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Space systems — Survivability of unmanned spacecraft against space debris and meteoroid impacts for the purpose of space debris mitigation

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Foreword

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The procedures used to develop this document and those intended for its further maintenance are described in the ISO/IEC Directives, Part 1. In particular, the different approval criteria needed for the different types of ISO document should be noted. This document was drafted in accordance with the editorial rules of the ISO/IEC Directives, Part 2 (see www.iso.org/directives).

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This document was prepared by Technical Committee ISO/TC 20, *Aircraft and space vehicles*, Subcommittee SC 14, *Space systems and operations*.

This second edition cancels and replaces the first edition (ISO 16126:2014), which has been technically revised.

The main changes are as follows:

— the provision of new impact risk analysis requirements and procedures aimed specifically at satisfying the high-level impact risk requirements defined in the top-level International Standard on space debris mitigation, ISO 24113;

— the provision of new informative annexes to assist in the implementation of the impact risk analysis procedures.

Any feedback or questions on this document should be directed to the user’s national standards body. A complete listing of these bodies can be found at www.iso.org/members.html.

Introduction

The purpose of this document is to help satisfy two of the high-level requirements defined in the top-level International Standard on space debris mitigation, ISO 24113. Specifically, this document aims to maximise the survival of critical equipment required to perform post-mission disposal of an unmanned spacecraft, and to limit the possibility of an impact-induced break-up of the spacecraft. The analysis procedures in this document are consistent with those defined in References [1] and [2].

In principle, this document can also be used to assess the impact survivability of an unmanned spacecraft in support of other mission objectives. However, careful adaptation of the document can be necessary if put to such use.

This document is part of a set of International Standards that collectively aim to reduce the growth of space debris by ensuring that spacecraft are designed, operated, and disposed of in a manner that prevents them from generating space debris throughout their orbital lifetime. All of the primary space debris mitigation requirements are contained in ISO 24113. The remaining International Standards, of which this is one, provide supporting methods and procedures to enable compliance with the primary requirements.

Space systems — Survivability of unmanned spacecraft against space debris and meteoroid impacts for the purpose of space debris mitigation

1 Scope

This document defines requirements and procedures for analysing the risk that an unmanned spacecraft fails as a result of a space debris or meteoroid impact.

2 Normative references

The following documents are referred to in the text in such a way that some or all of their content constitutes requirements of this document. For dated references, only the edition cited applies. For undated references, the latest edition of the referenced document (including any amendments) applies.

ISO 24113, *Space systems — Space debris mitigation requirements*

3 Terms and definitions

For the purposes of this document, the terms and definitions given in ISO 24113 and the following apply.

ISO and IEC maintain terminology databases for use in standardization at the following addresses:

— IEC Electropedia: available at https://www.electropedia.org/

— ISO Online browsing platform: available at https://www.iso.org/obp

3.1
ballistic limit
threshold of impact-induced failure of a structure

Note 1 to entry: A common failure threshold is the critical size of an impacting particle at which perforation occurs. However, depending on the characteristics of the item being hit, failure thresholds other than perforation are also possible.

3.2
catastrophic break-up
event that completely destroys an object and generates space debris

3.3
critical equipment
item(s) on a spacecraft whose failure would prevent the completion of one or more essential functions, such as post-mission disposal

3.4
high-energy SD/M
space debris or meteoroid object whose impact kinetic energy exceeds the threshold necessary to cause the *catastrophic break-up* (3.2) of a spacecraft

Note 1 to entry: The threshold is usually expressed in terms of the kinetic energy of an SD/M impact relative to the mass of the spacecraft, i.e. an energy-to-mass ratio (EMR). A typical value for the EMR threshold is 40 J/g.

3.5
project lifecycle
phases of a project from mission analysis through to disposal

Note 1 to entry: The phases of a project are summarised in Table 1. A more detailed description can be found in ISO 14300-1[3].

Table 1 — Summary of the phases of a project

Phase	Description
Pre-phase A	Mission analysis
Phase A	Feasibility
Phase B	Definition
Phase C	Development
Phase D	Production
Phase E	Utilization
Phase F	Disposal

3.6
small SD/M
space debris or meteoroid object whose size does not exceed one centimetre in its largest dimension

Note 1 to entry: This threshold is defined for two reasons. First, in impact risk analysis models it is difficult to characterise accurately the penetrative damage inside a spacecraft from an SD/M impactor larger than one centimetre in size. Second, it is difficult for current shielding technology to protect a spacecraft against an SD/M impactor larger than one centimetre in size.

4 Symbols and abbreviated terms

4.1 Symbols

A power law term

B power law term

C speed of sound of the material in a target wall (km/s)

D constant value

d_c critical diameter of an impactor at the threshold of failure of a wall, panel or shield (cm)

d_{LF} diameter of largest fragment in an in-line cloud ejection cone (cm)

d_p diameter of impacting particle or projectile (cm)

G constant value

H Brinell hardness of the material in a target wall

K factor that combines the material properties of a target

K_{CFRP} factor that combines the material properties of a CFRP target

K_f factor that distinguishes between different types of impact damage failure

K_1 factor that combines the material properties of a target

K_2 factor that combines the material properties of a target

K_3 factor that combines the material properties of a target

K_{3D} factor that combines the material properties of a target

K_{3S} factor that combines the material properties of a target

K_4 factor that combines the material properties of a target

k factor that distinguishes between different types of impact damage failure

L_1 adjustable coefficient to separate the ruptured and non-ruptured data points in an RLE

L_2 adjustable coefficient to separate the ruptured and non-ruptured data points in an RLE

L_3 adjustable coefficient to separate the ruptured and non-ruptured data points in an RLE

m_p mass of impacting particle or projectile (g)

p_{int} internal pressure in a pressurised tank (ksi)[1)]

p_0 constant value

r_o outer radius of a pressurised tank (cm)

S stand-off distance between the outer bumper of a shield and a back wall (cm)

t_{al} thickness of aluminium wall (cm)

t_b thickness of bumper shield (cm)

t_{CFRP} thickness of CFRP wall (cm)

t_{comp} thickness of composite material in a COPV (cm)

t_f thickness of foam core in sandwich panel (cm)

t_{hc} total thickness of honeycomb cell walls perforated by a projectile impacting at angle θ (cm)

t_{lin} thickness of liner material in a COPV (cm)

t_{tot} total thickness of cylindrical portion of COPV material overwrap, i.e. $t_{comp} + t_{liner}$ (cm)

t_w thickness of a single wall, or thickness of back wall in a multiple wall configuration (cm)

v impact velocity (km/s)

v_h high velocity limit for transition from fragmentation to hypervelocity regime (km/s)

v_l low velocity limit for transition from ballistic to fragmentation regime (km/s)

v_{LF} velocity of largest fragment in an in-line cloud ejection cone (km/s)

v_n normal component of impact velocity, i.e. $v \cos\theta$ (km/s)

α weighting coefficient

β weighting coefficient

γ weighting coefficient

1) 1 ksi = 6,895 MPa.

δ weighting coefficient

ζ_1 weighting coefficient

ζ_2 weighting coefficient

η weighting coefficient

θ impact angle with respect to surface normal (degrees)

κ weighting coefficient

λ weighting coefficient

μ weighting coefficient

ξ weighting coefficient

ρ_A areal density of one or more layers of material (g/cm^2)

$\rho_{A,f}$ areal density of foam core in sandwich panel (g/cm^2)

ρ_{al} density of aluminium wall (g/cm^3)

ρ_b density of bumper shield (g/cm^3)

ρ_{CFRP} density of CFRP wall (g/cm^3)

ρ_{comp} density of the composite material in a COPV (g/cm^3)

ρ_f density of foam core in sandwich panel (g/cm^3)

ρ_{hc} density of honeycomb core in a sandwich panel (g/cm^3)

ρ_p density of impacting particle or projectile (g/cm^3)

ρ_w density of a single wall, or density of back wall in a multiple wall configuration (g/cm^3)

σ_h hoop stress of a pressurised tank, i.e. $p_{int}\, r_o / t_{tot}$ (ksi)

σ_u ultimate tensile stress of the material in a pressurised tank (ksi)

$\sigma_{u,\,comp}$ unidirectional ultimate stress of the composite material in a COPV (ksi)

$\sigma_{u,\,lin}$ ultimate stress of the liner material in a COPV (MPa)

$\sigma_{y,\,lin}$ yield stress of the liner material in a COPV (MPa)

$\sigma_{y,\,w}$ yield stress of material in a single wall or the back wall in a multiple wall configuration (ksi)

ϕ angle between central axis of in-line cloud ejection cone and surface normal (degrees)

ψ spread angle of in-line cloud ejection cone (degrees)

4.2 Abbreviated terms

AIT	assembly integration and test
BLE	ballistic limit equation
CFRP	carbon fibre reinforced plastic
COPV	composite overwrapped pressure vessel
CVCM	collected volatile condensable material
EMR	energy-to-mass ratio
FTA	fault tree analysis
GEO	geostationary orbit
GVF	geometric view factor
HVI	hypervelocity impact
IADC	Inter-Agency Space Debris Coordination Committee
LEO	low Earth orbit
MLI	multi-layer insulation
MVF	modified view factor
REACH	registration, evaluation, authorisation and restriction of chemicals
RLE	rupture limit equation
RML	recovery mass loss
SD/M	space debris/meteoroid(s)
STENVI	standard environment interface
TT&C	telemetry, tracking, and command

5 Requirements for impact risk analysis

5.1 General

5.1.1 The top-level International Standard on space debris mitigation, ISO 24113, specifies two high-level SD/M impact risk assessment requirements that aim to:

a) ensure the post-mission disposal of a spacecraft;

b) limit the probability that a spacecraft experiences an SD/M impact-induced break-up before its end of life.

5.1.2 To satisfy these high-level requirements, the following two distinct analysis cases can be defined:

a) case 1: an analysis of the probability of SD/M impact-induced failure of the spacecraft, where failure is defined by an inability to perform successful disposal;

b) case 2: an analysis of the probability of SD/M impact-induced failure of the spacecraft, where failure is defined by a catastrophic break-up.

5.1.3 The analysis in case 2 can be subdivided by analysing the following two types of catastrophic break-up separately:

a) case 2a: a catastrophic break-up caused by the impact of a small SD/M on an equipment item containing a large amount of stored energy, such as a pressurised vessel;

b) case 2b: a catastrophic break-up caused by the impact of a high-energy SD/M on the spacecraft.

5.1.4 Detailed requirements to support the implementation of these analyses are provided in 5.2 and 5.3.

5.2 Failure probability thresholds

5.2.1 For case 1, during the design of a spacecraft for which a disposal manoeuvre has been planned, a threshold shall be specified for the probability that an SD/M impact prevents the disposal from being successful.

5.2.2 For case 2a, during the definition of a mission and the design of a spacecraft, a threshold shall be specified for the probability that the spacecraft experiences a catastrophic break-up before its end of life as a result of a small SD/M impacting an equipment item containing a large amount of stored energy.

5.2.3 For case 2b, during the definition of a mission and the design of a spacecraft, a threshold shall be specified for the probability that the spacecraft experiences a catastrophic break-up before its end of life as a result of a high-energy SD/M impacting the spacecraft.

NOTE The threshold in case 2b can be specified taking into account the significance of the mission, the mission requirements, and the expected severity of adverse effects on the orbital environment if a break-up occurs.

5.2.4 The failure probability thresholds shall be set by the approving agent responsible for requirements in the space debris mitigation plan.

NOTE Each of the probability thresholds can be expressed as a maximum value for the probability of failure, $P_{F\,max}$.

5.3 Failure probability analysis

5.3.1 To satisfy each of the failure probability thresholds in 5.2, an analysis shall be performed in which the corresponding probability of failure, P_F, is calculated and compared with the specified maximum value, $P_{F\,max}$.

5.3.2 If $P_F > P_{F\,max}$, then measures shall be taken to reduce P_F so that it is below the maximum value.

5.3.3 The analysis and reduction of P_F for each of the analysis cases shall follow a clearly defined procedure.

NOTE An example procedure for analysis case 1 is described in Clause 6. Example procedures for analysis cases 2a and 2b are described in Clause 7. For some types of spacecraft, such as small ones or those operating in GEO, simplified procedures for analysis cases 2a and 2b can be considered if the impact risks are sufficiently low.

5.3.4 The results of the impact risk analysis, the methodology used, and any assumptions made shall be approved by the approving agent of the spacecraft.

6 Impact risk analysis procedure for case 1

6.1 The consideration of SD/M at sub-centimetre sizes is particularly important when analysing the impact risks that can prevent the successful disposal of a spacecraft. An analysis of such impactors:

a) enables the probability of impact-induced failure of the spacecraft to be calculated, where failure is defined by not being able to perform a successful disposal;

b) allows any impact vulnerabilities in the spacecraft design to be identified;

c) guides the implementation of appropriate levels of impact protection in the spacecraft.

6.2 A procedure for performing a detailed analysis of the probability that a spacecraft cannot complete a successful post-mission disposal, as a result of impacts from small SD/M, is shown in Figure 1. The procedure is designed to be followed in phases B and C of the spacecraft project lifecycle.

NOTE It is also possible to perform a simple impact risk analysis during phase A for the purpose of defining key aspects of the proposed design of the spacecraft, such as its geometric characteristics. A procedure for such an analysis is described in Annex A.

6.3 During the preliminary design in phase B, the aim of an impact risk analysis is to be sufficiently detailed that it can suggest and enable efficient protection solutions which can otherwise be impossible during the final stages of development.

6.4 By contrast, in the late development stages a redesign of the general spacecraft architecture is not usually possible due to the complex subsystem interrelationships that are characteristic of spacecraft. Thus, during phase C, the main goal is to refine the impact risk analysis of the spacecraft and identify areas of its design where additional shielding is necessary.

Figure 1 illustrates the key steps in the procedure and the flow of information between the steps.

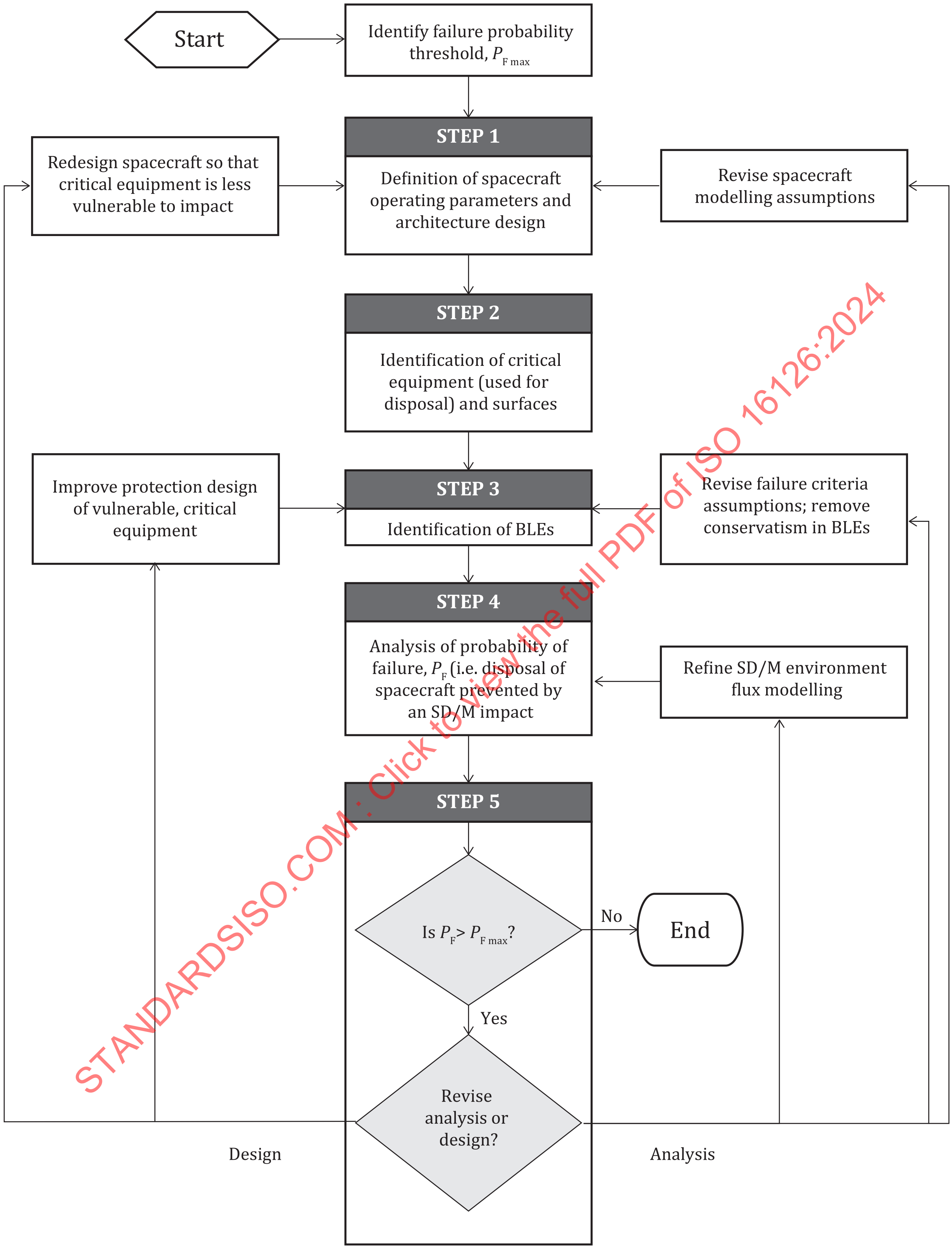


Figure 1 — Impact risk analysis procedure for case 1

Table 2 provides a more detailed description of each step in the procedure.

Table 2 — Impact risk analysis procedure for case 1

Step	Description	Further information
1	**Definition of spacecraft operating parameters and architecture design**	
1.1	Define the operating parameters of the spacecraft, such as its operational orbits and attitude orientation relative to the direction of motion.	
1.2	Define the architecture design of the spacecraft, such as its geometric characteristics and dimensions, the layout of all equipment, and the material properties of all surfaces, including any shielding.	B.2.2.2
2	**Identification of critical equipment**	
2.1	Identify every equipment item on the spacecraft that contributes to post-mission disposal.	
2.2	For each equipment item determine its redundancy, impact damage modes and any other design aspects that are pertinent, such as operating pressures.	
2.3	Use a reliability analysis technique, such as fault tree analysis or failure modes and effects analysis, to identify the system-level consequences that result when each of the equipment items is damaged by impact.	
2.4	Identify the critical equipment, i.e. those items which, when damaged by impact, can prevent post-mission disposal.	
2.5	On each critical equipment item, identify the critical surfaces, i.e. those surfaces which, when damaged by impact, cause the item to fail.	
3	**Identification of BLEs**	
3.1	Identify existing BLEs that are suitable for determining the ballistic limit of each surface or combination of surfaces on the spacecraft, especially the critical equipment.	C.3.1 to C.3.8, C.5.1 to C.5.3
3.2	If a suitable BLE cannot be identified for a particular surface or combination of surfaces, then perform a set of HVI tests, as well as hydrocode simulations if SD/M environment models indicate significant flux at velocities higher than the maximum velocity in the HVI tests, to adapt an existing BLE or derive a new one.	C.4.1 to C.4.3, C.5.1 to C.5.3
3.3	For each surface or combination of surfaces on the spacecraft, especially the critical equipment items, define an impact failure criterion, such as perforation.	B.2.2.3
4	**Analysis of probability of impact-induced failure**	
4.1	Select an SD/M impact risk analysis model that can evaluate the probability of impact-induced failure of a spacecraft.	B.3.1 to B.3.3
4.2	Select an SD/M environment model that is suitable for use with the chosen impact risk analysis model, and use it to produce a data set of directional impact fluxes on the spacecraft over the life of its normal operations.	B.2.2.4, B.2.2.5
4.3	Use the chosen SD/M impact risk analysis model to compute the impact and perforation fluxes on external surfaces of the spacecraft.	B.2.2.5, B.2.2.6
4.4	Use the chosen SD/M impact risk analysis model to compute the probabilities of impact and perforation for external surfaces of the spacecraft.	B.2.2.7
4.5	Use the chosen SD/M impact risk analysis model to compute the perforation fluxes on the surfaces of equipment inside the spacecraft.	B.2.2.8
4.6	Use the chosen SD/M impact risk analysis model to calculate P_F, i.e. the probability that one or more of the selected critical equipment items fail during the normal operations of the spacecraft as a result of an SD/M impact, thereby preventing the successful disposal of the spacecraft.	B.2.2.9
5	**Revision of the analysis or design**	
5.1	If $P_F > P_{F\,max}$, revise aspects of the analysis or design by considering the following (in order of preference): a) modify the analysis assumptions in terms of failure criteria or spacecraft modelling;	B.2.2.10

Table 2 *(continued)*

Step	Description	Further information
	b) compare the flux values obtained from the selected SD/M environment models with those from other models to characterize the differences due to inherent uncertainties in the models and, if appropriate, select alternative models for the analysis;	Reference [4]
	c) perform additional impact testing and, if necessary, hydrocode modelling to remove engineering conservatism in the BLEs;	C.4.1 to C.4.3
	d) identify those areas of the spacecraft design which are the greatest contributors to the spacecraft impact failure probability, and systematically apply one or more shielding modifications;	Annex D
	e) examine alternatives for designing the spacecraft so that it can be orientated in such a way that its most vulnerable, critical equipment does not face the direction of greatest impact flux.	Annex D

7 Impact risk analysis procedures for case 2

7.1 General

7.1.1 Impact risk analysis procedures for cases 2a and 2b are provided in 7.2 and 7.3, respectively. The procedures are designed to be followed in phases B and C of the spacecraft project lifecycle. Preliminary analysis of case 2b during phase A can also aid selection of the operational orbit of the spacecraft.

7.1.2 The overall probability of impact-induced failure for case 2 is calculated by combining the probability of impact-induced failure for case 2a and case 2b.

7.2 Case 2a

7.2.1 The consideration of SD/M at sub-centimetre sizes is particularly important when analysing the impact risks that can cause a catastrophic break-up. An analysis of such impactors:

a) enables the probability of impact-induced failure of a spacecraft to be calculated, where failure is defined by a catastrophic break-up;

b) allows any impact vulnerabilities to be identified in the design and location of spacecraft equipment containing large amounts of stored energy;

c) guides the implementation of appropriate levels of impact protection for spacecraft equipment containing large amounts of stored energy.

7.2.2 A procedure for performing a detailed analysis of the probability that a spacecraft fails as a result of a catastrophic break-up caused by the impact of a small SD/M on an equipment item containing a large amount of stored energy, is shown in Figure 2.

7.2.3 Since the impact risk analysis for case 2a can be thought of as a subset of the analysis for case 1, the steps in the procedure are almost identical to those described in Clause 6. Table 3 provides a more detailed description of each step in the procedure.

7.2.4 Alternatively, a much simplified version of the procedure, which does not necessitate the use of an SD/M impact risk analysis model, can be implemented as follows.

a) Select and use an SD/M environment model to calculate the most likely impact velocity and angle for an SD/M particle on a spacecraft in its particular orbit.

b) Select and use a BLE, with the information in a), to calculate the diameter of an SD/M particle that is most likely to cause the break-up of an equipment item containing a large amount of stored energy. For example, in the case of a metallic pressurised vessel, the RLE in C.3.7 can be used together with information on the vessel design and its operating pressure.

c) Use the chosen SD/M environment model to calculate the impact flux of SD/M particles, with diameter as calculated in b), on the spacecraft.

d) Use the equations in B.2.2.7, with the flux information in c), to calculate the probability that the equipment item breaks up.

e) In the case of an equipment item depleting its contents, repeat steps b) to d) to evaluate the effect of pressure change.

f) Repeat steps b) to e) for all equipment items containing a large amount of stored energy, and calculate the overall probability that the spacecraft fails as a result of a catastrophic break-up caused by the impact of a small SD/M.

NOTE This procedure provides a quick but approximate result. It can be useful when there is a need to perform multiple assessments to understand the effect of operational parameters changing over time, such as the pressure inside a vessel.

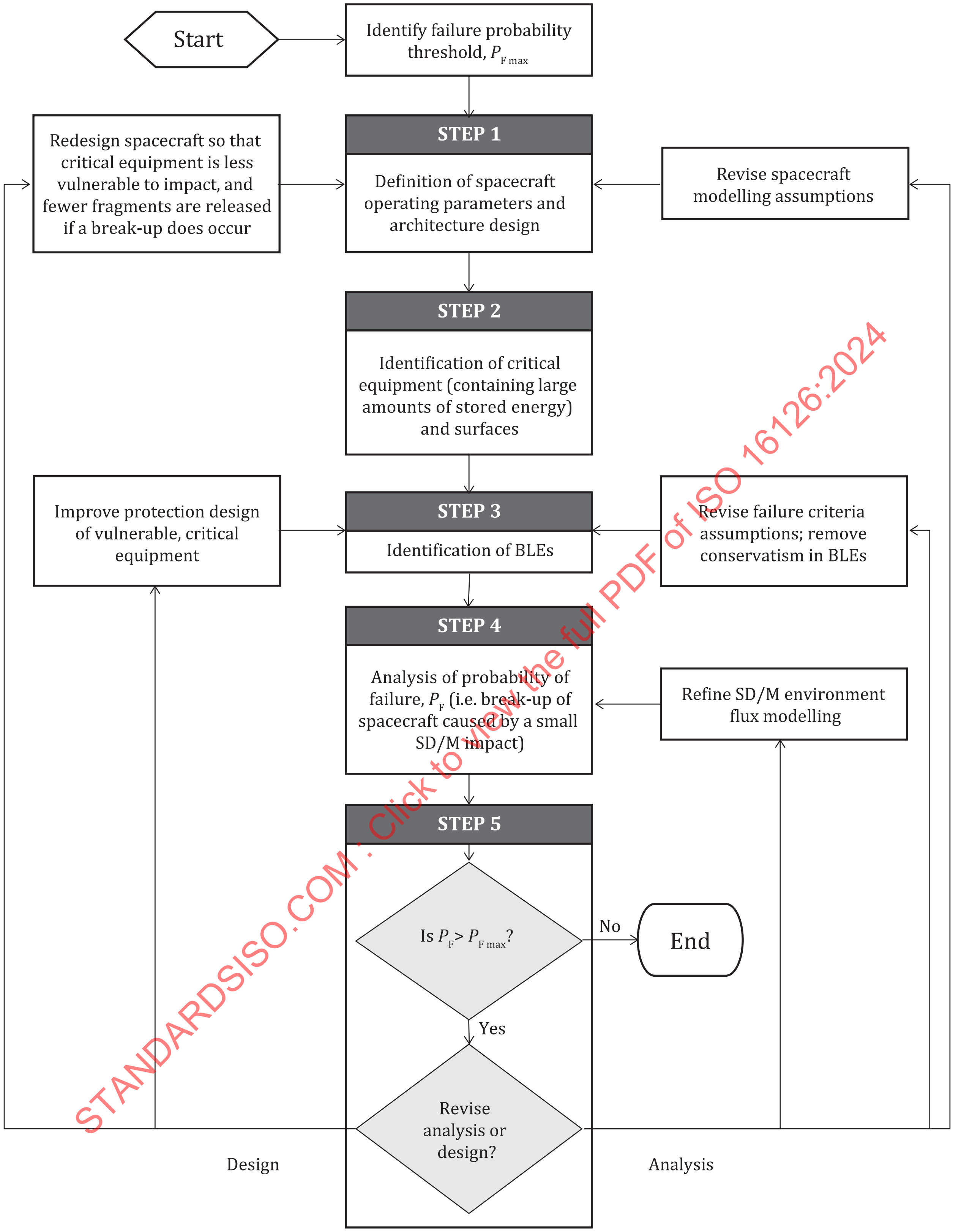


Figure 2 — Impact risk analysis procedure for case 2a

Table 3 — Impact risk analysis procedure for case 2a

Step	Description	Further information
1	**Definition of spacecraft operating parameters and architecture design**	
1.1	Define the operating parameters of the spacecraft, such as its operational orbits and attitude orientation relative to the direction of motion.	
1.2	Define the architecture design of the spacecraft, such as its geometric characteristics and dimensions, the layout of all equipment, and the material properties of all surfaces, including any shielding.	B.2.2.2
2	**Identification of critical equipment**	
2.1	Identify every equipment item on the spacecraft that contains a large amount of stored energy, including pressure vessels, high-pressure propellant tanks and high-pressure batteries.	
2.2	For each equipment item determine its redundancy, impact damage modes and any other design aspects that are pertinent, such as operating pressures.	
2.3	Use a reliability analysis technique, such as fault tree analysis or failure modes and effects analysis, to identify the system-level consequences that result when each of the equipment items is damaged by impact.	
2.4	Identify the critical equipment, i.e. those items which, when damaged by impact, would rupture causing a catastrophic break-up and, in so doing, make a conservative assumption that the surrounding spacecraft structure will not be able to contain the fragments and content of the ruptured items.	
2.5	On each critical equipment item, identify the critical surfaces, i.e. those surfaces which, when damaged by impact, cause the item to break-up catastrophically.	
3	**Identification of BLEs**	
3.1	Identify existing BLEs that are suitable for determining the ballistic limit of each surface or combination of surfaces on the spacecraft, especially the critical equipment.	C.3.1 to C.3.8, C.5.1 to C.5.3
3.2	If a suitable BLE cannot be identified for a particular surface or combination of surfaces, then perform a set of HVI tests, as well as hydrocode simulations if SD/M environment models indicate significant flux at velocities higher than the maximum velocity in the HVI tests, to adapt an existing BLE or derive a new one.	C.4.1 to C.4.3, C.5.1 to C.5.3
3.3	For each surface or combination of surfaces on the spacecraft, especially the critical equipment items, define an impact failure criterion, such as perforation or rupture.	B.2.2.3
4	**Analysis of probability of break-up due to a small SD/M impact**	
4.1	Select an SD/M impact risk analysis model that can evaluate the probability of impact-induced failure of a spacecraft.	B.3.1 to B.3.3
4.2	Select an SD/M environment model that is suitable for use with the chosen impact risk analysis model, and use it to produce a data set of directional impact fluxes on the spacecraft over the life of its normal operations.	B.2.2.4, B.2.2.5
4.3	Use the chosen SD/M impact risk analysis model to compute the impact and perforation fluxes on external surfaces of the spacecraft.	B.2.2.5, B.2.2.6
4.4	Use the chosen SD/M impact risk analysis model to compute the probabilities of impact and perforation for external surfaces of the spacecraft.	B.2.2.7
4.5	Use the chosen SD/M impact risk analysis model to compute the perforation fluxes on the surfaces of equipment inside the spacecraft.	B.2.2.8
4.6	Use the chosen SD/M impact risk analysis model to calculate P_F, i.e. the probability that one or more of the selected critical equipment items break-up catastrophically during the normal operations of the spacecraft as a result of an impact with a small SD/M.	B.2.2.9
5	**Revision of the analysis or design**	
5.1	If $P_F > P_{F\,max}$, revise aspects of the analysis or design by considering the following (in order of preference): a) modify the analysis assumptions in terms of failure criteria or spacecraft modelling;	B.2.2.10

Table 3 *(continued)*

Step	Description	Further information
	b) compare the flux values obtained from the selected SD/M environment models with those from other models to characterize the differences due to inherent uncertainties in the models and, if appropriate, select alternative models for the analysis;	Reference [4]
	c) perform additional impact testing and, if necessary, hydrocode modelling to remove engineering conservatism in the BLEs;	C.4.1 to C.4.3
	d) identify those areas of the spacecraft design which are the greatest contributors to the spacecraft impact failure probability, and systematically apply one or more shielding modifications;	Annex D
	e) examine alternatives for designing the spacecraft so that it can be orientated in such a way that its most vulnerable, critical equipment does not face the direction of greatest impact flux;	Annex D
	f) identify any aspects of the spacecraft design which can be modified to limit the release of fragments into the space environment if a catastrophic break-up occurs.	

7.3 Case 2b

7.3.1 The consideration of a high-energy SD/M is particularly important when analysing the impact risks that can cause a catastrophic break-up. An analysis of such impactors enables the probability of impact-induced failure of a spacecraft to be calculated, where failure is defined by a catastrophic break-up.

7.3.2 A procedure for performing a detailed analysis of the probability that a spacecraft fails as a result of a catastrophic break-up caused by the impact of a high-energy SD/M on the spacecraft, is shown in Figure 3.

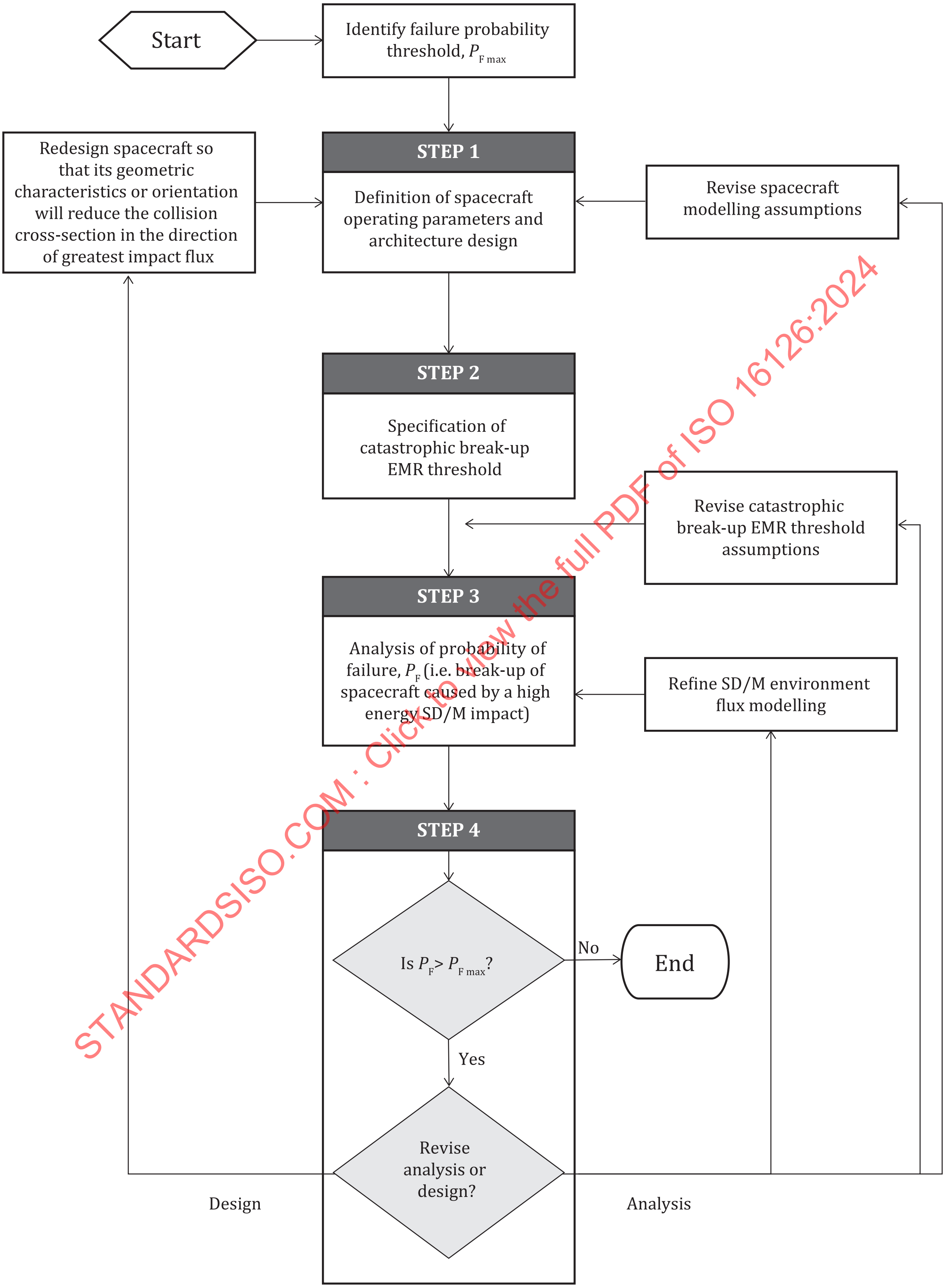


Figure 3 — Impact risk analysis procedure for case 2b

Table 4 provides a more detailed description of each step in the procedure.

Table 4 — Impact risk analysis procedure for case 2b

Step	Description
1	**Definition of spacecraft operating parameters and architecture design**
1.1	Define the operating parameters of the spacecraft, such as its operational orbits and attitude orientation relative to the direction of motion.
1.2	Define the architecture design of the spacecraft, such as its geometric characteristics and dimensions.
2	**Specification of catastrophic break-up EMR threshold**
2.1	Apply an EMR value for the threshold of an impact-induced catastrophic break-up.
3	**Analysis of probability of break-up due to a high-energy SD/M impact**
3.1	Select an SD/M impact risk analysis model that can evaluate the probability of impact-induced failure of a spacecraft.
3.2	Select an SD/M environment model that is suitable for use with the chosen impact risk analysis model, and use it to produce a data set of directional impact fluxes on the spacecraft over the life of its normal operations.
3.3	Use the chosen impact risk analysis model to calculate P_F, i.e. the probability that the spacecraft breaks up catastrophically during its normal operations as a result of an impact with a high-energy SD/M. During this analysis, if the EMR threshold is exceeded by SD/M objects of size greater than 10 cm, then these objects can be disregarded providing the spacecraft has a collision avoidance capability. Note that this is not conservative since the collision avoidance capability can also fail.
4	**Revision of the analysis or design**
4.1	If $P_F > P_{F\ max}$, revise aspects of the analysis or design by considering the following (in order of preference): a) modify the analysis assumptions in terms of catastrophic break-up threshold or spacecraft modelling; b) compare the flux values obtained from the selected SD/M environment models with those from other models, e.g. as discussed in Reference [4], to characterize the differences due to inherent uncertainties in the models and, if appropriate, select alternative models for the analysis; c) examine alternatives for designing the spacecraft in such a way that its geometric characteristics or orientation reduces the collision cross-section in the direction of greatest impact flux. d) examine alternative operational orbits with lower impact fluxes which still meet the mission requirements

Annex A
(informative)

Procedure for an impact risk analysis during phase A

For the feasibility studies in phase A of the spacecraft project lifecycle, a simple impact risk analysis can help with defining key aspects of the proposed design, such as operational orbit and spacecraft geometric characteristics.

A procedure for performing a simple impact risk analysis during phase A is listed in Table A.1. This can be used to provide a preliminary assessment of the spacecraft design with respect to case 1 and case 2.

Table A.1 — Impact risk analysis procedure during phase A

Step	Description
1	**Select an SD/M environment model and use it to compute the directional impact fluxes on the spacecraft for its proposed operational orbits until its end of life** ISO 14200 [4] provides guidance on the selection and use of suitable SD/M environment models for impact risk analysis. An example of an SD/M impact flux data file is provided in Reference [46]. This format, known as STENVI, was developed by the IADC as a standardised means of transferring flux data from SD/M environment models to impact risk analysis models. It lists the flux data in discrete bins. The fluxes can be aggregated in different ways to produce a variety of graphical plots, such as: a) the total flux of SD/M impacting the spacecraft from each azimuth and elevation direction; b) the flux of SD/M of a given size range or velocity range impacting the spacecraft from each azimuth and elevation direction.
2	**Incorporate the results of the impact flux analysis into the overall system engineering process to help define the spacecraft geometric characteristics and the approximate additional mass margins for impact protection** At this early stage in the assessment it can be necessary to treat the spacecraft as a sphere or a bounding box, if attitude laws are known. In some instances it is also possible to evaluate individual major surfaces of the spacecraft. The results of such an assessment can influence the preliminary layout of a spacecraft. For example, if there were a large flux from a particular direction, then the possibility of modifying the geometric characteristics of the spacecraft can be considered to reduce its projected area in that direction. Early analysis can also inform the choice of mission orbit by showing differences between SD/M fluxes in candidate orbits.
3	**Repeat the preceding steps if any of the proposed operational orbits of the spacecraft are changed significantly** For the purpose of an impact risk analysis, a significant orbital change can be considered as one in which the SD/M spatial density changes by at least 10 %.

Annex B
(informative)

Methods and models for analysing the impact risk from small SD/M

B.1 General

Case 1 (in Clause 6) and case 2a (in 7.2), respectively, describe impact risk analysis procedures for analysing the probability that:

a) a spacecraft is not able to complete a successful post-mission disposal as a result of impacts from small SD/M;

b) a spacecraft experiences a catastrophic break-up as a result of an impact from a small SD/M on an equipment item containing a large amount of stored energy.

This annex provides information on methods and models that can be used for these analyses.

B.2 Analysis methods

B.2.1 General

If the criterion for impact-induced failure of a spacecraft is defined as perforation of the external structure, then a logical consequence of this specification is that engineers will concentrate on applying any necessary impact protection to the external structure whilst giving little consideration to the equipment inside the spacecraft. On the face of it, this approach has a number of benefits, for example:

— the impact risk analysis is relatively simple;

— protection enhancements can be implemented quite easily by the addition of shielding to the external structure;

— it is not necessary to know anything about the design, operation or location of the underlying equipment, which can be difficult to identify early in the spacecraft design process.

For a manned spacecraft one can see that such an approach is reasonable since the perforation of a pressurised module would have serious consequences for the safety of the crew. However, the same is not necessarily true for an unmanned spacecraft. It is entirely possible that an unmanned spacecraft can survive an SD/M particle penetrating its external structure. There are several reasons why, including:

a) the resulting penetrative fragmentation cloud does not necessarily impact any sensitive or critical internal equipment;

b) if an equipment item is impacted, there is a possibility the cloud particles cannot penetrate its casing;

c) if perforation of the equipment casing occurs, there is a possibility that the resulting damage inside the equipment is insufficient to cause its failure;

d) a penetrated equipment item can sometimes survive if it has in-built redundancy;

e) if the equipment item does fail, then this does not necessarily mean termination of the mission. For example, if there is an undamaged back-up unit located elsewhere in the spacecraft then the mission can potentially survive.

Thus, by focusing protection enhancement on the external structure of an unmanned spacecraft, and ignoring the inherent robustness of the spacecraft design, it is likely that the structure will be heavier and bulkier than is strictly necessary. This suggests that, for unmanned spacecraft, a more sophisticated approach is necessary when analysing the impact risk and implementing protection, one in which damage to the spacecraft interior is considered.

B.2.2 Main steps

B.2.2.1 General

The procedures for analysing the risk to an unmanned spacecraft from small SD/M impacts, as described in case 1 and case 2a, comprise a clearly defined set of steps. Methods for accomplishing the main steps in the procedures are described in B.2.2.2 to B.2.2.10. They are derived from Reference [2].

B.2.2.2 Obtain the spacecraft geometry

This subclause relates to step 1.2 in case 1 and case 2a.

First, a 3D representation of the spacecraft physical architecture is constructed. The geometry of the entire spacecraft structure, including external panels, internal walls, shelves, etc., is modelled; and material properties are assigned to each of these structural elements. Each equipment item on the spacecraft is also defined in terms of its geometry, material properties, and internal/external position on the spacecraft. In terms of geometric modelling accuracy, it is usually not necessary to implement a high degree of precision. For example, when modelling an equipment item that has an irregular shape, it is usually sufficient to choose the simplest regular shape that approximates it. This helps to reduce the complexity of the impact risk analysis and the associated simulation run-times.

When defining the geometry information, it is necessary to select a reference frame for the main body of the spacecraft. This can be the same reference frame as used for the SD/M environment fluxes. All of the spacecraft surfaces can be located in this reference frame through the Cartesian coordinates of their geometric centres and three rotations around the coordinate axes; the unit normal to each surface can also be calculated. Furthermore, by calculating the applicable transformation matrixes it is possible to pass coordinate data from the main reference frame to local reference frames for the individual surfaces.

B.2.2.3 Define impact failure criteria for critical surfaces

This subclause relates to step 3.3 in case 1 and case 2a.

The definition of impact-related failure criteria for the different elements of an unmanned spacecraft is driven by a consideration of whether they are critical to some operational aspect of the spacecraft. In this document the concern is twofold:

— to minimise the possibility that an impact prevents post-mission disposal of the spacecraft;

— to avoid an impact-induced break-up of the spacecraft.

On a typical unmanned spacecraft, many of the equipment units contribute to its post-mission disposal, such as elements of the attitude and orbit control subsystem, the communication subsystem, and the power subsystem. Therefore, this equipment can be considered critical for the purposes of an impact risk analysis.

From an impact perspective, the main items of interest can be categorised as follows:

— electronics boxes;

— batteries;

— pressurised vessels;

— propellant pipes and heat pipes;

— solar arrays;

— wires and cables.

Several of these items are protected by a single-wall casing. The ballistic limit of a single wall is dependent on the criterion for failure. In many cases the failure criterion is defined to be perforation of the wall. However, other criteria are also possible, such as the production of detached spall or incipient spall at the rear side of the wall. In the example of an aluminium box protecting sensitive electronics equipment the choice of detached spall as a failure criterion, rather than perforation, can be preferable since the impact risk analysis will lead to a more conservative engineering solution. Annex C describes some ballistic limit equations where these failure criteria can be applied. The same failure criteria can also be applied to multi-wall shield designs, such as a sandwich panel placed some distance in front of an electronics box.

When assessing the risk that an impact causes the accidental break-up of a spacecraft, other failure criteria are important. Generally, a break-up can occur either when the kinetic energy of an impact between an SD/M object and a spacecraft is sufficiently high, or when a small SD/M object impacts an equipment item containing a large amount of stored energy.

In the former case, the analysis of the failure criteria for an impact-induced break-up can be performed using a variety of modelling techniques, examples of which can be found in References [9], [10] and [11]. Until these types of model become more mature and widely available, it is reasonable to apply a simple break-up failure criterion. Perhaps the most common criterion, albeit a rather crude one, is a threshold for the ratio of the impact kinetic energy to the mass of the spacecraft. A typical value for the EMR threshold is 40 J/g.

In the latter case, the impact of a small SD/M object on a pressurised vessel, such as a propellant tank, is of particular concern for most spacecraft. The impact of an SD/M particle on a pressurised vessel can result in one of two significant damage modes:

a) the creation of a perforation hole through which the contents of the vessel leak;

b) rupture of the vessel.

Tests have shown that a relatively narrow margin exists between the perforation and rupture of a pressurised wall under hypervelocity impact.

In the case of a perforation leak, it is possible for the spacecraft to spin-up uncontrollably and the resulting stresses to cause parts of the spacecraft to break off. There is also the possibility that the leaking contents, if reactive, can subsequently detonate and cause an explosion. In the case of a rupture, the sudden release of a large amount of stored energy can lead to catastrophic failure of the spacecraft and the generation of a significant quantity of space debris. All of these failure scenarios can loosely be described as accidental break-ups.

The failure response of a vessel is influenced by whether it is pressurised. That is, the threshold impact conditions for producing either of the above damage modes are different, and depend on the vessel's overpressure relative to the ambient pressure. If an impact risk analysis for accidental break-up takes a conservative approach, then it is reasonable to choose perforation as the failure criterion for the vessel. Alternatively, if there is confidence that a small impactor can only cause a break-up by rupturing the pressurised vessel, then the failure criterion can be set as rupture in the analysis. Annex C provides single wall ballistic limit equations for pressurised vessels using either of these failure criteria.

It is not common practice to expose a critical item, such as a pressurised vessel, directly to the space environment because of the SD/M impact risk. Usually, these items are located inside a spacecraft. For the case of a pressurised vessel placed some distance behind a spacecraft structure, if cloud modelling approach 1 is used in the impact risk analysis (see B.2.2.8), then vessel wall failure (perforation or rupture) can be determined by applying the relevant BLE in Annex C.

Alternatively, in the example of an impact on a CFRP or aluminium gas-filled high-pressure vessel placed a certain distance behind an MLI/sandwich panel, where the failure criterion is perforation, the SRL ballistic limit equation has shown good agreement with test data.[12] This equation has been developed specifically for such an arrangement (i.e. equipment behind a structure) and can be applied in cloud modelling approach 2, described in B.2.2.8.

B.2.2.4 Select an SD/M environment model

This subclause relates to step 4.2 in case 1 and case 2a.

An SD/M environment model is a key element in determining the magnitude of the impact threat that a spacecraft is likely to experience. ISO 14200 [4] provides information on the capabilities, selection and use of SD/M environment models to calculate the impact flux on a spacecraft with a given set of mission and orbital parameters. For a spacecraft in a given orbit, these models typically provide the number of impacts per unit area per unit time as a function of particle size, speed, and impact angle.

ISO 14200 notes that there are likely to be significant differences in the calculated fluxes among the available candidate models. This is particularly true for small SD/M, such as millimetre-size particles, where knowledge of the population is very limited. Therefore, when selecting an environment model, ISO 14200 recommends that the customer or supplier of the spacecraft compare the fluxes of several models. ISO 14200 also notes that to achieve adequate safety margin in the design of a spacecraft or its subsystems, it is reasonable to select the model with the highest flux values when analysing the risk caused by SD/M impacts.

B.2.2.5 Compute the SD/M impact fluxes on external surfaces

This subclause relates to steps 4.2 and 4.3 in case 1 and case 2a.

Typically, an SD/M environment model provides flux information in 'bins' according to particle size and the impact direction and velocity. It is commonplace also for the flux information to be provided relative to the spacecraft reference frame.

The transfer of flux data to an impact risk analysis model is usually accomplished in one of two ways:

a) through a bespoke interface between the environment model and the impact risk analysis model;

b) using the STENVI interface described in Reference [46].

After reading the flux data file, an impact risk analysis model can create impact flux vectors for each external surface of the spacecraft. Thus, when considering the entire spacecraft, the total impact flux can be represented as a two-dimensional $m \times n$ matrix of impact flux values, $\boldsymbol{F}_{\mathrm{I}}$, where each of the m rows in the matrix refers to a specific combination of particle diameter and relative impact velocity, azimuth angle and elevation angle, and where each of the n columns refers to a particular external surface of the spacecraft.

B.2.2.6 Compute the SD/M perforation fluxes on external surfaces

This subclause relates to step 4.3 in case 1 and case 2a.

To compute the portion of the total impact flux that perforates the spacecraft hull and penetrates inside the vehicle, suitable ballistic limit equations can be employed. A BLE is a type of empirical damage equation used to predict the size of an impacting particle that can cause the failure of a spacecraft structure or equipment item. It is usually expressed as a function of impact speed, impact angle, particle density and the design parameters of the target, such as wall thickness and material. Further information on BLEs is available in Annex C.

BLEs are applied on a case-by-case basis. For example, if one of the external surfaces is a sandwich panel, then it is possible to use a BLE such as the SRL equation [6] or the STP equation [7] to calculate if a particular combination of impact parameters, associated with a given impact flux value, results in the sandwich panel's perforation. Alternatively, if the external surface is a thin aluminium wall, such as used in the casing of an exposed electronics box, then the calculation can be performed using the Cour-Palais single wall BLE [8].

By following this method for each element of the impact flux matrix, then an $m \times n$ perforation flux matrix, $\boldsymbol{F}_{\mathrm{P}}$, can be constructed. Thus, $\boldsymbol{F}_{\mathrm{P}}$ is effectively a subset of $\boldsymbol{F}_{\mathrm{I}}$. That is, some of the flux values in the matrix, $\boldsymbol{F}_{\mathrm{P}}$, are zero because perforation does not occur.

B.2.2.7 Compute the probabilities of impact and perforation for external surfaces

This subclause relates to step 4.4 in case 1 and case 2a.

The following method for calculating impact probability is described in Reference [1].

For spacecraft that fly with a fixed orientation, SD/M fluxes are treated as vector quantities and the effects of directionality carefully evaluated. Most space debris impacts typically occur on the forward-facing, side-facing, and space-facing surfaces of an Earth-orbiting spacecraft, where the forward-facing surface is defined as the leading surface in the direction of motion of the spacecraft, i.e. the velocity direction or “ram” direction.

The number of impacts, N, from SD/M particles larger than a given diameter, increases linearly with exposed area (A), flux (F), and exposure time (Δt):

$$N = F\ A\ \Delta t \tag{B.1}$$

Impact fluxes for each external surface can be extracted from the matrix, $\boldsymbol{F}_{\mathrm{I}}$, described in B.2.2.5. It should be noted that impact fluxes are not constant in time. Therefore, for large values of Δt (e.g. more than two years) it is customary to use an averaged value of F or, alternatively, to sum N over smaller time steps.

The probability of exactly n impacts occurring in the corresponding time interval can be determined using Poisson statistics, provided N is sufficiently small (<~10):

$$P_{i=n} = \left(\frac{N^n}{n!} \right) \mathrm{e}^{-N} \tag{B.2}$$

Thus, the probability of no impacts, $P_{i=0}$, is given by:

$$P_{i=0} = \mathrm{e}^{-N} \tag{B.3}$$

And the probability of at least one impact is:

$$P_{i>0} = 1 - \mathrm{e}^{-N} \tag{B.4}$$

The same equations can be used to calculate the number of impacts, N, that cause the failure of each external surface of the spacecraft. In this case the perforation flux values for each external surface can be extracted from the matrix, $\boldsymbol{F}_{\mathrm{P}}$, described in B.2.2.6. Hence, the probability of perforation of each external surface can be calculated.

Knowing the impact probability and perforation probability of each external surface, it is then a relatively simple matter to calculate the overall probabilities of impact and perforation, i.e. for all external surfaces of the spacecraft.

B.2.2.8 Compute the SD/M perforation fluxes on internal equipment

B.2.2.8.1 General

This subclause relates to step 4.5 in case 1 and case 2a.

The next major step in the analysis process involves calculating the fraction of each of the perforation flux values in $\boldsymbol{F}_{\mathrm{P}}$ that directly hits, and subsequently causes the failure of, each surface of every internal equipment unit. This is the most difficult step to implement. It requires a careful balance to be struck between computational accuracy and simulation run-time.

When an SD/M impactor perforates the external structure of a spacecraft at a relatively low (ballistic) speed, typically less than 3 km/s, it will stay largely intact. However, at impact speeds between approximately 3 km/s and 7 km/s the particle usually breaks up into a cloud of solid fragments. Beyond ~7 km/s, a speed that is generally considered to be representative of a hypervelocity impact, the cloud comprises mostly fine vaporous material. Broadly, there are two different approaches that can be taken to quantify the damage such clouds can inflict when they propagate inside a spacecraft.

— Cloud modelling approach 1

  Employ a model that characterises the shape of the cloud and the distribution of mass and velocity within it. Then, using a geometrical method, propagate the cloud inside a spacecraft to identify the surfaces of internal equipment items that are hit by the cloud. For each cloud-surface interaction use a simple damage equation, such as a single-wall BLE, to calculate whether the equipment fails.

— Cloud modelling approach 2

  Employ a model that uses a multi-wall BLE in conjunction with a geometrical function to calculate whether an SD/M impactor can cause the entire combination of spacecraft external structure and one or more internal equipment walls to fail. For this approach, knowledge of the structure of a cloud is not required.

The following qualitative arguments have been put forward in support of the first approach.

— It is based on analyses of the physical interaction between penetrating SD/M and spacecraft internal equipment, and so it is inherently more accurate.

— It can be upgraded readily as and when new test data become available, thereby enabling the accuracy of the method to be improved continuously.

— It is ideally suited to investigate the impact vulnerability of any spacecraft design and any layout of internal equipment. Such freedom is not necessarily possible in the second approach, since multi-wall BLEs can only be used reliably when investigating the narrow range of geometric characteristics from which they were derived experimentally.

— The failure of internal equipment can be computed with reference to the damage on their cover faces using single-wall damage equations, whose reliability is high because of the large databases of test data on which they are based. Furthermore, with a single-wall equation it is easier to apply a failure criterion other than perforation.

— By propagating a cloud inside a spacecraft, the effects of mutual shadowing between equipment units can be considered with a high degree of accuracy.

The main arguments in support of the second approach are that it ought to be simpler to implement in a computer code and faster to run. Therefore, it is well suited to the early phases of a spacecraft mission when design changes are more frequent.

B.2.2.8.2 and B.2.2.8.3 provide examples of how the two modelling approaches can be applied to calculate the cumulative failure flux or failure probability for each internal equipment surface.

B.2.2.8.2 Example of cloud modelling approach 1

The first cloud modelling approach utilises knowledge of the structure of a cloud for an impact risk analysis. This includes the geometric properties of the cloud, and its mass distribution and velocity distribution, to calculate how the cloud propagates inside a spacecraft.

In such a model it is usually sufficient to characterise the geometry of an expanding cloud as the superposition of two cones, where one of the cones represents “in-line” fast, heavy fragments that follow a similar trajectory to the original impactor, and the other represents “dust” fragments. Thus, each cone is characterised by the direction of its axis, as measured from the surface normal, and the spread (or spray) angle around the axis. Figure B.1 illustrates the geometry of the in-line cloud cone, which is the most damaging part of a cloud.

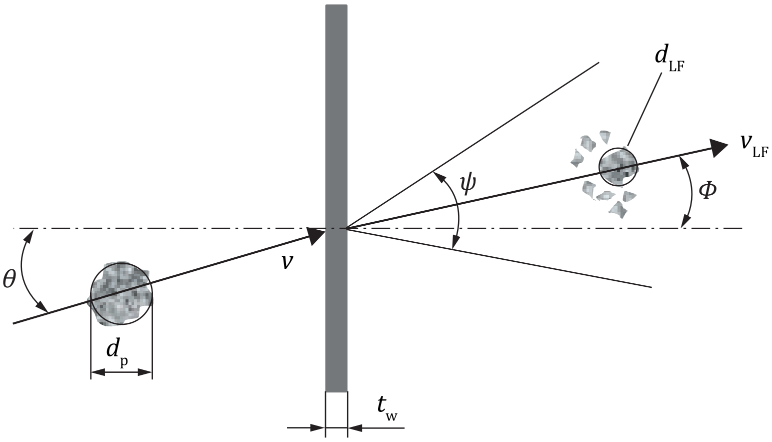


Figure B.1 — In-line cloud geometric model for a spacecraft single wall [14]

References [13] and [14] describe a cloud model for the case of an SD/M impactor perforating a single wall and for the case where it perforates a sandwich panel. Equations are provided for the angles ϕ and ψ, the diameter and velocity of the largest cloud fragment, the SD/M diameter at the ballistic limit of the wall / panel, the threshold velocity for SD/M fragmentation, and the total mass of the cloud.

The model also provides a geometric function for propagating the cloud inside a spacecraft to calculate the portion of the cloud that is intercepted by each of the internal equipment surfaces. One very accurate geometric method for this purpose is to use an anisotropic ray-tracing algorithm that fires rays only within the cloud cone. However, the main drawback of this approach is that it is computationally quite intensive. Instead the model makes use of so-called MVF. These are GVF, typically used for thermal calculations, which have been modified to take account of the cloud axis orientation and spread angle. This correction is necessary since GVF are normally obtained for surfaces that emit rays along every direction in the half-space containing the target, and hence they are not adequate to simulate the flux of fragments contained in the cloud cone. The MVF method is valid for two surfaces that are in direct view of each other. It has also been adapted for the situation where there is partial or total shadowing from other intervening surfaces.

Thus, returning to the discussion in B.2.2.6, for each element of the perforation flux matrix, $\boldsymbol{F}_\mathbf{P}$, the MVF is calculated from the relevant spacecraft surface towards each surface of the internal equipment units. Therefore, for an element $[i, j]$ in the matrix, $\boldsymbol{F}_\mathbf{P}$, the fraction of the perforation flux that directly hits the k^{th} internal surface is given by:

$$F_\text{P}[i, j]_k = F_\text{P}[i, j] \cdot V[i, j]_k \tag{B.5}$$

Where $V[i, j]_k$ is the MVF. To determine if the k^{th} internal surface fails when it is impacted by this fraction of the perforation flux, then a damage equation for a single wall can be applied using the corresponding information from the cloud model. For example, to calculate whether the largest fragment in a cloud will perforate the internal surface then a single wall BLE can be used. This is usually applicable for lower velocity impacts (approximately < 7 km/s). Alternatively, for hypervelocity impacts (typically > 7 km/s), a critical energy equation can be used to calculate if the energy of the cloud is sufficient to perforate the internal surface. Either way, the result can be recorded by entering the $F_\text{P}[i, j]_k$ value in a failure flux matrix, $\boldsymbol{F}_\mathbf{F}$ (at element $F_\text{F}[i, j]_k$), when perforation occurs, and the value zero when it does not. Finally, the cumulative failure flux for the k^{th} internal surface is achieved by summing $F_\text{F}[i, j]_k$ for all i and j.

Further details of this cloud model, including the assumptions made in its derivation, are available in References [13] and [14]. An example of a similar type of cloud model is given in Reference [15].

B.2.2.8.3 Example of cloud modelling approach 2

An alternative method for computing the penetrative damage inside a spacecraft is described in Reference [16]. At the heart of it is an orthographic projection technique. The vulnerable area of an internal equipment item is calculated in relation to the point of impact on a spacecraft external surface by being projected orthogonally along the cloud velocity vector. This unbounded projected area is then clipped by the projection of the external spacecraft surface with respect to the cloud velocity vector. Shadowing effects from other equipment items are also considered in the projected area calculation. The final projected area is then used to determine the "susceptibility" of the equipment item, which is one term in the computation of the item's failure probability. The other significant term in the equipment failure probability calculation is the "lethality" of the penetrating SD/M particle with respect to the equipment item. Physical damage effects are determined using BLEs, such as the SRL multi-wall equation, where the spacecraft external surface is described by the SRL's bumper terms and the internal equipment casing is the rear wall term in the equation.

B.2.2.9 Compute the probability of impact-induced failure of the spacecraft

This subclause relates to step 4.6 in case 1 and case 2a.

In the event that an equipment item fails as a result of a penetrating impact, the consequences of that failure for the operation of the spacecraft, either in terms of post-mission disposal or accidental break-up, can be determined. This requires consideration of the criticality of the equipment and the presence of any redundancy elsewhere in the spacecraft. To perform such an evaluation it is commonplace to use a technique such as FTA.

Generally, a fault tree is constructed using a Boolean logic approach to represent the functional relationships at each level of the spacecraft architecture, from equipment to subsystem and finally to the whole system. By following the logical connections the failure of an individual equipment item can be traced through the tree to reveal whether the spacecraft fails. In doing so, it is important to remember that a single debris cloud can impact and damage several equipment items. Therefore, it can be necessary to trace the consequences of multiple failures in the tree when assessing the damage from one penetrative impactor.

When this process is repeated for every penetration, the failures can be aggregated to quantify firstly the probability of impact-induced failure of each equipment item on the spacecraft, and secondly the probability of impact-induced failure of the spacecraft, P_F. It is necessary to implement the aggregation method with care in order to avoid the mistake of counting the same external SD/M impact event several times, which can happen when a single debris cloud damages multiple equipment items.

B.2.2.10 Identify impact vulnerabilities and protection solutions

This subclause relates to step 5.1 in case 1 and case 2a.

By combining flux data, damage equations and spacecraft geometry information, as described above, an impact risk assessment can be performed to help identify the main areas of vulnerability in a spacecraft design. Options can then be evaluated to reduce P_F, if necessary, so as to meet the failure probability threshold requirement, $P_{F\ max}$. For example, the benefit of implementing SD/M shields can be analysed, as well as other protection enhancement strategies, such as increasing equipment redundancy. These are discussed more fully in Annexes D to F.

Finally, it is worth noting that the steps in B.2.2.8 to B.2.2.10 can be bypassed if the probability of perforation of the spacecraft calculated in B.2.2.7 is less than $P_{F\ max}$.

B.3 Analysis models

B.3.1 General

This subclause relates to step 4.1 in case 1 and case 2a.

A number of statistically-based computer codes have been developed to perform detailed impact risk analyses of non-trackable SD/M particles. These allow a fully three-dimensional numerical analysis of a spacecraft, including directional and geometrical effects and shielding considerations. In doing so, they normally support the application of different environment flux models and damage equations. The codes provide a 3-D display of the results.

Typical user-specified input parameters for these tools are:

— the orbit and mission parameters;

— spacecraft attitude, geometry and shielding;

— the particle type, size, mass density and velocity range to be analysed;

— the damage equations and related parameters to be applied.

The computed output generally includes data such as:

— the distribution of impacts for a specified particle size range;

— the distribution of damaging impacts (e.g. perforations) taking into account the spacecraft shielding and damage assessment equations;

— the mean particle impact velocity (i.e. magnitude and direction);

— the number of craters of a specified size;

— the probability of failure.

B.3.2 Information on specific codes

Tables B.1 to B.4 provide summary information on several codes that can analyse the risk to a spacecraft from SD/M impacts. These codes have been validated in different ways. For simple test cases, such as a single wall and fixed impact velocity, the results have been compared to calculations done by hand. However, for more complex test cases, such as for when the full directional and velocity distribution of the impacting particles is included, the codes have been validated by comparing results against a set of benchmark test cases defined by the IADC [1].

Table B.1 — Accessibility and use of impact risk analysis models

Impact risk analysis model	Owner / developer	Country or region	Model accessible directly via licence to 3rd parties	Model accessible indirectly via analysis service to 3rd parties	Used for research studies
BUMPER [17]	NASA	USA	✓[a]	✓	✓
DRAMA/MIDAS [18]	ESA	Europe	✓	✓	✓
ESABASE2 / DEBRIS [19]	ESA / etamax space	Europe	✓	-	✓
MODAOST [20]	CAST	China	×	✓	✓
PIRAT [21]	Ernst Mach Institut	Germany	✓	✓	✓
S^{3}DE [22,23]	Harbin Inst. of Technology	China	×	✓	✓
SHIELD3 [24]	PHS Space	UK	×	✓	✓
SYSTEMA / DEBRIS [25]	Airbus	France	✓	×	✓
TURANDOT [26]	JAXA	Japan	✓	×	✓
[a] Requires US government contract.					

Table B.2 — Capability of impact risk analysis models to read flux data from the space debris environment models listed in ISO 14200 [4]

Impact risk analysis model	ORDEM 3.0	ORDEM 3.1	MASTER 2009 CPE file format	MASTER 8 CPE file format	SDEEM 2015	SDEEM 2019	Any model outputting STENVI SEI file format
BUMPER	✓	✓	×	×	×	×	✓
DRAMA/MIDAS	×	×	×	✓	×	×	×
ESABASE2 / DEBRIS	✓	×	✓	✓	×	×	-
MODAOST	×	×	✓	✓	✓	✓	✓
PIRAT	✓	✓	✓	✓	×	×	✓
S^{3}DE	×	×	✓	✓	✓	✓	✓
SHIELD3	×	×	×	×	×	×	✓
SYSTEMA / DEBRIS	×	×	×	×	×	×	✓
TURANDOT	✓	✓	✓	✓	✓	×	×

Table B.3 — Capability of impact risk analysis models to read flux data from the meteoroid environment models listed in ISO 14200 [4]

Impact risk analysis model	Grüen et al.	Divine-Staubach	NASA SSP 30425	MEM R2	Any model outputting STENVI SEI file format
BUMPER	×	×	✓	✓	✓
DRAMA/MIDAS	✓	✓	×	×	×
ESABASE2 / DEBRIS	✓	✓	×	✓	-
MODAOST	✓	✓	✓	×	✓
PIRAT	✓	✓	×	×	✓
S^{3}DE	✓	✓	✓	×	✓
SHIELD3	×	×	×	×	✓
SYSTEMA / DEBRIS	×	×	×	×	✓
TURANDOT	✓	✓	×	✓	×

Table B.4 — Special analysis features of impact risk analysis models

Impact risk analysis model	Spacecraft internal damage	Self-shadowing	Secondary ejecta	Optimise shielding[a]	Optimise equipment layout[a]
BUMPER	2	✓	×	✓	✓
DRAMA/MIDAS	2	✓	×	✓	✓
ESABASE2 / DEBRIS	2	✓	✓	✓	✓
MODAOST	1, 2	✓	×	✓	✓
PIRAT	2	✓	×	✓	✓
S^{3}DE	1, 2	✓	×	✓	✓

Key:

1 cloud modelling approach 1

2 cloud modelling approach 2

[a] Provides information to support this analysis.

Table B.4 *(continued)*

Impact risk analysis model	Spacecraft internal damage	Self-shadowing	Secondary ejecta	Optimise shielding[a]	Optimise equipment layout[a]
SHIELD3	1, 2	✓	×	✓	✓
SYSTEMA / DEBRIS	2	✓	×	✓	✓
TURANDOT	×	✓	×	✓	✓

Key:

1 cloud modelling approach 1

2 cloud modelling approach 2

[a] Provides information to support this analysis.

B.3.3 Uncertainty considerations

A major limitation with most impact risk analysis models is that they do not quantify the uncertainties associated with an impact risk analysis. Generally, uncertainty bounds or confidence intervals are not provided. The most significant sources of uncertainty in an impact risk analysis are:

— discrepancies between space debris environment models in determining impact fluxes, especially at millimetre particle sizes;

— knowledge about the mass density of space debris, as generated by environment models;

— inaccuracies in the derivation of damage equations, because they are often based on a statistically small number of impact tests or on numerical simulations for which there is limited confidence in the results;

— limitations in the application of damage equations, because they generally do not take account of variations in impactor shape and target temperature;

— limitations in analysing the damage inside a spacecraft as a result of a penetrative impact. In particular, the modelling of a cloud's interaction with internal equipment is usually simplified.

Uncertainties such as these can significantly influence the risk calculation, as discussed in Reference [27], and so careful consideration is necessary when deciding whether to modify the design of a spacecraft. One of the main problems in determining uncertainties has been the lack of available data to support the calculation. Traditionally, this has been overcome by doing multiple runs of an impact risk analysis code, whereby the program inputs for each run are varied by a certain amount based on engineering judgement. However, this approach lacks rigor since it does not properly take into account the size of the individual uncertainties.

Efforts are now underway to address the various uncertainties. For example, the MASTER 8 environment model provides uncertainty estimates for its population models which can be passed on to impact risk assessment tools. However, there is currently no recommendation on when to use the uncertainty estimates.

Annex C
(informative)

Ballistic limit equations

C.1 General

A BLE is a type of damage equation which defines the characteristics of an impacting SD/M particle on the threshold of failure of a wall, panel, or shield, where failure is defined by a user-specified criterion, such as perforation. That is, a BLE yields the critical size of an impacting particle at which the user-defined damage criterion for a structure is exceeded.

Note that BLEs are empirical in nature, i.e. generally derived from impact test programmes, and so they tend to have a limited range of application. Such limitations can include the range of impact velocities and angles, and the types of material in the target and projectile. Therefore, it is important to exercise care when using these equations, especially in situations where the limitations are exceeded.

A BLE is usually developed by drawing a curve through a set of pass/fail impact test data. Thus, it provides a simple means of assessing whether a surface (or collection of surfaces) can be penetrated by an impactor. As a consequence, BLEs represent a core element of almost every impact risk assessment code.

C.2 Generalised forms of ballistic limit equations

C.2.1 General

This clause describes generalised forms of a single wall BLE and a multiple wall BLE, as defined in Reference [19]. The BLEs comprise terms that describe the properties of a particle and the target that it impacts.

C.2.2 Single wall BLE

The single wall BLE defines the critical size of particle on the threshold of failure of a homogeneous structure such as an aluminium wall. It can be written in the following parametric form:

$$d_c = \left[\frac{t_w}{K_f K_1 \rho_p^{\beta} v^{\gamma} (\cos\theta)^{\xi} \rho_w^{\kappa}} \right]^{\frac{1}{\lambda}} \tag{C.1}$$

C.2.3 Multiple wall BLE

To determine the critical size of a particle penetrating a multiple wall shield design, the BLE can be written in the following parametric form:

$$d_c = \left[\frac{t_w + K_2 t_b^{\mu} \rho_b^{\zeta_2}}{K_1 \rho_p^{\beta} v^{\gamma} (\cos\theta)^{\xi} \rho_w^{\kappa} S^{\delta} \rho_b^{\zeta_1}} \right]^{\frac{1}{\lambda}} \tag{C.2}$$

In using Formula (C.2), three velocity regions are considered. At low velocities (typically below ~3 km/s), the ballistic region, an impactor is poorly fragmented by a bumper, and a very small number of solid particles are released. Between approximately 3 km/s and 7 km/s, an impactor is broken into several pieces, which can be solid or vaporous. This is the transition region. Above ~7 km/s, the hypervelocity region, an impactor is broken up into a dense cloud of numerous fine vaporous particles. The three velocity regions are often expressed in terms of the normal component of the impact velocity, as follows: $v_n \le v_l$, $v_l < v_n < v_h$, and $v_n \ge v_h$, where v_l and v_h are the lower and higher transition velocities between the three regions, respectively.

For impacts whose velocities are either in the ballistic region or the hypervelocity region, Formula (C.2) is used. However, for impact velocities in the transition region, i.e. between v_l and v_h, linear interpolation is used to calculate the critical particle diameter, as follows:

$$d_c = \left(\frac{v_h - v_n}{v_h - v_l}\right) d_c(v_l) + \left(\frac{v_n - v_l}{v_h - v_l}\right) d_c(v_h) \quad \text{(C.3)}$$

C.3 Specific examples of ballistic limit equations

C.3.1 General

This clause relates to step 3.1 in case 1 and case 2a.

This clause lists specific examples of BLEs for some typical spacecraft structures. Many more equations can be found in the literature. For example, References [1] and [28] provide comprehensive collections of BLEs. The limitations of the equations are also clearly identified.

C.3.2 Simplified BLE for a single wall or multiple walls

The areal density, ρ_A, of one or more layers of material along a given line of sight is its mass density multiplied by the total thickness of the layer(s).

For a simplified assessment of the critical diameter, d_c, of SD/M impactor that can perforate this arrangement, the BLE in Formula (C.4) can be used[29].

$$d_c = K \rho_A \quad \text{(C.4)}$$

where K has a value of 0,07 for a typical single layer of material, such as aluminium alloy 6061-T6, assuming that the units for d_c and ρ_A are cm and g/cm^2, respectively. Higher K values can be achieved for specially designed multiple wall shields, such as the Whipple shield (K = 0,35) and the multi-shock shield (K = 0,70). Note that these K values are only intended to give a rough estimate of shielding effectiveness. The calculation of the critical diameter, d_c, provides a lower bound on the size of impactor that can be expected to perforate the material layer(s), i.e. it is a conservative value for the ballistic limit.

It should be noted that while the BLE uses perforation as the criterion of failure, other failure criteria can also be applied.

It should also be noted that Formula (C.4) does not require any information on the properties of a typical SD/M impact, such as mean impact velocity, since these are embedded within the K term. For a more precise determination of ballistic limit, in which particle characteristics are explicitly considered, it is necessary to use a more sophisticated BLE, such as one of those in C.3.3 to C.3.8.

C.3.3 BLEs for a metallic single wall

The BLEs in Table C.1 assess the performance of a metallic single wall where the threshold of failure is defined by one of the following types of impact damage: perforation, detached spall, or incipient spall.

Table C.1 — BLEs for a metallic single wall

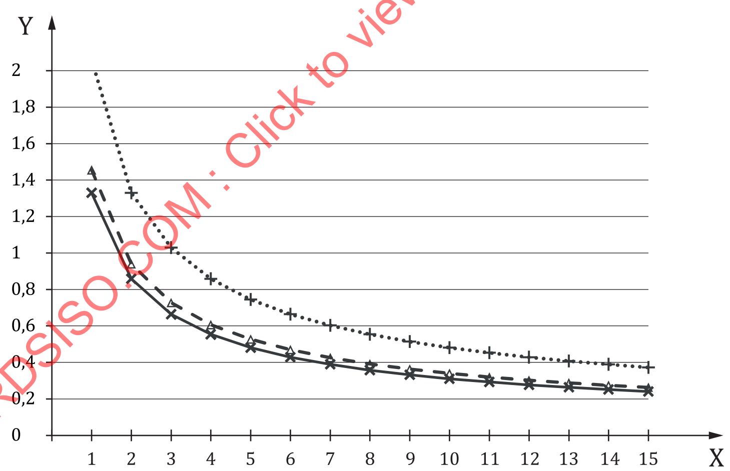
Wall type	Conditions	Ballistic limit equation
Aluminium single wall [8]	$\rho_p/\rho_w < 1,5$	$$d_c = \left[\frac{t_w}{k} \frac{H^{0,25}\left(\rho_w / \rho_p\right)^{0,5}}{5,24\left(v\cos\theta / C\right)^{2/3}} \right]^{\frac{18}{19}}$$
	$\rho_p/\rho_w \geq 1,5$	$$d_c = \left[\frac{t_w}{k} \frac{H^{0,25}\left(\rho_w / \rho_p\right)^{3/2}}{5,24\left(v\cos\theta / C\right)^{2/3}} \right]^{\frac{18}{19}}$$
Titanium single wall [30]	–	$$d_c = \left[\frac{t_w}{k} \frac{H^{0,25}\left(\rho_w / \rho_p\right)^{0,3}}{5,24\left(v\cos\theta / C\right)^{2/3}} \right]$$
Stainless steel single wall [28]	–	$$d_c = \left[\frac{t_w}{k} \frac{\left(\rho_w / \rho_p\right)^{0,5}}{K\left(v\cos\theta\right)^{2/3}} \right]^{\frac{18}{19}}$$

The following values for k and K can be used in the BLEs:

— $k = 3,0$; 2,2; 1,8 for incipient spall, detached spall, and perforation, respectively;

— $K = 0,345$.

Figure C.1 illustrates a typical plot of a metallic single wall BLE. In this example, the resulting ballistic limit curves are shown for an aluminium spherical projectile impacting a 1,5 cm thick aluminium wall at three different angles[28].



Key

X impact velocity (km/s)

Y projectile diameter (cm)

× impact at 0°

Δ impact at 30°

\+ impact at 60°

Figure C.1 — Example ballistic limit curves for an aluminium single wall

C.3.4 BLE for a CFRP single wall

The BLE in Table C.2 assesses the performance of a CFRP single wall where the threshold of failure is defined by one of the following types of impact damage: perforation or detached spall [31].

Table C.2 — BLE for a CFRP single wall

Wall type	Conditions	Ballistic limit equation
CFRP single wall	–	$d_c = \left[\frac{t_w}{k} \frac{\left(\rho_w / \rho_p\right)^{0,5}}{K_{CFRP} \left(v \cos\theta\right)^{2/3}} \right]$

The following values for k and K_{CFRP} can be used in the BLE:

— k = 3,0; 1,8 for detached spallation and perforation, respectively;

— K_{CFRP} = 0,62.

C.3.5 BLE for a sandwich panel with a honeycomb core

C.3.5.1 General

The ballistic limit of a honeycomb core sandwich panel is a function of its face-sheet thickness, cell dimensions and cell wall thickness, core depth, and the materials used. Few equations currently exist that combine these parameters. Instead, the most common approach is to adapt existing multi-wall BLEs for use with sandwich panels.

C.3.5.2 SRL BLE

A BLE called the SRL equation has been developed to determine the ballistic limit of any type of dual- or triple-wall shield design, including an equipment item located some distance behind a spacecraft structure, such as a sandwich panel[6][12]. The SRL equation is named after its originators, Schaefer–Ryan–Lambert. It is an extension of the ESA triple-wall equation and was derived following a programme of impact tests on a variety of typical equipment located behind spacecraft panels. The equipment included electronics boxes, batteries, wire harnesses, pressure vessels, propellant pipes and heat pipes.

The SRL equation was calibrated so that its ballistic limit curve corresponded with the onset of functional failure of the equipment. For pressurized equipment, such as propellant pipes and propellant tanks, failure was defined as rupture or leakage. For electrical equipment, such as harnesses or electronics boxes, equipment-specific failure criteria were established corresponding with observed functional anomalies, such as voltage spikes, data errors or power loss.

The SRL equation for a sandwich panel with honeycomb core (excluding equipment), i.e. a dual wall shield design, is shown in Table C.3. It assesses the impact performance of the shield over the three velocity ranges: high ($v_n \geq v_h$), low ($v_n \leq v_l$), and intermediate ($v_l < v_n < v_h$).

Table C.3 — SRL BLE for a dual-wall shield design

Panel type	Velocity regime	Ballistic limit equation
Sandwich panel with honeycomb core	$v_n \geq v_h$	$d_c = \left[\dfrac{1{,}155\, S^{1/3} t_w^{2/3} \left(\dfrac{\sigma_{y,w}}{70}\right)^{1/3}}{K_{3D}^{2/3}\, \rho_p^{1/3}\, \rho_b^{1/9}\, v^{2/3} (\cos\theta)^{\delta}}\right]$
	$v_n \leq v_l$	$d_c = \left[\dfrac{\left(\dfrac{t_w}{K_{3S}}\right)\left(\dfrac{\sigma_{y,w}}{40}\right)^{1/2} + t_b}{0{,}6\, \rho_p^{1/2} v^{2/3} (\cos\theta)^{\delta}}\right]^{\frac{18}{19}}$
	$v_l < v_n < v_h$	See Formula (C.3)

Table C.4 lists possible values for some of the terms in the SRL equation.

Table C.4 — Values for terms in the SRL equation

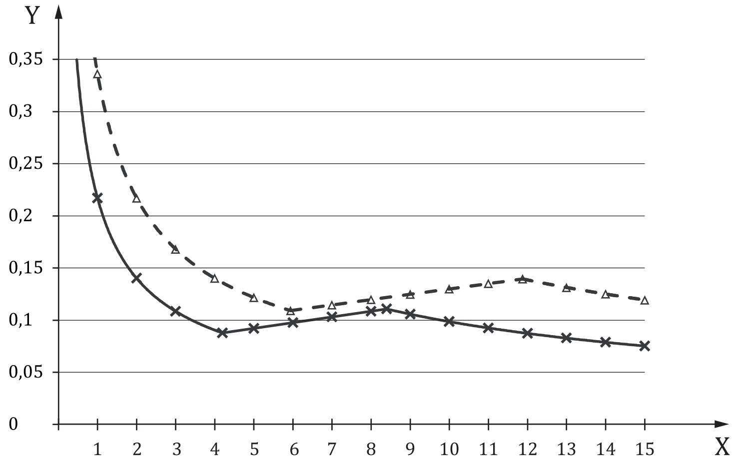
Outer layer	v_l	v_h	K_{3S}	K_{3D}	δ
Al	3	7	1,4	0,4	4/3 (if 45° ≤ θ ≤ 65°), else 5/4
CFRP	4,2	8,4	1,1	0,4	4/3
Other	3	7	1,4	0,4	4/3 (if 45° ≤ θ ≤ 65°), else 5/4

It should be noted that the thickness terms in the SRL equation are for aluminium walls. To convert the thickness of another material, such as CFRP, into the equivalent thickness of aluminium then equivalent areal densities are assumed, as shown in Formula (C.5).

$$t_{al} = t_{CFRP}\left(\frac{\rho_{CFRP}}{\rho_{al}}\right) \tag{C.5}$$

A more detailed consideration of the application of the SRL equation to CFRP sandwich panels is given in Reference [32].

By way of illustration, Figure C.2 provides a plot of the SRL BLE for a typical CFRP sandwich panel design[28]. Ballistic limit curves are shown for an aluminium spherical projectile impacting a dual wall shield at two different angles. In this example the shield is a sandwich panel with 0,145 cm thick CFRP face-sheets and a 5,06 cm thick aluminium honeycomb core. The influence of the three velocity regions is clearly seen.



Key

X impact velocity (km/s)

Y projectile diameter (cm)

× impact at 0°

Δ impact at 45°

Figure C.2 — Example ballistic limit curves for a dual wall CFRP sandwich panel

The SRL equation can be applied to a variety of multi-wall shield designs, including:

— MLI with a stand-off distance to an underlying equipment item;

— MLI placed directly on top of an equipment item;

— an equipment item placed behind a single wall shield;

— honeycomb sandwich panels (with or without MLI);

— an equipment item placed behind a dual wall shield, such as a sandwich panel (with or without MLI).

It is important to exercise care when applying the SRL equation to a particular shield design. Reference [6] provides several examples of how the equation can be used properly. It should also be noted that, given the generalised nature of the SRL equation, the calculated ballistic limit can sometimes be conservative.

C.3.5.3 STP BLE

The STP equation is named after its originators, Sibeaud–Thamié–Puillet[7]. It provides the critical perforation diameter for a projectile of density ρ_p impacting a honeycomb sandwich panel at velocity v and takes the form shown in Table C.5. It is applicable both to the high and low velocity regimes.

Table C.5 — STP BLE for a honeycomb sandwich panel

Panel type	Velocity regime	Ballistic limit equation
Sandwich panel with honeycomb core	$v_n \geq v_h$ $v_n \leq v_l$	$d_c = \left[\dfrac{K_3 \left(t_{hc} + t_w + K_2 t_b^{\mu} \rho_{hc}^{\zeta_2} \right)}{K_1 \rho_p^{\beta} v^{\gamma} (\cos\theta)^{\xi} \rho_w^{\kappa} S^{\delta} \rho_b^{\zeta_1}} \right]^{\frac{1}{\lambda}}$
	$v_l < v_n < v_h$	See Formula (C.3)

In this BLE the weighting coefficient ζ_2 is always 0, and so the term ρ_{hc} has no influence on the calculation of the ballistic limit.

The total thickness of the honeycomb cell walls, t_{hc}, is calculated in Formula (C.6).

$$t_{hc} = \left[K_4 \, r \, \mathrm{Int}\left(\frac{S \tan\theta}{q} \right) \right]^{\eta} \tag{C.6}$$

Where the “Int” function returns the integer part of any real argument and the terms r and q define the honeycomb cell geometry, as shown in Figure C.3:

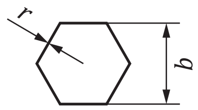


Figure C.3 — Definition of honeycomb cell geometry

The parameters in the equation have the values listed in Table C.6.

Table C.6 — BLE parameter values

Parameter	$v_n \leq v_l$ (=3)	$v_n \geq v_h$ (=7)
K_1	$0{,}6\ (\sigma_{y,w}/40)^{-0{,}5}$	$3{,}918\ [(\sigma_{y,w}/70)^{1/3}]^{-1{,}5}$
K_2	$(\sigma_{y,w}/40)^{-0{,}5}$	0
λ	1,056	1,5
β	0,5	0,5
γ	2/3	1
κ	0	0
δ	0	-0,5
ξ	5/3	1
ζ_1	0	0,167
ζ_2	0	0
μ	1	0
K_3	1,22	1,12
K_4	0,014	0,014
η	0,374 5	0,293

C.3.6 BLE for a sandwich panel with a metallic foam core

The BLE in Table C.7 assesses the impact performance of a metallic foam core sandwich panel over the three velocity ranges: high ($v \geq v_h$), low ($v \leq v_l$), and intermediate ($v_l < v < v_h$) [33].

Table C.7 — BLE for a metallic foam core sandwich panel

Panel type	Velocity regime	Ballistic limit equation
Metallic foam core sandwich panel	$v \geq v_h$	$d_c = 1{,}915 \left[\frac{\left(t_w + 0{,}5 \rho_{A,f} / \rho_w\right)^{2/3} t_f^{0{,}45} \left(\sigma_{y,w} / 70\right)^{1/3}}{\rho_p^{1/3} \rho_b^{1/9} v^{2/3} (\cos\theta)^{4/5}} \right]$
	$v \leq v_l$	$d_c = 1{,}83 \left[\frac{\left(t_b + t_w \left(\sigma_{y,w} / 40\right)^{0{,}5} + t_f^{1{,}1} \left(\rho_f / \rho_w\right)\right)}{\left(\rho_p^{0{,}5} v^{2/3} (\cos\theta)^{4/5}\right)^{18/19}} \right]$
	$v_l < v < v_h$	See Formula (C.3) (use v instead of v_n)

The following values for v_h and v_l can be used in the BLE:

— $v_h = 4{,}0\ (\cos\theta)^{-1/3}$

— $v_l = 2{,}25\ (\cos\theta)^{-1/3}$

C.3.7 BLEs for a metallic pressurised vessel

Whether or not a pressurised vessel ruptures on impact depends on its internal pressure. Some studies have shown that when the pressure is below 15 % to 25 % of a metallic pressure vessel's static burst pressure then rupture is unlikely to occur (see References [34] to [37]). This can be a reasonable assumption to make when analysing the impact-induced break-up risk of such a vessel. For higher pressures, though, it is necessary to use a BLE that can determine whether rupture occurs.

One such BLE has been derived from tests of unshielded spherical and cylindrical metallic, pressurised vessels impacted along their radial trajectory line, where the failure criterion is rupture[38]. The BLE, which is sometimes referred to as an RLE, is shown in Table C.8. The equation takes the form of a power law. According to the originators of the equation, "To render the equation as broadly applicable as possible, the operating conditions (x-axis) were parameterized as the hoop stress in the tank (non-dimensionalised by the ultimate tensile stress of the tank wall material), and the impact conditions (y-axis) were parameterized as impact energy (non-dimensionalised by a number of appropriate tank wall material properties)." It is worth noting that the ratio of hoop stress to ultimate stress decreases as a tank empties during its operational life, and so it is important for an impact risk analysis to take this into account.

Table C.8 — BLE for a metallic pressurised vessel

Wall type	Ballistic limit equation
Spherical metallic pressurised tank	$\dfrac{\frac{1}{2}m_p v^2}{\left(\rho_p t_w^3\right)C^2\left(\dfrac{\rho_p}{\rho_w}\right)^{-3\alpha} H^{3/4}} = A\left(\dfrac{\sigma_h}{\sigma_u}\right)^B$

The ballistic limit can be expressed in terms of the mass of the impacting projectile by rearranging the equation in Table C.8 as follows:

$$m_p = 2A\left(\frac{\sigma_h}{\sigma_u}\right)^B\left[\left(\rho_p t_w^3\right)C^2\left(\frac{\rho_p}{\rho_w}\right)^{-3\alpha} H^{3/4}\right]v^{-2} \qquad \text{(C.7)}$$

where

$\alpha = ½$, if $\rho_p/\rho_w < 1{,}5$

$\alpha = ⅔$, if $\rho_p/\rho_w > 1{,}5$

Thus, assuming the SD/M impactor is spherical, the critical diameter, d_c, is given by:

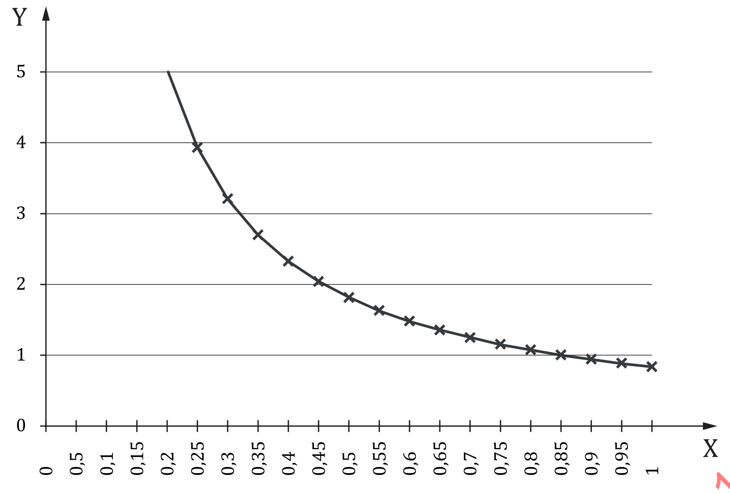
$$d_c = \left[\frac{\left(6m_p\right)}{\left(\pi\rho_p\right)}\right]^{1/3} \qquad \text{(C.8)}$$

Values for the power law terms are listed in Table C.9 according to the shape of the pressurised vessel.

Table C.9 — BLE power law values

Vessel shape	*A*	*B*
Spherical tank	0,838	-1,115
Cylindrical tank	0,327	-1,661

Based on these power law values, the ballistic limit curves for spherical tanks and cylindrical tanks can be plotted, as shown in Figures C.4 and C.5, respectively. In both figures the power law curve separates the regions of rupture and non-rupture.

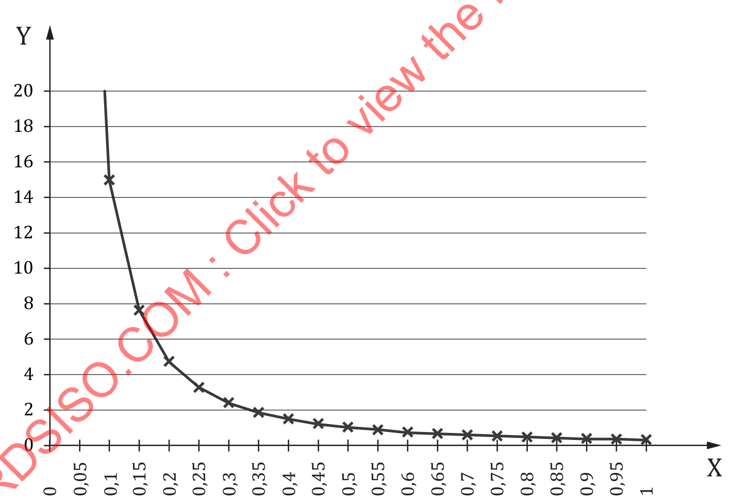


Key

X hoop stress/ultimate stress

Y non-dimensional projectile energy

Figure C.4 — Rupture/non-rupture response of radially impacted spherical metallic tanks (with hoop stress normalized by material ultimate strength at the test temperature)



Key

X hoop stress/ultimate stress

Y non-dimensional projectile energy

Figure C.5 — Rupture/non-rupture response of radially impacted cylindrical metallic tanks

Usually, a propellant tank operates with a stress ratio (σ_h/σ_u) of 0,5 at the beginning of the mission, and a smaller stress ratio of 0,2 at the end of operations. Following the simple procedure outlined in 7.2.4, a typical spherical titanium tank, with a wall thickness of 0,9 mm has an impactor rupture limit diameter of 5,8 mm at the beginning of operations and 8,1 mm at the end of operations. Thus, when the tank is operated at an altitude of 700 km, the cumulative flux of impactors with diameters larger than these limits is 0,000 70 impacts per square metre per year and 0,000 45 impacts per square metre per year, respectively, according to the SD/M environment model MASTER-8. Therefore, the average cumulative flux is 0,000 58 impacts per square metre per year.

When the tank cross-section is 0,24 m^2, and the operational life is 5 years, the average probability of rupture is 0,000 7. This probability is lower than the 0,001 threshold for the probability of break-up caused by stored energy, as specified in ISO 24113, and so it can be deemed low enough to be of limited concern.

A further consideration is that propellant tanks or high-pressure vessels are not usually exposed directly to outer space, particularly at the vulnerable front surfaces of a spacecraft. Normally, they are hidden inside the spacecraft behind various structural elements or other equipment. This means that an impacting object will most likely be fragmented, thereby decreasing the penetrative damage that it can cause to a tank or pressure vessel.

Finally, an alternative RLE, applicable to a spherical titanium alloy (Ti-6Al-4V) pressurised tank containing hydrazine, has been developed using numerical modelling techniques[39]. The equation is shown in Formula (C.9).

$$d_{\mathrm{c}} = K\, p_0\, v^{-0{,}865} \tag{C.9}$$

where the symbol, K, contains all of the parameters (material properties, etc.) except for p_0, the saturated vapour hydrazine pressure (in the range 1 bar to 5 bar)[2)], and v, the impact velocity. Although the results have not yet been experimentally validated, some interesting observations are made, including:

— at low pressures (< 0,2 bar), hydrazine reactivity has no additional detrimental effect compared to an inert fluid, i.e. there is no detonation of vapour hydrazine;

— at pressures in the range 1 bar to 5 bar, once detonation occurs, then pressure has no effect on the critical diameter, i.e. the RLE does not depend on pressure;

— the transition between 'non-reactive' and 'reactive' hydrazine occurs between 0,2 bar and 1 bar, possibly around 0,8 bar to 0,9 bar.

C.3.8 BLEs for a composite overwrapped pressure vessel

Preliminary power law BLEs have been developed for COPV using the failure criteria of rupture and perforation, respectively[40]. Both equations are listed in Table C.10, and the ballistic limit curve for rupture is plotted in Figure C.6. In this instance the impact conditions (y-axis) are parameterized as impactor momentum (non-dimensionalised).

Table C.10 — BLE for a composite overwrapped pressure vessel

Wall type	Failure criterion	Ballistic limit equation
Composite overwrapped pressure vessel	Rupture	$\left(\dfrac{m_{\mathrm{p}} v_{\mathrm{n}}}{\rho_{\mathrm{comp}} t_{\mathrm{tot}}^3 \left(\dfrac{\sigma_{\mathrm{u,comp}}}{\rho_{\mathrm{comp}}}\right)^{1/2}}\right)\left(\dfrac{\rho_{\mathrm{p}}}{\rho_{\mathrm{comp}}}\right)^{L_1}\left(1+\dfrac{\sigma_{\mathrm{u,lin}}}{100}\right)^{L_2}\left(1+\dfrac{\sigma_{\mathrm{y,lin}}}{100}\right)^{L_3} = A\left(\dfrac{\sigma_{\mathrm{h}}}{\sigma_{\mathrm{u,comp}}}\right)^{B}$
	Perforation of liner (front face)	$\left(\dfrac{m_{\mathrm{p}} v_{\mathrm{n}}}{\rho_{\mathrm{comp}} t_{\mathrm{tot}}^3 \left(\dfrac{\sigma_{\mathrm{u,comp}}}{\rho_{\mathrm{comp}}}\right)^{1/2}}\right)\left(\dfrac{\rho_{\mathrm{p}}}{\rho_{\mathrm{comp}}}\right)^{L_1}\left(1+\dfrac{\sigma_{\mathrm{u,lin}}}{100}\right)^{L_2}\left(1+\dfrac{\sigma_{\mathrm{y,lin}}}{100}\right)^{L_3} = p_0\,\dfrac{1-\mathrm{e}^{-D\left(1-\frac{\sigma_{\mathrm{h}}}{\sigma_{\mathrm{u,comp}}}\right)^{G}}}{1-\mathrm{e}^{-D}}$

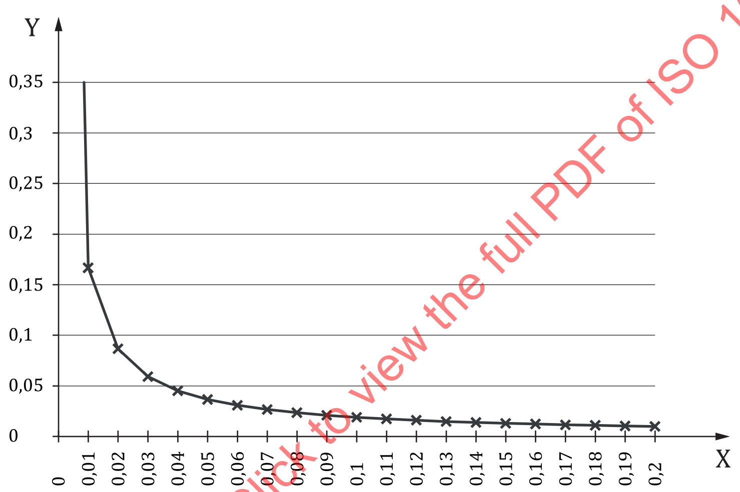
2) 1 bar = 0,1 MPa = 10^5 Pa; 1 MPa = 1 N/mm^2.

Care is needed when applying the BLEs in Table C.10. For example, some of the stress terms use the units of ksi whereas others use MPa.

Values for the power law terms are listed in Table C.11.

Table C.11 — BLE power law values

Term	Value
A	2,166 x 10^{-3}
B	-0,943
D	4,0
G	7,25
L_1	0,1
L_2	-1,0
L_3	-1,0
P_0	6,0 x 10^{-3}



Key

X hoop stress/composite ultimate tensile stress

Y non-dimensional projectile momentum

Figure C.6 — Rupture/non-rupture response of radially impacted composite overwrapped pressure vessels

C.4 Development of a ballistic limit equation

C.4.1 General

This clause relates to step 3.2 in case 1 and case 2a.

If a suitable ballistic limit equation cannot be identified for a particular surface or combination of surfaces, then it can be necessary to adapt an existing BLE or derive a new one. Adaptation of a BLE, such as one of those listed above, can be considered if the equation closely matches the particular shield / structure design of interest. Usually this is the case, but occasionally, if the design is sufficiently novel, a new BLE is necessary.

The most common way to satisfy either of these approaches is to perform a set of HVI tests. In the case of a BLE adaptation the number of impact shots is generally quite limited, since the purpose is just to provide a small number of ballistic limit reference data points against which the existing equation can be adjusted. By contrast, the development of a new BLE can require an extensive impact test program involving a wide variety of shots.

To adapt an existing BLE, one can consider the following limited impact test programme:

a) impact shots in each of the following three velocity ranges: the ballistic range (typically below ~3 km/s), the transition range (typically between ~3 km/s and ~7 km/s), and the hypervelocity range (typically above ~7 km/s);

b) impact shots at each of the following two angles: the angle perpendicular to the outermost surface and the mean angle of impact on the outermost surface, as determined from an analysis of the directional impact fluxes.

C.4.2 Hypervelocity impact testing

Ideally, a BLE should span the entire impact velocity range of on-orbit impacts, which is approximately 1 km/s to 16 km/s for space debris and 11 km/s to 72 km/s for meteoroids. Since laboratory hypervelocity launchers generally cannot accelerate millimetre-size projectiles much beyond 10 km/s, it is sometimes necessary to combine the laboratory experiments with numerical simulations (e.g. using hydrocodes) to characterize BLEs over the full velocity range. Therefore, HVI tests are necessary to (a) obtain the reference points of BLEs within the testable range and their verification, and (b) provide data for testing (i.e. verification, calibration) of the numerical codes (including models of material behaviour under HVI conditions).

The hypervelocity launchers normally used for impact testing are the following:

— one-stage powder guns;

— two-stage light-gas guns;

— electromagnetic launchers;

— electrostatic launchers;

— blast (explosive) launchers.

The following types of measurement technique can be employed:

— optical registration of the impact process (high frame-rate photography);

— X-ray registration of the impact process (if possible, multi-flash and multi-aspect X-ray);

— registration of dynamic pressures, stresses, and impulse by gauges placed into target;

— registration of time of arrival by contact gages;

— post-test study of damage (craters, holes, etc.).

Reference [1] provides information on several HVI launchers that are capable of simulating SD/M impacts on targets. These have been put through a series of calibration tests, defined by the IADC, to provide confidence in the results obtained.

C.4.3 Hydrocode modelling

In order to perform numerical simulation of fast transient events, innovative numerical methods have been under development since the early 1950s. These so-called hydrocodes or wave-propagation codes allow the study of the time-resolved progression of acoustic and shock wave propagation due to impact, penetration, or detonation in fluids and solids. This class of codes is fundamentally based on a spatial and time discretization of the impacting bodies into small elements to which the first principles or conservation equations for mass, momentum and energy are applied over small time steps. In hydrocodes, the first principles of physics are applied together with an equation of state to give the relationships between pressure, density, and internal energy. This provides a complete set of equations governing hydrodynamic behaviour.

Hydrocode modelling complements impact testing as a means of determining ballistic limit equations, particularly at the very high velocities that are characteristic of SD/M impacts. Before applying a hydrocode for this purpose, its applicability needs to be verified by comparison with experimental results. Reference [1] provides information on several hydrocode models which might be suitable. These have been undergoing a series of benchmark simulations defined by the IADC for the purpose of demonstrating that results obtained are comparable to those from impact tests.

C.5 Uncertainty considerations for ballistic limit equations

C.5.1 General

This clause relates to steps 3.1 and 3.2 in case 1 and case 2a.

The development of a ballistic limit equation based on hypervelocity impact tests is susceptible to a number of uncertainties. These include variations in the performance of the materials under test, changes in the test environment, and the accuracy of measurements. In the worst case, these uncertainties can give rise to completely contradictory results.

Ballistic limit equations are often derived by drawing lines through plots of impact test data to divide regions of penetration and non-penetration. Where there is no data, extrapolations are made based on certain assumptions. Thus, because BLEs are not statistical curve-fits, no information is available to calculate confidence intervals or uncertainty bounds. To overcome this problem, Reference [27] recommends that BLEs should be re-derived using a statistics-based approach to allow the uncertainty information to be extracted. For example, so-called damage predictor equations use statistical regression analysis of the test data to obtain a curve-fit. This means that uncertainty information can also be obtained when deriving the equation. However, there is still a limitation with the statistics-based approach; it does not provide uncertainty information outside the range of the available test data, e.g. for impact velocities beyond ~10 km/s.

It is also worth noting that the degree of uncertainty associated with a multi-wall BLE is likely to be significantly greater than that for a single wall BLE. Therefore, in terms of evaluating internal spacecraft damage, it is important to exercise caution when using such equations. Inevitably, a multi-wall BLE will produce conservative results, especially if it is designed to be used on a wide range of shield designs, such as the SRL equation. One obvious way to mitigate this problem, when using a BLE to investigate a particular multi-wall shield design, is to perform a small number of impact tests on the shield. The extra data provides anchor points thereby allowing the ballistic limit curve to be adjusted by an appropriate amount.

Generally, a BLE should only be used in circumstances consistent with the boundaries of the test conditions that led to its derivation. Use beyond those conditions can produce highly erroneous results. This is an important point to bear in mind when considering the suitability of a BLE. It is especially true when analysing a complex geometry such as a satellite. If the equation is used too far outside the range of test conditions from which it was derived, then the errors can be large enough to render the results useless.

A high level of precision in a BLE is generally desirable when the equation is to be used in an impact risk analysis. By contrast, a BLE that is biased towards engineering conservatism can be preferable when designing structures and shields since it guarantees a certain level of protection against SD/M impacts at the expense of a small amount of extra mass. Two important factors that can influence a BLE are impactor shape and the temperature at which a test is performed.

C.5.2 Particle shape

The shape of an SD/M particle can be described by its length-to-diameter (L/D) ratio. Generally, space debris particles of the order of a few millimetres have small L/D ratios. Thus, the particles have a reasonably compact shape and closely approximate a sphere. By contrast, space debris objects larger than approximately one centimetre in diameter have higher L/D ratios but are much less massive than a sphere of the same characteristic length.

Impact risk analyses generally assume that the shape of an SD/M particle can be described by its characteristic length, which is approximated by a sphere of the same diameter. This assumption is driven in part by the limitations of impact test facilities, which generally use spherical projectiles to derive the BLEs used in impact risk analyses. Unfortunately, the assumption is not well supported by analyses of the distribution of particle shapes from spacecraft fragmentation events [41] and recent laboratory impact test studies[42]. For example, Reference [43] shows that the shape of a particle and its orientation at impact can have a significant influence on the ballistic limit.

The effect of particle shape is currently an active area of research within the IADC, and should eventually lead to the development of new or modified particle shape-dependent BLEs. This will mitigate a long-standing source of uncertainty in impact risk analyses.

C.5.3 Temperature dependence

Impact tests have shown that the shielding performance of materials under hypervelocity impact varies with temperature. At cryogenic temperatures the impact damage on some aluminium alloys and composite materials is slightly less pronounced than at room temperature[44]. At higher temperatures, such as 110 °C and 210 °C, impact tests revealed that the hole diameters in bumpers were noticeably larger than those in bumpers at room temperature [45].

While some BLEs contain explicit terms to take into account the effects of temperature dependence, many do not. The effects of temperature are usually considered by modifying the values of material property terms in the equations.

Compared to particle shape effects, the uncertainty associated with temperature dependence is of less concern. Nevertheless, for some spacecraft there can be circumstances where it is necessary to consider this uncertainty.

Annex D
(informative)

Guidance for implementing impact protection on a spacecraft

D.1 General

This annex relates to step 5.1 in case 1 and case 2a.

The guidance in this annex is relevant for protecting unmanned spacecraft against impacts from small SD/M. It is based on information in Reference [1], a study called ReVuS, which is summarised in Reference [47], and knowledge of best practices in the industry. The guidance is given primarily from the point of view of implementing impact protection, and does not necessarily consider other aspects of the spacecraft design against which trade-offs might be necessary.

D.2 Considerations for evaluating impact protection solutions

The suitability of an SD/M impact protection solution for a spacecraft can be evaluated by considering:

a) its ability to reduce the effect of impacts on the spacecraft, e.g. by preventing penetrations that can cause the spacecraft to fail;

b) its affordability;

c) its level of implementation on the spacecraft, and the design phase at which it ought to be implemented;

d) its effect on other aspects of the spacecraft design, especially in terms of mass, complexity, geometric characteristics, structure, thermal control, and equipment layout;

e) its heritage, i.e. whether it has been implemented successfully on previous spacecraft;

f) its technical feasibility and maturity, i.e. whether it can be implemented in the near-term or long-term;

g) its effect on the spacecraft reliability, especially the level of redundancy;

h) its effect on the spacecraft assembly, integration and testing;

i) its applicability, i.e. the ease with which it can be designed and implemented on a spacecraft given the experience and capability of the manufacturer;

j) its ability to withstand the effects of the launch environment and the space environment;

k) its effect on spacecraft operations;

l) its demisability during uncontrolled re-entry of the spacecraft.

D.3 Guidance for implementing impact protection at the spacecraft level

D.3.1 General design measures

There are several possibilities for improving the survivability of a typical unmanned spacecraft without necessarily adding dedicated shielding mass. These involve design modifications at the spacecraft level. For example, the following general design measures can be considered:

a) modify the geometric characteristics of the spacecraft either to minimise the cross-sectional area of the most vulnerable surfaces in the direction of the highest SD/M impact flux or to change the angles of exposed vulnerable surfaces with respect to the flux;

b) take full advantage of the available structures on the spacecraft and add dedicated shielding layers only where necessary, for example, to protect those surfaces of the spacecraft, or its critical equipment, that are identified as being particularly vulnerable to impact;

c) optimise the layout of equipment, including harnesses, with respect to the direction of the highest impact flux, for example by locating critical or sensitive equipment items in the least vulnerable parts of the spacecraft, or hiding them behind other more robust items (shadowing), or positioning them further away from vulnerable surfaces, or re-orientating them;

d) compartmentalize the interior to mitigate the effect of a small SD/M impact on vulnerable equipment used in spacecraft disposal or on equipment containing a large amount of stored energy. The compartment structure can achieve this by acting as a bumper shield; it can also help to contain fragments resulting from the break-up of a pressurised vessel by a small SD/M impactor;

e) increase the redundancy of vulnerable equipment items used in spacecraft disposal to mitigate the effect of small SD/M impacts;

f) determine if it is better for an equipment item to have collocated redundancy or distributed redundancy to mitigate the effect of small SD/M impacts;

g) avoid placing redundant hardware in line with major flux vectors or close to one another if no major disadvantages on cable routing and thermal properties are expected; this is to reduce the risk of an impactor damaging both primary and back-up equipment items;

h) include automatic systems (e.g. automatic isolation valves and self-sealing bladders) to isolate the effects of impact damage, such as propellant leakage;

i) include an impact sensor network to detect impacts and guide operators in resolving any anomalies that can result;

j) include the capability to perform safe-mode operations to isolate and recover from the attitude control anomaly effect due to an SD/M impact;

k) include the capability to optimise the attitude profile of the spacecraft when it is exposed to an elevated impact risk, or when it is not required for operational purposes, to increase protection in the direction of the highest impact flux; note that normal attitude control operations might not be possible during such scenarios.

D.3.2 General design constraints

When designing the geometric characteristics of a spacecraft and the location and orientation of externally-mounted equipment, it is worth considering the possible damage effects from ejecta released by an oblique impact on an exposed surface. ISO 11227 [5] provides a test procedure for characterising the ejecta released from a surface when it is impacted by an SD/M particle.

When optimising the layout of equipment with respect to the impact flux direction, it is important to consider the following constraints:

a) maintaining a satisfactory centre of mass for the spacecraft;

b) ensuring the thermal requirements of equipment units are satisfied;

c) limiting the use of spacers and balance masses;

d) limiting the length of cables;

e) maintaining EMC between sensitive units or emitting magnetic units;

f) grouping units by functionality to reduce the overall harness mass and AIT complexity;

g) mounting heavy units on a stiff interface;

h) mounting moving parts that can induce vibrations as far as possible from the payload.

When considering physical separation of redundancies, it is important to balance the protection benefits against: a higher volume demand, additional harness mass, the necessity for supplementary connectors (thereby increasing mass and reducing reliability), more complex AIT sequences, and the necessity to protect switches relaying between the redundant units.

D.3.3 Subsystem-specific design measures

D.3.3.1 Propulsion

In view of the critical nature of a propellant tank, it can be advantageous to protect such an item from impacts by locating it in a part of the spacecraft that takes full advantage of the surrounding protection offered by other equipment, structures and any additional reinforcement that is deemed necessary.

If there are multiple tanks on a spacecraft, then it can be desirable to deplete and depressurise those in the more vulnerable locations first. Note, however, that there are several situations where this might not be feasible. For example:

a) when the mass balance properties of the spacecraft are adversely affected;

b) when the reliability of the propulsion system is adversely affected due to an increased risk of a propellant feeding line malfunction;

c) in the case of multiple bipropellant tanks it is not necessarily feasible to deplete one before another.

Alternatively, to reduce the break-up risk defined in case 2a, it can be better to decrease the maximum pressure across all of the tanks by using them equally. Careful study is necessary to select the correct option.

There can also be benefits in routing propulsion subsystem pipes to avoid vulnerable areas of a spacecraft providing certain design criteria are satisfied, such as: minimized length, residuals, pressure loss, position of valves and sensors, AIT and thermal aspects. If it is not possible to route pipes away from vulnerable areas, then it is worth considering whether to add shielding to protect the pipes.

The use of shut-off valves and containment to protect other equipment if a leak occurs is another important consideration.

Since thrusters are exposed items they are difficult to shield from SD/M impacts, and so the preferred protection approach for these items is to ensure there is sufficient redundancy.

D.3.3.2 Batteries

When pressurised battery cells are to be used on a spacecraft, it is important to subject them to the same impact design considerations as other pressurised vessels.

It is also important to select shielding for batteries based upon the type of cell to be protected, its failure modes, and the acceptability for loss of individual battery cells.

D.3.3.3 Avionics

It can be beneficial to decentralise the avionics functions by using a distributed architecture within the spacecraft.

D.3.3.4 Solar array

It is important to size a solar array to take account of the degradation in its performance, caused by the accumulation of damage from small SD/M impacts during the mission lifetime, so that the specified loss factor (x per year, in per cent) is satisfied.

When designing a solar array, it is worth considering the effect of different sizes of impact on the various parts of the solar array, such as an individual cell or a string of cells, so as to minimise the power loss.

It is important to incorporate a robust wiring architecture, with redundant electrical connections between cells, strings, modules, and arrays, and with suitable numbers of by-pass diodes to minimise power loss if a string of cells drops out of the power-generation circuit.

To enhance the survivability of a deployable solar array, one of two different approaches can be taken, either:

a) use thin, flexible arrays so that a penetrating particle can pass through causing relatively little damage; or

b) use toughened panels to absorb the kinetic energy of the particle. This can be achieved by toughening or laminating solar cell cover glasses, or adding MLI (or enhanced MLI) to the rear of the panel structure. MLI has demonstrated its ability to act as a protective bumper by absorbing the energy of small SD/M impactor. During impact, a small SD/M is progressively disrupted as it passes through the MLI layers, with the extent of failure increasing in the lower layers and a gradual reduction in the impactor's velocity.

D.3.3.5 Wire harness

If a wire harness is vulnerable to impact then its survivability can be enhanced either by adding shielding, designing in redundancy or routing it away from vulnerable areas of the spacecraft.

The following shielding methods merit consideration:

a) using thicker cable insulation;

b) adding a protective, flexible layer such as MLI or ceramic cloth, taking care not to compromise the thermal radiation requirements of the harness.

Within the constraints of mass and complexity, when designing a wire harness to have redundancy, it is worth considering the following:

— spatially separate the redundant cables to reduce the vulnerability to impact, especially in exposed areas where little or no additional protection can be provided;

— ensure the cables are of equal length (for resistance reasons);

— avoid inducing unwanted magnetic fields due to separate routing of the cables;

— since redundant cables meet at connection points on an equipment item, ensure these locations are well protected.

D.3.3.6 TT&C antenna

When positioning an antenna it is beneficial to consider the inherent protection provided by other parts of the spacecraft, whilst remaining in the proximity of associated hardware (to reduce cable length) and satisfying field-of-view and thermal requirements.

The use of redundant antennae, where feasible, also merits consideration.