable disposal orbit for spacecraft or launch vehicle orbital stages in lunar orbit. Lunar orbits are known to be highly unstable due to the Moon irregular gravitational field and third-body effects. Moreover, the absence of an atmosphere means that orbits do not decay naturally, meaning that gravitational perturbations are prevalent on the spacecraft and therefore able to build-up higher variations of orbital elements. Due to the chaotic nature of these disposal trajectories caused by the combined effect of perturbing forces described in Annex **A.2.8**, the long-term stability of these disposal orbits is assessed [RD058] [RD059].

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Analysis, to:

1. Identify tracking strategies and constraints for a space surveillance segment, similarly to Requirement 5.3.3.5.a: Trackability in 4.3.36 for Earth orbits;
2. Assess whether the selected disposal orbit remains within bounded variations of its orbital elements for at least 100 years, using higher order lunar gravitational field models, as described in Annex A.2.8;
3. Quantify the allowed bounds on the variation of orbital elements, which depend on the type of orbit, e.g.:
   (a) a spacecraft in a distant retrograde lunar orbit can vary its distance to the Moon between 40,000 to 60,000 km and still remain stable.
   (b) a 100 km x 100 km low lunar orbit (LLO) that oscillates up to 30 km x 170 km is considered bounded and can be suitable as a graveyard orbit when it does not pose a risk to potential other spacecrafts in that orbit.

Mitigation Measures

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

a. To dispose the spacecraft or launch vehicle orbital element into a Lunar graveyard orbit with shown relatively stable orbital elements.

b. To operate the spacecraft preferably in stable orbits, so that variations of the orbital elements in case of failures are minimised.

c. To avoid preferably the disposal into a lunar graveyard orbit by opting for a heliocentric disposal.

5 Verification and validation requirements

5.1 Models

5.1.1 Requirements 6.2.a and 6.2.b: Space debris and meteoroid models

Rationale for the Requirements

The requirements aim at harmonising the space debris and meteoroids impact assessment for different missions and having a solid empirical base for the results by using the so called "calibrations epochs". Space debris environment models are continuously updated. However, they contain a significant amount of data sources that are non-continuous. By applying models to epochs outside the validated data period, e.g. forecasts, several assumptions are introduced about the likely evolution of the debris environment. Calibration epochs are the latest available epoch of empirical reference data used to derive the status of the space debris environment.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Inspection, to ensure the calibration epoch of the selected space debris environment model is retrieved and reported.

b. Uncertainties at the calibration epoch are generally not used during the analysis, as using the calibration epoch implies the best fit possible.

MASTER [RD062]is the space debris and meteoroids environment model endorsed by ESA. The current calibration epoch is November 2016, and an update of the calibrated population is expected for 2025.

5.1.2 Requirement 6.2.c: Space object population for collision avoidance planning

Rationale for the Requirement

The requirement aims at specifying that as collision avoidance procedures can only address objects that can be tracked reliably and are part of space object catalogue, only such objects are included in the analyses related to collision risk management. When using a space debris environment model, a threshold is defined to consider the trackable objects, which are included in the analysis performed already during the design phase. This threshold is chosen based on the sensor performance of the selected SST provider.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Inspection, to ensure consistency between the selected threshold and the performance of the SST provider (as in requirements Requirement 5.3.3.5.a: Trackability in 4.3.36 and Requirement 5.3.3.5.b: Space surveillance segment in 4.3.37).

For reference, the trackability diameter estimated for the US SSN is reported in Figure 5-1.

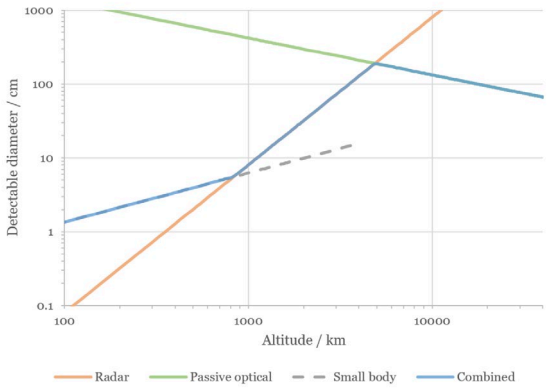

Figure 5-1: ARES trackable diameter as a function of altitude [SST-3]

5.1.3 Requirement 6.2.d: Space object population for SDMP analyses

Rationale for the Requirement

The requirement aims at checking the features of the space object populations used for the assessments. The population models can differ depending on the requirements to be verified (e.g. 1cm population at the calibrated epoch for assessments related to the cumulative collision probability as defined in 6.2.b, population of trackable objects at the epoch of operations for collision avoidance assessments). As the space debris environment can significantly evolve during the mission development, it is understood that the environment conditions identified at the moment of approval of the Space Debris Mitigation Plan are maintained for the assessments in the later mission phases.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Inspection, to ensure consistency:
   1. Between the selected population models and the applicable requirements;
   2. With the agreed baseline defined in the approved Space Debris Mitigation Plan.

5.1.4 Requirement 6.2.e: Cumulative collision probability assessment

Rationale for requirement

The requirement aims at considering in the definition of disposal strategies and in the assessment of the risk profile the collision risk generated with objects larger than 1 cm, which are able to cause the partial or complete fragmentation of a space system upon impact. Even at low velocity encounters in the GEO Protected region, a 1 cm sized object can penetrate the average spacecraft wall upon impact.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Inspection, to ensure that the modelling approach addresses:

1. Nominal conditions, both for the initial orbital state and for the attitude. For the computation during the operational phase (e.g. as for Requirement 5.3.3.2.c: Recurrent manoeuvre capability in 4.3.13), the handling of multiple attitudes can be needed: in that case the analysis can be performed considering a worst case scenario or an "averaged" attitude (i.e. cross-sectional area) considering the time spent in the different configurations; after the disposal phase, it is expected that the attitude is either randomly tumbling or following a well-characterised long-term attitude (e.g. gravity gradient) and that configuration is used for the assessment.
2. Large appendages, , that are known to create a significant amount of debris (e.g. solar panels, whose area is generally included in the total cross-sectional area for the cumulative collision probability assessment by considering the average cross-section of the long-term attitude motion of the spacecraft);
3. Appendages, that can be excluded from the overall cross-sectional area when not contributing to the collision risk (e.g. antenna wire-grids or foils that have been demonstrated to tear instead of fragment upon impact; more details on this aspect can be found in Annex Section C.2.5);
4. Trajectory evolution, when assessing the cumulative collision risk during the disposal phase, which is achieved through the orbit propagation and the consideration of the change in debris flux witnessed along the changing trajectory (the procedure is explained in detail in Annex Section C.3).

5.1.5 Requirements 6.2.f-g: Probabilistic assessment of the orbital lifetime

Rationale for requirement

The requirement aims at accounting for an entire solar cycle in a probabilistic manner such that the solar cycle variability can be assessed already during the design phase of a mission without making overly optimistic or pessimistic assumptions on the status of the atmosphere at the disposal time. The estimation of the orbital lifetime of a space object is indeed inherently uncertain due to the limitations of forecasting the space weather conditions over long periods of time, uncertainties in knowing the phase of the solar cycle at the disposal time, unknown interaction coefficient between the residual atmosphere and the spacecraft surfaces. Even in case of relatively well known and forecastable motions after disposal, large discrepancies between actual and forecasted lifetimes have been observed. In general, the distribution due to the relevant uncertainties is unimodal, with spread in the results when the interaction between the atmosphere and the space object is limited but significant fraction of an orbital revolution (e.g. GTO). The significant influence of the atmosphere on the spread in orbital predictions is captured for eccentric orbits with 90 % indicating conservative estimate leading to re-entry. For all circular orbits, 50 %, i.e. the median, is neither overly conservative nor optimistic.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Inspection, to:

1. Ensure that the lifetime assessment is computed by uniformly sampling the solar cycle for the cases where the solar activity plays a significant role in the determination of the orbit

evolution, i.e. the space weather conditions such as sunspot number, radio flux, or geometric index proxies over the duration of an 11-year cycle, that form the input parameters to atmospheric models used in propagation analyses as indicated in Annex A (yearly samples are recommended, together with the use of multiple solar activity models as described in A.2.8);

2. Verify that all relevant sources of uncertainty affecting the orbital lifetime are taken into account for all cases, e.g. sampling from different input distributions (e.g. drag coefficient or timing, positional, and velocity uncertainties on the disposal orbit) in a Monte Carlo process (a convergence criterion as described in Annex H is commonly used for the relevant percentiles).

5.2 Inputs

5.2.1 Requirement 6.3.a: List of constellations and inhabitable space objects

Rationale for the Requirement

The requirement aims at ensuring that the most updated information about the position and motion of known constellations and inhabitable space objects is taken into account. Constellations emerged as a major driver of current and predicted future space traffic. Maintaining a comprehensive list of known constellations is, therefore, important for near-term risk assessment and longer-term modelling efforts. Similarly, tailored strategies are also required to mitigate risks associated with human spaceflight activities, and thus a dedicated list of inhabitable space objects can also be maintained.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Inspection, to ensure that the applicable space environment data includes the latest updated information on known constellations and inhabitable space objects.

Information on known constellation is based on:

a. Constellations, whose members are released into orbit over more than two events, with more than one year from the first to the last event, with the orbits in which they are deployed directly related to the constellation objective, which include constellations started with systematic deployment, excluding precursor and technology demonstration missions.

b. Constellations, which are operational at the time of verification. A constellation is considered operational, i.e. functional, as long as at least one of its members is functional.

c. Constellations, which operate in LEO and MEO, as a minimum.

d. A constellation is considered to operate at fixed altitude if atmospheric decay is negligible, or it is compensated by propulsive station keeping.

A list of constellations satisfying these conditions, suitable for verification and versioned for reproducibility, is available from https://discosweb.esoc.esa.int/**.** A list of inhabitable space objects is also maintained at the same address.

5.2.2 Requirement 6.3.b: List of active spacecraft

Rationale for the Requirement

The requirement aims at ensuring that the most updated information about the distribution of active spacecraft is taken into account when assessing collision risk management approaches. Active spacecraft include all functioning spacecraft and not only manoeuvrable ones.

Methods to Assess Compliance

The verification methods used to assess the compliance are:

a. Inspection, to ensure that the applicable space population models include the latest updated information on active spacecraft.

A list of active spacecrafts can be retrieved from https://discosweb.esoc.esa.int/ through the Objects table and the Active column. Statistics on the distribution of such objects are regularly published through ESA's Space Environment Report [RD016].

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 manoeuvres.

b. Launch vehicle orbital stages after the operation phase and disposal manoeuvres (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. Force model for objects in lunar orbits (Section A.2.8).

h. Solar activity and geomagnetic index (Section A.2.9).

i. Solar radiation pressure (Section A.2.10).

j. Object cross-sectional area (Section A.2.11).

k. Object drag coefficient (Section A.2.12).

l. Object mass (Section A.2.13).

m. Object ballistic coefficient (Section A.2.14).

n. Solar radiation pressure reflectivity coefficient (Section A.2.15).

o. Propagation time and frequency (Section A.2.16).

p. Result accuracy margin (Section A.2.17).

q. Tool(s) use and acceptance (Section A.2.18).

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, Lagrange 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 likely trajectory propagation.

The output of the analysis includes:

a. Orbital parameters of the trajectory propagation.

b. Time spent in the LEO or GEO Protected Region after the operation phase.

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 manoeuvre 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 stochastic 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:

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

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 in [RD010]:

a. NRL-MSISE-00.

b. Jacchia-Bowman 2006 (JB-2006) / Jacchia-Bowman 2008 (JB-2008).

More recent versions of atmospheric models exist as well. The use of atmosphere models that were designed to fit a selected 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 the accuracy of the model density is the accuracy of the future predictions of the solar and geomagnetic activity data used as inputs, rather than the accuracy of the specific model in representing the density as a function of solar and geomagnetic activity. It is important for the orbital lifetime estimations the use of an atmospheric model with space weather indices forecasted for long periods of time.

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 [RD010]. As a minimum, the following approximations are recommended:

a. LEO, MEO, HEO, GTO, GEO vicinity:
   1. Zonal harmonics including J_2, J_3, J_4, J_{22};
   2. Zonal harmonics up to J_{15} 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 [RD010]. As a minimum, the following approximations are used:

a. LEO: lunar and solar central gravitational attraction.

b. MEO, HEO, GTO: lunar and solar central gravitational attraction.

c. GEO: lunar and solar central gravitational attraction.

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 Force model for objects in lunar orbits

For objects in lunar orbits, it is recommended to use a force model that includes, as a minimum, the following forces:

a. Lunar central gravitational attraction.

b. Earth central gravitational attraction.

c. Solar central gravitational attraction.

d. Moon expansion of perturbation potential (degree x order) depends on altitude:
   1. Periselene lower than 100 km: 100x100;
   2. Periselene between 100 and 2000 km: 50x50;
   3. Periselene above 2000 km: 10x10.

e. Solar radiation pressure.

The recommended degree and order of expansion of the gravitational potential are derived from lessons learnt from ESA mission analysis given that perturbations in the lunar gravitational field without atmosphere have a non-negligible effect on low lunar orbits, e.g. over 100 years, which is conventionally used as a time scale for analysis in Earth orbit. Any deviation from the above recommended values is object of a specific assessment through the related justification, analysis and documentation, which are provided.

A.2.9 Solar activity and geomagnetic index

Solar activity has an effect mainly on the orbital lifetime in LEO. The most commonly used proxies to measure the solar activity are 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. Further proxies do exist. Atmosphere models compatible with the solar activity proxies are interesting for future propagation, in contrast to atmosphere models that are a posteriori calibrated on variable proxies.

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

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 [RD063].

b. 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.

c. ECSS sample solar cycle: the method is based on repeated cycles for the solar flux taking from the ECSS solar cycle per [RD010], 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.

For the solar cycle method, the assessment of the expected presence in orbit is computed with a DRAMA/OSCAR orbit propagation analysis, using, as a minimum, the "ECSS sample solar cycle" and the "Monte Carlo Sampling with at least 5 samples cycles" (as in the example in Figure A-1). Given that the solar activity and geomagnetic index are dependent on the epoch of analysis, the orbit propagation analysis for lifetime assessment is performed considering the variability over a full solar cycle, in line with the Requirements 6.2.f-g: Probabilistic assessment of the orbital lifetime in 5.1.5).

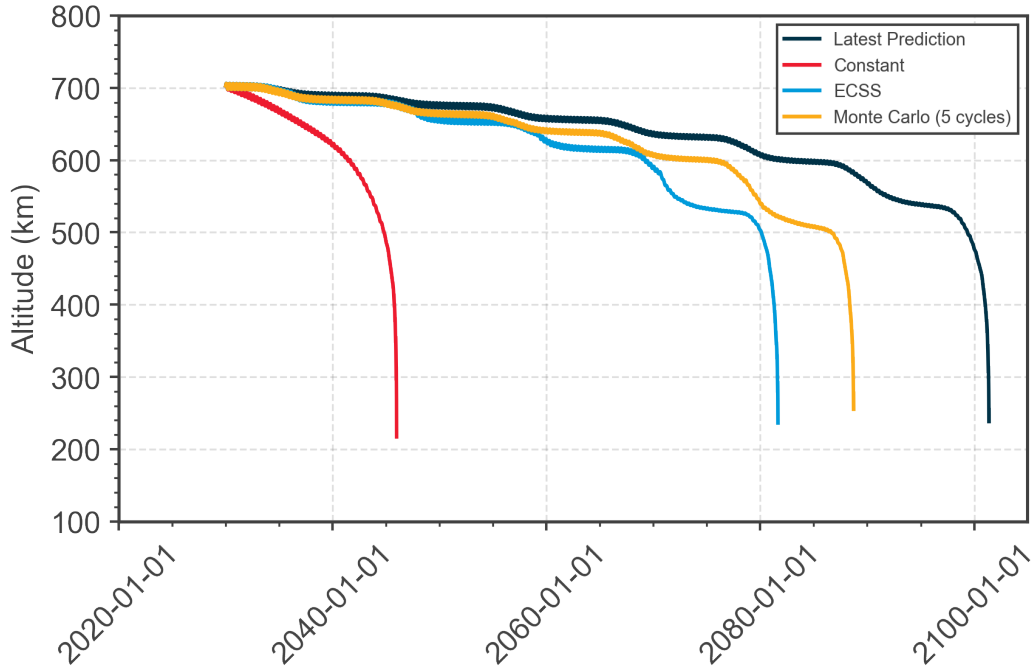

Figure A-1: An example of orbital lifetime assessment for different solar cycle scenarios, considering an initial circular orbit with semi-major axis = 7082 km, 98,2° inclination, epoch 01/01/2030 a cross-section 23,5 m² and a mass of about 2150 kg.

A.2.10 Solar radiation pressure

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

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

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 (Section A.2.11)

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.2.11 Object cross-sectional area

The cross-sectional area of an object is an input parameter to compute the atmospheric drag and the solar radiation pressure. 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 is acting, i.e.:

a. The cross-sectional area 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 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. 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, including the worst-cases, e.g.:

a. Object geometrical configuration at the beginning of the assessment, i.e. release time, end of the operation phase.

b. Object attitude, i.e. stabilisation, uncontrolled stabilisation, gravity gradient stabilisation or aerodynamic stabilisation effects, random tumbling, or any other damping effects.

When an object in orbit is non-functional, itis expected to be uncontrolled. Under certain conditions uncontrolled objects can be gravity gradient stabilised or aerodynamically stabilised.

An analysis is performed to determine the expected attitude state evolution after loss of control:

a. If specific justification and accurate quantification is available, the attitude can be assumed as:
    1. Stable (inertial or with respect to Earth orientation), or
    2. Rotation around one axis with known and constant motion vector.

b. If, as in most cases, the motion is unknown or chaotic, longer-term predictions on the rotation axis are uncertain, and damping effects are very unlikely quantifiable, the attitude can be assumed as:
    1. Randomly tumbling.

A cross-sectional area calculator for complex geometries is implemented in the ESA tool DRAMA (CROC) and NASA DAS.

A.2.12 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 an object 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) [RD091].

c. Summation of 6 or more simple-sided plates [RD092].

d. Direct Simulation Monte Carlo (DSMC).

Such assessments are particularly important for slender spacecraft [RD093] and spacecraft operating in very low Earth orbit.

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.2.13 Object mass

The mass (M) of an object is considered at the time or phase of the prediction. The predicted mass 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 is taken into account, e.g. 20 % at PRR/SRR, 10 % at PDR, and 5 % at CDR.

A.2.14 Object ballistic coefficient

The ballistic coefficient (M) is defined as:

$$B = \frac{M}{A_{drag} C_D} \qquad \text{[A-3]}$$

where:

M mass (Section A.2.13)

A_{drag} cross-sectional area for drag (Section A.2.11)

C_D drag coefficient (Section A.2.12)

A.2.15 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:

1. LEO, MEO, HEO, GTO: C_r = 1,2;
2. GEO: C_r =1,5.

A.2.16 Propagation time and output frequency

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

- Propagation time: at least 200 years, unless re-entry occurs before.
- Frequency of the output orbital states: 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.2.17 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 the 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.2.18 Analysis tool(s)

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

DRAMA/OSCAR is the tool endorsed by ESA to perform an orbit propagation analysis for bounded Earth orbits. Use of other different tools is also possible, pending a priori discussion and agreement of the selected tool with ESA.

For the propagation of lunar and libration point orbits, the following tools are recommended:

a. ESA GODOT (available from https://godot.io.esa.int/godotpy/ for ESA Member States users) for the long-term propagation of lunar orbits, e.g. to assess the variation of the orbital elements of the lunar graveyard orbit. This tool allows setting up a force model, or so called "dynamics" model, that includes the relevant higher order terms of the lunar gravitational field. A force model compatible with the recommendations in this Handbook is made available at https://debris-forum.sdo.esoc.esa.int/.

b. CUDAjectory (available from https://gitlab.space-codev.org/ for ESA Member States users) for the parallel computation of large numbers (>2000) of orbital states. This tool allows propagating orbits on a graphics processing unit (GPU) and is particularly useful for the long-term Earth re-entry risk analysis for spacecraft in libration point orbits as the analysis involves massive Monte Carlo sampling of initial orbital states and successive 100-year propagation of those samples while detecting Earth re-entry events.

A.2.19 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-2, which is based on a numerical propagator considering the NRLMSISE-00 atmosphere model, an 6th order and degree gravity model, lunisolar perturbations, solar radiation pressure, and solar activity predictions (best last update prediction) from SOLMAG [RD063]. Currently included in the ESA tool DRAMA. The start epoch is the 1$^{st\ of}$ January 2030. 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.

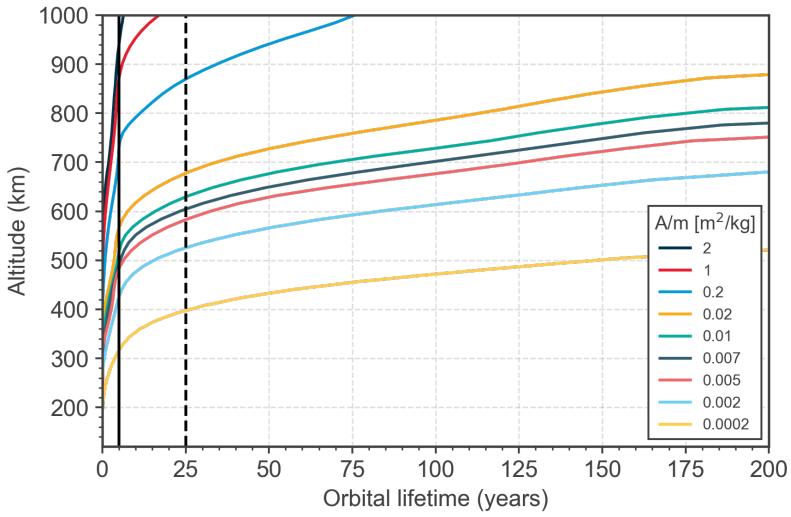

Figure A-2: Orbital lifetime and (initial) geodetic altitude for an object decaying from a circular orbit for different area-to-mass ratios

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 manoeuvres). 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 manoeuvres 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 objects at the Time of Closest Approach (TCA), is smaller than the sum of the maximum radius (including the longest appendages) of the two 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 objects. In order to reduce these uncertainties, accurate information about the trajectories of the possible 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 manoeuvres against tracked objects

Performance of collision avoidance manoeuvres 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 manoeuvres (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 manoeuvres per year.

i. Determination of the total expected number of collision avoidance manoeuvres over the operation phase duration.

j. Estimation of the amount of propellant for collision avoidance and returning manoeuvres 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 www.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 manoeuvres.

e. Strategy to access contact points to plan coordinated avoidance manoeuvres, 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 consider the overall dimensions of the space system, including solar panels and longest appendices (listed from the most to the least conservative estimators):

a. 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).

b. To consider the longest diagonal of the object to compute the radius of the equivalent object sphere.

c. 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:

a. 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.

b. 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 computing 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 Probability 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 detailed information that can be found in [RD020].

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:

a. 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.

b. 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 Analysis 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 Space Surveillance

B.3.1 Trackability

The trackability depends on the target orbit and physical properties, the sensor network geographic distribution and the detection performance and the sky coverage of the contributing sensors.

It is important that the spacecraft reflects enough light or radar energy to the optical or radar sensor to be detectable. It is also important that the reflected light (i.e. visible brightness) however does not violate any dark and quiet skies requirements. The reflective properties (i.e. brightness and radar cross section) are, in general, dependent on the orientation of the spacecraft with respect to the sensor and can be artificially amplified using special materials or components. The trackability can also be enhanced using active optical and RF transmitters.

The capability of the surveillance network can be described using a single trackability curve (as implemented in DRAMA ARES [RD020]), look-up tables, or a full simulation of the sensors using simplified or complex detection models. The trackability curve and look-up table are based on assessment of the catalogue maintained by the Space Surveillance segment. The orbit and size estimates in the catalogue allow estimating trackability thresholds for different altitudes or orbit regions. The trackable object size as derived for DRAMA ARES is shown in Figure 5-1.

Surveillance segments can lose track of spacecraft, e.g. due to manoeuvres or large observations gaps. It is important that operators support then the surveillance segment by sharing operational orbits and manoeuvre plans to allow recovering the trajectory. This information allows tasking new observations and/or identifying the spacecraft in uncorrelated surveillance data to update the trajectory estimate.

Another difficulty for surveillance systems are close-proximity operations of spacecraft e.g. in close formations. If the surveillance segment is not be able to uniquely identify the originating object of a new observation, then the trajectory estimate cannot be estimated. Shared orbits and special optical or RF signatures can help the surveillance segment to associate the new observations to the correct trajectory in the catalogue.

B.3.2 Estimated Position accuracy

An orbital state is considered a random variable modelled with a multivariate normal distribution around an estimated mean state $y(t_0) = \left(r^T(t_0), v^T(t_0)\right)^T$ and a covariance $C_y(t_0)$ at an orbit determination epoch t_0. The state vector contains position r and velocity v elements. The covariance is composed of a position and a velocity covariance and the correlation between the two components. Other uncertainty representations rely on Gaussian mixture or polynomial representations. However, the CCSDS ODM provides keywords for covariances only. The covariance describing the uncertainty of the predicted position $r(t)$ at an arbitrary epoch t is calculated by propagating it in time using a linear approximation:

$$C_r(t) = \frac{\partial r(t)}{\partial y(t_0)} C_y(t_0) \frac{\partial r(t)}{\partial y(t_0)}^T$$

This propagation is equivalent to the time update phase of a Kalman Filter (Section 8.3.2 of [RD064]) where the partial derivative of the full state with respect to the initial is called state transition matrix.

Additional parameters p are introduced that determine different forces acting on the satellite and their uncertainty using the covariance C_p. An example force parameter is the ballistic coefficient which determines the magnitude of the drag force. The partial derivative w.r.t. the parameter vector is called sensitivity matrix. The resulting covariance considering uncertain force model parameters is then:

$$C_r(t) = \frac{\partial r(t)}{\partial y(t_0)} C_y(t_0) \frac{\partial r(t)}{\partial y(t_0)}^T + \frac{\partial r(t)}{\partial p} C_p \frac{\partial r(t)}{\partial p}^T$$

Atmospheric density uncertainties are often modelled with a constant variance in the ballistic coefficient. Alternatively, covariances are also propagated using non-linear approaches, e.g. sigma-point methods [SST-9] or using higher order derivatives.

B.3.3 Position accuracy along the orbit determination arc

As show in the previous Section, the accuracy of the predicted position depends on the initial velocity accuracy. A single sensor can provide the expected position accuracies at single point in orbit but fails to provide the same accuracy on other points due to the poorly estimated velocity. A better velocity estimate is achieved if the spacecraft is observed frequently with observations covering the full orbit. The position accuracy is therefore estimated along the orbit determination interval covering at least one orbital revolution.

The accuracy capability of the surveillance network can be described using an interpolation or look-up tables [RD020], or a full simulation of the sensors using simplified or complex detection models and modelling the sensor observation accuracies. The interpolation or look-up table are based on assessment of the catalogue maintained by the Space Surveillance segment. Historic datasets of estimated covariances (e.g. from CDMs) allow estimating the achievable accuracy for different altitudes or orbit regions.

For a full simulation, the network of sensors is modelled. The observable passes as for Requirement 5.3.3.5.a: Trackability in 4.3.36 are predicted for a feasible orbit determination interval duration. The length of the orbit determination interval is dependent on the accuracy of the dynamical model and consequently differs between orbital altitudes and regions. For a spacecraft in LEO 7 days and in GEO 20 days are considered feasible.

The orbital state covariance is then estimated using a consider-covariance analysis [RD065] using

$$C_y(t_0) = (H^T R^{-1} H)^{-1} + S C_p S^T$$

where H describes the linear mapping of the modelled measurement errors R and the sensitivity matrix S the mapping of model errors C_p (parametrised with so-called consider parameters) to state errors. A complete derivation of the consider-covariance analysis is provided in Section 8.1.4 of [RD064]. The covariance is then mapped to the epochs along the orbit determination interval to identify the time epoch at which the resulting position uncertainty is maximum. Mapping can be performed by relying on the linearised approximation provided above.

The covariance assessment is performed for different scenarios for one year and to understand the impact of seasonal observation geometries and longer observation gaps. It can also test different uncertainty levels for the consider parameters to understand possible limitations and driving parameters, e.g. sensor time bias or atmosphere uncertainties [RD066].

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, sometimes also called micrometeoroids and orbital debris (MMOD). Such analyses are performed especially when impactors are untrackable 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 an untrackable object is about one order of magnitude higher than the one by the monitored space debris. Typically, such impacts are with objects 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 an on-orbit break-up and vulnerability analysis is considered to mitigate the generation of additional space debris and support the enhancement the spacecraft, or launch vehicle, making it more resilient to the space environment.

Note that an on-orbit vulnerability analysis is performed for different failure scenarios as especially:

a. Loss of manouvrability or collision avoidance capability.

b. Loss of disposal capability.

C.2 Methodology

C.2.1 General

The probability of an impact or collision with space objects is never zero. Collisions with space debris or meteoroids can cause:

a. Spacecraft or launch vehicle stage break-up leading to the release of space debris, which beyond a certain catastrophic collision threshold can result in complete fragmentation. Note that large break-up events are intended to be mitigated by collision avoidance manoeuvres.

b. Spacecraft or launch vehicle stage failures, which can be subcategorized as :
    1. Loss of manoeuvrability or collision avoidance capability (e.g. attitude and orbit control sensors and actuators, propellant leaks);
    2. Loss of disposal capability (e.g. propellant loss, drag-sail or tether damage).

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 (orbital) debris, including large trackable objects and small untrackable particles.

C.2.2 Collision cross-sectional area

The collision cross-sectional area of a space system or one of its surface elements is a measure of its probability to collide with an incoming space debris or meteoroid. In general, it depends on the object geometry and attitude and the size of the impactor. A simplified conservative representation for 2D analyses can be derived as indicated below.

The collision cross-sectional area (A_{Coll}) of a space system can be reduced to the envelope of the maximum projected area of the space system and the area of the impacting object (space debris or meteoroid). As a conservative estimate, considering the circular area with diameter equal to the maximum object extent has been used, for example by applying the expression:

$$A_{Coll} = \frac{\pi}{4}\left(D_t + D_p\right)^2 \qquad \text{[C-1]}$$

where:

D_t space system maximum extent

D_p debris or meteoroids diameter

C.2.3 Break-up

C.2.3.1 Catastrophic collisions threshold for complete break-ups

The currently applicable threshold to be considered for catastrophic collisions for complete break-ups is 1 cm of particle size. However, other thresholds definitions exist, as in particular the energy-to-mass ratio (EMR) [RD067].

As of today, these threshold definitions are Boolean (binary) and largely independent of the space system design, what makes them easy to apply. However, in reality the criticality of collision conditions is understood to depend at least on the spacecraft design and mission scenario (orbit/trajectory and attitude) and [RD068] (also see 5.3.3.2.e.2). It is anticipated that more detailed definition of a collision criticality can be introduced in the next years and in future evolutions of the standard.

C.2.4 Vulnerability

C.2.4.1 General

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

Vulnerability assessments are typically performed in various project phases, in some cases starting as early as Phase 0/A (if collisions are a design driver) and up to Phase C (CDR). The level of detail varies highly over the phases with regards to:

a. Geometrical topologies, i.e. assessments on cross-sectional, 2D or 3D spacecraft model.

b. Pointing / attitude, i.e. assessments on worst-case, solid angle averaged or detailed pointing scenario.

c. Logical levels, i.e. assessments on system (primary structure perforation), unit (batteries, tanks) or component (external harness) level.

C.2.4.2 Probability of damage or failure due to collisions

For a vulnerability assessment a list of targets needs to be defined which are required for a certain objective or function. Damage to such a target is then considered to potentially lead to a failure. For example, in the context of mission success, the explosion of a propellant tank can be considered a failure, while propellant leakage can still allow to meet objectives.

In the context of space debris mitigation, possible failure scenarios due to collisions are identified as:

a. Loss of manoevrability or collision avoidance capability.

b. Loss of disposal capability.

These two scenarios have in general a different set of critical components that need to be functional over different timescales. For loss of manoeuvrability or collision avoidance capability, for example, all or a part of the AOCS sensors and actuators can need to be available from launch to end of operational phase. For loss of disposal capability for example the disposal technology (as drag-sail, tether, thruster) and relevant passivation functions can be relevant.

The assessment includes the following steps:

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

b. Definition of the trajectory, free drift trajectories after orbit injection, end of mission, and disposal, and during normal operations, until re-entry or up to 100 years.

c. Definition of the pointing scenario for nominal orbit.

d. Definition of the space system design.

e. Identification of the critical components that, when damaged by impact, lead to a failure.

f. Identification of the ballistic limit equation and failure mode for each critical component (typically perforation of material layers), if necessary, broken down to specific critical component surfaces.

g. Determination of the at-risk surface areas for each critical component, considering that:
    1. In case the critical component is considered equally protected by other parts of the system, the at-risk area can be assumed as the total surface area of the critical component. Note that this can also be used when performing a conservative analysis with minimum protection in all directions;
    2. If the component is attached to the exterior of a system, the at-risk area can be the total area of the component, excluding the side attached to the outer wall.

h. Determination of the expected number of collisions causing failures and the corresponding risk for each critical component per relevant mission phase.

i. Determination of the share of the risk as function of impactor size, velocity and directionality. This can be used to collect a database of spacecraft vulnerability and to possibly mitigate risk by design.

j. Determination of the system level “Probability of No Failure” (PNF) including all critical components, considering redundancy where appropriate.

The ballistic limit for an impacted surface can be determined through Ballistic Limit Equations (BLE), which are equations based on experimental test and numerical simulation of hypervelocity impacts. The BLEs depend are typically analytical formulas depending on impactor properties (as density, shape), impact conditions (as incidence angle, relative velocity), failure mode (cratering, spallation,

detached spallation, perforation) and impacted surface properties, including shielding configuration (e.g. single wall, multiple walls), material properties (density, yield strength), type (e.g. homogeneous, composite, honeycomb, brittle, ductile), and geometry (thicknesses, spacing between walls). Information on the definition and use of BLEs can be found in DRAMA/MIDAS tool user manual [RD069][RD070].

C.2.5 Specific use-cases

C.2.5.1 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.2.4 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:

$$A_{Coll.T} = D_{Ti}L \qquad \text{[C-2]}$$

where:

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

L tether length

C.2.5.2 Booms

A boom is typically a deployable long structure hosting hardware, as for example payloads, instruments or thrusters. Hypervelocity impacts can affect electrical connections (harness), thermal insulation, integrity (break up) and mechanical properties.

Impact effects on electrical connections and thermal insulation can be assessed as in C.2.4. Break-ups of the boom can be assessed as in C.2.3, considering the impactor and the boom mass (excluding the spacecraft).

The effect of damages on the boom mechanical properties are use-case specific, in particular dependent on the boom design and mechanical loads. In general, a critical impact effect threshold is identified (e.g. crater size or perforation hole size) above which degradation of mechanical properties does not allow to meet functional requirements. More details on boom mechanical properties degradation can be found in [RD071].

C.2.5.3 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.2.3 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. In general, thin foils impacted by considerably larger particles than their thickness is considered to be punctured, effectively leading to a hole in the foil roughly of the size of the impactor.

Additional structural elements can also be present with a sail, like booms (see C.2.5.2).

C.2.5.4 Solar panels

In general space debris and meteoroid impacts on solar panels over long duration missions are unavoidable. Impacts on the front surface (cover glass), leading to cratering, are assumed to reduce the effective area of the solar panel. Therefore, the panel is typically oversized to compensate the impact related degradation over lifetime.

However, impacts on sensitive design elements as for example interconnectors, power harness or insulation, can lead to the loss of larger fractions of the panel. With a detailed design of the panel the vulnerability can be assessed as in C.2.4.

Furthermore, hypervelocity impacts also generate impact plasma which under specific conditions can lead to arcing. For more information, please refer to [RD072].

C.2.5.5 Active Debris Removal

In some active debris removal solutions, a fast connection (e.g. harpoon) is established between the spacecraft (chaser) and the target. In such dynamic processes space debris can be generated due to complete or partial break-up of the objects involved. The methods presented in C.2.3 are not developed for this use-case (e.g. shaped harpoon "rods") but can provide a reasonable starting point in assessing the order of magnitude of space debris generation, considering the masses of the target and the projectile in the collision.

C.2.6 Analysis tool(s)

The DRAMA/MIDAS and ESABASE2/Debris tools are used to determine the probability of damage and failures due to collisions with space debris or meteoroids (vulnerability) and are endorsed by ESA. However, other tools as e.g. Systema/Debris, PIRAT or BUMPER are available.

C.3 Cumulative collision probability

In order to assess the cumulative collision probability with objects larger than 1 cm (which is assumed to result in a catastrophic collision) the following steps are taken:

a. Propagation through the mission phase:

   1. If the spacecraft is performing station-keeping during its operational phase, the orbit can be assumed constant for the purpose of this analysis;
   2. If the spacecraft is not performing station-keeping, or the disposal phase is being analysed, a propagation of the orbit is performed. A tool such as ESA DRAMA/OSCAR can be used. In addition, the guidelines described in Annex A are followed.

b. Discretisation of the resulting trajectory:

   The trajectory obtained in step a can be discretised into a set of orbit steps, using different orbital parameters as discretisation criterion (e.g. epoch, semi-major axis). Based on this discretisation, different parts of the trajectory can be analysed for their contribution to the total cumulative collision probability;

   NOTE The discretisation criterion is chosen according to the orbital regime. e.g. for orbits within or crossing LEO, a change in perigee altitude of more than 10 km is recommended as discretisation criterion. For orbits outside LEO, a change in semi-major axis of more than 10 km is recommended.

c. For each of the resulting orbit steps, the collision probability of the spacecraft over this timeframe PoC(Δt) can be computed using tools such as ESA MASTER or ESA DRAMA/ARES, considering space objects larger than 1 cm. Note that objects larger than 10 cm are included in the analysis also for spacecraft with collision avoidance capabilities given that the assessment of the cumulative collision probability is meant to be a proxy of the fragmentation risk associated with analysed spacecraft once it is no longer operational.

d. In the computation of the collision probability, the orbit can be assumed constant over the previously determined discretisation period. The area used for this assessment generally includes all appendages, as described in 6.2.e (on Requirements 6.2.f-g: Probabilistic assessment of the orbital lifetime in 5.1.5).

e. The annual collision probability determined form DRAMA/ARES is scaled to the duration for the analysis (Δt) through the following formula:

$$P_{coll}(\Delta t) = 1 - (1 - ACP)^{\left(\frac{\Delta t}{yr}\right)} \quad \text{[C-4]}$$

   Where $\frac{\Delta t}{\mathrm{yr}}$ is the year fraction of the analysis interval and ACP is the annual collision probability resulting from the DRAMA/ARES analysis.

The resulting values of each orbit Section is aggregated to calculate the final cumulative collision probability using the multiplication rule as follows:

$$P_{cumulative}(t_{total}) = 1 - \prod_{i=1}^{n}\left(1 - P_i(\Delta t_i)\right) \quad \text{[C-5]}$$

According to verification and validation Requirement 6.2.c: Space object population for collision avoidance planning in 5.1.2, the analysis described in step 3 is always done using the space debris population at the latest reference epoch, meaning no predicted populations are used. For MASTER, the most recent reference epoch is named on the download page, available at https://sdup.esoc.esa.int/master/. In the context of the ARES tool, this means setting the epoch of the analysis to be equal to the reference epoch (not corresponding to the epoch of the trajectory).

Additionally, for the disposal analysis, the influence of the launch epoch is considered with respect to the forecasted solar activity. As requested by the Requirements 6.2.f-g: Probabilistic assessment of the orbital lifetime in 5.1.5, the variability along a whole solar cycle is analysed. Therefore, starting from the propagation, the analysis is repeated by sampling multiple epochs along a solar cycle (11 years), starting from the targeted disposal epoch. Based on this stochastic analysis, a disposal orbit is selected if the median value of the sampled trajectories is below the cumulative collision probability threshold. Alternatively, for verification purposes, the propagation initial epoch resulting in the lifetime closest to the median value can be used to generate the reference trajectory to be analysed in terms of cumulative collision probability.

Figure C-1 shows an example of computation of the cumulative collision probability as a function of the spacecraft mass and for different area-to-mass ratios. The computation was performed using the median disposal lifetime per solar-cycle from a sun-synchronous orbit and the calibrated MASTER population from 2016. For each case the decay duration corresponding to reaching the 10^{-3} cumulative collision probability value. The 5-year horizontal line helps identifying the mass value (m_T) corresponding to the change of driving parameter in the definition of the disposal strategy in LEO according to Requirement 5.4.2.3.a: LEO protected region clearance – objects operating in LEO in 4.4.21 For example, it can be observed that for the case with A/M=0,01 m²/kg, corresponding to the mean value for active satellites not belonging to constellations [RD016], the duration of 5 years is reached for mass values around 600 kg. This means that for spacecrafts larger than this size, it is expected that the criterion on the cumulative collision probability is the driving one in terms of LEO orbital clearance, whereas for smaller spacecraft the driving criterion is the lifetime limitation. The results in Figure C-1 can be used to state compliance to the Requirement 5.4.2.3.a: LEO protected region clearance – objects operating in LEO in 4.4.21 for spacecraft with mass values well below the m_T value, considering the cross-sectional area corresponding to the spacecraft's one, without the need for a dedicated assessment of the cumulative collision probability.

In addition, the cumulative collision probability criterion is also used to determine the need for design-for-removal features for spacecraft operating in LEO (Requirement 5.4.2.3.a: LEO protected region clearance – objects operating in LEO in 4.4.21). In particular, such features are requested for any spacecraft with natural orbital decay from the operational orbit longer than 5 years if the cumulative collision probability computed along the free drift trajectory from the operational orbit is above 10^{-3}. Figure C-2 gives an example of computation considering different CubeSat form factor, assuming that the spacecrafts have no appendages, so that the considered cross-sectional areas are respectively 1U: 0,015 m², 2U: 0,033 m², 3U: 0,053 m², 6U: 0,055 m², 12U: 0,079 m².

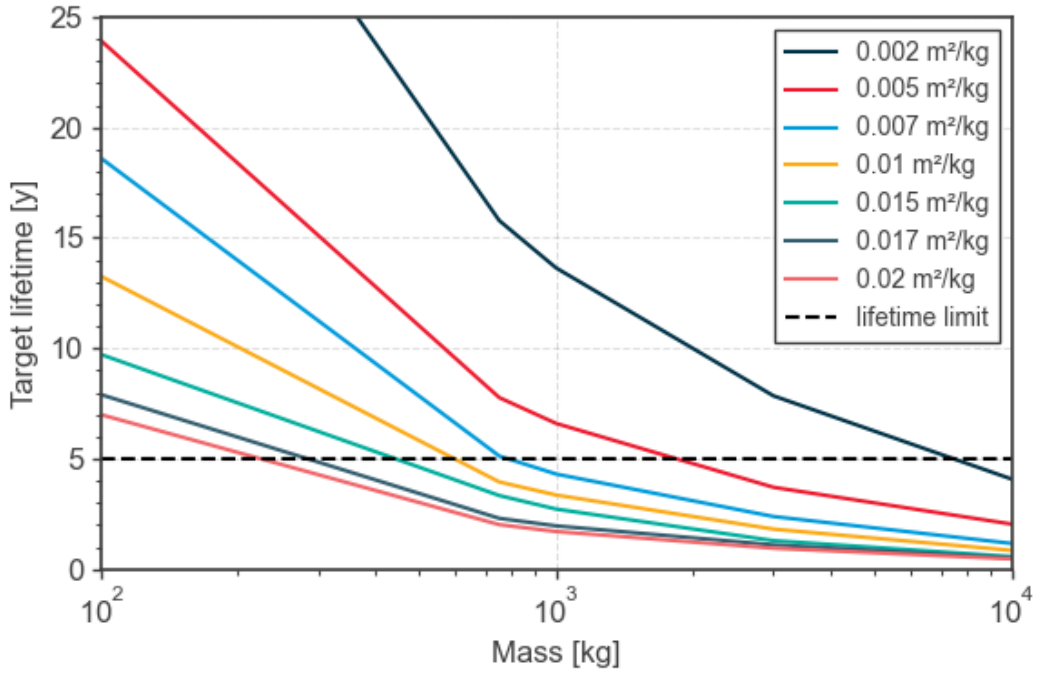

Figure C-1: Target lifetime to reach 0,001 cumulative collision probability threshold to as a function of spacecraft mass for different area-to-mass ratios. Using the median disposal lifetime per solar-cycle from a sun-synchronous orbit.

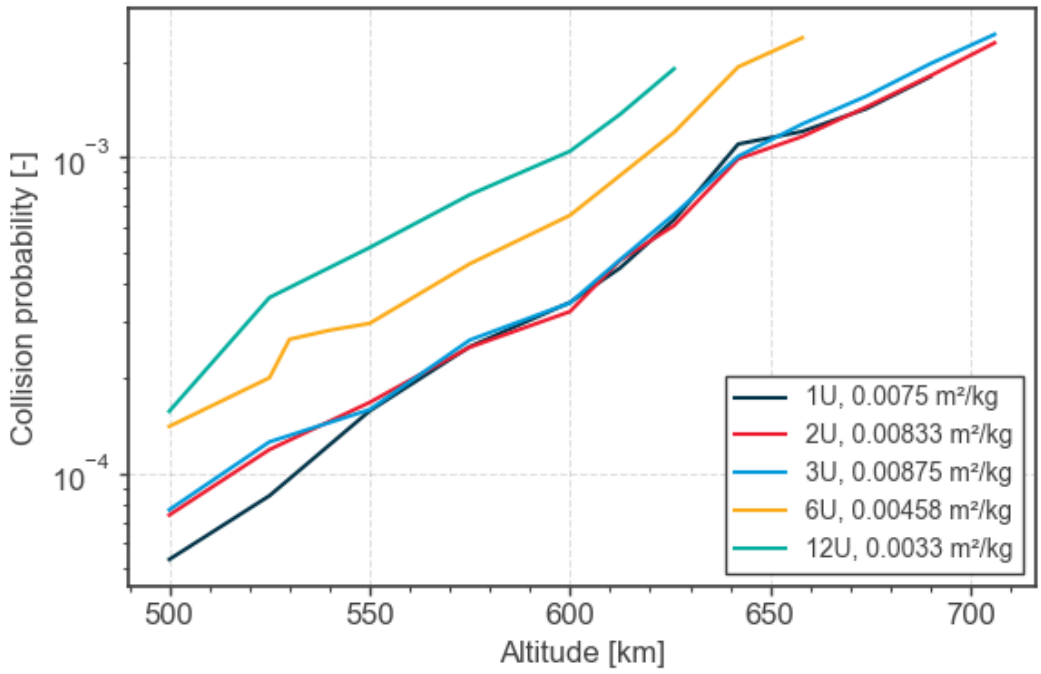

Figure C-2: Cumulative collision probability as a function of disposal altitude for different CubeSat configurations. Using the median disposal lifetime per solar-cycle from a sun-synchronous orbit and assuming 2kg/U.

Annex D
Re-entry casualty risk analysis

D.1 Objectives

Re-entry safety requirements and related verification methods are defined and explained in the ESSB-ST-U-004 [RD03]. This Annex integrates the guidelines provided in the Annex A of the ESSB-ST-U-004 [RD03] regarding the assessment of the expected number of human casualties (casualty risk) and design of controlled re-entry of a space system.

The guidelines for the verification of re-entry demisability and validation of "design for demise" solutions are available in [RD088] and in the D4D handbook (ESSB-HB-U-003) [RD086]. Section 4.2.4 of [RD023] provides parameters and minimum uncertainty ranges for consideration at system, equipment and material level.

D.2 Methodology

D.2.1 Re-entry probability

The calculation of the casualty risk takes into account the probability of a re-entry scenario ($P_{re\text{-}entry}$) to occur 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 G).

The probability of re-entry can be:

- $P_{re\text{-}entry} = 1$ for objects with permanent or periodic presence in LEO.
- $P_{re\text{-}entry} = 0$ for objects in non-Earth orbit (e.g. heliocentric orbits), HEOs, or orbits around Sun-Earth Lagrange Points, with periodic vicinity to Earth, when a precise long-term orbit propagation does not result in re-entry within the specified timeframe.
- $0 \leq P_{re\text{-}entry} \leq 1$ for objects where the initial conditions for long-term propagation are not precisely predictable, and a statistical approach (e.g. through Monte-Carlo techniques) can be used to assess accurately $P_{re\text{-}entry}$ within the specified timeframe. In this case, starting from an initial population of possible trajectories, which account for parameters dispersions (see also Annex H), $P_{re\text{-}entry}$ is the fraction of re-entry trajectories which reach the Earth atmosphere interface within at least 100 years of propagation.

D.2.2 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.2.3 Mission assessment: initial conditions and uncertainties

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 stabilisation, gravity gradient stabilisation, atmospheric drag stabilization.

To obtain accurate results, which are not biased by limited, or ill-posed, assumptions typical of a deterministic simulation, a stochastic simulation is performed, in line with the request from Req. Requirement 5.5.c: Re-entry casualty risk – probabilistic assessment in 4.5.3. The approach to define the initial conditions for a stochastic simulation, which considers trajectory uncertainties according to the type of re-entry and orbit, is summarised in Table D-1.

Table D-1: Initial conditions and uncertainties depending on re-entry type.

Re-entry type	Initial Conditions	Uncertainties
Uncontrolled re-entry from decaying circular orbit	Altitude: 130 km x 130 km Semi-major axis: 6501 km Eccentricity: ≤ $1x10^{-6}$ The eccentricity can be slightly non-zero for computational reasons.	Uniform variation in argument of perigee across an orbit.
Controlled re-entry from Low Earth Orbit	Sufficient number of possible state vectors (from mission analysis)	Initial conditions with an uncertainty profile at a suitable re-entry interface (see the example in D.4.3)
Re-entry from Highly Eccentric or Interplanetary Orbit	Sufficient number of possible state vectors (from mission analysis) to identify potential impact zones from the chords of the ground-track	As above.

D.2.4 Earth population density models

Earth population density data is used to assess the casualty risk. Depending on the type of re-entry, i.e. uncontrolled from a circular orbit, uncontrolled from a highly eccentric orbit, or controlled, an average or local value is calculated.

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

Earth population density data are based on:

a. Best estimation for the re-entry date.

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 $\Delta\phi$ around the latitude ϕ (e.g. Figure D-1).

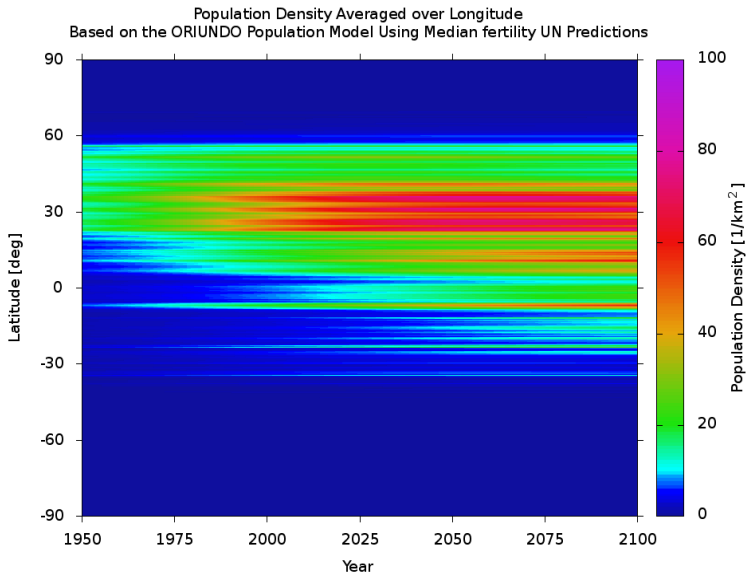

Figure D-1: Earth population density, latitude-dependent, using median UN predictions for the future growth rate (Rev. 2017)

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. Refer to par. 8.2 DRAMA-3.0.0 – SARA-TN – Final report [RD073] for further details on the population model.

D.2.5 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.2.5.1 Ground impact probability (1D)

When the average population density, i.e. mono-dimensional (1D) population density, is used to determine the expected number of casualties 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 (as detailed in the Sections D.2.5.2 and D.2.5.3).

D.2.5.2 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 (Figure D-2):

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

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

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

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

In any case, an object coming from an orbit with inclination i re-enter in the latitude range $[-i, i]$, and most likely close to the extreme of this interval.

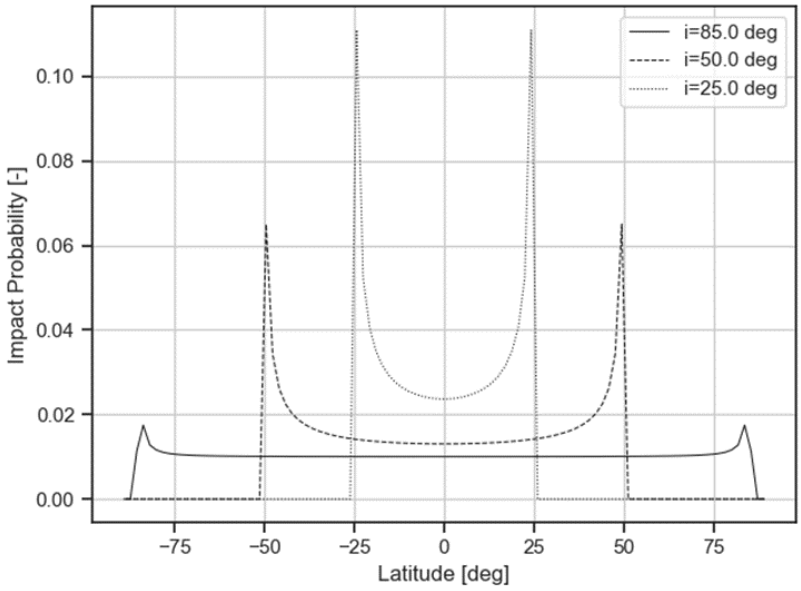

Figure D-2: Ground impact probability as function of latitude (approximation without effects of J2 Earth gravity parameter)

D.2.5.3 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 apogee altitude (e.g. some Molniya orbits, HEOs as for Integral, Cluster-II).

The main relevant effect of these lunisolar perturbations on re-entries is the 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 stabilising 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 uncertainty 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. For more detailed guidelines on the modelling complexity for re-entries from eccentric orbits refer to Table 3 of [RD073].

Figure D-3 shows an example of the re-entry prediction for HEO mission, with visible latitude band and limited longitude, performed three years before the targeted re-entry epoch. Further details on the procedure for the re-entry analysis of HEO orbits are available in Annex G and [RD074].

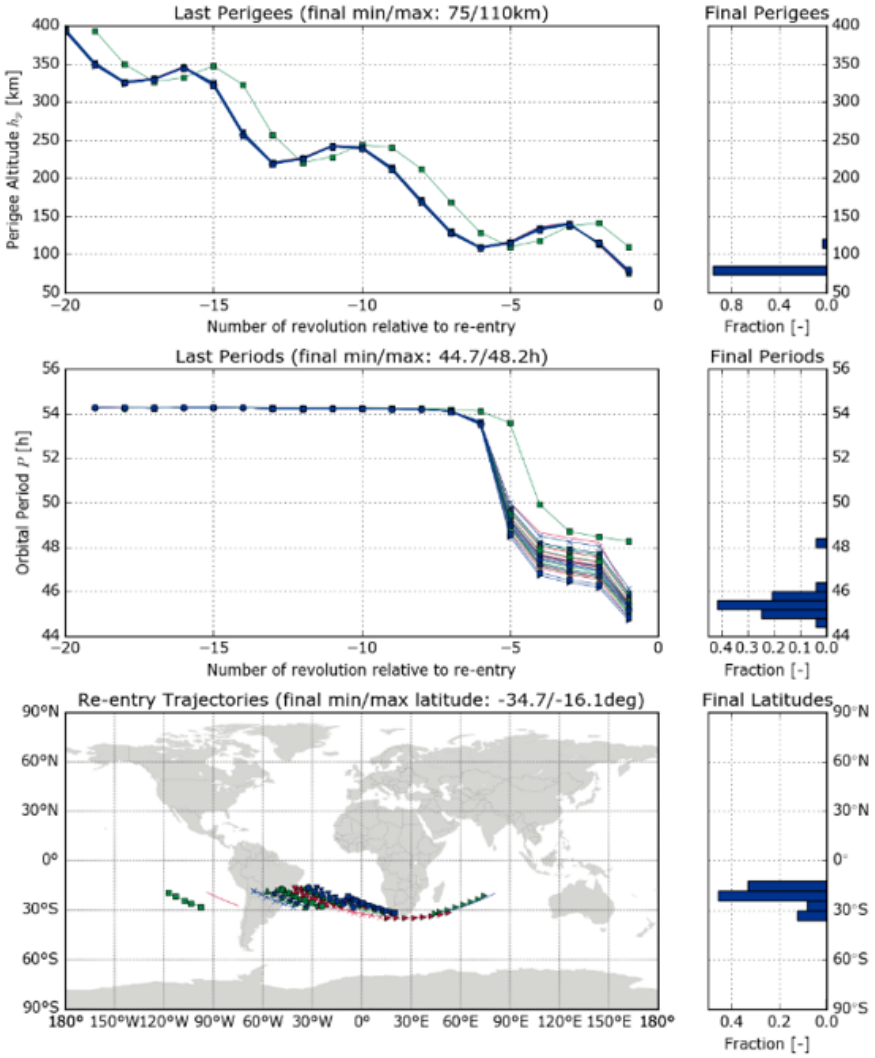

Figure D-3: Example of re-entry analysis for HEO mission with latitude band and delimited longitude

D.2.5.4 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 fragment 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.2.6 Explosion probability assessment

It is important to assess the probability and potential effects of an explosion during atmospheric re-entry, which can be caused by factors including, but not limited to, residual fuel, oxidizer, and pressurant.

The effect can be relevant especially for space systems performing controlled re-entry, when passivation is not performed in orbit. End of mission passivation and minimisation of residuals clearly allow to minimise the effect. If necessary, mitigation measures such as tank depletion can be

implemented. It is important to provide a rationale if the risk associated with potential explosions during re-entry is considered negligible; otherwise, the explosion scenario can be included in the worst-case re-entry casualty risk analysis.

An explosion model is implemented in the latest version of DRAMA/SARA (Section 7.2.3.3 of DRAMA 3.0.0 Final report [RD087]), based on NASA's EVOLVE 4.0 Standard Breakup Model [RD075]. It allows generating a list of fragments following an explosion event. Regarding the explosion triggers, there are two possibilities available, either based on altitude or based on temperature.

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

An example of higher fidelity assessment of the effects of explosions during a re-entry was performed for ATV, which is described in detail in [RD076]. The approach is summarised 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 pressure, temperature, density, velocity, G-load, convective heat flux, radiative heat flux.

b. A CFD simulation is performed to determine the flow-field around the spacecraft at different trajectory locations 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 maximum velocity, maximum temperature, maximum pressure, maximum partial pressure of diatomic and atomic oxygen, maximum heat flux.

d. 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.

e. 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.

f. 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.2.7 Casualty area and casualty risk

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 (Figure D-4):

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

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 m²

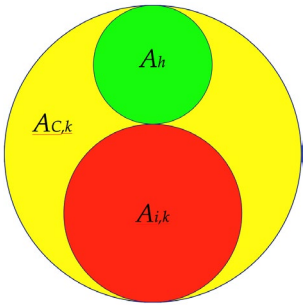

Figure D-4: Casualty area definition

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

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

The 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:

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

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

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

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 (D.2.5). 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 expectancy 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 (D.2.5), 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 ($E_{C,prob,re\text{-}entry}$), 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 ($P_{re\text{-}entry}$):

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

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

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

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, $P_{non\text{-}nom,,k}$ 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), $E_{C,non\text{-}nom,r}$ 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.2.8 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-5).

This can be used in conjunction with a population density model based on the Gridded Population of the World (GPW) v4 for the year 2017, 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-6). A simple tool (ORIUNDO) for this last step of computation is made available by ESA at https://sdup.esoc.esa.int/oriundo/.

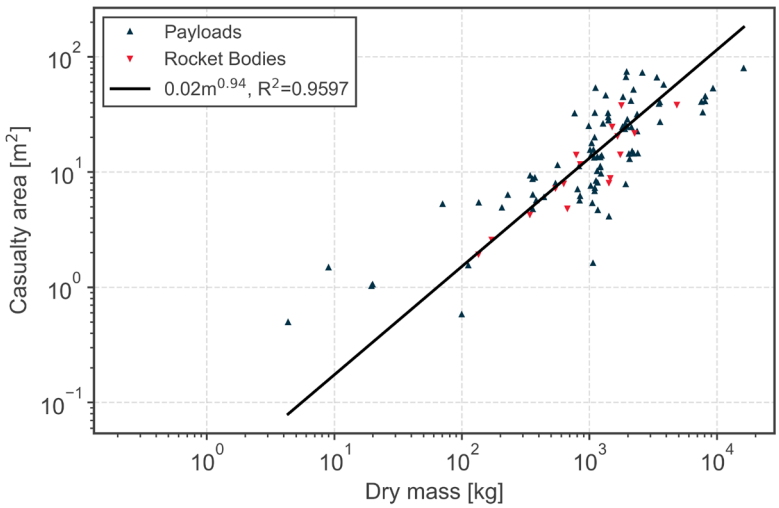

Figure D-5: Fit of historical re-entry assessment for the casualty area as a function object mass for an uncontrolled circular re-entry from LEO

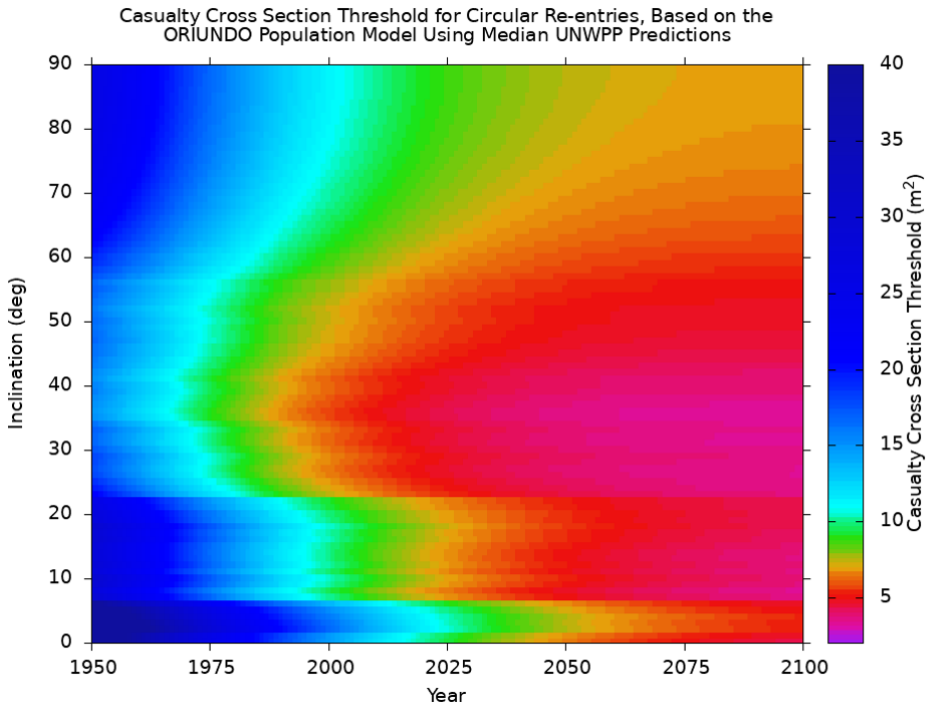

Figure D-6: Casualty cross-section threshold for a 10^{-4} casualty risk and uncontrolled re-entries from circular orbits, using median UN predictions for the population growth (Rev. 2017)

D.3 Destructive re-entry analysis tools

D.3.1 General

Numerous methods and tools have been developed to simulate spacecraft break-up during atmospheric re-entry, predict the characteristics of the surviving fragments, and finally estimate the expected number of casualties.

The basic framework of these tools, known as destructive re-entry or re-entry survivability analysis tools, is to simulate the break-up of an object by reconstructing its geometry and its trajectory, modelling the atmospheric profile, aerodynamics, aerothermodynamics and thermo-structural aspects.

DRAMA/SARA is a tool made available by ESA and is accepted to assess the expected number of casualties per re-entry or ground casualty risk. 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. ESSB-ST-U-004 Issue 1 – ESA Re-entry Safety Requirements [RD03] clause 6 provides further details, and Section 4.1 of DIVE [RD088][RD023] provides the minimum capabilities required by a tool in order to be comparable to DRAMA.

The destructive re-entry analysis tools are classified as follows, based on the way the spacecraft is represented, with the required computational effort depending on this representation:

- Object-oriented.
- Component-oriented.
- Spacecraft-oriented approach.

Information on these approaches is provided in Table D-2.

Table D-2: Characteristics of the different re-entry modelling approaches

	Object-oriented approach	Component-oriented approach	Spacecraft-oriented approach
Geometry modelling	Set of simple (primitive) geometric objects (spheres, cylinders, plates and boxes), with parent object as a container for the child/internal components	Set of simple geometric objects, with connection and inclusion relations	As close as possible to the real design, using mesh and not predefined shapes
Aerodynamic	Aerodynamic coefficients of the primitives pre-computed based on engineering methods, or, in recent versions, on CFD generated databases	As in the object-oriented approach	Aerodynamic parameters of the real geometry based on engineering methods, based on the local panel inclination method
Flight Dynamics Trajectory	Stable attitude motion or random tumbling (3 DoF) ballistic re-entry	Assumed attitude motion or random tumbling (3 DoF or partly 6 DoF)	Full attitude motion (6 DoF)
Aerothermal	Aerothermal analysis for each object separately, heating based on shape specific heat transfer	Aerothermal analysis for each object accounting for connections, heating based on space specific heat transfer	Aerothermal analysis for the complete panelised geometry, panel-wise melting analysis
Structural analysis / Break-up / Ablation	Break-up altitude pre-determined and leading to the release of all components with approximated ablation method for components, e.g. based on melting for metals, charring for CFRPs, from the outside layer-by-layer, while maintaining their shape	Break-up triggers (including melt or force) and subsequently calculated exposure trajectories	Break-up based on ablation, stress and structural integrity checks
Response time	Minutes	Minutes	Days (depending on model complexity)
Use	Certification tools Parametric and statistical analyses	Certification tools Parametric and statistical analyses	For detailed assessment at system or equipment level

	Object-oriented approach	Component-oriented approach	Spacecraft-oriented approach
Tools	DRAMA/SARA 2 (ESA, DEIMOS Space, HTG GmbH, ILR/TUBS) ASTOS/DARS & DIA (ESA, Astos Solutions GmbH) DAS (NASA) ORSAT (NASA) ORSAT-J (JAXA) DEBRIS (DEIMOS Space) FOSTRAD (University of Strathclyde)	DRAMA/SARA 3 (ESA, DEIMOS Space, HTG GmbH, ILR/TUBS, Belstead Research Limited, R. Tech) SAMj (Belstead Research) DEBRISK (CNES) RADID (ASTOS Solutions GmbH)	SCARAB (ESA, HTG GmbH) PAMPERO (CNES, R.Tech) TITAN (University of Strathclyde) LS-DARC (JAXA)

All the aforementioned tools in Table D-2 provide the characteristics of the fragments and the impact area as results, but not all of them include the calculation of the expected number of casualties.

It important to mention that not all the mentioned tools fall perfectly into one of the three categories and that the distinction between object/component-oriented and spacecraft-oriented is becoming increasingly blurred with the recent upgrades.

Finally, it is important to note that all the tools above were developed to assess the demisability of a spacecraft during re-entry and, therefore, the modelling assumptions are oriented towards being conservative in such assessment. For this reason, they are usually not suitable for demonstrating the efficacy of so-called "design-for-containment" approaches, where instead the objective is to prove that the connected components do not demise.

D.3.2 Object-oriented tool approach

Analysis with an object-oriented tool (Figure D-7) 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 selection of shape type (sphere, box, plate, cylinder), dimensions, materials.

e. Material properties of all relevant components, including at least density, heat capacity, melting temperature, heat of melting, 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.

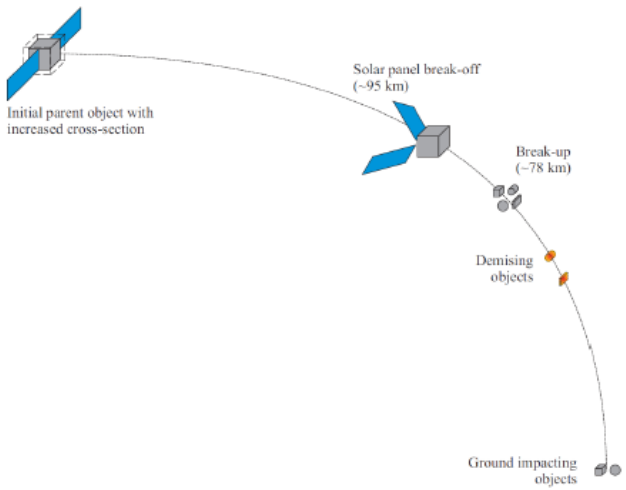

Figure D-7: Object-oriented tool concept

D.3.3 Component-oriented approach

The component-oriented approach is intended as an extension of the object-oriented approach. 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 recommended 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 recommended 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 uncertainties on the re-entry casualty risk and better quantify a confidence level.

D.3.4 Spacecraft-oriented approach

Spacecraft-oriented tools consider the spacecraft geometry and moments of inertia in a full-force and torque six degree of freedom analysis (Figure D-8). 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 probabilistic 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, HTG GmbH) and PAMPERO (CNES, R.Tech) are examples of tools based on the spacecraft-oriented approach.

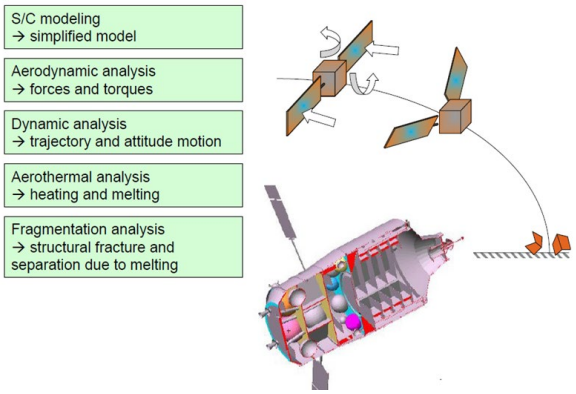

Figure D-8: Spacecraft-oriented tool concept

The re-entry model of the spacecraft–oriented tool is as representative as possible 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 shapes, dimensions, 2D drawings with readable or measurable dimensions or 3D CAD file, masses, centres of mass, moments of inertia, materials, maximum structural loads for major connection elements.

c. Material properties of all relevant components, including at least density, specific heat capacity, melting temperature, melting heat, emissivity. 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.3.5 Coupling of higher-fidelity physics-based numerical tools

Although the level of simplification can vary depending on the adopted geometric representation, both the categories, object or component and spacecraft-oriented tools, apply approximation methods in the assessment of the involved physical quantities, the so-called engineering methods. Simplifications are applied for the estimation of the aerodynamic, aerothermodynamic and thermo-structural parameters.

Higher fidelity methods consist in the coupling of accurate flow computation (Direct Monte Carlo Simulations, Computational Fluid Dynamics), thermo-structural computation of the fragmentation process (Finite Element methods), 6 DoF flight dynamics propagation.

This approach is extremely expensive in terms of running time (in the order of weeks), but the only precise for the solution of the flow conditions surrounding the object. These methods allow for obtaining more precise estimates of the material degradation of objects and fragments with complex shape (e.g. concavities) or in presence of complex physical phenomena, such that the shock impingement, shock-shock interactions.

D.3.6 DRAMA/SARA Guidelines and Technical Documentation

SARA is part of the DRAMA tool suite, which is made available by ESA and is accepted to assess the expected number of casualties per re-entry or ground casualty risk. It can be downloaded from the Space Debris User Portal [RD078] upon registration. A DRAMA Python Package is also available at the same link and particularly useful to perform stochastic assessments and, in general, repeated assessments.

For detailed background information on DRAMA/SARA and practical guidelines to perform a representative analysis, the following technical documentation is available:

- DRAMA Software User Manual [RD078] for the operations environment (hardware and software configuration), the external view on the installation folder and the workspace, and the operations manual (installation and description of the graphical user interface)
- Upgrade of DRAMA's Spacecraft Entry Survival Analysis Codes [RD087];
- Guideline on DRAMA Spacecraft Modelling (SARA) [RD073] for rule-based re-entry modelling procedure designed to ensure that the analysis is independent of the individual engineer modelling the spacecraft
- Guideline on DRAMA Materials (SARA) [RD079] for practical implementation and usage of the material database for re-entry risk verification with SARA.

Concerning material modelling, it is important to note that for composites, CFRP or GFRP, models for demise or fragmentation are an area of active research and are not generalizable outside tested and experimental ranges. For sandwich panels with CFRP facesheets and aluminium honeycomb core, DRAMA employs two different models, with either 4 or 8 CFRP plies, derived from experimental activity based on HC-AA7075 material and calibrated through testing. These models account for the

fact that the failure mechanism of CFRP sandwich panels is primarily due to the detachment of the facesheets from the honeycomb core, rather than the failure of the CFRP material itself. The model is suitable for the common CFRP sandwich panels used in spacecraft structures, such as M55J/EX1515 with Al5056 honeycomb core.

For monolithic CFRP structures, such as tertiary structure brackets, struts, and large booms, the model currently implemented in DRAMA is conservative, having been developed for Composite Overwrapped Pressure Vessel (COPV) overwraps. However, the model in DRAMA can be adjusted to fit experimental results, so equipment or structural elements composed mainly of CFRP are evaluated on a case-by-case basis. The behaviour of CFRP during re-entry is strongly influenced by the matrix, which is a key factor in composite demisability due to its low char yield (i.e. the amount of material left over after being subjected to high temperature pyrolysis). For further details, the user is referred to Guideline on DRAMA Materials (SARA) [RD079].

For additional information related to execution of Monte Carlo simulations with SARA, the reader is invited to refer to paragraph 7.3.5.4 of DRAMA-Software-User-Manual [RD078]. In addition, ESA makes available an example script for Monte Carlo analysis, which are based on the use of the DRAMA Python Package. The script is available on ESA's Space Debris Forum [RD080].

D.4 Controlled re-entry

D.4.1 Methodology

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 decreases 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 [RD03].

The controlled re-entry operation sequence is determined by considering several factors, including:

a. Attitude and orbit control modes.

b. Number and position of the thrusters.

c. Available and depleted propellant amounts.

d. Propulsion tank pressurisation level (i.e. minimum level needed for thrust performance).

e. Available power supply from power generators, and energy storages.

f. 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.

g. 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.4.2 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 (> 100) simulation runs (Monte Carlo), which are based on the dispersions of the relevant variables to cover all uncertainties of the model, where the amount of runs yields stable confidence intervals (Figure D-9):

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.

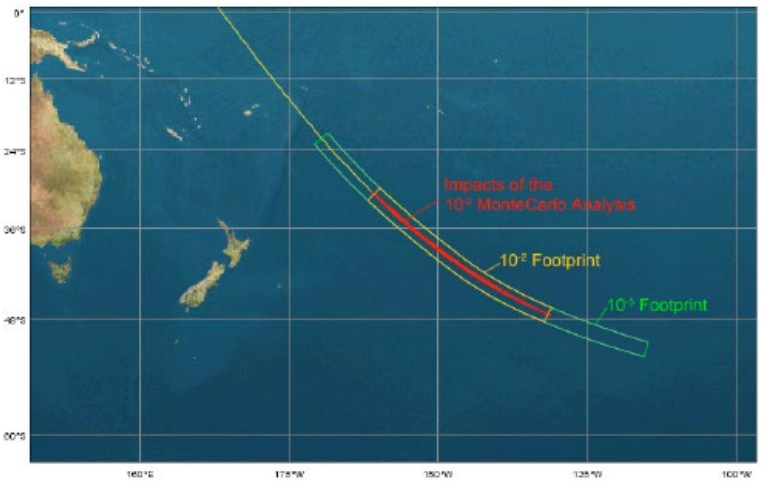

Figure D-9: Example of DRA (10^{-2} Footprint) and SRA (10^{-5} Footprint)

The definition of the DRA and SRA, which to the first order includes 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.4.3 Uncertainties for controlled re-entries. Nominal and off-nominal scenarios

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

a. Nominal case i.e. controlled re-entry.

b. Off-nominal cases i.e. 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, from 5,4 % to 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 *a* to *k* 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.

In order to evaluate the re-entry of a LEO spacecraft, DRAMA/SARA can be used to perform a Monte-Carlo simulation contemplating uncertainties on, for instance:

a. Spacecraft orbit after the last burn.

b. Atmospheric density.

c. Fragments properties (e.g. ballistic coefficient).

d. Break-up altitude.

e. Lift over drag ratio.

Dedicated sensitivity analyses can be used to conclude about the impact of a certain parameter on the re-entry results and about its importance for the Monte-Carlo simulation. The calculation of the dispersion of the spacecraft orbit after the last burn takes into account the eventual uncertainties on, for instance:

a. Spacecraft mass and CoG.

b. True latitude of the last burn.

c. Duration of the last burn.

d. Total impulse of the last burn.

e. Thrusters alignment.

f. Thrusters off-modulation during the last burn.

g. Attitude errors during the last burn.

h. Navigation errors.

For example, the following dispersion parameters have been used for a controlled re-entry of a spacecraft in LEO, in a past mission:

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.4.4 Off-nominal scenarios for non-destructive re-entries

During the design of the controlled re-entry mission phase of a vehicle designed to survive the re-entry, the expected number of casualties is computed for all the off-nominal scenarios. Examples are the cargo return vehicles and more generally reusable vehicles.

In case of Fragmentation Re-entry assessment for Space Vehicle designed to survive Re-entry phase, the following general guidelines are applicable:

a. The use of high-fidelity tools for the destructive re-entry analysis is recommended (e.g. spacecraft-oriented tools, CFD/DSMC based tools). Component oriented tool (e.g. DRAMA/SARA) can provide a preliminary estimation of the casualty risk but the maximum number of allowed items and the available shapes can be not appropriate for a space vehicle designed to survive the re-entry as thermal protection systems are currently not considered in the destructive re-entry analysis tools.

b. It is recommended to perform material characterisation of protective system (e.g. ceramic/ablative Thermal Protection System) if not present in the database.

c. Different off-nominal mission scenarios need to be selected for the Re-entry analysis and the relevant occurrence probability assessed. Mission analysis and Failure Mode and Effect Analysis (FMECA) drive the selection of the flight parameters (inclination, initial orbit) and fragmentation

event triggers (failure event during nominal re-entry scenario). Example of different mission scenarios to be considered is reported hereafter:

1. Triggered fragmentation event at different altitudes (e.g. 45, 55, 65, 75, 85 km) during nominal re-entry;
2. Failure event at 120 km during descending phase leading to tumbling attitude;
3. Uncontrolled re-entry event (flight path angle zero);
4. Uncontrolled re-entry event (flight path angle zero), failure event at 120 km.

D.5 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 as reported in Table D-3.

Table D-3: 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.2.8) (for circularised LEO re-entry orbits)
b. First assessment of E_C (D.2.7) using likely re-entry conditions (D.2.3) and a first space system model using object-oriented tools (for all orbits) (D.3)</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 uncontrolled re-entry using a more elaborate space system model and object-oriented tools (D.3)</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.2.5.2)</td><td>Eccentric re-entry (D.2.5.3)</td><td>First sizing of debris fall-out footprint (D.4.2)</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.2.1)
2. Final assessment of a space system model in object-oriented tools, including uncertainty quantification (D.3.2)
3. Establishment of a model in a spacecraft-oriented tools (for confirmation) (D.3.4), determination of explosion likelihood and effects (D.2.6), and passivation measures</td></tr>
<tr><td colspan="2">Uncontrolled re-entry</td><td>Controlled re-entry</td></tr>
<tr><td>Circular re-entry (component-based or spacecraft-oriented tool)</td><td>Eccentric Re-entry (component-based or 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.2.7)</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="2">PDR
Phase B</td><td colspan="5">1. Refinement of the re-entry model for the component-based or 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</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
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</td><td>$E_{C,nom}R_{nom}$
Mitigation options:
1. Lower re-entry perigee
2. Alternate target area
3. Passivation</td><td>$E_{C,off\text{-}nom}P_{off\text{-}nom}$
Mitigation options:
1. Improvement of system reliability
2. Passivation</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 violation of 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>
<tr><td>EOL</td><td colspan="2">Uncontrolled re-entry</td><td colspan="3">Controlled re-entry</td></tr>
<tr><td colspan="2">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-</td><td>Controlled re-entry
1. Inform sea traffic authorities for NAVAREA</td><td>Degraded Controlled re-entry
1. Same actions as for controlled re-entry</td></tr>
</table>

Review/ Phase	Action				
		entry epoch and location 2. Notify national alert centers and supply them with the prediction results 3. Confirm the re-entry	messages at least 6 days before 2. Inform air traffic authorities for NOTAM messages at least 2 days before	

Annex E
Passivation methods

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

Table E-1: Passivation measures

Subsystem	Unit	Passivation Measures	Critical Issues and Remarks
GNC	Attitude Control Sensors and Actuator	• Disconnection from power supply sources	• A dedicated GNC mode can be implied.
GNC	Cold Gas Thruster	• Depletion of gas supply source	• A dedicated GNC mode can be implied.
GNC	Control Moment Gyro	• Disconnection from power supply sources • De-spin or 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	• Disconnection from power supply sources • De-spin or 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	• Fix and block the relative movements	• Mobile parts can lead to mechanical ruptures due to fatigue.
Mechanism	Electro-explosive or pyrotechnic device	• Disconnection from power supply sources • Disarming of the electrical barrier • Isolation by design (mechanical barrier) of the primary pyrotechnic component (e.g.	

Subsystem	Unit	Passivation Measures	Critical Issues and Remarks
		European/NASA Standard Initiator) from the secondary explosive chain • Disconnection from the commanding unit (i.e. PCDU)	
Power	Battery	• Self-protection • Discharge • 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 likelihood of debris generation is very low. • 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 demonstrate 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: o Available information on the battery cell procurement to check if they come from well-reputed supplier.

Subsystem	Unit	Passivation Measures	Critical Issues and Remarks
			○ 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). ○ 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. ○ Adequacy of measures foreseen by design for the end of mission passivation to withstand the thermal and radiation environment for the entire on-orbit duration (e.g. in LEO, 25 yrs before re-entry). ○ 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). ○ 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). ○ 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.

Subsystem	Unit	Passivation Measures	Critical Issues and Remarks
Power	Fuel Cell	• Self-protection • Discharge • Disconnection from any charging source (e.g. solar array) • Depressurization of the cells (if needed)	• See Battery.
Power	Power Conditioning and Distribution Unit (PCDU)	• Isolate power storages from power generators • Switch-off all possible circuits	• The PCDU can include a function to short-circuit the spacecraft bus to derive battery discharge.
Power	Solar Array	• Disconnection from power bus or batteries • Short-circuit	• See Battery.
Propulsion	Pipeline	• Venting (as far as possible) • Scavenging of residual propellants actively (though pressurization) or passively (by slow evaporation) • 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.

Subsystem	Unit	Passivation Measures	Critical Issues and Remarks
Propulsion	Pressurant Tank	• Venting (as far as possible) • 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	• Venting (as far as possible) • Depletion burn(s) • 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

Subsystem	Unit	Passivation Measures	Critical Issues and Remarks
			hypergolic propellants; pressure build-up that can cause tank explosion (e.g. to be prevented through thermal protection).
TC	Telemetry	• Switch-off the telemetry transmitter with monitoring RF signal	
Thermal Control	Heat Pipe	• 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-30-02 [RD011], clause 5.3), or a qualitative hazard risk matrix as in Table E-2. In absence or insufficiency 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 E-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.:

a. Very low, e.g.: event extremely remote to happen; $P \leq 10^{-5}$.

b. Low, e.g.: event not expected to happen; $10^{-5} < P \leq 10^{-4}$.

c. Moderate, e.g.: event not likely to happen; $10^{-4} < P \leq 10^{-3}$.

d. High, e.g.: event likely to happen; $10^{-3} < P \leq 5 \cdot 10^{-3}$.

e. Very high, e.g.: event very likely to happen; $P > 5 \cdot 10^{-3}$.

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

a. Minor (negligible), e.g.: no or minor damages to the space system; no debris release in orbit.

b. Moderate, e.g.: moderate damages to the space system; no debris release in orbit.

c. 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.

d. 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.

e. 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:

a. ESA Technical Authority for Space Debris Mitigation recommendations.

b. Subject Matter Experts judgements.

c. Project assessments (e.g. assessment of the probability of explosion/burst given the space environment conditions and design qualification data).

d. ESA Alerts applicable to the perimeter of the hazard.

e. State-of-the-Art knowhow available in ESA.

Annex F
Disposal reliability, diagnostic and prognostic methods

F.1 Objectives

This Annex provides guidelines for the assessment of the disposal reliability and adoption of diagnostic and prognostic methods.

F.2 Disposal reliability assessment

F.2.1 General

A successful disposal of a space system can be assured by performing the following assessment of the probability of successful disposal (focusing on the reliability contribution):

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.

F.2.2 Disposal reliability assessment during the development phase

The reliability 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 model for the assessment of the reliability 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.

It is normally desirable 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 is 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.

F.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 are mandatorily considered in an update to the disposal reliability prediction, if a conservative assessment is maintained. However, they can be useful contribution to a rationale for a potential extension of the space system 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.

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.

In order to allow an efficient re-assessment of the disposal reliability it is important to build the prediction models during the development phase such that they can be used as risk monitors along the life of the spacecraft.

F.3 Diagnostic and prognostic methods

The following approaches have been investigated with respect to their capability to improve the assessment of the probability of successful disposal:

a. Approach 0: Reliability as per CDR design.

b. Approach 1: Current (in-orbit).

c. Approach 2: Diagnosis and REX:
   1. Health monitoring;
   2. REX.

d. Approach 3: Prognostics:
   1. Stochastic models, or Weibull laws;
   2. Model-based models;
   3. Data trends.

Summary of the objectives, benefits and limitations of proposed approaches are provided in Table F-1, together with a list of the degradation phenomena and impact on mission extension in Table F-2, and a summary of the benefits and drawbacks of the different approaches applied to different units in Table F-3. The colours in Table F-2 indicate the impact on the spacecraft life extension (i.e. green = low, orange = medium, red = high). The colours in Table F-3 indicate the expected benefits, with dark green corresponding to the highest level.

Table F-1: Summary of objectives, benefits, and limitations of proposed approaches (extracted from [RD044])

Approach	Objective	Methods	Inputs	Outputs	Main benefits and improvements	Main limitations
Approach 2.a: **Health monitoring**	Update and better evaluation of unit failure rate with real operating conditions	Simple exploitation of TMs	Real operating conditions (e.g. temperature, duty cycles, stress)	Updated units failure rate	More realistic and less worst-case assumptions for units failure rate	Some parameters not always monitored or available (e.g. electrical stress of the components, number of cycles or activations)
	Monitoring of unit performance to identify symptoms of anomaly, failure, or performance degradation	Identification of abnormal behaviour from TMs, by comparison with expectations	Units TMs	Inputs for prognostic methods	Existing margins (positive/negative) to update the redundancy schemes (e.g. acceptable failures) Non-nominal behaviour identifiable and units recoverable before failure occurrence	Additional data, time and workload Better quantity, quality and frequency of data and additional TMs needed for some units
Approach 2.b: **REX**	Computation of unit failure rate	Chi-Square	Cumulated operating hours and failures occurred during test or in-orbit operations	Unit failure rate based on REX	Failure rate from in-flight data	Large amount of data to derive a relatively low (and accurate) failure rate (less feasible on unique spacecraft design) Assumptions on failures accounted or design models considered Incorrect or risky approach for units experiencing wear out phenomena
	Update and re-assessment of unit failure rate through test or in-flight data	Chi-Square Bayesian techniques	Initial unit failure rate Cumulated operating hours and failures occurred	A posteriori failure rate from a priori value and REX	More realistic (and ideally lower) failure rate taking from heritage or experience of the unit and its initial reliability assessment	
	Better understanding of units anomalies, failures, and impact on the mission	Statistical analyses on a database of anomalies and failures	Database of anomalies and failures occurred in orbit (internal or public database)	Statistics on occurrence, severity, time distribution and impact on the mission extension or EOL disposal	Identification of behaviours different from a constant failure rate (e.g. infant mortality or wear out) Additional element supporting EOL decision (e.g. avoiding mission extensions in a failure scenario leading to loss of past similar spacecraft)	Statistics merging different spacecraft designs, suppliers, orbits, technologies. Availability, accuracy, and completeness of publicly disclosed failures

Approach	Objective	Methods	Inputs	Outputs	Main benefits and improvements	Main limitations
Approach 3.a: **Prognostic based on stochastic models**	Statistical prediction at any time of future units status and RUL	Mortality model (lognormal), Weibull, Bertholon, Pseudo Failure Time, Gamma and Weiner process	Test/in-flight data (health monitoring)	RUL or unit survival probability	No longer a constant failure rate for units experiencing wear out phenomena More accurate and realistic reliability assessment beyond the nominal lifetime	Large amount of data to estimate the parameter and with good confidence Engineering judgment to estimate the parameters of the law
Approach 3.b: **Model-based prognostic**	Qualitative prediction at any time future units status and RUL	Exploiting engineering models of the unit	Test or in-flight data (health monitoring) Engineering model of the unit	Future performance of the unit and RUL	Evaluation of a time dependent failure rate (no longer constant assumption) from unit RUL Wear out modelled Functional failures (unit no longer functional) taken into account in addition to random physical failures	Performance or degradation model not available for all units Accuracy of the models, especially outside the nominal/qualified behaviour of the unit RUL not necessarily exploitable in the reliability model Access to proprietary tools
Approach 3.c: **Prognostic based on data trends**	Identification of degradation trends to unit failure at the end of RUL	Several Data Trend Analysis methods	In-flight data Data trend analysis tool	Time dependent failure rate		Degradation trends leading to the failure of the unit not common in space application Large amount of data and workload needed
	Identification of variation in TM from nominal condition	Several Data Trend Analysis methods	In-flight data Data trend analysis tool	Identification of unit anomalous behaviour	Suggestion about the possibility to perform dedicated analyses and monitoring of the unit	TM variation to be analysed and possibly compared to reference conditions (either nominal or failure) or other similar units Large amount of data needed for training and workload if done on ground, or computation capabilities and memory needed if done on-board

Table F-2: Summary of the degradation phenomena and impact on mission extension/EOL disposal (extracted from [RD044])

Spacecraft unit	Involved function	Degradation phenomena	Main causes / factor	Degradation timeframe	Observables	Recovery / corrective actions	Impact on the spacecraft life extension / EOL disposal	Comments (differences / similarities)
Battery	Electrical power storage and supply during eclipse	Aging resulting in energy and power loss (long-time storage) Fading (mainly from positive electrode resulting in capacity loss and internal resistance increase)	Storage conditions, mainly temperature and State of Charge (SoC) Battery operating conditions (temperature, Depth of Discharge, charge rate)	Slow, continuous, well-known process (modelled)	Battery voltage, current and power from TM Battery capacity decrease and internal resistance increase derivable from TM	Degradation considered in the design Redundant cells usually available (otherwise, spacecraft power consumption, HW matrix or modes to be adapted)	Low to medium since wear out known, monitored and mastered Spacecraft power margins (payload usually OFF, EPS sized on worst-case scenarios) No or few failures observed in orbit (at least from Li-ion cells)	Higher risk for spacecraft with electrical propulsion (requiring power supply) Different operating conditions (e.g. DoD, cycles) depending on the orbit (e.g. LEO, GEO) and different degradation impact and timeframe
Solar array (SA)	Electrical power generation from the solar energy	Damage mainly from cumulative radiation and MMOD impact Failure of components (diodes)	Radiation (non-ionizing, atomic displacement effects, minor effect from ionization) Solar flares and MMOD impact High cycling temperatures or simply high temperatures	Slow, continuous, well-known process (modelled) Higher and unpredictable degradation from solar flares and MMOD impact	Short-circuit current (Isc), open circuit voltage (Voc) and maximum power (Pmax), generated power Thermal sensors for aging characterization (if implemented forth and backward on the panels)	Degradation taken into account in the design, power margin generally covering loss of strings. Spacecraft power consumption reduction or modes adapted (recovery) Ultimately, additional power provided by battery	Low to medium since wear out known, monitored and mastered. Spacecraft power margins (payload usually OFF, EPS sized on worst-case scenarios)	Higher risk for spacecraft with electrical propulsion (requiring power supply) Higher risk in case of completely loss of one solar array (e.g. failure of the Solar Array Drive Mechanism) Higher risk if low power margins (e.g. small spacecraft)
Solar Array Drive Mechanism (SADM)	Electrical power transmission and rotation of solar arrays	Bearings degradation (wear) Motor degradation (isolation, open or short-circuit)	Mainly lubricant wear out Thermo-mechanical cycling in components Contamination or isolation failures (low probability) MMOD (low probability)	Slow and continuous for the bearings Unpredictable generally if contamination is concerned Random for motor failure	Motor currents, potentiometers position, temperatures	Lubrication homogenization via complete arrays rotations Mostly no recoverable actions for lost functions Ultimately, additional power provided by the battery if one wing power not sufficient (not applicable if only one SADM)	Medium to high since loss of SADM rotation function implies diminution of available power Indirect impact on AOCS and propellant consumption	Risk dependent on thermal conditions (e.g. in LEO) SADM current affecting temperature Worst-case in LEO (rotation conditions and mechanical charging compared) Higher risk for spacecraft with only one SADM (only one SA)

Spacecraft unit	Involved function	Degradation phenomena	Main causes / factor	Degradation timeframe	Observables	Recovery / corrective actions	Impact on the spacecraft life extension / EOL disposal	Comments (differences / similarities)
Chemical propulsion (THR)	Attitude and orbit control	Degradation of catalyst granules leading to lower thruster force and Isp Thermal shock destroying catalyst granules when rapidly heated Thermal choke (propellant vaporization in capillarity feed tube leading to a reduced, or no, propellant flow)	Trend strongly depending on firing mode (pulse on time and pulse cycle period) and on the catalyst bed temperature and thermal cycles Linked to cold starts Linked to THR design and low pressure and high temperature conditions	Slow and predictable evolution with the number of thruster activation	Thruster temperature profile during the burn Thruster force and Isp evolution over time, derived from realised Delta-v with respect to the required one or by evaluating the duty cycles and the number of THR actuations	Thruster qualified with (expected) real operating conditions and multiplication factor on the lifetime Usually available redundant thrusters (if not sufficient, to adapt EOL disposal manoeuvres strategy)	Medium to high since if THR indispensable for EOL manoeuvres and lower performance or failures leading to degraded or emergency disposal	Degradation phenomenon affecting mainly mono-propellant THRs (no major degradation in bi-propellant systems) Higher risk in case of a THR used at operating conditions different from the tested ones or beyond its qualified lifetime
Electrical propulsion (HET)	Attitude and orbit control (or RWs desaturation)	Erosion of the ceramic walls of the anode chamber leading to the end of the life of the HET when the magnetic circuit is eroded and the magnetic field interrupted Oxidation of the emitting elements of the cathode	Erosion caused by the ion sputtering when the thruster is used Contamination of the propellant	Slow, continuous and well-known process (modelled)	Reference potential of the cathode (CRP), discharge current, current oscillation, thruster force and Isp	Thruster qualified with (expected) real operating conditions and multiplication factor on the lifetime Usually available redundant thrusters (if not sufficient, to adapt EOL disposal manoeuvres strategy)	Medium to high if HET indispensable for manoeuvres (station keeping or disposal) Lower performance or failures leading to additional Xenon consumption or contingency manoeuvres (longer manoeuvre duration)	Higher risk if only one HET (e.g. small and "low cost" spacecraft)
Other Propulsion units (SAPT)	Monitoring of tank pressure and temperature, and PVT method for propellant mass estimation	Ageing and radiation effects	TID for electronics	Slow and continuous	Pressures Temperatures	Bookkeeping or thermal gauging instead of PVT method	Low since SAPT information not always indispensable (alternatives remaining propellant mass estimation)	
Other Propulsion units (valves, regulator)	Monitoring, management and regulation of	Seat mechanical wear	Cycling exceeded Material flaw in seat	Slow and continuous, but sharp increase of	State of valve Pressures	Redundancy (if applicable)	Medium since necessary for the propulsion subsystem (although no major degradation effects and	Risk not changing with respect to the altitude, but stronger for high cycles units (admissible domain of cycles)

Spacecraft unit	Involved function	Degradation phenomena	Main causes / factor	Degradation timeframe	Observables	Recovery / corrective actions	Impact on the spacecraft life extension / EOL disposal	Comments (differences / similarities)
	propellant flux and pressure	Motor degradation isolation Open or short-circuit Contamination	Environment temperature exceeded Particles trapped TID for electronics	damage at a triggering level	Temperatures Motor current	Design margins and qualification Fluidic branch isolation (if applicable)	probability expected, or typically observed in orbit)	
Reaction wheel (RW)	Attitude control (kinetic momentum)	Degradation of the ball bearings linked to deterioration of lubrication over time and leading to increased friction torque (dry and viscous frictions) Motor degradation (isolation, open of short-circuit) Electronics wear out	Insufficient or unstable film thickness leading to metal-on-metal contact between bearing balls and races Lubrication deterioration and zero-speed crossings damaging wheel and limiting lifetime (wheel speed, number of remaining active wheels) Radiation effects	Assumed linear with the time (although cases of rapid and premature degradation observed in orbit)	Higher friction torque, equivalent to higher torque (higher power demand) and higher temperature for the same commanded torque Torque and friction derived from motor current and speed measured values	Usually a redundant RW is available (if not sufficient, to adapt the attitude control and manoeuvres strategy, e.g. to use less RWs or alternative AOCS actuators)	Medium since RWs baseline actuators for the AOCS and EOL disposal strategies (adapted to use less RWs or alternative AOCS actuators, if needed)	Valid only for RW with ball bearing system (RW using magnetic bearing not experiencing wear out effects) Feasibility of reduced de-orbiting mode using other AOCS sensors (R&D studies)
Magneto torquer (MTQ)	Attitude control and RWs desaturation	No major degradation expected	Ageing effects of thermal cycling	If any, very slow and unknown process	Comparison of commanded and measured current	MTQ internally redundant Use of another AOCS actuator	Low since not usually used as nominal AOCS actuator No major degradation expected	
Magnetometer	Estimation of the Earth magnetic field direction and intensity	No major degradation, except ageing and radiation effects on the electronics	Mainly radiation effects as per a typical electronic unit	Slow, continuous process	Comparison of measured and expected Earth magnetic field direction / intensity	Usually a redundant Magnetometer available (or other AOCS sensors) Earth magnetic field model for MTB commanding	Low since not usually used as nominal AOCS sensor for the extension of the lifetime or de-orbit	

Spacecraft unit	Involved function	Degradation phenomena	Main causes / factor	Degradation timeframe	Observables	Recovery / corrective actions	Impact on the spacecraft life extension / EOL disposal	Comments (differences / similarities)
Sun acquisition sensor	Estimation of the Sun direction for spacecraft attitude	Change of performance and response of photovoltaic cell of the sensor	Aging phenomena due to the radiation environment	Slow, continuous and known process	Current output for a given and known Sun position decreasing over time	Adjustment of the parameters used by the AOCS to derive the position of the Sun to compensate wear out and have consistent attitude information	Low since reliable unit and wear out known and considered in the AOCS control loop Other AOCS sensors usually used	
Star Tracker (STR)	Attitude measurements (accurate determination)	Lower accuracy of STR data Impact on the attitude determination of AOCS nominal mode	Radiation Thermal cycles Contamination Ageing	Slow, continuous process	Internal unit health check (quality index) STR number used in the field of view Voltage and current, operating temperature of CCD and equipment	Usually, redundant STR available Use of other AOCS sensors (if available) although less accurate	Medium to high since STRs indispensable for EOL manoeuvres and lower performance or failures leading to degraded or emergency disposal	Feasibility of reduced de-orbiting mode using other AOCS sensors (R&D studies)
Gyroscope (GYRO)	Feeding on-board attitude estimation filter	Depending on the technology Drift and lower accuracy of the spacecraft angular rates estimation	Depending on the technology Radiation Aging	Rapid and unknown degradation	Health monitoring by comparing measured angular rates with the ones derived from other AOCS sensors (STR) Technology specific gyro TMs	Usually, a redundant GYRO available STRs alternative for angular rates estimation (gyro-less mode)	Medium since GYROs not completely indispensable for the attitude control Other AOCS sensors available	Maintenance operation (calibration) performed to correct or reduce gyro bias. Some GYROs having limited lifetime wrt spacecraft mission duration, therefore, are not always ON Other technologies less impacted by degradation
GNSS	Estimation Of accurate position, velocity and reference time using GPS or Galileo signals	Degradation of orbit determination accuracy (position, velocity) Drift of the receiver clock	Radiation Aging Thermal stress	Slow, continuous process	Several TMs available (NOF_SV, GDOP, Time Quality Index Clock Frequency)	Usually, a redundant GNSS available Orbit restitution possibly from ground	Low since wear out known and considered in the AOCS control loop Other means for orbit determination typically available	No significant degradation phenomena expected for the RF section of GNSS GNSS generally integrated in LEO and MEO only (more recently in GEO for autonomous Electrical Orbit Raising)
Earth sensor	Determination of spacecraft roll and pitch angle	Depending on the technology ("telescope" degradation similar	Depending on the technology Radiation	Slow, continuous process	Depending on the technology	Depending on the technology and on the mission needs (e.g.	Low since reliable unit and wear out having a minor impact on the attitude accuracy	Mainly used in the past (no longer used in recent missions as STR usually preferred for higher accuracy)

Spacecraft unit	Involved function	Degradation phenomena	Main causes / factor	Degradation timeframe	Observables	Recovery / corrective actions	Impact on the spacecraft life extension / EOL disposal	Comments (differences / similarities)
		to STR or electronic units wear out)	Thermal cycles Contamination Ageing			nominal or back-up sensor)	Other AOCS units possibly usable	
Thermal control	Control of spacecraft temperature range for the nominal operation and limit the temperature gradients	Change of thermo-optical properties of the passive means for the temperature control (increase of absorptivity and decrease of emissivity)	Aging phenomena due to the external environment (e.g. UV, photonic, radiation, atomic oxygen)	Rapid increase during the first years in orbit and that then slower degradation trend from radiation dose accumulation	Thermo-optical properties not directly derived from in-orbit TM (increase of units temperatures observable)	Degradation considered in the design of the thermal control subsystem	Low since wear out known and considered in the design of the thermal control subsystem High margins between the observed range of temperatures in orbit and the maximum acceptable values for the spacecraft units	Recent R&D studies addressing new materials and technical solutions not or less affected by the ageing effects and leading to lower impacts on the spacecraft thermal control
Rotary actuators mechanisms (RA)	Rotation of elements like thrusters, antennas.	Bearings degradation (wear) Motor degradation (isolation, open or short-circuit)	Cycling exceeded Environment estimation weak MMOD	Slow and continuous for the bearings Random for motor failure or MMOD	Motor currents, potentiometers position, temperatures	Design, including safety margins, and qualification tests Redundancy for electrical motor parts Lubrication homogenization (full range cycling for the mechanisms)	From low to high depending on the RA (high for RA for thrusters, low for RA for antennas with no life extension or EOL impact)	Higher risk for thruster arms exposed to highly changing thermal environment (both MMOD and thermal control relying on shielding thermal passive structure)
Other electronic units	Data Handling, Power conditioning Telemetry and Telecommand	No major degradation except ageing and radiation effects on the electronics (TID, TNID)	Mainly radiation effects and component ageing	Slow, continuous process	Depending on the unit Usually temperature, voltage, current.	Usually, a redundant avionic equipment available Radiation design margin accounted (at least 1.2 times the expected radiation dose, from radiation analyses and WCA)	Medium since, although indispensable, usually low/negligible impact from degradation, if correctly designed units	Radiation environment significantly depending on orbital regime
Payload units	Depending on the mission application and objectives	Depending on the technology (covered by the effects mentioned above)	Depending on the technology (covered by the aspects mentioned above)	Depending on the specific payload unit	Depending on the specific payload unit	Depending on the specific payload unit, its redundancy schemes (if any)	Depending on the specific mission and payload unit	

Table F-3: Benefits and drawbacks of the different approaches applied to different units and recommendations (extracted from [RD044])

Spacecraft unit	Approach 2.a: Health monitoring	Approach 2.b: REX and Bayesian techniques	Approach 3.a: Prognostic based on stochastic models	Approach 3.b: Model-based prognostic	Approach 3.c: Prognostic based on data trends	Conclusions and recommendations (per orbit or type of mission, if applicable)
Battery	Useful to check the correct behaviour and performance of the batteries but estimation not always possible or accurate	Useful to derive less pessimistic failure rate but extended amount of cumulated hours is needed Not necessarily or directly reusable for different or new technologies or mission with different operating conditions	Interesting, but requiring a huge amount of data Current examples mainly based on engineering judgment (questionable representativeness)	Very useful to predict future performance and RUL Currently more often used approach	Surely improving health monitoring and the investigation of anomalies Currently few or no practical use mainly because of the amount of data needed	Approach 3.b and 2.a (health monitoring) currently used operationally Approach 3.c seen as very promising solution to further improve health monitoring and decision process
Solar array (SA)	Useful to check the correct behaviour and performance of the solar arrays Specific actions needed in case of anomalies Currently more often used during operations	Failure rate of cell usually already derived with a REX Valid and useful only for the same design on the same orbit (very good for constellation, but less for single missions)	Approach potentially leading to more accurate reliability figures since wear out effects are taken into account Usable only on spacecraft having the same orbit and technology because of the amount of data needed to derive a correct model	Currently the approach used to predict the performance degradation and size the SA accordingly Major drawback from lacking the statistical data, and hence confidence interval.	Allowing more easily detection of anomalies to anticipate actions before severe failure occurrence	On a short-term, risk assessment based on health status and simple model prognostic are complementary A combined approach is recommended for best decision (although new, promising since incorporating the benefits of model prognosis, random failures on design and MMOD, and the possibility to have Bayesian updating)
Solar Array Drive Mechanism (SADM)	Currently more often used during operations Not applicable if TM sampling is not adapted or if useful TMs not available (as in several current missions)	Evaluated and useful only for the same design on the same orbit (very good for constellation, but less for single missions)	Approach leading to more accurate reliability figures but limited applicability because of the complexity and amount of data needed to apply this method	Physics of failure good for new applications but to be focused on dominant failure modes to limit the complexity Difficult to validate the model because of the lack of data	Not evaluated yet and few data usually available (can be not really feasible)	Currently Approach 2.a (Health monitoring) is mostly often used during operations Generally, need of larger and more accurate data to apply the approaches Additional monitoring probably needed in some cases
Chemical propulsion (THR)	Useful to check the correct behaviour and performance of the THR via direct or indirect TMs	Currently not applied since difficult to gather enough data on similar units. THR usually not always ON (difficult to achieve a number of cumulated hours	Lower benefits expected compared to other approaches, especially because of the complexity and amount of data needed to apply this method	Useful to predict the THR performance but accuracy and validity questionable in case of a THR used at operating conditions different from the tested	Surely improving health monitoring and investigation of anomalies	Approach 2.a (health monitoring) is the one currently used operationally. Approach 3.c recommended as well (very promising solution to further improve health monitoring and decision process)

Spacecraft unit	Approach 2.a: Health monitoring	Approach 2.b: REX and Bayesian techniques	Approach 3.a: Prognostic based on stochastic models	Approach 3.b: Model-based prognostic	Approach 3.c: Prognostic based on data trends	Conclusions and recommendations (per orbit or type of mission, if applicable)
	Specific actions needed in case of anomalies or rapid degradation Currently more often used during operations	leading to reasonable failure rates)		ones or beyond its qualified lifetime	Currently few/no practical use mainly because of the amount of data needed	
Electrical propulsion (HET)	Useful to check the correct behaviour and performance Specific actions needed in case of anomalies or rapid degradation Currently more often used during operations	Approach currently used by some suppliers since unit not really covered by reliability standards. Useful to derive less pessimistic failure rate and to choose and optimise the architecture but not useful for the decision on the life extension or EOL disposal.	Lower benefits are expected compared to other approaches, especially because of the complexity and amount of data needed to apply this method Operating principles quite complex to be simply modelled	Useful to predict the HET performance but accuracy and validity questionable in case of a THR used at operating conditions different from the tested ones or beyond its qualified lifetime	Surely improving health monitoring and investigation of anomalies Currently few or no practical use mainly because of the amount of data needed	Approach 2.a (health monitoring) currently used operationally Approach 3.c recommended as well (very promising solution to further improve health monitoring and decision process)
Other propulsion units	Currently more often used during operations Not applicable if TM sampling is not adapted or if useful TMs not available (as in several current missions)	As a large fleet of units exist, a supplier REX useful to better evaluate the reliability and lifetime of these units Limitation in having to keep a coherent design	Approach leading to more accurate reliability figures but limited applicability because of the complexity and amount of data needed to apply this method	Physics of failure for space application can be not preponderant (not among the preferred approach)	Not evaluated yet and few data usually available (can be not really feasible)	Currently Approach 2.a (Health monitoring) more often used during operations Approach 2.b (REX) recommended for heritage technologies Generally, need of larger or more accurate data to apply the approaches Additional monitoring probably needed in some cases
Reaction wheel (RW)	Currently more often used during operations Useful to check the correct behaviour and performance of the RWs Specific actions needed in case of anomalies or rapid degradation	Useful to derive less pessimistic failure rate and to choose and optimise the architecture but not useful for the decision on the life extension or EOL disposal	Interesting approach but requiring a huge amount of data to be followed and especially to provide accurate results Current examples mainly based on engineering judgment (questionable representativeness)	Unit supplier developing mathematical model to simulate and evaluate the performance of unit (useful to predict the RW performance) Not a clear or complete view on the accuracy and validity of the models	Surely improving health monitoring and investigation of anomalies Currently few or no practical use mainly because of the amount of data needed	Currently Approach 2.a (Health monitoring) more often used during operations Approach 3.c recommended as well (very promising solution to further improve health monitoring and decision process) Approach 3.b can be further evaluated with the involvement of the supplier

Spacecraft unit	Approach 2.a: Health monitoring	Approach 2.b: REX and Bayesian techniques	Approach 3.a: Prognostic based on stochastic models	Approach 3.b: Model-based prognostic	Approach 3.c: Prognostic based on data trends	Conclusions and recommendations (per orbit or type of mission, if applicable)
Magneto torquers (MTQ)	Currently more often used during operations Useful to check the MQTs correct behaviour and performance	Not really needed because of the already low failure rate of this unit (huge amount of samples needed to derive a failure rate lower than the basic one)	Not really needed and applicable (degradation phenomenon negligible, unit already high reliable)	Not really needed and applicable to this unit since the degradation phenomenon negligible, and unit already high reliable	Not really needed and applicable to this unit since the degradation phenomenon negligible, and unit already high reliable	Currently Approach 2.a (Health monitoring) more often used during operations No major improvements needed for decision-making process for life extension or disposal
Magnetometer	Currently more often used during operations Useful to check the correct behaviour and performance	Not really needed because of the already low failure rate of this unit (huge amount of samples needed to derive a failure rate lower than the basic one)	Not really needed and applicable (degradation phenomenon negligible, unit already high reliable)	Not really needed and applicable to this unit since degradation phenomenon negligible, and unit already high reliable	Not really needed and applicable to this unit since the degradation phenomenon negligible, and unit already high reliable	Currently Approach 2.a (Health monitoring) more often used during operations No major improvements needed for decision-making process for life extension or disposal
Sun sensor	Currently more often used during operations Useful to check the correct behaviour and performance	Not really needed because of the already low failure rate of this unit (huge amount of samples needed to derive a failure rate lower than the basic one)	Not really needed and applicable (degradation phenomenon negligible, unit already high reliable)	Similar to SA Useful to predict the performance degradation but not needed to improve decision-making process for life extension or disposal	Not really needed and applicable (degradation phenomenon negligible, unit already high reliable)	Currently Approach 2.a (Health monitoring) more often used during operations No major improvements needed for decision-making process for life extension or disposal
Star Tracker	Currently more often used during operations Useful to check the correct behaviour and performance and to improve the accuracy of the reliability models with real operating temperatures (better than at CDR)	Useful to derive less pessimistic failure rate and to choose and optimise the architecture but not useful for the decision on life extension or EOL disposal	Lower benefits expected, especially because of the complexity and amount of data needed Wear out phenomena not so evident, or at least severe	Unit suppliers developing mathematical model to simulate/evaluate the performance Can be useful to predict the RW performance Not a clear or complete view on the accuracy and validity of the models	Surely improving the health monitoring and investigation of anomalies Currently few or no practical use mainly because of the amount of data needed	Currently Approach 2.a (Health monitoring) more often used during operations Approach 3.c recommended as well as (very promising solution to further improve health monitoring and decision process) Approach 3.b can be to be further evaluated with the involvement of the supplier
Gyroscope (GYRO)	Currently more often used during operations Useful to check the correct behaviour and performance and to improve the accuracy of the reliability	Useful to derive less pessimistic failure rate and therefore to choose and optimise the architecture but not useful for the decision on life extension or EOL disposal	Lower benefits expected, especially because of the complexity and amount of data needed Wear out phenomena not so evident, or at least severe	Not evaluated yet (no valid model found describing the degradation phenomenon)	Surely improving health monitoring and investigation of anomalies Currently few or no practical use mainly because of the amount of data needed	Currently Approach 2.a (Health monitoring) more often used during operations Approach 3.c recommended as well (very promising solution to further improve health monitoring and decision process)

Spacecraft unit	Approach 2.a: Health monitoring	Approach 2.b: REX and Bayesian techniques	Approach 3.a: Prognostic based on stochastic models	Approach 3.b: Model-based prognostic	Approach 3.c: Prognostic based on data trends	Conclusions and recommendations (per orbit or type of mission, if applicable)
	models with real operating temperatures (better than at CDR)					
GNSS	Currently more often used during operations Useful to check the correct behaviour and performance and to improve the accuracy of the reliability models with real operating temperatures (better than at CDR)	Useful to derive less pessimistic failure rate and therefore to choose and optimise the architecture but not useful for the decision on life extension or EOL disposal.	Lower benefits expected, especially because wear out phenomena not so evident, or at least severe	Lower benefits expected, especially because wear out phenomena not so evident, or at least severe	Surely improving health monitoring and investigation of anomalies Currently few or no practical use mainly because of the amount of data needed	Currently Approach 2.a (Health monitoring) more often used during operations Approach 3.c recommended as well (very promising solution to further improve health monitoring and decision process)
Earth sensor	Currently more often used during operations Useful to check the correct behaviour and performance	Not really needed because of the already low failure rate (huge amount of samples needed to derive a failure rate lower than the basic one)	Not really needed and applicable (degradation phenomenon negligible, unit already high reliable)	Not really needed and applicable (degradation phenomenon negligible, unit already high reliable)	Not really needed and applicable (degradation phenomenon negligible, unit already high reliable)	Currently Approach 2.a (Health monitoring) more often used during operations No major improvements needed for decision-making process for life extension or disposal
Thermal control	Currently more often used during operations Useful to check the correct behaviour and performance and to improve the accuracy of the reliability models with real operating temperatures (better than at CDR)	Useful to derive less pessimistic failure rate for the heaters but not useful for the decision on life extension or EOL disposal No need to re-assess the failure rate of the heaters in the future (already very low)	Not really needed and applicable (degradation phenomenon negligible, unit already high reliable)	Mathematical models built during the development process in order to define the design of the thermal control subsystem and to guarantee the correct temperature ranges even in worst case scenarios	Surely improving health monitoring and investigation of anomalies Currently few or no practical use mainly because of the amount of data needed	Currently Approach 2.a (Health monitoring) more often used during operations Approach 3.c recommended as well as (very promising solution to further improve health monitoring and decision process) Can be exploited also by thermal engineers to refine and update the parameters taken in their models
Rotary actuators mechanisms	Currently more often used during operations Can be not applicable if TM sampling not adapted or if interesting TMs not	Evaluated and useful only for the same design on the same orbit (very good for constellation, but less for single missions)	Approach leading to more accurate reliability figures but limited applicability because of the complexity and amount of data needed to apply this method	Physics of failure good for new applications but to be focused on dominant failure modes in order to limit the complexity	Not evaluated yet and few data are usually available (Can be not really feasible)	Currently Approach 2.a (Health monitoring) more often used during operations A combined approach is recommended for best decision (although new, promising since incorporating the benefits of model prognosis,

Spacecraft unit	Approach 2.a: Health monitoring	Approach 2.b: REX and Bayesian techniques	Approach 3.a: Prognostic based on stochastic models	Approach 3.b: Model-based prognostic	Approach 3.c: Prognostic based on data trends	Conclusions and recommendations (per orbit or type of mission, if applicable)
	available (as in several current missions)			Can be difficult to validate the model because of the lack of data		random failures on design and MMOD, and the possibility to have Bayesian updating) Generally, need of larger or more accurate data to apply the approaches Additional monitoring probably needed in some cases
Other electronics units	Currently more often used during operations Useful to check the correct behaviour and performance Specific actions needed in case of anomalies	Useful to derive less pessimistic failure rate for the Avionic units but not useful for the decision on life extension or EOL disposal	Lower benefits expected, especially because wear out phenomena not so evident, or at least severe	Promising on radiation drifts but to be validated with real WCA in order to conclude on its validity / interest	Surely improving health monitoring and investigation of anomalies Currently few/no practical use mainly because of the amount of data needed	Currently Approach 2.a (Health monitoring) often used during operations Approach 3c recommended as well (very promising solution to further improve health monitoring and decision process) The evaluation of the lifetime of electronics units because of radiation effects (TID) to be further evaluated on real cases

Annex G
Guidelines for missions outside the protected regions

G.1 Highly Eccentric orbits

Missions in Highly Eccentric Orbits (HEO) fall under the category of Earth's orbits and, depending on the perigee altitude, in the category of near-Earth orbit. In addition, given the conventions on the orbital regions in ESSB-ST-U-007 [RD02], a spacecraft on a HEO with perigee altitude below 2000 km is considered to be operating in LEO or crossing LEO if active.

Due to the effect of perturbations, spacecraft in HEO can cross the protected regions, and, for some cases, uncontrolled re-entries can also occur. For these reasons, disposal actions can be foreseen to limit interference with the protected regions [RD081] [RD082] and to limit the casualty risk on ground due to re-entry [RD083]; alternatively, the missions can be designed in such a way that, by selection of the orbit, it can be demonstrated that no crossing with the protected regions and with the orbits of known constellations is expected during operations and for 100 years after the end of life.

Specifically for what concerns the assessment of re-entry conditions, thanks to the dominant effect of lunisolar perturbations (with respect to atmospheric drag) HEO trajectories up to re-entry are more predictable and it is usually possible to target a certain re-entry latitude by changing the geometry of the orbital evolution at the re-entry epoch, e.g. by placing the apogee over the Southern Hemisphere to reduce the chance of re-entry over populated areas. Similarly, the re-entry of the Salsa satellite from the Cluster-II mission [RD084] has shown the feasibility of targeting also the re-entry longitude for further reduction of the casualty risk [RD074]. Such considerations on the re-entry location for missions in HEO orbit hold only in case that no circularisation of the orbit occurs before re-entry as, in that case, the footprint of the surviving fragments can become significantly larger and affected by a large degree of uncertainty, especially in the case of repeated perigee passes with altitude below 150 km, where the uncertainty on the atmosphere properties is high.

In terms of simulation tools, if and only if it is demonstrated that circularisation does not take place, SARA tool from DRAMA can be run with the method *Latitude-band-limited* method [RD078], meant specifically for re-entry from HEO, where the impact locations for the surviving fragments are used with 1D population data (i.e. as a function of the latitude only). If circularisation occurs, then the circular orbit method is used also for re-entries from HEOs.

For what concerns the execution of a probabilistic assessment of the casualty risk, once a baseline manoeuvre strategy is defined, an assessment of its robustness can be performed by perturbing the solution by applying stochastic accelerations. This can be done to consider

- Typical levels of uncertainty associated with the orbit determination process;
- Specific failure modes that can results in spurious Delta-v.

as shown in [RD074].

G.2 Sun-Earth Libration points 1, 2, and 3 orbits

Orbits at Sun-Earth libration points fall in the definition of Earth orbit in ESSB-ST-U-007 [RD02], so all requirements applicable to missions crossing or operating in Earth Orbit are applicable to this class of missions (e.g. limitation of the generation of mission related objects, passivation capability, trackability, identification, use of standard formats for the ground segment products, disposal and probability of successful disposal, re-entry safety). In particular, 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.

Historically, the driving requirements are the ones related to demonstration of the compliance with the casualty risk threshold, given that the space systems in orbits around SEL-1, 2, 3 are typically large and embarking components difficult to demise (e.g. optical units).

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 [RD085]. 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 probability to return to Earth orbits, which can be conservatively assumed to be 0,50.

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), considering:
        (a) orbit propagation with stochastic simulations (e.g. Monte Carlo, Annex H).
        (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 Annex H;
    3. Assess the re-entry casualty risk to ensure the compliance with the re-entry safety requirements (ESSB-ST-U-004 [RD03]), if re-entry is planned or possible.

For what concerns the assessment in point 1, the statistical analysis contains the following tasks:

- Distribute 20,000 points along the operational orbit evenly in time from the first considered epoch of the disposal manoeuvre to the last one. Each point represents a different initial state for the statistical analysis.
- For each of the 20,000 orbital states, apply the desired Delta-v increment along the unstable direction of the libration point orbit away from the Sun. This creates 20,000 samples that are escaping into a heliocentric trajectory.
- These 20,000 orbital states form the basis of the Monte Carlo analysis. Each of these orbital states are propagated for 100 years or up to impact with either the Moon or the Earth. See A.2.18 for the recommended analysis tools.
- The above steps are repeated for disposal manoeuvres of different magnitudes.

This analysis results in an Earth-return probability as a function of the epoch of the disposal manoeuvre and the magnitude of the disposal manoeuvres. This allows selection of the disposal manoeuvre epoch and manoeuvre magnitude that meets the required threshold.

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 maximise the resources (propellant mass, Delta-v) allocated for the disposal manoeuvres to a stable orbit with no return to bounded Earth orbits.

c. To optimise the Delta-v budget allocation from lessons learnt in the operations of similar missions in order to possibly increase the allocation for disposal manoeuvres, 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 3σ-trajectory accuracy (where a Delta-v of about 5 m/s is typically used), and for station keeping (where a Delta-v of about 1 m/s per year is typically used).

G.3 Sun-Earth Libration points 4 and 5 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. Therefore, with respect to the case of missions at the SEL-1, 2, 3, re-entry is no longer a concern once a spacecraft is operating in SEL-4, or 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 in orbital regions of possible interest (e.g. for science).

The presence in SEL-4 and 5 orbits of space systems can be beneficially minimised by performing disposal manoeuvres. A Delta-v allocation of about 42 m/s allows 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) 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).

G.4 Launchers with insertion into an escape trajectory

Escape orbits fall in the definition of Earth orbit in ESSB-ST-U-007 [RD02], so all requirements applicable to missions crossing Earth Orbit are applicable to this class of missions (e.g. limitation of the generation of mission related objects, passivation capability, trackability, disposal and probability of successful disposal, re-entry safety), with specific considerations on requirements related to launch elements (e.g. on the collision probability after release).

The verification of compliance is based on:

a. Analysis, to:

   1. Assess the probability of successful disposal, i.e. 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), considering relevant uncertainties as mentioned in Annex H;
   2. Assess the re-entry casualty risk to ensure the compliance with the re-entry safety requirements (ESSB-ST-U-004 [RD03]), considering also degraded cases.

G.5 Interplanetary missions injected into an escape trajectory

The applicable requirements are the ones related to re-entry (4.5). It is understood that the compliance with Requirement 5.5.b: Re-entry casualty risk threshold in 4.5.2 for this class of missions can be achieved with strategies different from the point 1 and 2 listed in the requirement 5.5b., such as heliocentric disposal. For Requirement 5.5.c: Re-entry casualty risk – probabilistic assessment in 4.5.3, the recommended verification method is to perform an analysis to assess the probability of no revisit closer than 1,5 million km to Earth (or negligible probability to interfere with the Earth Sphere of Influence) within 100 years after the end of life, considering common sources of uncertainty such as cross-sectional area, disposal epoch, state dispersion (Annex H).

Passivation of the spacecraft at the of life and the availability of passivation functions (e.g. in case of failure) is always recommended.

G.6 Earth fly-by

Spacecraft in Earth fly-bys cross Earth orbit according to [ESSB-ST-U-007 [RD02]]. While most requirements can have a very limited impact because of the short duration of the crossing phase, it is important to consider the requirements related to collision risk management as fly-by trajectories are screened for potential close approaches with other spacecraft and space debris objects. For those requirements, the verification of compliance is based on:

a. Review-of-design, to:

   1. Check reaction thresholds based on collision probability (Requirement 5.3.3.3.a: Acceptable collision probability threshold in 4.3.17, Requirement 5.3.3.3.e: Collision probability computation during operation in 4.3.21) and corresponding actions (Requirement 5.3.3.3.o: CAM coordination in 4.3.29). In case the design of a possible CAM be not feasible during flyby operations, multiple trajectory scenarios can be prepared in advance. Such scenarios are designed according to the

estimated uncertainty of the spacecraft, derived from own computations, and the expected uncertainty of potential chasers, as computed e.g. statistical analyses of available data (e.g. CDMs);

3. Verify that operational procedures are in place for the distribution of planned ephemerides and other relevant information (Requirements 5.3.3.3.k-m: Collision avoidance procedure information in 4.3.27). It is important that shared data is compliant with formats commonly used for satellites in Earth protected regions, in terms of time scale, reference frame, and ephemeris span;
4. Ensure that the spacecraft can be tracked by a space surveillance system, which supports the ground segment (Requirement 5.3.3.5.a: Trackability in 4.3.36, Requirement 5.3.3.5.b: Space surveillance segment in 4.3.37, Requirement 5.3.3.5.f: Ephemerides frequency in);
5. Confirm that the ground segment can generate predicted ephemerides (Requirement 5.3.3.5.h: Ephemerides forecast) and quantify the associated position and velocity accuracy while in proximity of the Earth (Requirement 5.3.3.5.c: State vector quantification frequency in 4.3.38);
6. Attest that the ground segment can generate and process products with standard formats (Requirement 5.3.3.5.i: CCSDS format in 4.3.44).

b. Analysis, to:

1. Estimate the probability of conjunctions and its sensitivity to design parameters, if relevant Screenings against high-fidelity catalogues of objects can be done in advance, to have an a priori assessment of the possible conjunction events;
2. Simulate relevant conjunction scenarios.

More detailed guidelines can be found under the specific requirements.

Annex H
Stochastic approach

A deterministic analysis of a single trajectory is not always sufficiently representative to verify the compliance with the Space Debris Mitigation 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 (e.g. disposal from the MEO region), 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. 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. The relevant parameters include:

a. Solar cycle and geomagnetic activity, e.g. with sensitivity as for orbit propagation analysis (Annex A).

b. Cross-sectional area (can be different for collision risk and orbit perturbation analysis), 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 values is enough to define the envelope of evolution).

c. Reflectivity coefficient, e.g. with distribution between the minimum and maximum value, for regions where the solar radiation pressure is a high relative magnitude perturbing force, e.g. for high lunar orbits

d. 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.

e. Separation dispersion, e.g. sampling of the covariance matrix depending on the launch vehicle trajectory.

f. 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.

g. Initial state dispersion, e.g. sampling the knowledge dispersion matrix of the initial state due to orbit determination uncertainties or random failures.

h. Uncertainty in the force models used, e.g. sampling with different values for the spherical harmonic terms of a gravitational field that represent their uncertainty.

In cases where 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:

$$\hat{p} - z_{1-\alpha/2}\sqrt{\frac{1}{n}\hat{p}(1-\hat{p})} \leq p \leq \hat{p} + z_{1-\alpha/2}\sqrt{\frac{1}{n}\hat{p}(1-\hat{p})} \qquad \text{[H-1]}$$

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:

$$\frac{\hat{p} + \frac{z^2{}_{\alpha/2}}{2n} - z_{\alpha/2}\sqrt{\frac{\hat{p}(1-\hat{p})}{n} + \frac{z^2{}_{\alpha/2}}{4n^2}}}{1 + \frac{z^2{}_{\alpha/2}}{n}} \leq p \leq \frac{\hat{p} + \frac{z^2{}_{\alpha/2}}{2n} + z_{\alpha/2}\sqrt{\frac{\hat{p}(1-\hat{p})}{n} + \frac{z^2{}_{\alpha/2}}{4n^2}}}{1 + \frac{z^2{}_{\alpha/2}}{n}} \qquad \text{[H-2]}$$

Equation [H-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 $n > 50$):

$$\hat{p} - \left(\frac{1}{2n} + z_{\alpha/2}\sqrt{\frac{\hat{p}(1-\hat{p})}{n}}\right) \leq p \leq \hat{p} + \left(\frac{1}{2n} + z_{\alpha/2}\sqrt{\frac{\hat{p}(1-\hat{p})}{n}}\right) \qquad \text{[H-3]}$$

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.

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).