



BSI Standards Publication

Space engineering — Spacecraft charging

National foreword

This British Standard is the UK implementation of EN 16603-20-06:2020. It supersedes BS EN 16603-20-06:2014, which is withdrawn.

The UK participation in its preparation was entrusted to Technical Committee ACE/68, Space systems and operations.

A list of organizations represented on this committee can be obtained on request to its committee manager.

This publication does not purport to include all the necessary provisions of a contract. Users are responsible for its correct application.

© The British Standards Institution 2020

Published by BSI Standards Limited 2020

ISBN 978 0 539 16096 3

ICS 49.140

Compliance with a British Standard cannot confer immunity from legal obligations.

This British Standard was published under the authority of the Standards Policy and Strategy Committee on 30 September 2020.

Amendments/corrigenda issued since publication

Date	Text affected
30 November 2020	Implementation of CEN/CENELEC correction notice 9 September 2020: correction to supersession information in European foreword

English version

Space engineering - Spacecraft charging

Ingénierie spatiale - Charges électrostatiques des engins spatiaux

Raumfahrttechnik - Aufladung von Raumfahrzeugen

This European Standard was approved by CEN on 3 May 2020.

This European Standard was corrected and reissued by the CEN-CENELEC Management Centre on 14 October 2020.

CEN and CENELEC members are bound to comply with the CEN/CENELEC Internal Regulations which stipulate the conditions for giving this European Standard the status of a national standard without any alteration. Up-to-date lists and bibliographical references concerning such national standards may be obtained on application to the CEN-CENELEC Management Centre or to any CEN and CENELEC member.

This European Standard exists in three official versions (English, French, German). A version in any other language made by translation under the responsibility of a CEN and CENELEC member into its own language and notified to the CEN-CENELEC Management Centre has the same status as the official versions.

CEN and CENELEC members are the national standards bodies and national electrotechnical committees of Austria, Belgium, Bulgaria, Croatia, Cyprus, Czech Republic, Denmark, Estonia, Finland, France, Germany, Greece, Hungary, Iceland, Ireland, Italy, Latvia, Lithuania, Luxembourg, Malta, Netherlands, Norway, Poland, Portugal, Republic of North Macedonia, Romania, Serbia, Slovakia, Slovenia, Spain, Sweden, Switzerland, Turkey and United Kingdom.



CEN-CENELEC Management Centre:
Rue de la Science 23, B-1040 Brussels

Table of contents

European foreword.....	9
Introduction.....	11
1 Scope.....	13
2 Normative references	14
3 Terms, definitions and abbreviated terms.....	15
3.1 Terms defined in other standards	15
3.2 Terms specific to the present standard	15
3.3 Abbreviated terms.....	18
3.4 Nomenclature	19
4 Overview.....	21
4.1 Plasma interaction effects.....	21
4.1.1 Presentation.....	21
4.1.2 Most common engineering concerns.....	21
4.1.3 Overview of physical mechanisms	22
4.2 Relationship with other standards	24
5 Protection programme	26
6 Surface material requirements	27
6.1 Overview	27
6.1.1 Description and applicability	27
6.1.2 Purpose common to all spacecraft	28
6.1.3 A special case: scientific spacecraft with plasma measurement instruments	28
6.2 General requirements	28
6.2.1 Maximum permitted voltage	28
6.2.2 Maximum resistivity	29
6.3 Electrical continuity, including surfaces and structural and mechanical parts.....	29
6.3.1 Grounding of surface metallic parts.....	29
6.3.2 Exceptions	30

6.3.3	Electrical continuity for surface materials	31
6.4	Surface charging analysis.....	35
6.5	Deliberate potentials	35
6.6	Testing of materials and assemblies.....	35
6.6.1	General	35
6.6.2	Material characterization tests.....	37
6.6.3	Material and assembly qualification.....	37
6.7	Scientific spacecraft with plasma measurement instruments	38
6.8	Verification.....	38
6.8.1	Grounding	38
6.8.2	Material selection	39
6.8.3	Environmental effects.....	39
6.8.4	Computer modelling	39
6.9	Triggering of ESD	40
7	Secondary arc requirements	41
7.1	Description and applicability	41
7.2	Solar arrays	42
7.2.1	Overview	42
7.2.2	General requirement	42
7.2.3	Testing of solar arrays.....	43
7.3	Other exposed parts of the power system including solar array drive mechanisms	47
8	High voltage system requirements	48
8.1	Description	48
8.2	Requirements	48
8.3	Validation.....	48
9	Internal parts and materials requirements	49
9.1	Description	49
9.2	General.....	49
9.2.1	Internal charging and discharge effects.....	49
9.2.2	Grounding and connectivity.....	49
9.2.3	Dielectric electric fields and voltages.....	50
9.3	Validation.....	51
10	Tether requirements	55
10.1	Description	55
10.2	General.....	55

10.2.1	Hazards arising on tethered spacecraft due to voltages generated by conductive tethers	55
10.2.2	Current collection and resulting problems	55
10.2.3	Hazards arising from high currents flowing through the tether and spacecraft structures.....	56
10.2.4	Continuity of insulation.	56
10.2.5	Hazards from undesired conductive paths	56
10.2.6	Hazards from electro-dynamic tether oscillations	56
10.2.7	Other effects	56
10.3	Validation.....	57
11	Electric propulsion requirements	58
11.1	Overview	58
11.1.1	Description.....	58
11.1.2	Coverage of the requirements.....	58
11.2	General.....	60
11.2.1	Spacecraft neutralization.....	60
11.2.2	Beam neutralization	61
11.2.3	Contamination.....	62
11.2.4	Sputtering	62
11.2.5	Neutral gas effects	62
11.3	Validation.....	63
11.3.1	Ground testing	63
11.3.2	Computer modelling characteristics	63
11.3.3	In-flight monitoring.....	63
11.3.4	Sputtering	63
11.3.5	Neutral gas effects	64
Annex A	(normative) Electrical hazard mitigation plan - DRD	65
A.1	DRD identification	65
A.1.1	Requirement identification and source document.....	65
A.1.2	Purpose and objective.....	65
A.2	Expected response	65
A.2.1	Scope and content	65
A.2.2	Special remarks	66
Annex B	(informative) Tailoring guidelines.....	67
B.1	Overview	67
B.2	LEO	67
B.2.1	General.....	67

B.2.2	LEO orbits with high inclination	68
B.3	MEO and GEO orbits.....	68
B.4	Spacecraft with onboard plasma detectors	68
B.5	Tethered spacecraft.....	69
B.6	Active spacecraft	69
B.7	Solar Wind	69
B.8	Other planetary magnetospheres.....	69
Annex C (informative)	Physical background to the requirements	70
C.1	Introduction.....	70
C.2	Definition of symbols.....	70
C.3	Electrostatic sheaths.....	70
C.3.1	Introduction	70
C.3.2	The electrostatic potential	71
C.3.3	The Debye length.....	71
C.3.4	Presheath	72
C.3.5	Models of current through the sheath.....	73
C.3.6	Thin sheath – space-charge-limited model.....	73
C.3.7	Thick sheath – orbit motion limited (OML) model	74
C.3.8	General case.....	75
C.3.9	Magnetic field modification of charging currents.....	75
C.4	Current collection and grounding to the plasma	75
C.5	External surface charging	76
C.5.1	Definition.....	76
C.5.2	Processes	76
C.5.3	Effects.....	77
C.5.4	Surface emission processes	77
C.5.5	Floating potential.....	78
C.5.6	Conductivity and resistivity	79
C.5.7	Time scales.....	81
C.6	Spacecraft motion effects	81
C.6.1	Wakes.....	81
C.6.2	Motion across the magnetic field	84
C.7	Induced plasmas	85
C.7.1	Definition.....	85
C.7.2	Electric propulsion thrusters	86
C.7.3	Induced plasma characteristics	86
C.7.4	Charge-exchange effects	87

C.7.5	Neutral particle effects	88
C.7.6	Effect on floating potential.....	88
C.8	Internal and deep-dielectric charging	88
C.8.1	Definition.....	88
C.8.2	Relationship to surface charging	89
C.8.3	Charge deposition	90
C.8.4	Material conductivity.....	90
C.8.5	Time dependence	93
C.8.6	Geometric considerations.....	93
C.8.7	Isolated internal conductors	94
C.8.8	Electric field sensitive systems	94
C.9	Discharges and transients	95
C.9.1	General definition	95
C.9.2	Review of the process.....	95
C.9.3	Dielectric material discharge	96
C.9.4	Metallic discharge	98
C.9.5	Internal dielectric discharge.....	99
C.9.6	Secondary powered discharge	100
C.9.7	Discharge thresholds	100
Annex D (informative)	Charging simulation.....	102
D.1	Surface charging codes	102
D.1.1	Introduction	102
D.2	Internal charging codes	104
D.2.1	DICTAT.....	104
D.2.2	ESADDC.....	104
D.2.3	GEANT-4	105
D.2.4	NOVICE	105
D.3	Environment model for internal charging.....	105
D.3.1	FLUMIC	105
D.3.2	Worst case GEO spectrum.....	105
Annex E (informative)	Testing and measurement.	106
E.1	Definition of symbols.....	106
E.2	Solar array testing.....	106
E.2.1	Solar cell sample.....	106
E.2.2	Pre-testing of the solar array simulator (SAS)	107
E.2.3	Solar array test procedure.....	109
E.2.4	Other elements	113

E.2.5	The solar panel simulation device	114
E.3	Measurement of conductivity and resistivity	116
E.3.1	Determination of intrinsic bulk conductivity by direct measurement	116
E.3.2	Determination of radiation-induced conductivity coefficients by direct measurement	117
E.3.3	Determination of conductivity and radiation-induced conductivity by electron irradiation.....	118
E.3.4	The ASTM method for measurement of surface resistivity and its adaptation for space used materials.....	118
References		120
Bibliography		124

Figures

Figure 6-1: Applicability of electrical continuity requirements	32
Figure 7-1: Solar array test set-up	45
Figure C-1 : Schematic diagram of potential variation through sheath and pre-sheath.	72
Figure C-2 : Example secondary yield curve	78
Figure C-3 : Schematic diagram of wake structure around an object at relative motion with respect to a plasma.....	82
Figure C-4 : Schematic diagram of void region.....	83
Figure C-5 : Schematic diagram of internal charging in a planar dielectric.....	89
Figure C-6 : Dielectric discharge mechanism.	97
Figure C-7 :Shape of the current in relation to discharge starting point.....	97
Figure C-8 : Example of discharge on pierced aluminized Teflon® irradiated by electrons with energies ranging from 0 to 220 keV.	98
Figure C-9 : Schematic diagram of discharge at a triple point in the inverted voltage gradient configuration with potential contours indicated by colour scale.	99
Figure E-1 : Photograph of solar cells sample – Front face & Rear face (Stentor Sample. Picture from Denis Payan - CNES®).....	107
Figure E-2 : Schematic diagram of power supply test circuit.....	108
Figure E-3 : Example of a measured power source switch response.....	108
Figure E-4 : Example solar array simulator.....	109
Figure E-5 : Absolute capacitance of the satellite	110
Figure E-6 : Junction capacitance of a cell versus to voltage.....	112
Figure E-7 : The shortened solar array sample and the missing capacitances	113
Figure E-8 : Discharging circuit oscillations	114
Figure E-9 : Effect of an added resistance in the discharging circuit (SAS + resistance)	114
Figure E-10 : Setup simulating the satellite including flashover current	115

Figure E-11 : Basic arrangement of apparatus for measuring dielectric conductivity in planar samples.....	116
Figure E-12 : Arrangement for measuring cable dielectric conductivity and cross-section through co-axial cable	116
Figure E-13 : Arrangement for carrying out conductivity tests on planar samples under irradiation	118
Figure E-14 : Basic experimental set up for surface conductivity	119

Tables

Table 4-1: List of electrostatic and other plasma interaction effects on space systems.....	23
Table 7-1: Tested voltage-current combinations	42
Table 7-2: Typical inductance per unit length for cables	46
Table C-1 : Parameters in different regions in space	72
Table C-2 : Typical plasma parameters for LEO and GEO	83
Table C-3 : Plasma conditions on exit plane of several electric propulsion thrusters	87
Table C-4 : Emission versus backflow current magnitudes for several electric propulsion thrusters.....	87
Table C-5 : Value of E_a for several materials	91

European foreword

This document (EN 16603-20-06:2020) has been prepared by Technical Committee CEN-CENELEC/TC 5 "Space", the secretariat of which is held by DIN.

This document (EN 16603-20-06:2020) originates from ECSS-E-ST-20-06C Rev.1.

This European Standard shall be given the status of a national standard, either by publication of an identical text or by endorsement, at the latest by March 2021, and conflicting national standards shall be withdrawn at the latest by March 2021.

Attention is drawn to the possibility that some of the elements of this document may be the subject of patent rights. CEN [and/or CENELEC] shall not be held responsible for identifying any or all such patent rights.

This document supersedes 16603-20-06:2014.

The main changes with respect to 16603-20-06:2014 are listed below:

- Addition of definition for the term "flashover current"
- Addition of abbreviated terms "RIC" and "SPIS"
- Addition of the "Nomenclature" in clause 3.4
- Addition of informative text in 6.1.3 about neutralizers after deletion of requirement 6.7e
- Changes to maximum permitted voltages and acceptance of higher voltages if the effect of worst-case ESD would be acceptable
- Simplification of the grounding requirements for surface materials
- Change in the permitted ESD energy where surface ESD cannot be excluded
- Change to explicitly allow surface charging analysis to justify acceptance
- For surface charging analysis, removal of acceptance by similarity and statement of the need for material testing
- For internal charging, completely embedded floating metals may be accepted under specified conditions

This document has been prepared under a standardization request given to CEN by the European Commission and the European Free Trade Association.

This document has been developed to cover specifically space systems and has therefore precedence over any EN covering the same scope but with a wider domain of applicability (e.g. : aerospace).

According to the CEN-CENELEC Internal Regulations, the national standards organizations of the following countries are bound to implement this European Standard: Austria, Belgium, Bulgaria, Croatia, Cyprus, Czech Republic, Denmark, Estonia, Finland, Former Yugoslav Republic of Macedonia, France, Germany, Greece, Hungary, Iceland, Ireland, Italy, Latvia, Lithuania, Luxembourg, Malta, Netherlands, Norway, Poland, Portugal, Romania, Serbia, Slovakia, Slovenia, Spain, Sweden, Switzerland, Turkey and the United Kingdom..

Introduction

The subject of spacecraft plasma interactions has been part of the spacecraft design process since spacecraft surface charging was first encountered as a problem in the earliest geostationary spacecraft. However, spacecraft surface charging is only one of the ways in which the space environment can adversely affect the electrical state of spacecraft and satellite technology has evolved over the years.

A need was identified for a standard that is up to date and comprehensive in its treatment of all the main environment-induced plasma and charging processes that can affect the performance of satellites in geostationary and medium and low Earth orbits. This standard is intended to be used by a number of users, with their own design rules, and therefore it has been done to be compatible with different alternative approaches.

This document aims to satisfy these needs and provides a consistent standard that can be used in design specifications. The requirements are based on the best current understanding of the processes involved and are not radical, building on existing de-facto standards in many cases.

As well as providing requirements, it aims to provide a straightforward brief explanation of the main effects so that interested parties at all stages of the design chain can have a common understanding of the problems faced and the meaning of the terms used. Guide for tailoring of the provisions for specific mission types are described in Annex B. Further description of the main processes are given in Annex C. Some techniques of simulation, testing and measurement are described in Annex D and Annex E.

Electrical interactions between the space environment and a spacecraft can arise from a number of external sources including the ambient plasma, radiation, electrical and magnetic fields and sunlight. The nature of these interactions and the environment itself can be modified by emissions from the spacecraft itself, e.g. electric propulsion, plasma contactors, secondary emission and photoemission. The consequences, in terms of hazards to spacecraft systems depend strongly on the sensitivity of electronic systems and the potential for coupling between sources of electrical transients and fields and electronic components.

Proper assessment of the effects of these processes is part of the system engineering process as defined in ECSS-E-ST-20. General assessments are performed in the early phases of a mission when consideration is given to e.g. orbit selection, mass budget, thermal protection, and materials and component selection policy. Further into the design of a spacecraft, careful consideration is given to material selection, coatings, radiation shielding and electronics protection.

This standard begins with an overview of the electrical effects occurring in space (Clause 4). The requirements, in terms of spacecraft testing, analysis and design that arise from these processes (Clause 5 to Clause 11) form the core of this document. Annex B holds a discussion of types of orbits and how to tailor the requirements according to the mission. Annex C discusses the quantitative assessment of the physical processes behind these main effects. Annex D describes computer simulations and Annex E describes testing and measurement.

1

Scope

This standard is a standard within the ECSS hierarchy. It forms part of the electrical and electronic engineering discipline (ECSS-E-ST-20) of the engineering branch of the ECSS system (ECSS-E). It provides clear and consistent provisions to the application of measures to assess, in order to avoid and minimize hazardous effects arising from spacecraft charging and other environmental effects on a spacecraft's electrical behaviour.

This standard is applicable to any type of spacecraft including launchers, when above the atmosphere.

Although spacecraft systems are clearly subject to electrical interactions while still on Earth (e.g. lightning and static electricity from handling), these aspects are not covered, since they are common to terrestrial systems and covered elsewhere. Instead this standard covers electrical effects occurring in space (i.e. from the ionosphere upwards).

This standard may be tailored for the specific characteristic and constraints of a space project in conformance with ECSS-S-ST-00.

2

Normative references

The following normative documents contain provisions which, through reference in this text, constitute provisions of this ECSS Standard. For dated references, subsequent amendments to, or revision of any of these publications do not apply. However, parties to agreements based on this ECSS Standard are encouraged to investigate the possibility of applying the more recent editions of the normative documents indicated below. For undated references, the latest edition of the publication referred to applies.

EN reference	Reference in text	Title
EN 16601-00-01	ECSS-S-ST-00-01	ECSS system - Glossary of terms

Terms, definitions and abbreviated terms

3.1 Terms defined in other standards

For the purpose of this Standard, the terms and definitions from ECSS-S-ST-00-01 apply.

3.2 Terms specific to the present standard

3.2.1 aluminium equivalent thickness

thickness of aluminium with a mass density per unit area equal to that of the material being described

NOTE The mass density is normally measured in (g cm^{-2}).

3.2.2 auroral zone

region at a latitude between 60 and 70 degrees north or south where aurorae are formed

3.2.3 deep-dielectric charging

electrical charge deposition within the bulk of an external or internal material

3.2.4 dielectric

pertaining to a medium in which an electric field can be maintained

NOTE Depending on their resistivity, dielectric materials can be described as insulating, antistatic, moderately conductive or conductive. The following gives a classic example of classification according to the resistivity:

- more than $10^9 \Omega \text{ m}$: insulating
- between $10^2 \Omega \text{ m}$ and $10^9 \Omega \text{ m}$: antistatic
- between $10^3 \Omega \text{ m}$ and $10^6 \Omega \text{ m}$: static dissipative
- between $10^{-2} \Omega \text{ m}$ and $10^2 \Omega \text{ m}$: moderately conductive
- less than $10^{-2} \Omega \text{ m}$: conductive

3.2.5 dose

energy absorbed locally per unit mass as a result of **radiation** exposure

3.2.6 downstream

<relating to an object at relative motion with respect to a **plasma**>
on side of an object in the same direction as the **plasma** velocity vector

3.2.7 electrostatic

pertaining to static electricity or electricity at rest

3.2.8 electrostatic breakdown

failure of the insulation properties of a **dielectric**, resulting in a sudden release of charge and risk of damage to the **dielectric** concerned

3.2.9 electrostatic discharge

rapid, spontaneous transfer of electrical charge induced by a high **electrostatic** field

3.2.10 external charging

electric charge deposition on external materials

3.2.11 flashover current

current arising from a surface propagating discharge

3.2.12 fluence

time-integration of the flux

3.2.13 insulator

insulating **dielectric**

3.2.14 internal charging

electrical charge deposition on internal materials shielded at least by the spacecraft skin due to penetration of charged particles from the ambient medium

NOTE Materials can be conductors or dielectrics.

3.2.15 internal dielectric charging

internal charging of **dielectric** materials

3.2.16 ion engine

propulsion system which operates by expelling ions at high velocities

3.2.17 L shell

parameter of the geomagnetic field

NOTE 1 It is also referred as L, and is used as a co-ordinate to describe positions in near-Earth space.

NOTE 2 L or L shell has a complicated derivation based on an invariant of the motion of charged particles in the terrestrial magnetic field. However, it is useful in defining plasma regimes within the magnetosphere because, for a dipole magnetic field, it is equal to the geocentric altitude in Earth-radii of the local magnetic field line where it crosses the equator.

3.2.18 omnidirectional flux

scalar integral of the flux over all directions

NOTE This implies that no consideration is taken of the directional distribution of the particles which can be non-isotropic. The flux at a point is the number of particles crossing a sphere of unit cross-sectional surface area (i.e. of radius). An omnidirectional flux is not to be confused with an isotropic flux.

3.2.19 outgassing rate

mass of molecular species evolving from a material per unit time and unit surface area

NOTE The units of outgassing rates are $\text{g cm}^{-2} \text{s}^{-1}$. It can also be given in other units, such as in relative mass unit per time unit: (g s^{-1}), ($\% \text{s}^{-1}$) or ($\% \text{s}^{-1} \text{cm}^{-2}$).

3.2.20 plasma

partly or wholly ionized gas whose particles exhibit collective behaviour through its electromagnetic field

3.2.21 primary discharge

initial electrostatic discharge which, by creating a temporary conductive path, can lead to a secondary arc

3.2.22 radiation

transfer of energy by means of a particle (including photons)

NOTE In the context of this Standard, electromagnetic radiation below the UV band is excluded. This therefore excludes visible, thermal, microwave and radio-wave radiation.

3.2.23 radiation belt

area of trapped or quasi-trapped energetic particles, contained by the Earth's magnetic field

3.2.24 ram

volume adjacent to the spacecraft and located in the same direction of the spacecraft motion where modification to the surface or **plasma** can occur due to the passage of the spacecraft through the medium

3.2.25 secondary arc

passage of current from an external source, such as a solar array, through a conductive path initially generated by a **primary discharge**

3.2.26 surface charging

electrical charge deposition on the surface of an external or internal material

3.2.27 tether

flexible conductive or non-conductive cable linking two spacecraft or two parts of the same spacecraft not mechanically attached in any other way

3.2.28 thruster

device for altering the attitude or orbit of a spacecraft in space through reaction

NOTE E.g. rocket, cold-gas emitter, and electric propulsion.

3.2.29 triple point

<relating to onset of electrostatic discharge>
point where **dielectric**, metal and vacuum meet

3.2.30 upstream

<relating to an object at relative motion with respect to a **plasma**>
on the side of the object in the opposite direction to the **plasma** velocity vector

3.2.31 wake

volume adjacent to a spacecraft and located in the opposite direction to the spacecraft motion where the ambient **plasma** is modified by the passage of the spacecraft through the medium

3.3 Abbreviated terms

The following abbreviated terms are defined and used within this Standard.

Abbreviation	Meaning
AOCS	attitude and orbital control system
DGD	direct gradient discharge
EMC	electromagnetic compatibility
emf	electro-motive force
EP	electric propulsion
ESD	electrostatic discharge
ETFE	ethylene-tetrafluoroethylene copolymer
eV	electron volt (also keV, MeV)
FEPP	field emission electric propulsion
FEP	fluoroethylene-propylene

GEO	geostationary Earth orbit
HEO	highly eccentric orbit
ISS	International Space Station
IVG	inverted voltage gradient
IVGD	inverted voltage gradient discharge
LEO	low Earth orbit
MEMS	micro-electromechanical system(s)
MEO	medium (altitude) Earth orbit
MLI	multi-layer insulation
MLT	magnetic local time
NASA	National Aeronautics and Space Administration
NGD	normal gradient discharge
PCB	printed circuit board
PEO	polar Earth orbit
PTFE	poly-tetrafluoroethylene
PVA	photo-voltaic assembly
RIC	radiation induced conductivity
r.m.s.	root-mean-square
SAS	solar array simulator
SPIS	spacecraft plasma interaction simulation
SPT	stationary plasma thruster
SSM	second surface mirror
UV	ultra-violet light

3.4 Nomenclature

The following nomenclature applies throughout this document:

- a. The word “shall” is used in this Standard to express requirements. All the requirements are expressed with the word “shall”.
- b. The word “should” is used in this Standard to express recommendations. All the recommendations are expressed with the word “should”.

NOTE It is expected that, during tailoring, recommendations in this document are either converted into requirements or tailored out.

- c. The words “may” and “need not” are used in this Standard to express positive and negative permissions, respectively. All the positive permissions are expressed with the word “may”. All the negative permissions are expressed with the words “need not”.
- d. The word “can” is used in this Standard to express capabilities or possibilities, and therefore, if not accompanied by one of the previous words, it implies descriptive text.

NOTE In ECSS “may” and “can” have completely different meanings: “may” is normative (permission), and “can” is descriptive.

- e. The present and past tenses are used in this Standard to express statements of fact, and therefore they imply descriptive text.

4

Overview

4.1 Plasma interaction effects

4.1.1 Presentation

Most spacecraft engineers have come across the term space plasma because of the potential risk it represents in leading to high-level spacecraft charging and related electrostatic discharges. Such a phenomenon is a critical environment hazards for satellites, especially in high altitude orbits (including in geostationary orbit). However, there are numerous other plasma interaction effects of which it is also important to be aware, in order to improve engineering design and to optimize the behaviour of space systems in the space plasma environment. Not all spacecraft-plasma interactions are adverse. For example, electrodynamic tethers use current collection from the plasma to complete their electrical circuit and generate power or thrust. Scientific instruments can exploit spacecraft plasma interactions to make plasma measurements, and a magnetospheric propulsion system has been proposed to use plasma interactions with an artificial magnetic field as a propulsion system.

This clause briefly introduces the various effects covered by this standard. The most common engineering issues are first briefly described in clause 4.1.2 while a short overview of the physical mechanisms involved is given in the second of this section.

4.1.2 Most common engineering concerns

The following are the most common engineering concerns related to electrical charging of a spacecraft:

- **Surface charging** due to charge accumulation on spacecraft surface is a potential cause of spacecraft anomalies. High levels of differential potential can lead to ESDs which can couple to spacecraft electronics and damage electronic components. Initial ESDs can also trigger the generation of secondary arcs on high voltage systems. Several occurrences of permanent solar array power loss in space are believed to be related to such phenomena.
- **Internal charging** due to penetrating electrons is another cause of spacecraft anomalies. ESDs can be generated within the spacecraft Faraday cage and in close proximity to vulnerable components.

- **Current leakage** affects exposed high voltage systems which can experience significant power loss due to current leakage sustained by freely moving particles (from the plasma or photo-electrons).
- **Environment modification** occurs as a bi-product of surface charging and/or particle emission by surfaces. Electric charge and plasma density are affected in a volume that can extend very far from the spacecraft (many times its typical dimension). This is usually a critical problem for scientific plasma measurements that cannot retrieve the natural parameters to be measured without taking these alterations into account.
- **Electric propulsion** actively modifies the local environment and creates a local plasma population and modifies the current balance of the satellite that otherwise can occur. It can affect other spacecraft systems, e.g. by increased contamination.
- **Electrostatic tethers** make use of current collection from the ambient medium to allow a current to pass. These also change the currents passing through the system and can lead to high potentials.

Table 4-1 contains a more complete list of the various effects in which space plasma plays an important role.

A quantitative description of the most important processes in spacecraft-plasma interactions is given in Annex C but a brief qualitative overview follows here.

4.1.3 Overview of physical mechanisms

Surface charging occurs because electric charges (electrons and ions) of the plasma are free to move and eventually get trapped on material surfaces when they hit them. Electrons and ions provide negative and positive current respectively. Both electrons and ions can also expel electrons when they hit the surface, providing an additional positive current. UV and soft X-ray photons coming from the Sun also have enough energy to expel electrons from materials, thereby providing a positive current. The accumulation of charge (positive or negative) leads to the creation of a potential that eventually prevents further charge accumulation by repelling charges of a given polarity and attracting charges of opposite polarity. Different surfaces on the same spacecraft can charge to very different potentials, resulting in strong local electric fields. Photo-emission and secondary yield are important factors in determining occurrence of high level differential charging. Charging effect is highly dynamic as the plasma populations can change rapidly.

Internal charging occurs because higher energy electrons are able to penetrate thin surfaces and deposit charge in or on materials within the spacecraft. Internal charging involves lower currents but potentials can build up over longer periods on materials with low resistivity because current through the structure is the only mechanism by which charge can be removed. Strong electric fields again are the result.

An important hazard related to charging (surface and internal) is sudden electrostatic discharge (ESD). ESDs are most likely initiated by field effect emission and avalanche process at so-called 'triple-points', where metal, dielectric and vacuum are found together. This can eventually result in the injection of high current electrical transients into the spacecraft electrical

system. This can cause transient state changes in electronics or permanent damage of material coating or electronic components. In the vicinity of high potential differences, plasma released in an ESD can trigger a secondary discharge or even a sustained arc if the potential is maintained e.g. by the solar array.

Table 4-1: List of electrostatic and other plasma interaction effects on space systems

Spacecraft reference potential change (e.g. so-called spacecraft high voltage charging)
Differential surface charging
Internal surface charging
Deep-dielectric charging
Surface charging effects on e.g. plasma instruments
Charging interactions with thruster plumes
Charging induced sputtering
Charging induced contamination
Dust charging and sticking
Electrostatic sticking (SA, inflatables, solar sails, dusts)
Charging differences in docking procedures and EVA
Material modifications (e.g. electric field induced conductivity, plasma-contaminant modifications, cracking e.g. of ITO)
Tether systems and $\mathbf{v} \times \mathbf{B}$ induced voltages and currents
Interference induced by electrostatic sheath and induced plasma on EM systems
Spontaneous discharges
Triggered discharges: - Thruster-triggered - Low voltage discharges (e.g. Paschen and secondaries) - Sunlit/shadow triggered discharges - triggered by micro particles - triggered by natural high density plasma
Synergistic secondary low-voltage discharges (e.g. on solar array)
Power loss and current flow from exposed elevated voltage systems
Photoelectron cloud induced charging and disturbance of plasma sensors
Plasma perturbation from microparticle impacts (e.g. antennas)
Active plasma sources (e.g. emitters, thrusters, contactors)

If a surface is maintained at high potential, it draws a current from the plasma, particularly in the dense LEO environment. This is an issue with exposed solar array interconnects because this leakage current drains power from the array. In addition the floating potential of the spacecraft can be altered as a result.

The magnitude of spacecraft plasma interaction effects varies greatly with the changes in the ambient plasma from one type of Earth orbit to another. Also the plasma environment can exhibit temporal changes along the orbit and as a result of so-called 'Space Weather'. In addition, the spacecraft itself can modify the plasma environment e.g. via thruster firings, active experiments, or debris impacts. Other aspects of the space environment besides plasma can also influence the plasma interaction e.g. radiation, contamination, and temperature.

In any case, it is important to have access to a good quantitative description of the space environment. Suitable descriptions and models of the space environment, including aspects that control charging effects, can be found in ECSS-E-ST-10-04.

4.2 Relationship with other standards

This standard is a standard within the engineering branch. It relies on the following ECSS standards:

ECSS-E-ST-10 "Space engineering – System engineering general requirements"

ECSS-E-ST-20 "Space engineering - Electrical and electronic"

A number of ECSS standards are related to this Standard:

- The environment, defined by ECSS-E-ST-10-04 "Space engineering - Space environment", is an input to the analysis of charging and related effects described in this document.
- Radiation transport as described in ECSS-E-ST-10-12 "Space engineering - Method for the calculation of radiation received and its effects, and a policy for design margins" has relevance to internal charging effects described in this document.
- Secondary electron emission is a feature of surface charging effects described in this document and in multipaction as described in ECSS-E-ST-20-01 "Space engineering – Multipactor design and test".
- Testing of electrostatic behaviour shares many techniques with general electromagnetic cleanliness. Hence ECSS-E-ST-20-07 "Space engineering – Electromagnetic compatibility" interfaces with this standard.
- Charging of solar arrays is an important charging hazard. Hence ECSS-E-ST-20-08 "Space engineering – Photovoltaic assemblies and components" interfaces with this standard.
- This standard specifies the selection of materials and components according to their electrical behaviour, in order to avoid hazardous effects. Electrical properties are not the only consideration and ECSS-Q-ST-70-71 "Space product assurance - Materials, processes and their data selection" describe complementary information.
- ECSS-E-ST-10-02 "Space engineering – Verification" specifies the concepts behind verification processes in general, the criteria for defining a verification strategy and rules for its implementation. It includes a description of the documentation associated to the verification process, and provisions on specific aspects of the verification process.

- ECSS-E-ST-10-03 “Space engineering – Testing” gives standard environmental and performance test requirements for a space system and its constituents. It defines the general test requirements for products and systems that are generally applicable to all projects. It also defines the documentation associated with testing activities.

5

Protection programme

- a. For any spacecraft, the supplier shall establish, at the start of the project, a plan for the assessment and mitigation of hazards associated with electrical interactions with the environment in conformance with Annex A.
- b. The compliance of the design with the design requirements shall be verified through measurement, inspection and testing.

6

Surface material requirements

6.1 Overview

6.1.1 Description and applicability

The phenomenon of surface charging leads to requirements to control the electrical behaviour of materials directly in contact with the ambient plasma. These materials can become the source of electrostatic discharges whose principal hazardous effect is to cause damage to electronic systems elsewhere on the spacecraft and of undesired electric fields, whose principal effect is to interfere with measurements of the ambient medium. The components affected include any spacecraft material on the outer surface of the spacecraft, e.g. thermal blankets, solar cell cover glass, sensor optics, exposed cables.

In addition, thin films, e.g. thermal blankets, sometimes do not shield underlying materials from the electrons mainly responsible for high level surface charging (~ 10 keV to 50 keV). Hence components affected include any spacecraft material covered by only a thin film, with thickness equivalent to less than 100 micro-meters Aluminium ($0,03 \text{ g/cm}^2$), including multi-layer insulation. The requirements in this clause also apply to surfaces partly exposed to space through small apertures (e.g., venting holes), although holes can be designed so as to avoid direct exposure.

Another factor influencing charging levels is the conduction path from each surface to ground and to other parts on the spacecraft. Hence these requirements cover all components concerned with the grounding and electrical continuity of the spacecraft structure, subsystems and electronics.

In addition to considering components that affect the build-up of charge, the effects of any resulting electrostatic discharges, i.e. coupling to sensitive devices, are controlled in order to prevent discharges causing damage and interference to systems elsewhere. This is an EMC issue that is common to electrical transients, however generated. Because this is part of general EMC design, specific requirements are not included here but standard EMC procedures are applied to control the transmission of transients.

These requirements also cover solar arrays and there are additional specific requirements for these due to the possibility of secondary discharges, fed by the power of the array itself, in clause 7.

6.1.2 Purpose common to all spacecraft

The main purpose of the requirements in this clause is to reduce differential charging and therefore the risk of electrostatic discharge or its power below an acceptable level. This is achieved principally by effective grounding throughout the spacecraft.

6.1.3 A special case: scientific spacecraft with plasma measurement instruments

Scientific spacecraft, which have the measurement of ambient electric field or of low energy particles as part of their mission, have more stringent requirements in terms of differential charging, absolute spacecraft potentials and contamination by spacecraft generated particles. For this reason, clause 6.7 provides specific requirements for this case.

Since quantitative requirements are often mission specific, the requirements in clause 6.7 are not intended to be exhaustive, but to cover the common aspects that can be expected in most of the scientific missions.

For each science mission the scientific community assesses the need for a neutralizer (ion or electron emitter) is used to mitigate the positive or negative floating potential of the spacecraft. This device can be supplied as part of the scientific payload or included in the supplier contract.

Plasma contactors are generally avoided on scientific spacecraft because of the large scale disturbance they induce on scientific measurements.

In practice the reduction of photo-electron and secondary electron emission is often achievable by active potential control techniques only.

6.2 General requirements

6.2.1 Maximum permitted voltage

- a. In order to avoid electrostatic discharge, the following critical voltages and electric fields shall be used:
 1. To avoid electrostatic discharges, a maximum electric field within the dielectric of 10^7 V/m.
 2. To avoid electrostatic discharges in normal potential gradient a maximum differential voltage between a dielectric and an adjacent conductor of -2000 V.
 3. To avoid an electrostatic discharge in inverted voltage gradient, a maximum differential voltage between a dielectric and an adjacent conductor exposed to the vacuum of +300 V.

NOTE 1 Maximum permitted voltages specified for scientific missions including field or plasma measurement can be lower than the limits above to limit contamination of environmental measurements.

NOTE 2 An inverted voltage gradient is between a dielectric and a conductor at a more negative potential.

NOTE 3 A normal voltage gradient is between a dielectric and a conductor at a more positive potential.

NOTE 4 These voltage limits do not include margin for the possible inaccuracy of analysis by simulation.

NOTE 5 Electric fields can be locally enhanced by sharp edges.

- b. If the limits specified in 6.2.1a are not met, the assembly shall be submitted to charging tests as described in clause 6.6.
- c. The test specified in requirement 6.2.1b may be replaced by an ESD analysis under the condition that the ESD responsible agrees that the result of this analysis demonstrate that the effect of such a worst-case ESD is acceptable.

6.2.2 Maximum resistivity

- a. In order to limit differential voltage, a maximum current density of $J = 10 \text{ nA/cm}^2$ through the dielectric shall be used to calculate the maximum resistivity of dielectric materials.

NOTE This current is a recommended upper bound of the current through dielectric material to other parts on Earth orbiting spacecraft. It is not the electron irradiation current used for testing.. For example, a small sunlit area of aluminium coupled to a larger area of material in eclipse can lead to a current flow through a dielectric beneath the aluminium as large as the photocurrents ($\sim 4 \text{ nA/cm}^2$ for aluminium) plus any difference in secondary emission current.

- b. The method used for resistivity measurement shall be justified.

NOTE Some methods are described in Annex D.3.

6.3 Electrical continuity, including surfaces and structural and mechanical parts

6.3.1 Grounding of surface metallic parts

- a. To prevent ESD events, all structural and mechanical parts shall be electrically bonded to each other, ensuring that, in order to control differential charging, there is a resistance of less than $10^6 \Omega$ at each bond.

NOTE 1 For example, for electrical boxes.

NOTE 2 EMC requirements can be more restrictive than this.

For EMC requirements, see ECSS-E-ST-20-07.

- b. <<deleted>>
- c. Where structural ground is carried across slip-rings on a rotating joint, at least two slip-rings shall be dedicated to the structural ground path, some at each end of the slip-ring set.

NOTE This is to provide redundancy.

- d. <<deleted>>
- e. <<deleted>>
- f. Circuits and conductive elements shall remain grounded in all possible situations

NOTE E.g. contacts and cables in switched circuits and moveable conductors.

- g. The grounding of surface conductors shall be implemented in one of the following two ways:
 1. Make the grounding to the main spacecraft of external and internal metallic parts and intrinsically conductive parts, like carbon, by a resistance of a value that satisfies requirement in clause 4.2.11 of ECSS-E-ST-20-07.
 2. Select the resistance to spacecraft ground not to be exceeded, such that it ensures that metal parts do not float by over V_{MAX} absolute potential with respect to spacecraft ground as specified in requirement 6.2.1a.

NOTE This normally leads to a requirement that the product of the resistance R and the area exposed to the environment A , is such that

$$R \cdot A \leq \frac{V_{MAX}}{J}$$

For a V_{MAX} of 300 V and $J = 10 \text{ nA cm}^{-2}$,

$$RA \leq 3 \times 10^6 \Omega \text{ m}^2$$

For example:

- For a one square metre MLI, the maximum resistance is $3 \text{ M}\Omega$
- For a scientific mission where $V_{MAX}=10 \text{ V}$,
 $RA \leq 10^5 \Omega \text{ m}^2$

6.3.2 Exceptions

- a. Except in the case specified in 6.3.2c, floating conductive parts may be used if they have an area less than 1 cm^2 and have a total capacitance less than 50 pF with respect to the structure and not be part of a multi-conductor elements structure and are more than 20 mm away from a power line.

NOTE 1 This is to limit the energy available to discharge.

NOTE 2 Multi-conductor structures exist where at least two conductive elements are separated by a dielectric surface of less than 100 cm².

- b. The configuration shall be such that energy involved in any possible discharge is less than one corresponding to a discharge of a capacitance of 50 pF under 6×10^3 V.
- c. In case of multiple elements, in which case the empirical rule in 6.3.2a does not apply, tests shall be performed.

NOTE Examples are patched antenna and solar arrays are multiple elements.

6.3.3 Electrical continuity for surface materials

6.3.3.1 Applicability

- a. The applicability of the electrical continuity requirements shall be established in accordance with the diagram in Figure 6-1.

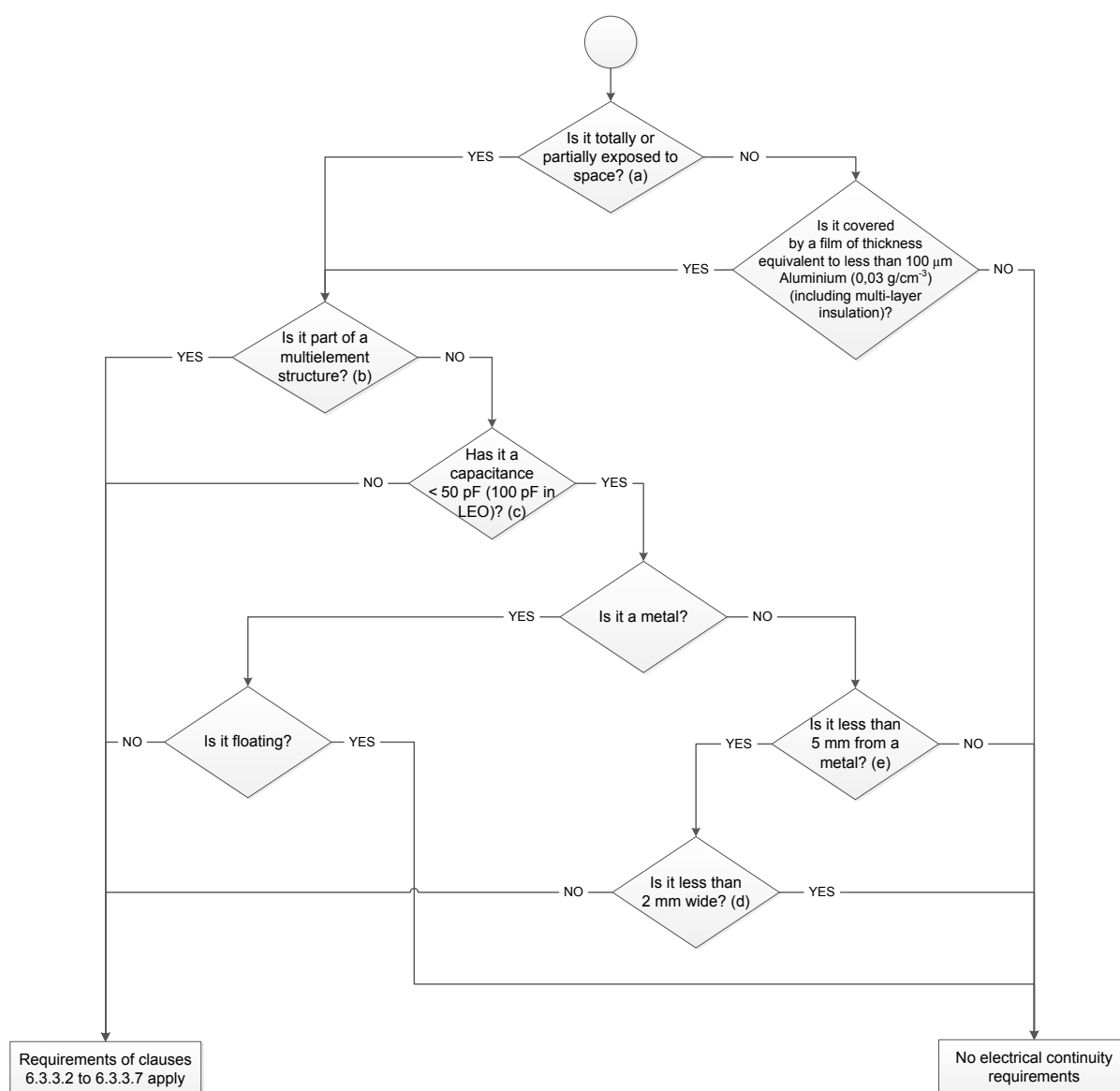


Figure 6-1: Applicability of electrical continuity requirements

6.3.3.2 General

- a. All spacecraft surface materials shall be conductive to the extent that they lead to surface voltage below the maximum voltage specified in clause 6.2.1.
- b. If a surface material is used which is not conductive, to the extent specified in 6.3.3.2a, then it shall be verified by testing or appropriate charging analysis in accordance with clause 6.4, whether hazardous charging can occur under representative worst case conditions.

- c. The determination of dc resistance of the surface material shall include the change over the service life of the bond in the environmental conditions which it is expected to experience including vacuum, temperature and mechanical stress.

6.3.3.3 Blankets

- a. All external and internal metallic layers of a thermal blanket shall be grounded to the structure with at least 2 bonding straps directly to ground (no daisy chain configuration).
- b. Any point on a blanket shall be within 1 m of a bonding strap.
- c. Adjacent blankets greater than 100 cm² in area shall be separately grounded to structure.

6.3.3.4 Material assemblies

- a. Material assemblies composed of different materials that are not separately grounded, shall be analysed or tested in their entirety.

NOTE 1 An example is optical solar reflectors.

NOTE 2 A complicated assembly, e.g. with high number of different layers, or not well represented by dielectrics coating planar surfaces, can be difficult to simulate in the analysis tools and then testing is needed in accordance with this requirement.

- b. Metallic layers within material assemblies shall not be exposed to the plasma.

6.3.3.5 Conductive coatings

- a. The thickness of conductive coatings shall be such that there is no degradation of their performance as a conductor when exposed to the expected erosion due to sputtering and atomic oxygen.

6.3.3.6 Dielectric materials

- a. The following materials shall not be used unless it can be demonstrated by charging analysis with a proper 3D computation tool that they pose no threat to the spacecraft:

1. Perforated metallized FEP or ETFE (SSM)
2. Polyimide material with thickness above 50 µm if no ESD conductive coating is used on the space-exposed side
3. FEP, ETFE, C₂F₄
4. Epoxy glass
5. Silica cloth

NOTE The reason is that they have a known association with ESD risks.

- b. For dielectric materials, laid on top of a more conductive material, the resistivity of the dielectric shall be such that the electric field anywhere within the dielectric does not exceed $E_{MAX}=10^7$ V/m.

NOTE Normally, this is fulfilled through the requirement that the resistivity is less than the ratio of E_{MAX} by the current density J .

$$r \leq \frac{E_{MAX}}{J}$$

For a J of 10 nA cm⁻², $r \leq 10^{11}$ Ω m.

- c. For a dielectric material, adjacent to a more conductive material and both exposed to vacuum, hereby constituting a triple point, the resistivity of the dielectric shall be such that the dielectric does not charge at any location to more than 2 kV negative or to more than 300 V positive with respect to the adjacent conductor.

NOTE 1 Normally, for a slab geometry, this is fulfilled through the requirement that the resistivity r is such that

$$r \leq \frac{1}{t} \frac{V_{MAX}}{J}$$

or

$$r \leq \frac{t}{d^2} \frac{V_{MAX}}{J}$$

where t is the material thickness and d is the minimum distance at the surface of the dielectric to the adjacent grounded conductor.

or

$$\rho \leq \frac{V_{MAX}}{J} \frac{2}{d^2}$$

where ρ is surface resistivity in Ω/square.

NOTE 2 The first equation in NOTE 1 is when the main conduction path is through the thickness of the material, e.g., to an underlying conductor while the second and third equations are when the main conduction path is parallel to the surface, e.g. to an adjacent conductor.

For a V_{MAX} of 100 V and J of 10 nA cm⁻², $V_{MAX}/J = 10^6$ Ω m².

For example, a 100 μm thickness dielectric must have a resistivity of less than 10¹⁰ Ω.m if on top of a conductor and a resistivity of less than 10⁶ Ω.m if on top of a very resistive material and grounded every 1 cm.

For a scientific mission where, say $V_{MAX}=10$ V, $V_{MAX}/J = 10^4$ Ω m² usually implies that all surface materials must be good conductors.

NOTE 3 No margin is included in the values in NOTE 2.

6.4 Surface charging analysis

- a. A surface charging analysis shall be performed, including:
 - 1. A systematic inventory of all surface materials, their resistivity, area and thickness.
 - 2. A systematic inventory of triple-points, floating conductors and multiple-conductor element components.
- b. In the event that some dielectric surfaces are left more insulating than specified in 6.3.3.6, requirements in 6.3.3.2 shall be verified by 3D surface charging analysis.
- c. Computational modelling at spacecraft scale shall emulate the effects of the phenomenon related to surface particle emission, secondary emission and photo-emission, conduction and collection processes.
 - NOTE 1 Annex D describes some charging codes that can form part of this analysis.
 - NOTE 2 Many materials suitable for space use are found in ECSS-Q-ST-70-71.
- d. In case the material parameters needed for 6.4c are not known and cannot be extracted from a recognized material database accepted by the customer, testing shall be performed to obtain the data.
- e. The charging analysis may consist of an SPIS analysis.

6.5 Deliberate potentials

- a. The perturbation of surface electrostatic potential arising from potentials that are deliberately maintained on a surface shall be analysed.
 - NOTE This can for example be the case where a scientific experiment has a deliberately exposed surface held at a potential. Recessing and shielding of the potential surface are strategies for reducing such perturbations.

6.6 Testing of materials and assemblies

6.6.1 General

- a. If the requirements of 6.3.3 and 6.4 are not satisfied, tests shall be performed on a representative sample of the concerned surface material or assembly.
- b. The rationale behind the test, the details of the procedure and the test results shall be submitted to the customer for approval.
- c. The tests shall either:
 - 1. reproduce the physical process of concern, i.e. charging, in a worst-case environment, or

2. yield electrostatic material properties enabling the materials response to such an environment to be assessed.
- d. The tests shall be performed under vacuum (i.e. < 1 mPa) and electron irradiation.
- e. The tests shall be performed after outgassing of the material under vacuum for at least 48 hours.

NOTE Changes to material conductivity can occur as volatile substances are lost by outgassing.

- f. In cases in which the effect of dose-rate and dose is determined by test, it shall be performed via irradiation by electrons that have energy sufficient to pass right through the sample.
- g. A mono-energetic spectrum with energy from 20 keV to 40 keV shall be used to deliver the surface charging current.
- h. The flux intensity applied to the samples under test shall be adapted to realistic rate of charging as experienced in orbit.

NOTE For GEO testing it is generally 250pA/cm² for low energy electrons (0 to 50) keV and (0,1 to 50) pA/cm² for high-energy electrons (400 keV).

- i. For LEO, the current shall be delivered in pulses representative of auroral crossing durations, typically 2 minutes for circular LEO orbits with inclination above 50°.
- j. Tests shall be performed at temperature of use in space.

NOTE This is because temperature has a large effect on conductivity in dielectric materials. As temperature falls, resistivity rises.

- k. Qualification tests for materials shall:
 1. be performed with multi-energetic electron beams, and
 2. reproduce the dose-rate and charging current of the environment.

NOTE 1 This is because of the radiation and electric field induced conductivity. These effects influence the surface voltage and consequently the risk of discharge.

NOTE 2 Reproduce ageing effect due to high energetic electron irradiation of the environment corresponding to the entire life in orbit.

- l. The number of samples to be tested shall be agreed with the customer.

NOTE Because of microscopic differences between samples, several samples are normally tested to allow for sample variability.

6.6.2 Material characterization tests

- a. A material characterization test need not be performed for a material, if it is only used in assemblies for which tests have been performed at assembly level.
- b. Except in the case specified in 6.6.2a, each new non-conductive material that has never flown on a spacecraft shall be tested under realistic irradiation conditions in terms of electron spectrum and across the temperature range to be used, to prove that there is no ESD risk unless test results under similar conditions are available.
- c. Materials shall be tested separately in order to evaluate electrostatic risk through knowledge of their electrical properties, including:
 1. bulk (intrinsic) conductivity,
 2. surface conductivity,
 3. radiation-induced conductivity,
 4. relative permittivity, and
 5. discharge threshold level
- d. The minimum size of the sample under test shall be above 30 mm × 30 mm.

NOTE A good quality measurement of the surface voltage can be made with this size.

- e. Sample thickness and physical properties shall be representative of flight configuration.

6.6.3 Material and assembly qualification

- a. Tests shall be performed for both normal voltage and inverted voltage situations.
- b. It shall be agreed with the customer whether or not inverted voltage situation is simulated through potential biasing.
- c. The assembly testing under the worst case environment at all temperatures to be used in the mission shall lead to an assessment of ESD risk as follows:
 1. No ESD is observed and potential remains below 100 V in inverted gradient or 1000 V in direct gradient: No ESD risk.
 2. No ESD is observed but potential between 100 V and 1000 V in inverted voltage gradient or potential above 1000 V in normal gradient mode under multi-energetic irradiation: moderate ESD risk.
 3. ESD is observed: high ESD risk.

NOTE Definition of the worst-case environment is part of the customer specification and can refer to the ECSS-E-ST-10-04 "Space environment".