General

JERG-2-211A

DESIGN STANDARD

SPACECRAFT CHARGING AND DISCHARGING

May 10, 2012　Revision A

Japan Aerospace Exploration Agency

This is an English translation of JREG-2-211A. Whenever there is anything ambiguous in this document, the original document (the Japanese version) shall be used to clarify the intent of the requirement.

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1.Purpose and Scope of Application

In a space environment, there are various ions and electrons (plasma) with different energy levels, densities, and particle species at a specific space and time, thus exerting a variety of influences on spacecraft. First, this chapter describes “passive plasma interaction” based on the quantity of the electric charge that flows into and out of a spacecraft. When high-energy electrons enter the spacecraft, it begins to be negatively charged because of the excessive negative charge. The charging process continues until the subsequent electrons fail to flow into the spacecraft. The spacecraft, however, is only slightly charged when the electron emission mechanism is dominant because of the effect of sunlight or the properties of its surface materials. When the spacecraft is charged too much, a material that has supported a high electrical potential difference begins to lose its insulation and finally allows a large transitional electric current to flow (electrostatic discharge) across the material. Thus, if there is a weak electronic circuit along the current pathway, it will quickly lose its function, or electromagnetic noise may cause the circuit to break down for a short time.

Second, this chapter describes an example of “active plasma interaction.” As space activities have been expanded and power demand has been heightened, the higher voltage has been used for the corresponding higher efficiency power generation. When light-weight solar array panels with many exposed electrodes are used, plasma current flows constantly through the electrodes to maintain the generated electrical potential. From the view point of power generation, this current flow refers to power leakage to outer space. Ions having the kinetic energy corresponding to generation voltage flows into the solar panel, causing sputtering erosion of the material. An electric spark generated on the surface of a solar panel can trigger the inflow of the generated current, inducing an arc discharge. The arc discharge, if maintained, may lead to thermal breakdown. On the other hand, the plasma current has been actively studied from the positive side; researchers have been studying on the interaction between the plasma current and the Earth’s magnetic field, aiming at using the current to generate propulsion force or electric power.

When the first man-made equipment reached space, it discovered active phenomena induced by space plasma in space although space had been considered to be only a dark, empty space. The active space environment caused much damage to artificial satellites and shortened their functions. Space engineers, however, learned a lot from these cases and gradually cultivated the technology that can help cope with these problems. Nowadays, they not only can break a chain of space plasma interaction that could lead to an accident and prevent the chain from occurring, but also can be actively involved in the space environment by using such means as artificial plasma-jet injection.

This document aims to familiarize people with a space environment that a spacecraft may encounter, to achieve people’s understanding of the mechanism of space plasma interaction, and to provide means to cope with problems induced by space plasma interaction so that spacecraft with higher reliability and certainty can be embodied.

This standard applies to general “spacecraft.” With the temporal, spatial, and territorial development of space activities, the concept of spacecraft has been widened. In addition, spacecraft can encounter a wide range of space environment conditions. In addition, consideration should be given to the fact that spacecraft is based on various methods depending on purposes for participating in space activities, implementing organizations, or policies. Thus, the purpose of this document should be fully understood, this standard should be properly applied, and furthermore, advice from experts should be sought whenever necessary.

2. Related Documents

2.1 Applicable documents

The following document is a part of the standard within the scope of this standard. Note that when the standard and the following document conflict with eath other, this standard prevails.

(1) JERG-2-141: Space environment standard

2.2 References

The following document is referred to during the process of establishing the standard.

(1)ECSS-E-20-06 Draft 1 Spacecraft charging

3. Terminology, Definition, and Abbreviation

3.1 Terminology and definition

The following describes the technical terms used in this standard and their meanings.

3.1.1 Aurora zone

The aurora zone is a space region where aurora can be formed. It is located between a magnetic latitude of 60°and 75° north or south.

3.1.2 Radiation dose

Radiation dose refers to the amount of energy per unit mass locally absorbed as a result of exposure to radiation.

3.1.3 Wake

trail of rarefied plasma left behind by a moving spacecraft [ISO-11221]

3.1.4 Electrical breakdown

Failure of the insulation properties of a dielectric, resulting in a sudden release of charge with possible damage to the dielectric concerned [ISO-11221]

3.1.5 Electrostatic discharge

Electrical breakdown of dielectric or gas or vacuum gaps, and also of surface interface of dissimilar materials, caused by differential charging of parts of dielectric materials and their interfaces [ISO 11221]

3.1.6 Internal charging

Internal charging refers to the accumulation of electric charge on internal materials shielded by the surface layer of a spacecraft because of the penetration of high-energy charged particles.

3.1.7 Ion engine

An ion engine is a propulsion system ejecting ions at a high speed to product thrust force.

3.1.8 L-shell

L-shell refers to a parameter of the Earth’s magnetic field

Note: L-shell is simply denoted as “L”, which is used as a coordinate to indicate the position in the near-earth outer space.

3.1.9 Outgassing rate

Outgassing rate refers to the mass of a molecular species that evaporates, sublimates, and separates from surfaces (per unit time and unit surface area)

Note: Outgasing rate is measured in kg m^{-2} s^{-1}. In addition, the outgasing rate can be represented in other units, for example, relative mass unit per unit time (kg s^{-1}, % s^{-1}, or % s^{-1} m^{-2}).

3.1.10 Plasma

Plasma refers to a group of partially or fully ionized particles that collectively respond to an external electromagnetic field.

3.1.11 Radiation

Radiation refers to high-energy particle rays including photon beams.

Note: In view of the purpose of this standard, this definition excludes the electromagnetic waves with frequencies lower than ultraviolet light, including visible light, heat waves, and microwaves.

3.1.12 Radiation belt

Radiation belt refers a field of high-energy particles contained by a magnetic field.

3.1.13 Ram
space in front of and adjacent to a spacecraft in which the plasma density can be enhanced by the motion of the spacecraft. [ISO-11221]

3.1.14 Surface charging
Deposition of electrical charges onto, or their removal from, external surfaces [ISO-11221]

3.1.15 Tether
A tether is a conductive or insulating flexible cable that can be extended for a long distance in order to connect multiple spacecraft or component parts, or a space system with the flexible cable.

3.1.16 Thruster
A thruster is a drive unit that uses reaction force in space to change the attitude or the orbits of a spacecraft.
Example: Rocket, cold gas discharge device, electric propulsion unit, and iron engine

3.1.17 High-energy electron
A high-energy electron is an electron having an energy range from about 1 kV to 100 kV that contributes to electrical charging.

3.1.18 Sputtering
Sputtering is a phenomenon in which, when the surface of a solid is irradiated with high-energy ions or neutral atoms, these ions or atoms cause atomic collisions one after another in the vicinity of its surface, thereby inducing the atoms in the solid to jump out of its surface into the vacuum of space.

3.1.19 Precipitating electron
A precipitating electron is a high-energy electron that falls into the aurora zones along magnetic field lines

3.1.20 Differential voltage
Potential difference between any two points in a spacecraft during spacecraft charging, especially between the insulator exterior surface potential and the spacecraft chassis potential [ISO-11221]

3.1.21 Inverted potential gradient
Inverted voltage gradient, Result of differential charging where the insulating surface or dielectric reaches a positive potential with respect to the neighbouring conducting surface or metal [ISO-11221]

3.1.22 Normal potential gradient
Normal voltage gradient, Result of differential charging where the insulating surface or dielectric reaches a negative potential with respect to the neighbouring conducting surface or metal

3.1.23 Primary discharge
initial electrostatic discharge which, by creating a conductive path, can trigger a secondary arc [ISO 11221]

3.1.24 Primary arc
trigger arc
developed phase of a primary discharge, under an inverted potential gradient, which is associated with cathodic spot formation at a metallic or semiconductor surface [ISO-11221]

3.1.25 Secondary arc
passage of current from an external source, such as a solar array, through a conductive path initially generated by a primary discharge [ISO-11221]

3.1.26 Non--sustained arc
Passage of current from an external source through a conductive path that lasts only while the

primary discharge current flows [ISO-11221]

3.1.27 Temporary sustained arc

Passage of current from an external source through a conductive path that lasts longer than a primary discharge current pulse but terminates without leaving a permanent conductive path [ISO-11221]

3.1.28 Permanent sustained arc

Passage of current from an external source through a conductive path that keeps flowing until the external source is intentionally shut down [ISO-11221]

3.1.29 Paschen discharge

Paschen discharge refers to a phenomenon in which an electrical discharge occurs at a certain minimum voltage between two opposite parallel plate electrodes made of a specific material, separated by a specific distance, and in a specific gas species of a specific pressure. The minimum voltage is given as a function of the pressure and the distance between the electrodes.

3.1.30 Insulator

In this document, an insulator refers to a dielectric with a volume resistivity of more than 10^{14} Ωm (refer to Chapter 5.2.6).

3.1.31 High-resistance material

In this document, a high-resistance body refers to a dielectric with a volume resistivity of 10^{14} Ωm or less (refer to Chapter 5.2.6).

3.1.32 Dielectric

In this document, “dielectric ” is a general term for insulators and high-resistance material.

3.2 Abbreviations

The following shows the abbreviations used in this document and their meanings.

Abbreviation	Meaning
AM0	Air mass zero (AM0 refers to no reduction in the amount of sunlight due to the earth atmosphere.)
CFRP	Carbon Fiber Reinforced Plastics
CME	Coronal Mass Ejection (x-rays, flare particles, or plasma lumps emitted during solar flares)
DMSP	Defense Meteorological Satellite Program
EMC	Electromagnetic Compatibility
Emf	Electro-motive Force
ESD	Electro-Static Discharge
FEEP	Field Emission Electric Propulsion (a type of electric propulsion methods)
GEO	Geostationary Equatorial Orbit
ISS	International Space Station
LANL	Los Alamos National Laboratory (New Mexico, U.S.A.)

LEO	Low Earth Orbit
MEMS	Micro-electromechanical System
MEO	Medium (altitude) Earth Orbit
MLI	Multi-Layer Insulator
MLT	Magnetic Local Time
MPD	Magneto-Plasma Dynamic (magnetoplasmadynamic thruster, a type of electric propulsion methods)
MUSCAT	Multi-Utility Spacecraft Charging Analysis Tool (spacecraft charging analysis program in Japan)
NASCAP	NASA Charging Analysis Program (spacecraft charging analysis program in U.S.A.)
OSR	Optical Solar Reflector
PCB	Printed Circuit Board
PEO	Polar (low) Earth Orbit
RIC	Radiation Induced Conductivity (irradiation-induced electrical conductivity of a dielectric)
RTV	Room Temperature Vulcanization (room temperature vulcanizing adhesive)
SAS	Solar Array Simulator (solar array panel simulator as a power source)
SPIS	Spacecraft Plasma Interaction Modeling Software (spacecraft charging analysis program in Europe)
SPT	Stationary Plasma Thruster (hall thruster, a type of electric propulsion methods)
SSJ	Special Sensor Precipitating Electron and Ion Spectrometer (satellite-mounted precipitating electron and ion measuring instrument)
TJ	Triple Junction (triple junction at which electrode, insulator, and vacuum come together)
TSS	Tether Satellite System (Italian tether experiment on board a space shuttle)
FLUMIC	Flux Model for Internal Charging (a radiation model)
PA	Primary Arc
NSA	Non-Sustained Arc
TSA	Temporary Sustained Arc
PSA	Permanent Sustained Arc

4. General Requirements

4.1 Principle

A plan to implement the necessary preventive measures including analysis, test, and verification should be established for every spacecraft to prevent damage due to the electrical interaction between a spacecraft and the surrounding space environment, from the standpoint of charging and discharging through the understanding of the orbit and the space environment, as well as the electrical characteristics of the spacecraft.

4.2 Surface materials

Surface materials in this document refer to any surface materials for a spacecraft, for example, thermal blanket, solar-cell coverglass, optical equipment for sensors, and exposed cables. Reducing surface charging requires controlling electric properties of the materials that directly contact with the surrounding plasma. Thus, it is necessary to reduce the possibility of local charging and the resultant electrostatic discharge.

4.3 Spacecraft for plasma measurement

Spacecraft designed for the measurement of surrounding electrical field or low-energy charged particles must take into account the conditions required from the standpoint of the measurement, in terms of suppression of local charging, suppression of photoelectron and secondary electron emission, and active control of spacecraft potential.

4.4 Solar array panel

Discharges on the solar panel should be prevented because of the principle of eliminating the insulation breakdown of the solar cell circuit as well as eliminating one of the physical processes leading to power loss. In addition, it should be proved that the amount of reduction in the electrical output from the solar panel, predicted at the end of the operation period, is within an acceptable range and that the secondary arc does not lead to the insulation breakdown of the solar panel circuit. If the breakdown due to the secondary arc is inevitable, it should also be proved that the solar cell circuit is still capable of supplying electric power to spacecraft loads even after the occurrence of the insulation breakdown. Furthermore, when positive charging is predicted, it should be demonstrated that the reduction in electric power due to the power leakage into the plasma is within an acceptable range.

4.5 High-voltage system

(1) It should be proved that the possibility of the occurrence of Paschen discharge in all spacecraft-mounted equipment is within an acceptable range in all possible gas environments from the launch of a spacecraft to the end of its operation.

(2) Utmost attention should be given to the design of electrical insulation when high-voltage systems that cause outgassing are installed or when they are installed closed to the equipment that causes outgassing.

4.6 Internal parts and materials

Related components to be considered are all dielectrics (or combinations of floating conductors and dielectrics) that may cause insulation breakdown due to electrostatic charging. These substances must satisfy the following requirements:

(1) When a spacecraft is launched into GEO or into the orbit exceeding an L-value of 1.5 in the B-L coordinate system near the geomagnetic equator, the internal charging effects and the following discharging effects on the spacecraft should be considered at the earlier stages of designing. Furthermore, the thickness of the shielding structure should be included into the consideration because the amount of charging varies depending on the thickness of shielding structure.

(2) There is no need for conducting special evaluations of any spacecraft intended to be launched

into its LEO (altitude: less than 1,200 km including PEO) on the condition of obtaining client's consent as long as proven, conventional space technologies are used.

4.7 Tether

A tether is a conductive or non-conductive flexible cable provided from a spacecraft or connecting two points on the spacecraft. For a spacecraft with the tether system, the following requirements related to (1) consideration to the maximum voltage (conductive tether), (2) consideration to the maximum current (conductive tether), and (3) consideration to current pathways (conductive tether) should be considered because of the electrical interaction between the spacecraft and the space environment.

4.8 Electrical propulsion

Consideration should be given to the effect of plasma environment on the operation of the electrical propulsion system of a spacecraft, as well as the effect of the electrical propulsion system on the other systems of the spacecraft through the interaction between the propulsion system and the plasma environment.

5. Design Requirements and Design Means

5.1 Principle

5.1.1 Program planning

A plan to implement the necessary preventive measures specified in item 5.1.1.4 should be established for every spacecraft to prevent damage due to the electrical interaction between a spacecraft and its space environment.

5.1.1.1 Understanding of orbit and space environment

Information on the following items should be clarified: the orbits of all spacecraft predicted during its mission and its space environment, and, in particular, duration time in the regions in which electron flux is high in the keV range. The following items show the regions that require special attention. Refer to Appendix 1.2 for the nominal environment and the worst environment for these individual regions.

LEO : Polar aurora regions during magnetic storm
MEO : Radiation belt during magnetic storm (high-energy electrons and protons)
GEO : Nighttime-side magnetospheric equatorial region during magnetic storm
In magnetosphere : Nighttime-side magnetospheric equatorial region during magnetic storm
Interplanetary : During the occurrence of CME (high-energy electrons and protons)
Orbit around the planet : Each planet should be considered individually.

The charged state in the above-mentioned worst environment should be predicted and evaluated. During the prediction, care should be given to the fact that a large electrical potential difference can be observed because the floating conductors and insulators on the nighttime-side of a three-axis control spacecraft provide shadowed side electrical potential, whereas the whole spacecraft provide daytime electric potential. Table 5.1-1 through Table 5.1-3 show the worst environmental conditions (during substorm) for three-axis control satellites operating in GEO and those for three-axis control satellites operating in PEO.

Table 5.1-1 Three-axis control satellites in GEO: The worst environmental conditions (during substorm)

Initial orbit: GEO

Phenomena	Photoelectron ($\mu A/m^2$)	keV electron ($\mu A/m^2$)	Plasma ion ($\mu A/m^2$)	Charged particle emission (A)	Charged potential (V)
During daytime	10 (Shadowed side: 0)	-10 (Several keV)	0	0	+ several V to - several kV (Shadowed-side floating conductor: - several kV)
During nighttime	0	-10 (Several keV)	0	0	- Several kV

Table 5.1-2 Three-axis control satellites in GEO: The worst environmental conditions (during substorm)

Operational orbit: GEO

Phenomena	Duration time	Photoelectron ($\mu A/m^2$)	keV electron ($\mu A/m^2$)	Plasma ion ($\mu A/m^2$)	Charged particle emission (A)	Charged potential (V)
During daytime	4 hours	10 (Shadowed-side: 0)	−10 (Several keV)	0	0	+ several V to - several kV (Shadowed-side floating conductor: - several kV)
During nighttime	72 minutes	0	−10 (Several keV)	0	0	- Several kV
Electrical propulsion during daytime (+plasma emission)	6 hours	10 (Shadowed-side: 0)	-10 (Several keV)	0	1 (1 k eV)	+ several V to - several V (Shadowed-side floating conductor: - several kV)

Table 5.1-3 Three-axis control satellite in PEO: During the worst space environment (during substorm)

Operational orbit: PEO

Phenomena	Duration time	Photoelectron ($\mu A/m^2$)	keV electron ($\mu A/m^2$)	Plasma ion ($\mu A/m^2$)	Charged particle emission (A)	Charged potential (V)
During daytime	60 minutes	10 (Shadowed side: 0)	0	1,000	0	+ several V
During daytime and at poles	20 minutes	10 (Shadowed side: 0)	-10 (Several keV)	1,000	0	+ several V to - several V (Wake-side floating conductor: - several kV)
During nighttime	40 minutes	0	0	1,000	0	- several V
During nighttime and at poles	20 minutes	0	-10 (Several keV)	1,000	0	+ several V to - several V (Wake-side floating conductor:- several kV)

5.1.1.2 Understanding of electrical characteristics of a spacecraft

The electrical characteristics of a spacecraft in relation to charging and discharging should be understood. The following shows the items that require special attention during the measurement of these characteristics:

(1) Sections that may potentially produce an electrical potential diffence of 70 V or more between these sections and surrounding plasma
  Floating conductors and insulators exposed to surrounding plasma
  Floating conductors inside a spacecraft
(2) Charged particle emission devices such as electrical propulsion systems
(3) Solar cell exposed to surrounding plasma
(4) Dangerous sections to which a voltage of 30 V is applied and that may produce secondary arc
(5) High-voltage sections which are 50 V higher in potential than the structure of a spacecraft.
(6) Mission requirements for spacecraft charging
(7) Systems that require considerations related to charging and discharging

5.1.1.3 Evaluation of parts in need of preventive measures

The following items that require consideration should be evaluated on the basis of the above items.

(1) Ungrounded surface conductors and exposed insulators (refer to Item 5.2 and Item 5.3)
Presence or absence of grounding and conductive processing
Evaluation of discharging (maximum potential difference and possible discharge channels and their effects)

(2) Solar cell (refer to Item 5.4)
Electrical potential difference between adjacent cells and validity of gap distance
Validity of exposed electrodes in terms of electrical potential and position
Evaluation of discharging
Power leakage into plasma

(3) High-voltage electrodes (refer to Item 5.5)
Evaluation of discharging (possible discharge channels and their effects)

(4) Internal ungrounded conductors and dielectrics (refer to Item 5.6)
Presence or absence of grounding processing
Evaluation of discharging (possible discharge channels and their effects)

(5) Charged particle emission (refer to Item 5.8)
Neutralizing functions and effect of beam emission

The equipment and sections that do not satisfy the standard as a result of the above evaluation should be risk-assessed in view of the “degree of damage” and the “event probability during a mission period” shown in Fig. 5.1-1 in order to develop preventive measures, spread preventive measures, contingency plans, and operation control measures.

The following items (a) and (b) as well as Fig. 5.1-1 show an example of risk assessment.

(a) Level of damage

Ⅰ:	Breakup	Example: Loss of Minimum Mission Success
Ⅱ:	Serious	Example: Loss of nominal Mission Success
Ⅲ:	Local	Example: Partial loss of nominal Mission Success
Ⅳ:	Minor	Example: Minor loss of nominal Mission Success

(b) Event probability during a mission period

A:	Frequently	10% or more
B:	Sometimes	10 - 1%
C:	Rarely	1 - 0.1%
D:	Few	0.1 - 0.0001%
E:	None	0.0001% or less

Fig. 5.1-1 Risk assessment (PI matrix)

		LIKELIHOOD				
		A	B	C	D	E
Level of Impact	Ⅰ					
	Ⅱ					
	Ⅲ					
	Ⅳ					

	Preventive measures required
	Right or wrong judgment required
	Acceptable range

5.1.1.4 Planning of analysis, testing, and verification programs

The following analysis, testing and verification programs should be drawed up.

(1) The equipment and sections that satisfy thisa standard should be should be confirmed the suitability of their installation by measuring and inspection .

(2) The equipment and sections that do not satisfy the standard should be checked by using the following procedure.

(a) Design change, preventive measures, spread preventive measures, contingency plans, and operation control measures must be developed on the basis of computer modeling and analysis results (refer to Item 6.2.2.4).

The judgment of whether or not to conduct the evaluation and analysis should be based on the risk assessment. At the same time, design change, preventive measures, spread preventive measures, contingency plans, and operation control measures should be considered.

(b) Those that do not satisfy the standard should be measured and checked to confirm how they are mounted.

The judgment of whether or not to conduct the confirmation of the how they are mounted should be based on risk assessment.

5.2 Spacecraft surface

5.2.1 Explanatory comments and Applicability

A spacecraft is charged because of the inflow of high-energy charged particles coming from outer space. These particles produce large, local electrical potential difference, causing electrostatic discharge (ESD). The ESD can, in turn, cause serious problems such as damage to electronic equipment, surface contamination, and interference in observation and measurement. This Section defines the following design standard to prevent these problems.

(1) To prevent the occurrence of ESD, the surface electrical potential of a spacecraft should be kept evenly distributed, so that no large electrical field can be produced. To be more precise, the following items should be conducted.

(a) The electrical characteristics of all surface materials exposed to outer space should be controlled properly.
(b) The electrical potential of all electrically conductive parts exposed to space should be set up properly.
(c) All electric conductors and high-resistance bodies in a spacecraft should be properly grounded.

(2) If there is no choice but to permit a spacecraft to be electrically charged, design consideration should be given to the sections in which charge can be accumulated, so that the effect of EDS can be lowered. To be more specific, the following two items should be considered.

(a) Possible discharge pathways should be properly controlled to prevent the discharge on the surface of important electronic equipment or the inflow of electric current into the surface or the equipment.

(b) Electromagnetic compatibility (EMC) should be ensured in an electrical transient state induced by current pulses involved in the discharge.

(3) Item 5.4 defines a design standard to prevent the “secondary arc” that can be maintained by the eclectic power supplied by the solar panel.

5.2.2 Basic design requirements

(1) The surface of a spacecraft should be designed so that they cannot be charged to a potential higher than the following maximum absolute electrical potential V_{MAX} in reference to the grounding points of the spacecraft. The potential V_{MAX} may be changed to a lower value according to mission requirements.

(a) 500 V for electric conductor surfaces
(b) 1 kV for high-resistance material surfaces

(2) A current density of 10 μA/m^2 is commonly used as the worst-case current density due to keV electrons. Since this item depends on the orbits, different worst-case current density values may be adopted on the basis of analytical results. (A typical value of 10 μA/m^2 is used in Table 5.1-1 through Table 5.1-3.)

(3) If the above items (1) and (2) are not applicable, due reasons should be presented.

5.2.3 Conductivity of structural bodies and mechanisms

Electrical conductivity should be ensured between all structures and mechanisms of a spacecraft and its grounding points.

(1) All structural and component parts should be grounded through a resistance of 1 MΩ or less. If this condition is not satisfied, any structural and component part with surface area A, withstanding voltage V, and electron-impact secondary electron emission coefficient γ should be

grounded through the following resistance R unless other means are available.

$$R < \frac{V}{J(1-\gamma)A} \qquad (5.2\text{-}1)$$

(Example) When V = 500 V, J = 10 μA/m^2, A = 1 m^2, and $\gamma = 0$, R < 50 MΩ.

(2) Joint or hinge parts are recommended to be connected over to the ground through ground straps. When they are grounded through rotary joint-based slip rings, two or more slip rings should be used.

5.2.4 Conductivity of the surface of a spacecraft

(1) Electric conductors and high-resistance bodies (volume resistivity $\rho \leqq 10^{14}$ Ωm) should be used for spacecraft surface materials

Note 1: Assuming that ρ is volume resistivity, ε is dielectric constant; t is thickness; and A is surface area, the capacitance and the resistance is given by the following formulae, respectively:

$$C = \varepsilon A/t$$
$$R = \rho t/A$$

Here, the relaxation time constant of the charge accumulated on the surface is given by
RC = ερ.
When a spacecraft is periodically charged because of the exposure to external current in a special space in its orbit and when the relaxation time constant is shorter than its orbital period, at least, no further charge accumulation occurs on the surface of the spacecraft.

(Example) LEO satellite (orbital period = 90 min) : $\rho < 3 \times 10^{14}$ Ωm (assuming $\varepsilon = 2\varepsilon_0$)

Stationary satellite(orbital period = 24 h) : $\rho < 3 \times 10^{15}$ Ωm (assuming $\varepsilon = 2\varepsilon_0$)

Note 2: Notwithstanding materials that comply with the above Note 1, they are subject to the limitation that the maximum electrical potential for the surface of electric conductors is 500 V, and that for high-resistant bodies is 1 kV" (refer to 5.2.2).

Note 3: In addition to the volume or surface resistance of a material, the concept of surface conductivity includes the phenomenon in which thermal electrons induced by electron-impact secondary electron emission are moved by the electrical field along the material surface.

(2) All surface materials should be grounded to the structure of a spacecraft. The equipment and sections that cannot be grounded should be properly treated as required by the rules and regulations described in Item 5.2.10.

(3) The resistivity of surface materials should satisfy acceptable values including those predicted and deteriorated because of damage during operational period.

5.2.5 Materials: Electric conductors

(1) All the electric conductors exposed to outer space should be grounded to the main body of a spacecraft; they include metal parts (small parts such as metal labels and connector brackets) and electric conductive parts (e.g., carbon products).

(2) They should be grounded through an appropriate resistance so that the electrical potential of these parts in reference to the grounding points of a spacecraft does not exceed an absolute

voltage of 500 V. The grounding resistance R of an electric conductor with exposed area A in reference to the grounding points should satisfy the following formula unless other methods are available (refer to Formula (I. 6-16)).

$$RA \leq \frac{V_{MAX}}{J(1-\gamma)} \qquad (5.2\text{-}2)$$

(Example) When V_{MAX} = 500 V, J = 10 μA/m^2, and γ = 0, RA ≦50 MΩm^2 (when A = 1 m^2, R ≦50 MΩ).

5.2.6 Materials: High-resistance materials

A high-resistance material is a dielectric with a volume resistivity (ρ) of 10^{14} Ωm or less.

(1) The following insulating materials should not be applied to the sections exposed to outer space. The ESD risk of these materials has been known because they have a high resistance and require a long time for reducing the internally penetrated and accumulated electric charge.

(a) Standard teflon (FEP, PTFE, and ETFE)
(b) Epoxy glass
(c) Silica-fiber cloth

Note 1: Electronic-bridge ETFE electric wire (Spec 55 electric wire manufactured by Raychem for space applications) and kapton are high-resistance materials and do not come under the category of this item.

Note 2: It is generally difficult to define the volume resistivity of an insulator.
(Example) Although it is reported that kapton has a volume resistivity of 10^{13} Ωm when the time is 10 seconds or shorter [1], a volume resistivity of 10^{16} Ωm is generally recommended. It can be thought that kapton has been used without trouble for space applications because its electrical conductivity increases when placed under RIC or high electrical field.

(2) High-resistance materials stacked on a material with a higher conductivity should have the volume resistivity at which the absolute electrical potential in reference to the grounding points of a spacecraft does not exceed 1 kV. The volume resistivity ρ and material thickness t should satisfy the following formula unless other methods are available (refer to Formula (I. 4-12)).

$$\rho t < \frac{V_{MAX}}{J(1-\gamma)} \qquad (5.2\text{-}3)$$

(Example) When V_{MAX} = 1,000 V, J = 10 μA/m^2, and γ = 0, ρt ≦100 MΩm^2 (when t = 100 μm, ρ ≦10^{12} Ωm)

(3) If the previous item (2) cannot be satisfied, it should ensured that the surface materials are not charged to an electrical potential of 1 kV or higher in reference to the grounding point during the finite occurrence time of a charging event. Assuming that T is the elapsed time required for an orbiting spacecraft to pass through an high-energy electron existence region, ε is dielectric constant, t is material thickness, A is surface area, and γ is the secondary electron emission coefficient induced by electronic irradiation, the following formula should be satisfied [$J(1-\gamma)TA = Q < CV_{MAX} = \varepsilon V_{MAX}A/t$].

$$\frac{Tt}{\varepsilon} < \frac{V_{MAX}}{J(1-\gamma)} \qquad (5.2\text{-}4)$$

(Example) When V_{MAX} = 1,000 V and J = 10 μA/m^2, T t/ε ≦100 MΩm^2 (when T = 100 s, ε = 3ε_0, and γ = 0.5, thickness t is 53 μm)

(4) High-resistance materialsstacked on a material with a lower conductivity and those with grounded

ends should have the volume resistivity at which the absolute electrical potential in reference to the grounding points of a spacecraft does not exceed 1 kV. The volume resistivity ρ, material thickness t, and the maximum distance to the nearest grounding point should satisfy the following formula unless other methods are available.

$$\frac{\rho d^2}{t} \le \frac{V_{MAX}}{J(1-\gamma)} \qquad (5.2\text{-}5)$$

(Example) When V_{MAX} = 1,000 V, J = 10 μA/m^2, and γ = 0, ρd^2/t = 100 M Ωm^2 (when t = 100 μm, and the maximum distance d = 1 cm, ρ ≦10^8 Ωm)

(5) If the above items from (1) through (4) are contradictory to each other, or if their geometrical shapes are too complex to be shown in plane shapes, either of the following items should be conducted for safety evaluation.

(a) Select and test a sample that represents appropriate surface materials
(b) Conduct three-dimensional charging analysis

(6) When high-resistance material surfaces need to have higher insulation values, charging analysis of high-resistance material surfaces should be conducted to show that the electrical potential of the insulators does not exceed 1 kV.

Note 1: Appendix II shows the calculation program for the charging analysis.

5.2.7 Thermal blanket

(1) All thermal blankets (multi-layered heat insulating materials) inside and outside a spacecraft should be grounded to its structure.

・At least two ground straps should be used for the grounding.
・Every section on a thermal blanket should be within 1.4 m from the grounding straps.
・Grounding straps connected to the same thermal blanket should be grounded to different points.
・When the blanket has an area of more than 2 m^2, a single ground strip should be added to it for every excess area of 1m^2.

(2) Adjacent thermal blankets should be grounded to the structure of a spacecraft in a best possible distributed manner.

(3) All the layers of a thermal blanket should be grounded.

5.2.8 Conductive coating

(1) When surface-exposed high-hesitance materials are predicted to be charged in a possible environment (refer to 5.2.6), use conductive coating, conductive paint, conductive covering, and conductive materials to prevent the charging from happing. Here, volume resistivity ρ and surface resistivity (Ω/square) (=ρ/t) should comply with Formulae (5.2-1) and (5.2.5).

(Example) When V_{MAX} = 1,000 V, J = 10 μA/m^2, and γ = 0, the surface resistivity ρ/t ≦ 100 MΩ/square. Furthermore, volume resistivity ρ ≦10^4 Ωm for a material with a thickness of 100 μm.

(2) Conductive coatings should be thick enough so as not to deteriorate in performance during the entire operational period because of erosion (due to sputtering and atomic oxygen molecules).

(3) To reduce charging amount, it is preferable to use conductive coatings with a high secondary electron emission coefficient.

5.2.9 Intentional electrical potential

When the surface electrical potential of a spacecraft is intentionally maintained at values higher than those specified in Item 5.2.2, a region to apply a voltage should not have a protrusion for ESD suppression, or its electrical field should be shielded by mesh and grid electrodes or plasma so that effective surface electrical potential can be suppressed.

5.2.10 Preventive measures for charged sections

Sections that have an surface area of more than 2 cm^2 and are inevitably subject to charging should be risk-evaluated to see if ESD is present or not and make sure that its ripple effects are within an acceptable range. Use the Risk Evaluation sheet based on the Manual and Computer-Based Analysis for right or wrong judgment.

(1) Suppression of charging

To decrease the probability of the occurrence of ESD and to suppress its effects, the following items should be considered in terms of the exposed insulating sections with an area of more than 100 cm^2. Additionally, similar consideration is recommended to be given to smaller exposed insulating sections.

(a)Reduction in discharging probability—part 1:

The exposed area of a continuous insulator should be divided into smaller ones.

(Example) The exposed area should be divided into smaller ones with an area of 100 cm^2 or smaller. Adjacent insulators should be separate so that they are at least 2 cm apart. Furthermore, place an electric conductor or a high-resistance material in the gap between them.

(b)Reduction in discharging probability—part 2:

The exposure viewing field of an insulator should be narrowed.

(Example) A “conductor hood” grounded to the structure of a spacecraft can be used to narrow the exposure viewing field.

(2) Suppression of effects caused by discharging

Discharge pathways on individual charged sections should be risk-evaluated in terms of possible effects. The following measures should be taken to unacceptable risks.

(a) Possible discharge pathways should be properly controlled to prevent the discharge on the surface of important electronic equipment or the inflow of electric current into the surface or the equipment.

(Example) Local current should be discharged by forming proper discharge pathways, for example, by mounting a “discharge guard” (an electric conductor, e.g., the above mentioned “conductive hood”) connected to the structure of a spacecraft.

(Example) Any equipment very sensitive to discharge, such as power-supply systems or signal systems should not be placed close together.

(b) Proper electromagnetic compatibility (EMC) should be maintained against an electrical transient state induced by current pulses that occur in connection with discharge.

(Example) Use “shielded wires” for the harnesses near discharge pathways. Furthermore, take necessary measures in case when discharge current that flows through the shield.

(Example) Preventive measures against malfunction due to discharge current should be introduced to the current channels that form discharge pathways.

5.3 Satellites for plasma measurement

5.3.1 Overview

A spacecraft designed to measure the surrounding electrical field or low-energy charged particles should satisfy some requirements from the perspective of observation, with respect to suppression of local charging, suppression of photoelectron and secondary electron emission, and active control of spacecraft electrical potential. This section shows qualitative guidelines for these spacecraft. Quantitative levels should be defined in each mission in response to observation requirement.

5.3.2 Guideline

5.3.2.1 Suppression of local charging

Levels of local charging on all the surfaces of a spacecraft should be lowered, if possible, to several volts (defined on a mission basis). To this end, all exposed surface materials, where possible, should have an electrical conductivity that satisfies target values.

The area of exceptional sections (including floating conductor sections and insulators) should be minimized.

Special care should be given to the surfaces in the vicinity of the entrance to and in the visual vicinity of the sensors for low-energy charged particles.

5.3.2.2 Suppression of charged particle emission

Negative electrodes should not be exposed to outer space to prevent accelerated emission of photoelectrons and secondary electrons. In the vicinity of the entrance to and in the visual vicinity of the sensors for low-energy charged particles, consideration should be given to suppress the inflow of photoelectrons and secondary electrons emitted from the surfaces of a spacecraft into the sensors.

5.3.2.3 Active control of spacecraft electrical potential

Consideration should be given to charged particle emission devices and plasma contactors used for active control of spacecraft electrical potential so that they do not interfere with observation purposes (e.g., suppression of contamination, suppression of large-scale electrical potential variation, etc.).

5.4 Solar panel

5.4.1 Explanatory comments and applicability

The intended systems in this section are high-output solar panels and solar panel paddle drive mechanisms. All requirements related to the solar panel stated in this section apply to every solar panel consisting of solar cells, substrates, coverglass, and interconnectors. Furthermore, in addition to secondary arc, items to consider for the solar panel include electrical output performance reduction and power leakage into plasma.

An insulator, if located in the immediate vicinity of a metallic material, can cause primary discharge, and the existence of an electrical field parallel to the surface of an insulator can also lead to the flash-over that spreads along the insulator surface. Plasma, when it is fully grown, can cause a short circuit between solar cell rows or between solar cells and their substrates. Secondary arc will not occur if either of these problems is eliminated. Furthermore, suppressing the occurrence and the development of trigger arc allows us to avoid and reduce adverse effect of primary discharge on electrical output. Note that coverglass for solar cells, adhesives used for bonding the coverglass and solar cells to the substrate, as well as insulating sheets to be mounted on the substrate surface are dealt with only in this section as multi-layered high-resistance materials consisting of solar cells and conductive face sheets (refer to 5.2-6 for the definition of a high-resistance material).

The requirements for solar panel paddle drive mechanisms apply to electric motors, gear devices, slip rings, and their drive systems used to maintain the direction of the solar panel.

The following shows the design standard required for the prevention of primary and secondary arc that may occur on the solar panel.

5.4.2 Basic principle

(1) All metal parts, regardless of their sizes, should be grounded to the panel structure except that grounding is judged to be difficult, no adverse effects are found for the function and performance of the solar panel, and client approval is obtained.

(2) Metal sections with surface areas too small to be grounded, such as empty connector pins should be left electrically floating. The exposed section should be covered with a high-resistance body to prevent the development of the accumulation of electric charge.

(3) Adjacent layout should be avoided if the boundary between metal and insulator sections is exposed to outside. If such adjacent layout cannot be avoided, these sections should be arranged through a high-resistance material with a minimum width of 5 mm. In that case, risk evaluation should be performed from the viewpoint of EMC and ESD.

(4)High-resistance materials should be used for the parts and materials other than metal parts. Follow Item 5.2.6 for the rules and regulations required for high-resistance matelials. Use of insulators requires client approval.

(5) Kapton layers used for the insulation between solar cells and substrates should be thick enough so that they can retain predetermined functions during a mission period in view of various factors that the solar panel might be affected throughout manufacturing processes or while in orbit. (Kapton layers are recommended to have a thickness of 50 μm or more.) Note that these various factors include time-related insulation resistance degradation in orbit, damage to surface kapton skin due to the tip of carbon fibers, and human-related damage during other manufacturing processes.

(6) Do not use standard teflon (refer to Item 5.2.6) as a covering material for power cables. If standard teflon has to be used, they should be further wrapped with kapton insulation.

(7) All the alignment cubes with ungrounded metal faces should be removed before the launch of a spacecraft. If they cannot be removed, it is desirable to cover them with kapton tape.

(8) Electrical conductivity related to surface materials should comply with Item 5.2.4. If their surface electrical conductivity cannot be achieved, it is desirable to evaluate the effect of static discharge on related equipment before using the materials.

(9) When rows of solar cells are grounded for secondary arc suppression, good defoaming processing should be given during blending so that no air voids can remain. Good baking processing should also be given so that grounding materials cannot be a contaminant source in orbit.

(10) It is desirable to keep the voltage between two solar cells next to each other as low as possible. When the voltage is 30 V or lower, the probability of the occurrence of secondary arc is very low.

(11) It is desirable to make the parallel circuits of the solar panel as small as possible so as to reduce the amount of supply current at the occurrence of secondary arc. Duration time of secondary arc is highly dependent on the amount of electric current. The shorter the duration time, the lighter the damage.

(12) It is desirable to keep the gap between rows of solar cells as wide as possible. The wider the gap, the lower the probability of the occurrence of secondary arc.

5.4.3 Solar panel paddle drive mechanism

(1) To suppress the secondary arc due to the primary power-supply for a spacecraft in slip ring

assembly, a good consideration should be given to the distance between the slip rings, the height of the barriers between the rings, materials, and ring configuration.

(2) External exposure harness to be connected to the solar panel should comply with Item 5.4.2.

5.5 High-voltage system

In this document, the equipment that deals with a voltage of 70 V or more is defined as a high-voltage system (refer to Appendix I. 8.3).

5.5.1 Normative rules and regulations

(1) All the devices installed in a spacecraft should be proved that the possibility of the occurrence of Paschen discharge is within an acceptable range in all possible gas environments from the beginning of spacecraft launch to the end of its operation.

(2) If high-voltage systems have to be installed in the vicinity of the high-voltage systems or devices that will cause outgassing, the greatest care should be given to electrical insulation design.

(a) Right values
It is desirable to exactly understand the increase in the outgassing gas pressure in the vicinity of high-voltage systems through numerical calculation.

(b) If analysis predicts that the voltage level is higher than the Paschen discharge threshold at the worst gas pressure, consideration should be given to equipment layout and exposed electrode sections. Furthermore, the possibility of the occurrence of Paschen discharge should be lowered within an acceptable range.

5.6 Internal parts and materials

5.6.1 Overview

The parts to be considered are all dielectrics (or combinations of floating conductors and dielectric) that can cause insulation breakdown due to electrostatic discharge, including the following items:

(1) Cable insulator
(2) Printed-circuit board
(3) Surface-mount package for integrated circuits
(4) Ungrounded spot shield

In addition to these items, every internal system containing insulators or floating conductors can cause similar electrostatic breakdown, including the following items:

(5) Three-axis acceleration meter
(6) MEMS (Micro Electro Mechanical System: A device consisting of machine element parts and sensors integrated on a silicon substrate)

5.6.2 Requirements

5.6.2.1 Grounding and conductivity

(1) In principle, every metal component should be designed and manufactured so that they are equipped with a grounding section. The intended items in these requirements include structural elements, spot shields, transformer cores, metal component packages, and unused circuit patterns on PCBs. Furthermore, they include the cables that have passed ground tests but have not yet used in orbit, as well as the cables that will be temporarily isolated when a relay (e.g., antenna feed) works in the redundancy mode.

(2) In principle, conductors should be grounded. In particular, unused floating conductors must be grounded or removed. The intended items include unused connector pins and unused wiring patterns on printed-circuit boards. Note that any component that cannot be directly grounded to a

conductor should be connected to a grounding point through a high-resistance material. In that case, the resistance of the high-resistance material should be selected so that the charge relaxation time of the component can be shorter than the orbital time of a spacecraft.

(3) When cables are placed inside of electronic equipment, surge countermeasures should be taken to the positions through which cables are pulled from the outside of the main body of a spacecraft in order to prevent the electrical disturbance due to charging and discharging in covering materials for external exposure cables, to signal lines or the electronic equipment connected to the signal lines. Specific countermeasures include the insertion of limiting resistors, ferrite cores, capacitors, and current limiters.

(4) It is desirable to pass cables through the places where they can take advantage of the shielding effect provided by other components.

(5) Cables should not be passed across an opening of a spacecraft to avoid the cables from being exposed to outer space.

(6) Every cable placed outside of the main body of a spacecraft should be wrapped with a high-resistance amterial that is at least thick enough to provide good insulation.

5.6.2.2 Internal charged electrical field

Internal charging can be grouped into two cases: (1) Surface charging of the dielectric used in the structure of a spacecraft and (2) charging due to the inflow or accumulation of charged particles into dielectric materials (not only those used inside the structure but also those used as surface materials). The following explains the criteria for judgment concerning the charged electrical field resulting from both cases:

(1) Internal charging is caused by charged particles captured by a spacecraft in orbit. The range of the captured particles depends on their energy. Thus, it is necessary to consider whether they stay in the surface materials of a spacecraft or whether they pass through the surface materials and infiltrate into the structure of the spacecraft. The range can be calculated by using various empirical formulae or by the computer simulation method shown in Item (3).

In this section, the range of an electron is calculated as an example, assuming that a kapton film with a thickness of 25μm is irradiated with electrons, each having an energy level of 35 keV. By using the empirical formula given by Katz and Penfold, the range is calculated to be about 10μm [[2]]. In this case, all electrons immediately after irradiation come to a stop at 10 μm from the irradiated surface in the thickness direction.

(2) To minimize the electrical discharge caused by the internal charging of dielectrics , the strength of the electrical field in the dielectrics should be kept less than 10^7 V/m. Furthermore, care should be given to the amount of the electric charge accumulated in the dielectrics . In this section, the amount of the electric charge accumulated in the dielectrics is calculated as an example assuming that a kapton film with a specific dielectric constant of 3.4 is irradiated with electrons; each having an energy level of 35 keV. Assuming that the range of electron 10 μm, irradiated electrons are distributed and accumulated evenly from the exposed surface, and the strength of the electrical field is 10^7 V/m, the electric charge density is calculated to be about 30 C/m^3.

(3) It should be shown that the above item (1) is observed by applying numerical electrical field simulation methods such as the finite element method, difference method, or electric charge superposition method, or by conducting experimental tests.

(4) Determination of charging environments should be based on either of the following worst-case environments:

(a) The worst-case model in which outer-zone electron fluence and orbit calculation software are combined

(For example, environmental models such as FLUMIC or AE-8 can be used. However, care should be given during the actual simulation to the fact that time-averaged environmental models such as AE-8 do not provide a model for the worst-case environment simulation.)

(b) The worst-case spectrum related to intended orbit

(c) The worst-case environment presented by clients

(5) Systems such as MEMS and three-axis acceleration meters sensitive to internal electrical field and containing insulators or floating conductors should be carefully designed at the system design stage of the electrical field that could be induced by internal charging.

5.7 Tether

5.7.1 Definition

A tether is a conductive or non-conductive flexible cable provided from a spacecraft or connecting two points on the spacecraft. It serves as an electric interface with a space environment. This section describes qualitative guidelines for a spacecraft with tether systems. Quantitative levels are defined in each mission as needed.

5.7.2 Guideline

5.7.2.1 Consideration to the maximum voltage (conductive tether)

(1) The evaluation of the maximum voltage should be based on the following items: (1) the electromagnetic induction voltage associated with the orbital motion across a magnetic field, and (2) the charged potential difference “between the surfaces of a spacecraft” connected by thethers or “between independent spacecraft.” Consideration should also be given to the evaluation results.

(2) Tether covering insulating materials should have a withstanding voltage larger than the predicted maximum voltage.

5.7.2.2 Consideration to the maximum current (conductive tether)

(1) The evaluation of the maximum current should be based on voltage and power collecting area. Consideration should be given to the evaluation results.

(2) The maximum current predicted from a power collecting area should not exceed the amount of transmittable electric current.

(3) When a large current flows, consideration should be given to plasma heating, ohmic heating, and the interaction with outer magnetic field. The amount of the electric current should be limited as needed.

5.7.2.3 Consideration to electric pathways (conductive tether)

(1) Care should be given to conductive pathways other than planned pathways in the structure of a spacecraft so that they have so high a resistance that electric current cannot pass through them. Care should also be given to the effects due to insulation damage in the structure of a spacecraft.

(2) When insulator damage occurs, electric current flows in a spacecraft or through unexpected pathways via plasma. Thus, consideration should be given to insulators so that they cannot be damaged while they are treated or deployed.

5.7.2.4 Consideration to damage

(1) Consideration should be given to the damage to tethers due to mechanical and electrodynamic vibration and the damage to deployment and repair mechanisms. To suppress electrodynamic vibration, the vibration should be detected, and a vibration control system should be installed as needed.

(2) Consideration should be given to mechanical hazards in a spacecraft, which could be caused by tether fracture due to electric burnout.

5.7.2.5 Consideration to electrostatic stiction

Consideration should be given to not only vacuum stiction but also electrostatic stiction.

Note: The following items are not listed under the coverage of this section.

(1) Mechanical strength, thermal deformation, and torsional stress
(2) Damage due to dust or debris collision and resultant debris particles
(3) Outgases
(4) Deterioration due to radiation and atomic oxygen

5.8 Electrical propulsion

5.8.1 Explanatory comments

Electrical propulsion systems are thrusters used for a spacecraft. Although their thrust force is weak, they are used for attitude control, station keeping, orbit-raising, and interplanetary space mission. The requirements in this section apply to all the systems that produce thrust force by discharging charged particles, including the following thrusters:

(1) Field emission electrical propulsion (FEEP) thruster
(2) Ion engine with a grid system (e.g., micro-wave discharge, high-frequency discharge, and direct current discharge types)
(3) Ion engine without a grid system (e.g., hall thruster)
(4) MPD thruster
(5) PPT (Pulsed Plasma Thruster)

Most of these systems require neutralization electron discharge devices or neutral plasma discharge devices as an essential component. Thus, the same design and testing processes are used when they are considered.

Furthermore, neutralization devices can also be applied to the control of the spacecraft electrical potential that can be induced by interaction with a space environment. Thus, the requirements in this section apply to the following neutralization devices.

(6) Plasma contactor (e.g., hollow-cathode or micro-wave discharge neutralization device)
(7) Electrical field emission-type neutralization device
(8) Electron beam discharge device
(9) Heated filament-type electron discharge device

Most of the requirements stated in this document do not apply to electrical propulsion systems that do not discharge charged particles, for example, DC arc jets or resist jets. This standard, however, applies to the aspect that these systems also discharge the contaminants or gases that can affect the electrostatic interaction outside a spacecraft.

5.8.2 Requirements

5.8.2.1 Grounding of the power-supply for electrical propulsion equipment

Most electrical propulsion systems deal with high voltage. The high-voltage power-supply should be grounded to a spacecraft—either positive, negative, or center tap should be grounded. Grounding methods include simple, resistance insertion, and zener diode insertion [3].

5.8.2.2 Neutralization of a spacecraft

(1) To prove that the beam current produced by an electrical propulsion system is completely neutralized, it should be shown that the performance of neutralization devices matches the maximum beam current and neutralization current in the worst-case plasma environment including time-related changes.

Note 1: The amount of charged-particle current induced by environment is generally insufficient for the neutralization of a spacecraft with an electrical propulsion system installed in it. Thus, the spacecraft should be equipped with neutralization devices capable of supplying a greater amount of electric current than the beam discharged from the electrical propulsion system.

Note 2: Inferior neutralization induces spacecraft charging, making its electrical potential shift toward negative as time advances. The injected-ions move backward and jump into the spacecraft at high speed, causing interference with other systems, damage to the spacecraft, drastic decrease in electrical propulsion efficiency, contamination on the surface of the spacecraft, and sputtering errosion.

(2) A hollow thruster is generally equipped with a hollow cathode used as a part of a beam neutralizing and ionization process. Note that the requirements for the neutralization devices shown in the above item (1) can apply to the hollow cathode.

(3) Thrusters should not be operated in a spacecraft unless they are equipped with related neutralization devices.

(4) A mechanism that can release electron current that matches ion beam current or other positive current should be clearly specified, or the mechanism should have the control mechanism that achieves the function

(5) A mechanism that can detect the reduction in neutralizing capacity should be installed.

(6) A safety treatment logic circuit for detecting the reduction in neutralizing capacity should be installed.

(7) An electric wire line that connects neutralization devices and discharge devices should be installed so that a complete circuit including the flow of electrons from neutralization devices and thruster ions can be established.

Note 1: It is worried that EMC problems will occur as long as neutralization current passes through other on-board systems.

Note 2: The ion current discharged from thrusters and the electron current from neutralization devices constitute a current loop. Thus, it is worried that if these devices are separately installed, they will emit magnetic components, causing EMC or geomagnetic torque problems. In particular, care should be given when multiple on-board thrusters are simultaneously operated and neutralized by a smaller number of neutralization devices.

(8) Several low-power thrusters such as FEEP for LEO can be neutralized by using ionosphere plasma current. However, this technique is classified as a very special method. Note that neutralization devices may be eliminated when the worst-case outer current (each current component in Formula (I. 3-8)) for thruster operation has been considered and the resultant method is proved to be appropriate.

(9) Neutralization devices that serve to control spacecraft charging in a spacecraft without electrical propulsion thrusters should be installed directly to the ground section of the spacecraft with the exception that the electrical potential of the spacecraft is intentionally biased to a certain voltage. In this case, a neutralization device can be installed to the ground section through a power-supply.

(10) When neutralization devices that serve to control spacecraft charging are installed, they should a current capacity greater than the maximum external plasma current.

(11) No active control and measurement of spacecraft electrical potential is necessary for the neutralization devices that discharge low-energy neutral plasma.

Note: However, the measurements can be useful for diagnosing the operation of thrusters.

(12) Neutralization devices (e.g., electron gun) that discharge positive or negative high-energy charged particles should not be used because they could increase charging levels as time passes, and furthermore, tit is worried that the potential barrier will cause malfunction.

Note: The discharge device current of these discharge devices should be actively controlled for the maintenance of a constant electrical potential in light of real-time electrical potential measurement (Langmuir probe or ion energy analyzer)

(13) The following items can be regarded as exceptions of the above item (12).

(a) When the surface is a complete conductor, or when it is proved that the surrounding plasma has such a low level of energy and thermal velocity that the time-varying surface charging remain at a very low level.

(b) When scientific measurement requires the surrounding plasma not to be disturbed

5.8.2.3 Beam neutralization

The space potential distribution within a beam and the spatial volume that the beam occupies depend on how efficiently electrons from neutralization devices can infiltrate into the beam and cause it to be neutralized. If beams around a spacecraft cannot be neutralized, it is worried that the electric charge exchange ions produced in the beams will cause serious contamination and contribute to the dispersion of the beams.

(1) In spacecraft design processes, consideration should be given to beam neutralization.

(2) It is desirable to locate the neutralization devices of a spacecraft as closely as possible to its electrical propulsion system so that charged beams can be neutralized in the closest vicinity of the spacecraft. Note that the neutralization devices are usually installed in the same unit as the thrusters.

(3) These requirements, however, are the matters that can trade off between the advantages brought by the simplified design of electrical propulsion systems and the disadvantages brought by the effect of the above items. For example, placing a single neutralization device between two or among more electrical propulsion systems can contribute to weight-reduction. Note that the presented method needs client approval. Furthermore, evidence should be shown to prove that beams can be continuously neutralized.

5.8.2.4 Effect of low-speed, low-temperature plasma cloud

A spacecraft will be covered with low-speed, low-temperature plasma because of the interaction with neutral gases discharged from the spacecraft and its electrical propulsion system, high-speed beams, and plasma. Electric current can be induced through the plasma between the exposed high-voltage electrodes, for example, of electrical propulsion systems and high-voltage solar panels. Thus, the effect should be considered.

Note: Neutralization electron current, if it is constantly flowing into other sections of a spacecraft, can induce spacecraft charging because of the loss of the neutralization current by that amount.

5.8.2.5 Contamination

It is worried that neutral particles discharged from the electrical propulsion system of a spacecraft are partially ionized, return to the spacecraft, and serve as a contaminant source. These neutral particles are, for example, atoms of the materials separated from propellants and thrusters due to erosion (atoms emitted from electrostatic grids or plasma acceleration sections).

(1) Propellants for thrusters and neutralization devices as well as thruster structural materials should be selected with the effect of contaminants in mind.

Note: Because of the above reason, rare gas is typically selected as a medium for propellants and neutralization devices.

(2) Consideration should be given to the type of atoms discharged from the electrical propulsion system and the spatial distribution of these atoms.

(3) The contamination due to discharged atoms should be decreased to the level approved by clients.

Note: Acceptable contamination levels are diversified and determined on the basis of individual cases. Note that acceptable levels from the viewpoint of optical engineering and thermal control can be stricter than those in other areas.

5.8.2.6 Plasma-jet irradiation

(1) The orbit of high-speed ions discharged from the electrical propulsion system of a spacecraft should be designed so that the ions do not collide with the other surfaces of the spacecraft or their speed is reduced to the lowest possible level.

(2) The effect of transient ion orbit at start and end of acceleration or during on-and-off operation on a spacecraft should be considered and shown to be within an acceptable range.

Note: Neglecting the above requirements can lead to the occurrence of high-level surface material sputtering (which results in surface erosion), function and performance degradation in the irradiated area, and undesirable thrust torque.

(3) If the ion orbit collides with the other sections of a spacecraft, it should be proved that sputtering and torque is lower than the level agreed upon by the client.

5.8.2.7 Effect of neutral gas

Evaluation should be conducted as to the density level of the plasma located around a spacecraft and caused by natural gas emission. Furthermore, consideration should be given to the discharge that occurs through the natural gas.

Note: A spacecraft and its electrical propulsion systems could discharge a massive amount of neutral gases. These gases, for example, are the gases discharged from electric thermal thrusters or chemical thrusters as well as those discharged at start-up from ion engines with a grid system and MPD thrusters before they start internal discharge. In addition, a spacecraft tends to emit a great amount of gas immediately after it is launched. These neutral gas emissions are worried to induce Paschen discharge in high-voltage equipment and sections.

Natural gases discharged from a spacecraft can cause unforeseen discharge, increase in surrounding plasma density due to charge-exchange reactions, and increase in leakage current to acceleration grids. In addition, they can drastically increase the density of the plasma around the spacecraft after they have been photoionized. Furthermore, the resultant effects can include the discharge induced on charged surfaces or between various surfaces that are actively maintained at different electrical potentials.

5.8.2.8 Peripherals around installed electrical propulsion system

Once neutralization failure occurs for even a very short time, the injected-ions move backward and jump into the spacecraft at a high speed. Thus, care should be given to the local charging of the spacecraft due to the high-energy ions. The inflow is limited to the area around the electrical propulsion thruster. Thus, all the surfaces of the related sections should be covered with conductive materials or should be grounded.

5.8.3 General statement

This section covers the plasma, electrical field, and magnetic field (including the field produced by neutralization devices) outside a spacecraft, as well as requirements related to the interaction with the spacecraft. Furthermore, the requirements are related to (1) the effect of the plasma environment on the operation of the electrical propulsion system and (2) the effect of the electrical propulsion system on the other systems of the spacecraft through the interaction with the plasma environment.

The following shows the items not covered in this section in terms of the interaction between the electrical propulsion system and the environment.

(1) Neutralization devices can be powerful EMC interference sources, and plasma contactors are basically exposed discharge sources. Thus, it should be confirmed by implementing the EMC standards that the plasma contactors do not adversely interfere with the other systems of the spacecraft.

(2) Ion engines, ion neutralization devices, and discharged beams are the sources of visible light, ultra-violet rays, and heat rays. Thus, consideration should be given to the effect of these sources on optical systems (e.g., star tracker) and the heat balance of a spacecraft.

(3) Electrical propulsion plasma beams are worried to interrupt the transmission and reception through radio communication [4]. Thus, consideration should be given to the antenna layout based on these plasma beams.

(4) Extreme care should be given to designing electrical insulation because electrical propulsion systems often treat with high-voltage. The sections that serve as high-voltage insulators should be identified, and, to prepare for unforeseeable circumstances, if an insulation failure occurs at any of the sections, the short-circuit pathway should be clarified. Thus, care should be given to the design of these sections so that no short-circuit current flows into other equipment through another subsystem, further spreading damage to other equipment.

6 Testing and Verification Methods

6.1 Principle

Confirming the soundness of a spacecraft by using the verification plan established in Item 5.1.1.4.

6.2 Surface materials

6.2.1 Testing of exceptional materials

6.2.1.1 Testing

(1) If the requirements described in the items (1)-(3) of Item 5.2.2 are contradictory from the perspective of design, or if the analysis described in the item (5) of Item 5.2.6 cannot be applied because the shape of the intended spacecraft is too complex to be represented in two-dimensional figures, or if the electrical properties of the materials in use are unknown, use the relevant surface materials for testing.

(2) The rationale, detailed testing procedure, and test results involved in testing should be presented to the clients for approval.

6.2.1.2 Material test conditions

(1) Testing should ensure the reproduction of the charging in the worst-case environment as well as the precise evaluation of the response of the materials under the relevant environment.

(2) Tests should be conducted in a vacuum state as well as in an electronic irradiation state.

(3) Tests on a testing material should not be conducted before it finishes outgassing.

Note: The electrical conductivity of the material can vary as it loses volatile substances because of outgassing.

6.2.1.3 Parameters that affect testing

6.2.1.3.1 Temperature dependency
Tests should be conducted within the operating temperature range in outer space

Note: Temperature can significantly affect the electric conductivity of dielectric materials: the lower the temperature, the higher the volume resistivity.

6.2.1.3.2 Effect of electrical field and radiation
Test should be conducted with electron beams—the electron current in the relevant environment should be reproduced. Furthermore, it is desirable to cause the dosage-rate of radiation to be reproduced.

Note: During the electron beam irradiation, the change in electrical conductivity during measurement affects the surface electrical potential. It also affects the risk evaluation of discharging.

6.2.1.4 Test-result acceptance criteria
It should be shown by direct experimental simulation or the calculation based on material parameters that the surface electrical potential and the electrical field of test samples are constantly lower than the specified electrical potential and the electrical field to be applied to the test samples over the whole temperature range predicted in the worst-case environment and during missions.

6.2.1.5 Material charging test
(1) New materials that have not been used in outer space should be tested with the following conditions for discharge threshold values.

(2) Test conditions
(a) The accuracy of test samples should be better than that of the instruments used for measuring the surface electrical potential and the electrical field. Typical test samples should have an area of 30 mm × 30 mm or larger.
(b) Take the target orbital environment into account and determine the energy and the flux of radiation (electrons and ions). Note that the energy spectrum in the orbital environment is desirable, but the energy spectrum based on a single energy can be used as an alternative, provided that the simulation of the spectrum is based on the worst-case environment.
(c) Temperature should be changed over the whole operating temperature range.
(d) Refer to, Chapter 6-9 of JERG-2-14 for orbit-based radiation and temperature environment.

(3) Electrostatic discharge risk evaluation should be based on the following electric properties:
(a) Volume resistivity
(b) Surface resistivity
(c) Radiation-induced conductivity (RIC)
(d) Specific dielectric constant
(e) Discharge generating threshold
(f) Secondary electron emission coefficient
(g) Photoelectron emission coefficient

6.2.2 Verification: Space Environment Standard

6.2.2.1 Grounding
(1) Electrical conductivity and grounding should be tested, examined, and evaluated.

(2) Examination and test plans on grounding should be prepared as a quality verification testing item

for spacecraft development.

6.2.2.2 Material selection

(1) Material parameters related to charging, such as volume resistivity and surface resistivity should be confirmed. This confirmation should be performed by material suppliers or performed as a part of a material selection process. Furthermore, it is also necessary to understand the effect of materials on physical parameters in a space environment.

(2) Physical values should be determined by conducting the tests according to the standards agreed upon with clients.

6.2.2.3 Environmental effects

It should be demonstrated that the electrical conductivity of a current pathway cannot be damaged by the following environmental effects:

(a) Vibration
(b) Change in heat
(c) Material deterioration

Note: The deterioration shown in the above item (3) refers to that due to atomic oxygen in a space environment, radiation, ultra-violet rays, and heat.

6.2.2.4 Computer model and analysis results

(1) Specifications of computer analysis models and resultant analysis results should be shown.

(2) The validity of analysis results should be compared and analyzed with reference to the past analysis results of space vehicles.

(3) The analysis of exposed parts that are too small to be simulated can be neglected, provide that the charging and discharging at the locations do not affect peripheral equipment. If the charging and discharging are too significant to be neglected, they should be evaluated according to the risk evaluation shown in item 5.1.1.3.

(5) The change (deterioration) in the physical parameters of surface materials in a space environment should be considered, and the effect of the change on their charged state should be analyzed qualitatively and/or quantitatively.

6.3 Spacecraft for plasma measurement

6.3.1 Overview

The qualitative test and verification corresponding to Item 5.3.2 is described. Quantitative levels are defined in each mission in response to risk evaluation results.

6.3.2 Test and verification

The following items should be confirmed during unit testing, or overall testing, or both.

6.3.2.1 Suppression of local charging

The electrical conductivity of the structure of a spacecraft should be actually measured on all exposed surfaces to confirm that they satisfy their design target values.

Use material measurement and bonding resistance measurement methods as an alternative for the positions where actual measurement of electrical conductivity is difficult.

Confirm the implementation of the “Area Minimization Measures” prepared at the design stage in exceptional places including floating conductor sections and insulators.

6.3.2.2 Suppression of charged particle emission

Confirm the implementation of the “Non-Exposure Measures for Negative Electrodes” prepared at the design stage.

Confirm the implementation of the “Suppression Measures against the Inflow of Photoelectrons and Secondary Electrons” prepared at the design stage in the vicinity of and in the visual vicinity of the entrance of the low-energy electron energy analyzer.

6.3.2.3 Active control of spacecraft electrical potential

Confirm the implementation of the “Measures for Suppressing Contamination and Large-Scale Electrical Potential Variation” prepared at the design stage with ion and electron emission devices and plasma contactors.

6.4 Solar panel

6.4.1 Solar panel test (negative charging)

6.4.1.1 Overview

There are two sets of requirements that apply to the solar panel: (1) the surface charging test requirements explained in Items 6.2.1 for elemental samples of each solar panel component (e.g., a single coverglass plate or a single connector) and (2) the following test requirements related to the degradation of electrical performance of solar cells and control of secondary arc in the assembled state as a solar panel:

6.4.1.2 Test items

The following three items should be tested:

(1) Threshold measurement of electrical potential difference between a coverglass plate and a solar cell at the occurrence of primary discharge

(2) Degradation of electrical output performance of solar cells caused by the contamination or breakdown of PN conjunctions attributed to primary discharge. Here, the degradation of electrical output performance refers not to the loss of electrical output from a single serial circuit of a solar panel, but to gradual reduction in output voltage because of repeated primary discharge.

(3) Tests related to sustained secondary arc (temporary sustained arc or permanent sustained arc) resulting from primary discharge

6.4.1.3 Cases that do not require testing

No testing is required for the following items:

(1) Testing of the items (1)-(3) of Item 6.4.1.2 can be skipped, provided that the same type was designed in the past and the durability has been confirmed by the test conducted on the assumption of the same orbit (GEO, PEO, etc.).

(2) Testing of the items (2) and (3) can be skipped for LEO satellites with orbital inclinations that do not intersect with the aurora zone, provided that the maximum generation voltage at the timer when the satellite moves into daytime from nighttime is smaller than the threshold obtained in Item 6.4.1.2 because primary discharge is very unlikely to occur.

(3) Testing of the item (3) of Item 6.4.1.2 can be skipped, provided that the probability of the occurrence of sustained secondary arc can be determined by the design to be far below the experimental values set by reference to types of solar cell, the voltage between adjacent cells, and the gaps between solar cells (refer to Appendix-Ⅲ).

(4) To evaluate the discharge by the negative charging attributable to the failure of the neutralization of ion beams from a spacecraft with an electrical propulsion system, firstly estimate the potential difference between the coverglass surfaces and the spacecraft. Testing of the items (2) and (3) of Item 6.3.2.3.2 can be skipped, provided that the maximum potential difference is equal to or

lower than the threshold at which primary discharge occurs. Even if the potential difference is higher than the threshold value, the item (2) of Item 6.3.2.3.2 can be skipped because repeated primary discharge is quite unlikely to occur.

6.4.1.4 Testing conditions

Before starting the test of a solar panel, estimate the number of primary discharge on the solar panel during its operation period. On this occasion, pay attention to the following items:

(1) Take advantage of charging analysis tools to find the environmental conditions under which the potential difference between the coverglass and the solar cells exceeds the threshold for primary discharge to occur, and estimate the number of occurrences of the primary discharge using the occurrence probability of the environmental conditions and the charging time for the potential difference to exceed the threshold.

(2) Estimate the number of occurrences of primary discharge that occurs both across a whole spacecraft and on the surface of a test-coupon.

(3) With respect to GEO satellites, estimate the number of occurrences of primary discharge in the daytime and nighttime states.

(4) With respect to PEO satellites, estimate the number of occurrences of primary discharge in a state when the solar cell-mounted side of the solar panel is facing the direction of movement and in a state when it is not facing the direction of movement.

(5) Estimate the number of occurrences of primary discharge in the inverted potential gradient state (surface potential of coverglass is higher than that of cell electrode) and in the normal potential gradient state (surface potential of a coverglass plate is lower than that of a cell electrode).

(6) With respect to the neutralization device failure of the electrical propulsion system, estimate the number of occurrences of primary discharge using the total time from neutralization device failure to beam cut-off and the time for the potential difference between the coverglass and the spacecraft to reach the threshold.

6.4.1.5 Intentional generation of primary discharge

(1) It is desirable to generate a primary discharge by using the potential gradient states obtained from charging analysis.

(2) The most practical method for GEO satellites is to use “keV electron beams” to charge the coverglass surface for the generation of primary discharge.

(3) When the solar cell-mounted side of the solar panel is facing the direction of movement, the most practical method for LEO satellites is to use low-energy plasma. On the other hand, when the side is facing the wake side, a practical method is to use “keV election beams” to charge the glass surface for the generation of primary discharge.

(4) Using low-energy plasma is a practical method for the tests that simulate the neutralization device failure of the electrical propulsion systems.

(5) When test items are in the category of item (3) of Item 6.4.1.2 (secondary arc), and when it has been confirmed by some ways that the probability the transition from primary discharge to secondary arc remains the same regardless of any primary discharge pulse generating means, effective primary discharge generation methods can be used without attention to the items (2) and (3) of Item 6.4.1.2. Use of low-energy plasma, electron beam focusing, and laser pulse triggering are known as such effective approaches to generate a primary discharge between the rows of solar cells.

(6) When test items are in the category of item (2) of Item 6.4.1.2 (degradation of electrical

performance of solar cells due to primary discharge), it is desirable to generate primary discharge pulses according to the means described in the above items (2) and (3). Furthermore, generate a primary discharge the same number of times per test-coupon as the number estimated in the item (2) of Items 6.4.1.4. Then, estimate the amount of decrease in electrical power at the end of operation period on the basis of how the electrical performance has degraded. Make sure to consider the error that may be involved in charging analysis so that a sufficient margin can be ensured.

(7) To examine the probability of occurrence of the secondary arc between the solar panel and the underlying conductive substrate, cause a primary discharge by using the test-coupon used to simulate the intended discharge positions a specific number of times that corresponds to the sum of the number of the occurrences of primary discharge predicted between the cells with the maximum potential difference.

(8) With respect to the probability of occurrence of the secondary arc between the adjacent rows of the solar panel, generate a primary discharge between the solar cells of a test-coupon that serves to simulate the adjacent rows a specific number of times that corresponds to the sum of the number of occurrences of primary discharge predicted to occur between the solar cells that have the maximum potential difference.

(9) The sum of the number of occurrences of primary discharge in the above (7) and (8) refers to the summation of all the number of occurrences of primary discharge at the pertinent sections of each circuit in the solar panel.

6.4.1.6 Explanatory comments on testing

(1) Electron beam tests should be conducted in a test room with the following characteristics:

(a) Vacuum environment in which “keV electron beams” can be stably irradiated: normally about 3×10^{-3} Pa.

(b) Power-supply (SAS) powerful enough to reproduce the dynamic reaction of the solar panel in response to a transient short-circuit state. It should have the least possible inter-terminal output capacity, and its incoming current should be suppressed as low as possible.

(c) High-voltage feed-through cables to connect to power supplies and test-coupons (maximum: 20 kV)

(d) Devices to simultaneously monitor and record the transient state of discharge current and inter-row voltage

(a) Visual inspection, photo shooting, and video recording of test samples during a test

(e) High-voltage power-supply (about -10 kV)

(f) Electron beam (20 keV)

(g) Noncontact surface potential meter for x-y scanning: 0-20 kV

(2) Low-energy plasma tests should be conducted in a test room with the following characteristics:

(a) Vacuum environment in which no glow discharge is generated on the surface of the solar panel even when a negative voltage of -1 kV is applied to the solar panel. When the degree of vacuum is lower than 0.1 Pa, a primary discharge usually occurs at the end of the insulator in contact with an exposed electrode or at a triple junction, and no glow discharge occurs on the surface of an exposed electrode.

(b) Power-supply (SAS) powerful enough to reproduce the dynamic reaction of the solar panel in response to a transient short-circuit state. It should have the least possible inter-terminal output capacity, and its incoming current should be suppressed as low as possible.

(c) High-voltage feed-through cables to connect to power supplies and test-coupons (maximum: 3 kV)

(d) Devices to simultaneously monitor and record the transient state of discharge current and inter-row voltage

(e) Visual inspection, photo shooting, and video recording of test samples during a test

(h) High-voltage power-supply (about -1 kV)

(f) Low-energy plasma source such as an electron impact-type direct current plasma source and an electron cyclotron resonance-type plasma source. Compared with the voltage applied to a test-coupon, electron temperatures should be kept quite low—normally several eV or lower. Plasma is required to have a level of density that allows debye shielding to prevent the sheath around a test-coupon from reaching the wall.

(g) Langmuir probes for background plasma measurement

(3) The relevant test coupons have the following main characteristics:

(a) The test-coupon should serve as a mock flight model. A test-coupon can be smaller than the actual model because it is not practical to test the actual solar panel under sunlight irradiation and in a vacuum environment.

(b) A sufficient number of test-coupons should be prepared in view of the variation during the production of flight products.

(c) There is not need to irradiate solar cells with light during a test

(d) It is desirable to use a microscope to observe the details on the surface state of test-coupons before and after conducting a test.

(e) It is desirable to use a video during a test and record the occurrence position of discharge.

(4) When test items are in the category of item (3) of Item 6.4.1.2 (secondary arc), care should be taken to the following items on test-coupons:

(a) At least, a set of four solar cells are required, which consists of two strings of two cells connected in series. Each string has two lead wires: one connected to its positive (P) terminal and the other connected to its negative (N) terminal

(b) The gap between adjacent strings should be the same as the minimum design interval.

(5) The following shows the main characteristics of the test circuit.

(a) Use the external capacitance C_{SAT} to simulate the capacitance between the structure of a spacecraft and outer space. The capacitance is normally 1 nF, or less. Thus, if C_{SAT} is almost the same as the floating capacitance included in the cables of the test circuit, there is no need to insert the external capacitance.

(b) Use the external capacitance C_{ext} to simulate the capacitance of insulators such as the coverglass mounted on the solar panel or that of large area insulators such as the thermal control materials near the solar panel. With respect to the energy injection through the external capacitance, adjust the external capacitance C_{ext} so that the energy of primary discharge pulses can be equivalent to the energy predicted in orbit. Furthermore, the energy injection is due to the spread of the plasma with a finite velocity. Thus, it is desirable to use delay circuits based on the following two assumptions so that the time waveform of the current supplied from C_{ex} to primary discharge is equivalent to the waveform predicted in orbit.

○ Assume the sum of each coverglass capacitance of a single solar array panel as the capacitance contributing to temporary discharge.

○ Assume a plasma growth rate of 10 km/s for inverted potential gradient in stationary orbit and 100 km/s for inverted potential gradient in low orbit, and 100 km/s for normal potential gradient.

(6) When test items are in the category of the item (3) of Item 6.4.1.2, care should be given to the following items on test circuits:

(a) It is desirable to generate a primary discharge in the state in which an electric current is flowing through solar cells. When the solar cells have no bypass diode function and at the same time they are not irradiated with light, the direction of the current flowing through the solar cells is the opposite to that when electricity is generated. When the solar cells have bypass diode function, current can be flown in any direction, provided that flowing a large current is permitted. If not permitted, flow the current through the bypass diode.

(b) When the blocking diode of the SAS is not used to separate each string, the current flowing through the solar cells before the occurrence of primary discharge corresponds to the amount of the current flowing through a single string. However, after the occurrence of primary discharge, the current can increase to the amount larger than the current flowing through a single string. In testing the whole section consisting of multiple strings in which two or more SAS units are connected in parallel, flowing the current equivalent to two strings or more through the solar cells may cause the destruction of the PN junction. If flowing a large amount of current is not permitted, keep the current flowing through the solar cell to the level equivalent to one string in advance, and then attempt to flow the current equivalent to two or more strings immediately after the occurrence of a primary discharge.

Note: Flowing an excessive amount of current in the travelling direction from the positive pole to the negative pole may cause the destruction of the PN junction because the current concentrates on one location to cause local heating.

(c) When the current from other strings is additionally supplied by the SAS as previously described, it is desirable to reduce the inductance associated with the cables from the SAS so that the additional current can be promptly supplied.

(c) Minimize the length of cable s in the vacuum chamber and in the external power-supply to prevent electric current from being delayed or deformed by undesirable inductance. Use stranded wires or coaxial cables if possible (refer to Fig. 6.4-1).

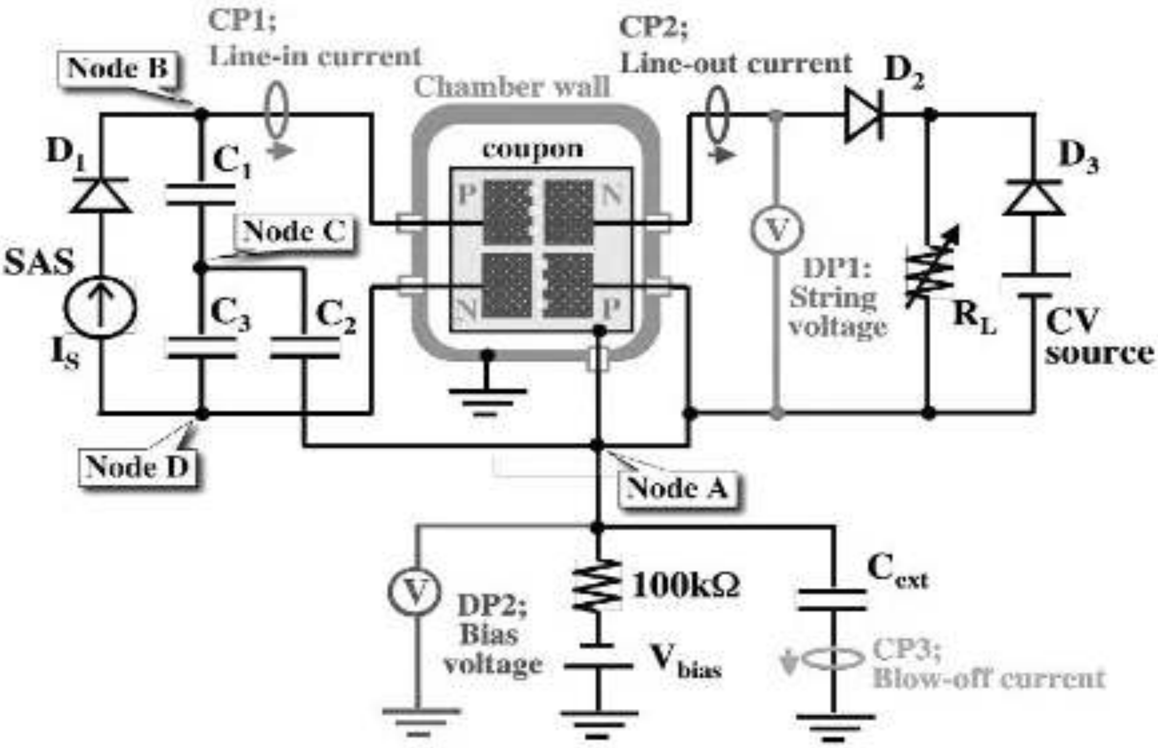


Fig. 6.4-1 Schematic diagram of the test circuit (for inverted potential gradient in plasma)

(d) To consider the capacitance C_j of the PN junctions of the solar cells contained between the positive and negative terminals, as well as the capacitance C_k between the facesheet and the adhesive used between the solar cells and the substrate, it is desirable, as shown in Fig. 6.4-1, to introduce the three capacitances C_1, C_2, and C_3, which are connected in π- or T-configuration to form an analogous circuit of the above capacitances. The three capacitances C_1, C_2, and C_3 are given by the following formulae based on the distributed constant circuit in which a single solar cell is taken as a single calculation unit:

$$C_1 = C_3 = N_p \sqrt{C_j C_k} \frac{1}{\tanh\left(\frac{N_s}{2}\sqrt{\frac{C_k}{C_j}}\right)} \tag{6.4-1}$$

$$C_2 = N_p \sqrt{C_j C_k} \sinh\left(N_s \sqrt{\frac{C_k}{C_j}}\right) \tag{6.4-2}$$

where, N_p stands for the number of parallel circuits of the solar panel, and N_s for the number of solar cells connoted in series in a single circuit of the solar panel.

(f) As shown in Fig. 6.4-1, it is desirable to introduce the constant voltage source V_2 through the diodes D_2 and D_3 so that the maximum supply current of the solar panel circuit can promptly flow through between the rows. When the arc resistance between the rows is reduced and thereby the voltage between the rows is reduced to a value smaller than V_2, D_2 prevents the current from I_1 from flowing into R_L, causing the total current I_1 to flow between the HOT row (the upper-side row of the solar cells in Fig. 6.4-1) and the RTN row (the lower-side row of the solar cells in Fig. 6.4-1). The voltage V_2 simulates the bus voltage in the power-supply system of a spacecraft.

6.4.1.7 Test-result acceptance criteria

(1) When test items are in the category of item (2) of Item 6.4.1.2 (degradation of electrical performance of solar cells due to primary discharge), it should be demonstrated that the amount of reduction in the electrical output from the solar panel, predicted at the end of the operation period, is within an acceptable range.

(2) When test items are in the category of item (3) of Item 6.4.1.2 (secondary arc), it should be demonstrated that a single secondary arc or an accumulated secondary arcs does not lead to the insulation breakdown of the solar panel circuit. If the insulation breakdown due to secondary arc cannot be avoided, it should be demonstrated that electrical power can be supplied to spacecraft load from the circuit even after the occurrence of the breakdown.

6.4.1.8 Additional tests

When test items are in the category of item (3) of Item 6.4.1.2 (secondary arc) and after these test items have been cleared, it is desirable to increase the inter-row voltage or the string current to a value larger than the nominal value until a secondary arc occurs to find the secondary arc generation conditions. These conditions will help enable the understanding of design margins and examine secondary-arc generation conditions at unexpected places, leading to the improvement of insulation measures. This test should be conducted after the completion of all other tests because it is likely to destroy test-coupons.

6.4.2 Solar panel test (positive charging)

6.4.2.1 Overview

There are three sets of requirements that apply to the solar panel: (1) surface charging test requirements explained in Item 6.2.1, (2) negative charging test requirements explained in Item 6.4.1, and the following test requirements related to the power leakage to the surrounding plasma when the potential of the solar panel is higher than the plasma.

6.4.2.2 Test items

Investigation tests should be carried out to find the possibility of the leakage of the power generated by the solar panel into the surrounding plasma when the region near the positive terminal of the solar panel excessively collects electrons from the plasma.

6.4.2.3 Cases that require testing

The test items in the category of Items 6.4.2.2 mainly apply to LEO satellites or GEO satellites with electrical propulsion systems and are necessary when their solar panels generate a voltage of more than 50 V. However, the testing of these items can be skipped when it is clarified by numerical analysis that the potential of the positive terminal of the solar panel relative to the potential of the surrounding plasma is +50 V or less.

Note: A potential of more than +50 V should be adopted as a guideline to determine whether or not the testing should be conducted, on the basis of the experience of failures in spacecraft power-supply systems frequently occurred concurrently with the time when the voltage generated by solar cells increased from 50 V to 100 V, as well as the ground that the generation of glow discharge was confirmed at a lowest voltage of 90 V under an extremely back pressure of $10^{19}m^{-3}$ in a past ground test (refer to Fig. I.5-1).

6.4.2.4 Test conditions

Before testing, conduct numerical analysis to estimate the following items.

(1) With respect to LEO satellites, the electrical potential of the structure of a spacecraft and the positive terminal of the solar panel under power generation operation to the potential of the

surrounding plasma. A special care should be given to the analysis when plasma contactors are used for potential control because the positive terminal is likely to have a potential equivalent to the generation voltage.

(2) With respect to a spacecraft with electrical propulsion systems, the electrical potential difference between discharged plasma during the operation of the electrical propulsion system and the positive terminal of the solar panel

(3) Plasma density in the vicinity of the solar panel. Care should be given to the increase in the density due to the discharged plasma for a spacecraft with the electrical propulsion system.

(4) The maximum density and the type of the neutral gas predicted during operation in the vicinity of the solar panel. Consideration should be given to the increase in density due to the injection by control thrusters.

6.4.2.5 Explanatory comments on testing

(1) Tests should be conducted in a vacuum state in a test room with the following characteristics

(a) Low-energy plasma sources such as an electron impact-type direct current plasma source and an electron cyclotron resonance-type plasma source. Compared to the voltage applied to test-coupons, electron temperatures should be quite low— normally several eV or lower. The plasma should have a density close to the value estimated in Item 0.

(b) Suction capacity that enables a vacuum of 0.01 Pa or better even in a state in which low-energy plasma is in operation

(c) Feed-through cables to connect to power supplies and test-coupons (maximum: 1 kV)

(d) Devices to simultaneously monitor and record test-coupon potential and collected current

(e) Visual inspection, photo shooting, and video recording of test samples during a test

(f) Direct-current power-supply (with a capacity of about 500 V and 3 A)

(g) Langmuir probes for background plasma measurement

(2) The following shows the main characteristics of the test-coupon:

(a) The test-coupon should serve as a mock flight model. A test-coupon can be smaller than the actual model because it is not practical to test the actual solar panel under sunlight irradiation and in a vacuum environment.

(b) Care should be given to the number of the solar cells in a single test-coupon because if the number is too small, it leads to the collection of electric current larger than the actual current due to the focusing effect of electrons. To reproduce a solar cell surrounded by solar cells in all directions, it is desirable to use a combination of at least three cells connected in series time three cells connected in parallel.

(c) During a test, there is no need to irradiate the cells with light during a test.

(d) It is desirable to use a microscope to observe the details on the surface state of test-coupons before and after conducting a test.

(4) The following shows the main characteristics of the test circuit. Cables can be connected to the positive (P) pole and the negative (N) pole of a test-coupon, which then can be bundled and connected to the direct-current power-supply; the voltage of the power-supply can be determined as the single voltage. It is desirable to use the power-supply equipped with current limiting

functions against the occurrence of glow discharge, or equipped with protective limiting resistors (refer to Fig. 6.4-2).

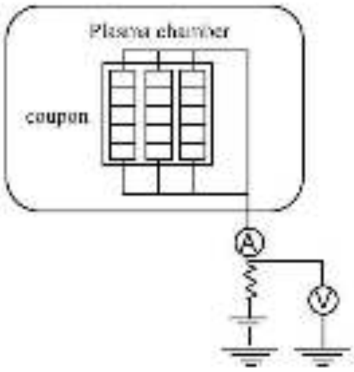


Fig. 6.4-2 Schematic diagram of the test circuit

6.4.2.6 Test-result acceptance criteria

It should be demonstrated that the amount of reduction in electrical output due to the power leakage into the plasma is within an acceptable range by estimating the amount of electronic current collected from the entire solar panel.

6.4.2.7 Additional tests

To understand design margins after test items have been cleared, it is desirable to increase the gas density in the chamber or the coupon potential to a value larger than its nominal value until a arc glow discharge occurs on the surface of the test-coupon in order to examine glow-discharge generation conditions.

6.5 High-voltage system

It is desirable that the high-voltage systems are demonstrated by conducting unit-based tests or system-based comprehensive tests under possible environments (in terms of degree of vacuum, temperature, and vibration).

6.6 Internal parts and materials

6.6.1 Verification

The following rules and regulations should be observed for the measures against a dangerous level of internal charging to function effectively:

(1) Inspect the structure and cable harnesses and verify that no metal components are grounded. However, not that when they are grounded through high-resistance bodied, they do not have to be grounded, provided that they satisfy the requirements described in the item (2) of Item 5.6.2.1.

(2) Resistance tests should be conducted for grounded metal components to verify that the grounding meets the requirements described in Item 5.6.2.1.

(3) During the calculation of internally charged electrical field, the volume resistivity of insulators or high-resistance materials should be the actual value at predicted operating temperature. At least, the resistivity at the maximum and the minimum temperatures should be used for the consideration. Note that when the resistance variation with varying temperature is on the same order as that of the values at room temperature, the values at room temperature can be used, on the condition that client approval is obtained. Furthermore, the resistivity should be measured by the method accepted by the client as appropriate in terms of internal charging.

(4) When the resistivity of a material is very low, there is no need to conduct a dielectric-considered detailed computer simulation of electrical field. this is when no high electrical field can be produced even if electrical field-induced conductivity or radiation-induced conductivity is not

considered because of the inherent low resistivity—lower than 1.0×10^{11} Ωm—of the materials (at the lowest operation temperature) used for the intended parts

(5) The systems containing the insulators or floating conductors sensitive to the effect of the internal field should be evaluated in terms of how much effect they are affected by the internal charging. Monte Carlo simulation programs such as GEANT4, EGS-4, or SRIM can be used for the evaluation of the infiltration and movement of charged particles—they cause internal charging—in an insulator.

6.7 Tether

Quantitative tests and verification are defined by each mission in response to risk evaluation. They should be evaluated by conducting unit-based tests or system-based comprehensive tests or both.

6.8 Electrical propulsion

6.8.1 Verification

6.8.1.1 Basic principle

(1) Validity should be confirmed mainly by conducting ground tests and by computer model analysis. Note that both approaches have limitations. Thus, experimental simulation and computer simulation should be used in a complementary manner so that the validity can be reliably confirmed.

(2) For the systems equipped with flight recording functions, further validity confirmation can be conducted by using in-flight performance monitoring.

6.8.1.2 Characteristics of ground tests

When electrical propulsion systems are tested on the ground, they are usually tested in vacuum chambers. They are larger than the size of thrusters so that the effect of the walls on beam characteristics can be minimized. In addition, the degree of vacuum in the chamber is constantly monitored. However, the accuracy of measurements is limited because of the physical restrictions on these vacuum chambers. For example, the density of background neutral particles or plasma in a vacuum chamber is normally increased to a larger value than that observed in orbit because plasma beams are neutralized when they collide with the walls of the vacuum chamber or target objects. Furthermore, sputtering contamination in the vacuum chamber is much more serious than that in a space environment.

The effects due to the limitations of the vacuum chamber should be determined by using evaluation models (e.g., by computer simulation models).

Detailed requirements related the ground test, such as the size of the vacuum chamber or diagnosis accuracy depend on thrusters. Furthermore, thrust range, plasma density, and ion energy greatly differ depending on each thruster.

6.8.1.3 Characteristics in creating computer models

One way to gain a deep insight into the behavior of spacecraft thrusters is create a computer simulation model of the electrical propulsion systems and the interaction model with the plasma around the spacecraft. Computer simulation generally involves simplification and generalization but is regarded as the most practical means within the range of limited resources.

(1) Computer simulation models should be created in consideration of the geometrical shape of a spacecraft and its operational environment.

(2) Creating a thruster plume model should entail the description plasma velocity distribution, electrical field around a spacecraft, and boundary state on the surface of the spacecraft

(3) The validity of computer simulation models should be confirmed by using the data obtained by ground tests.

6.8.1.4 In-flight monitoring

To ensure reliable design, ground tests, and computer simulation, measuring the performance of the electrical propulsion systems serves as an important means.

It is desirable to monitor the potential of spacecraft potential and its plasma environment in any mission that use the electrical propulsion systems.

Note: These requirements are determined because experiments in a vacuum chamber or computer simulation analysis do not provide a whole picture of the interaction process of the plasma in outer space.

6.8.1.5 Neutralization of a spacecraft

(1) When a spacecraft is neutralized by using neutralization devices, it should be proved by ground tests that the electrical propulsion system and the neutralization devices can operate at the same time, in the state in which exact current is used and the thruster/neutralization device assembly is separated from the ground section of a vacuum chamber, or after confirming that the current that flows backward from the vacuum chamber to the power-supply circuit of the electrical propulsion systems is negligible.

Note: In reality, no special additional testing seems to be necessary because this type of ground test is required as a part of the life test of electrical propulsion systems and neutralization devices.

(2) It should be shown that the maximum current of the neutralization devices is larger than the maximum beam current and the maximum current provided by the external current components (e.g., space plasma or photoelectron).

(3) Assume the worst-case environment for the operation of the electrical propulsion system when the current provided by the external current components (e.g., space plasma or photoelectron) is determined.

(4) In a particular case when external plasma current is used to achieve the neutralization, it should be proved that the current of low-energy particles resulting from the surrounding environment is larger than any combination of beam current and other external plasma current.

(5) When the electrical propulsion is accelerated, attempt to stop the neutralization current intentionally to verify, on the ground, the failure detection performance and the safety treatment performance of the neutralization devices. At this time, the soundness of the surface conductivity treatment around the thrusters should be evaluated at the same time.

6.8.1.6 Neutralization of beams

Neutralization of beams should be proved by conducting ground tests (including the measurement of beam potential and beam dispersion) in the state in which neutralization devices are operated and thruster/neutralization device assembly is separated from the ground section of a vacuum chamber.

6.8.1.7 Low-speed plasma interference around a spacecraft

It should be proved that the amount of current flowing toward the high-voltage exposure sections such as the high-voltage solar panel does not affect the operation of a spacecraft by evaluating the current on the basis of the plasma density obtained by computer simulation. It is desirable to add demonstration experiments by using miniature models [5] or full-scale models, if needed.

6.8.1.8 Contamination

(1) Ground tests should be conducted to determine the degree of contamination around thrusters.

(2) In a vacuum chamber, beams collide with the walls of the vacuum chamber and the inserted target objects, producing sputtering products with a large fluence because of the infinite length of the vacuum chamber. In this case, the contamination level cannot be sufficiently evaluated only by ordinary ground tests because of the excessive contamination due to these factors. If this contamination happens, conduct a ground test, and measure the degree of erosion on the surface of thrusters (including electrostatic grids when they have) by sputtering to calculate the amount or the rate of emitted contaminants assuming that the wear amount is dispersed into the surrounding environment. Furthermore, estimate the degree of contamination by calculation or computer simulation.

6.8.1.9 Plasma-jet injection

(1) The effect of plasma-jet injection or sputtering during normal operation should be determined by computer simulation analysis of ion orbits or by using other computer models.

(2) Evaluation should be conducted to determine the effect of ion orbits on spacecraft during transient operations such as at the start or end of acceleration or during on/off operation

6.8.1.10 Effect of neutral gases

(1) In the initial period of operation immediately after the launch of a spacecraft, the following items should be clarified:
(a)Processing plans for outgases from the spacecraft and the electrical propulsion systems
(b) Operating procedures and operating conditions for electrical propulsion systems

(2) Evaluation should be conducted in terms of the density of the plasma around a spacecraft and caused by the neutral gases emitted from a spacecraft and electrical propulsion systems.

Note: It seems impossible for experimental simulation to properly evaluate the density level of the plasma around a spacecraft and caused by the neutral gases emitted from a spacecraft and electrical propulsion systems.

7. Appendix

Appendix I Quantitative evaluation

I.1 Introduction

This chapter describes various basic items required to quantitatively evaluate the charging phenomena of a spacecraft. Refer to the reference [6] for details.

I.2 Plasma environment

Table I.2-1 shows charged particles to be considered with respect to LEO, PEO, and GEO.

Table I.2-1 Typical energy and current density by charged particles in each orbit.

Orbit	Particle	Typical energy (eV)	Typical current density (A/m^2)
LEO	Ionospheric electron	0.1-0.2	10^{-4}-0.1
	Ionospheric ion	0.1-0.2	10^{-5}-10^{-3}
PEO	Ionospheric electron	0.1-0.2	10^{-5}-0.1
	Ionospheric ion	0.1-0.2	10^{-6}-10^{-3}
	Precipitating electron (aurora zone)	100-10 k	10^{-8}-10^{-3}
GEO	Magnetospheric electrons	100-25 k	10^{-6}
	Magnetospheric ion	100-25 k	10^{-8}
	Radiation belt electron	100 k-10 M	10^{-10}
Interplanetary (near the earth)	Solarwind electron	10	10^{-7}-10^{-6}
	Solar wind ion	10	10^{-8}-10^{-6}

Care should be given to the fact that these values can vary by several orders of magnitude depending on unexpected variation of solar activities on the magnitude. Furthermore, models with these values and past measured data are available from the database on the Internet [7].

Figure I.2-1 and Figure I.2-2 show the probability distribution of the precipitating electron current in PEO and the magnetospheric electron current in GEO obtained from the statistical analysis of the actually measured satellite data [8] [9]. These tow figures show the probability of the electric current density higher than the value shown in the horizontal axis.

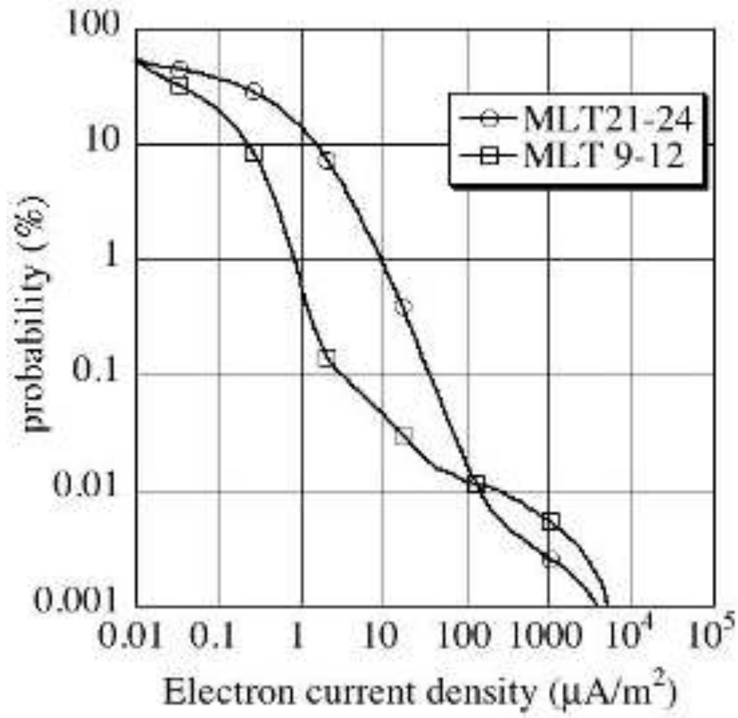


Fig. I.2-1 Distribution density of precipitating electron current density in the region of geomagnetic latitude from 60 º to 75º at an altitude of 840 km, which is based on the analysis of SSJ/4 data [8] of the DMSP satellite (energy range: 1 keV-30 keV).

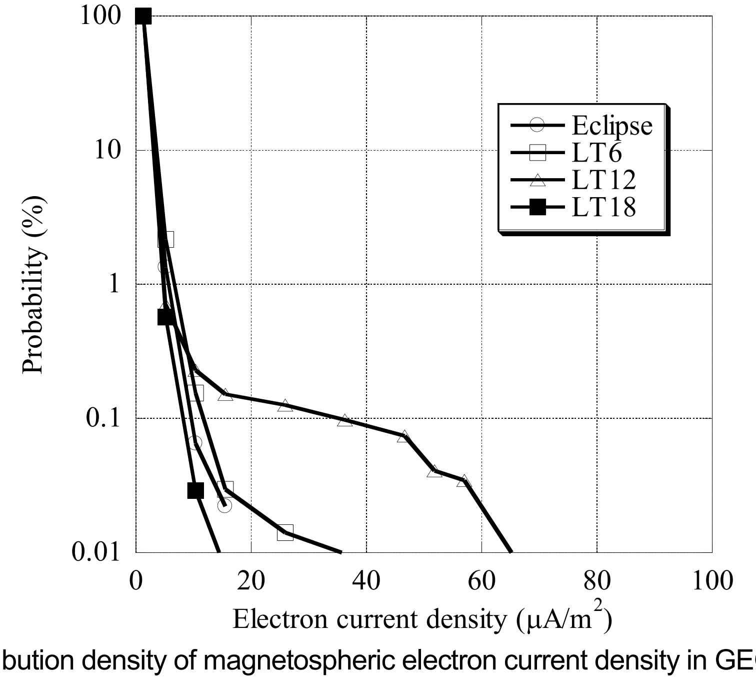


Fig. I.2-2 Distribution density of magnetospheric electron current density in GEO based on the analysis [9]of MPA data (energy range: 30 eV-45 keV).

I.3 Principle of spacecraft charging

I.3.1 Electrostatic sheath

I.3.1.1. Sheath boundary

The evaluation of spacecraft charging is based on the calculation each charged particle current that inflows into or outflows from a spacecraft, and it is the concept of sheath that becomes important when the motion of the substances around the charged particles is considered.

When a spacecraft is in plasma, the information on the existence of the spacecraft can only penetrate only a finite distance into the plasma because of the electrostatic screening effect of the plasma. In most cases, spacecraft potential differs from plasma potential, and the electrical field around a spacecraft is perpendicular to its surface. The negatively and positively charged particles in the surrounding plasma are attracted or repelled by the electrical field. The electrical field produced by the spacecraft is confined within an infinite distance by the electrostatic screen, and a “sheath” refers to an infinite region of the electrical field. In a sheath, quasi-neutrality—a fundamental characteristic of plasma—is not preserved: positive electric charge becomes predominant when the spacecraft is negatively charged, whereas negative electric charge becomes predominant when the spacecraft is positively charged.

The upper limit of the current that a spacecraft attracts from external plasma is determined by the current particles that inflows into and outflows from the sheath boundary. When the distribution function of the plasma is given, the density of the current inflowing only in a single direction from beyond a boundary is given by the following formula:

$$j_{so} = q_s \int_0^{\infty} v f_s(v) dv \qquad \text{(I.3-1)}$$

where, q_s stands for electric charge of s-species [C], v for the velocity perpendicular to the sheath boundary [m/s], and $f_s(v)$ for one-dimensional distribution function of s-species [/m^3]. Assuming that the external plasma is distributed in accordance with the Maxwell-Boltzmann distribution, the density of the current that crosses the sheath boundary when a spacecraft is negatively charged is shown as follows:

$$j_{i0} = en\sqrt{\frac{kT_e}{m_i}}\exp\left(-\frac{1}{2}\right) \qquad \text{(I.3-2)}$$

Furthermore, the density of the current that crosses the sheath boundary when the spacecraft is positively charged is shown as follows:

$$j_{eo} = en\frac{1}{\sqrt{2\pi}}\sqrt{\frac{\kappa T_e}{m_e}} \qquad \text{(I.3-3)}$$

where, n stands for plasma density [/m^3], k for Boltzmann constant [J/K], T_e for electron temperature [K], m_i and m_e for the mass of an ion and an electron [kg], respectively. There is a buffer region called a “pre-sheath” between the plasma and the sheath, which gradually accelerate or decelerate the electric charge while keeping its quasi-neutrality. Thus, the plasma density on the sheath boundary is somewhat lower than that of the outer plasma density (by about 50%). The reason that the electron current density differs from that of ion current density is because the ions are accelerated by the pre-sheath until they reach the Bohm velocity on the sheath boundary.

I.3.1.2 Debye length

The Debye length, which is given by the following formula, is an important parameter to estimate the thickness of the sheath that covers around a spacecraft.

$$\lambda_d = \sqrt{\frac{\varepsilon_o \kappa T_e}{n_o e^2}} \qquad \text{(I.3-4)}$$

The Debye length is a measure related to the length of a screen used for the electrostatic plasma screening. The following sections show two theories of current collection. To select one the theories, compare the Debye length with a characteristic length of a spacecraft. If the former is much shorter than the latter, use the thin-sheath theory. If the condition is the opposite, use the thick-sheath theory.

Table I.3.1 shows typical Debye length values in various plasma regions.

Table I.3-1 Parameters in various regions in outer space

Plasma region	Density $[m^{-3}]$	Temperature [eV]	Debye length [m]
Interstellar	10^6	10^{-1}	1
Solar corona	10^{13}	$1\text{-}10^2$	$10^{-2}\text{-}10^{-3}$
Solar wind	$10^3\text{-}10^9$	$1\text{-}10^2$	$1\text{-}10^2$
Magnetosphere	$10^6\text{-}10^{10}$	$10\text{-}10^3$	$1\text{-}10^2$
Ionosphere	$10^8\text{-}10^{12}$	10^{-1}	$10^{-1}\text{-}10^{-3}$

Note that electron volt (eV) is an energy unit used to express the temperature of space plasma (1 eV = 11,605 K = 1.6 x 10^{-19} J. Furthermore, care should be given to the fact that electrons or secondary particles emitted from a spacecraft can change the properties of the plasma near the spacecraft. Note that the plasma environment in these regions is so dynamic that the properties can fall out of the range shown in Table I.3-1.

I.3.1.3 Current collection within a thin-sheath

When the thickness of a sheath is much smaller than the curvature radius of the surface of a spacecraft, the sheath can be considered locally to be flat. Thus, assuming that the electrical field in the sheath is one-dimensional only and perpendicular to the surface of the spacecraft, the motion of the charged particles in the electrical field can be approximated by the simple motion in one-dimensional direction. What is needed for the analysis are only three equations: a one-dimensional equation of continuity, an energy-related equation, and a Poisson's equation. The following Child-Langmuir space-charge limited current gives the relationship among sheath thickness d, spacecraft electrical potential V, and the density of the current j that inflows from the sheath boundary.

$$j = \frac{4}{9}\varepsilon_o \sqrt{\frac{2e}{m}} \frac{V^{3/2}}{d^2} \qquad \text{(I.3-5)}$$

When the density of the current inflowing from the sheath boundary is given by Formula (I.3-5), the sheath thickness can be approximated by the following formula as a function of the Debye length.

$$d \approx \lambda_d \left(\frac{|qV|}{kT_e} \right)^{3/4} \qquad \text{(I.3-6)}$$

I.3.1.4 Current collection within a thick-sheath

When the thickness of a sheath is much larger than the curvature radius of the surface of a spacecraft, it depends on the shape of the spacecraft whether charged particles actually can penetrate the sheath and reach the surface of the spacecraft surface. In this case, the motion in the sheath cannot be approximated by simply applying one-dimensional equations. However, if the shape of the spacecraft can be dealt with as a simple shape such as a cylinder or a sphere, the conditions for the charged particles inflowing from the sheath boundary to reach the spacecraft surface can be derived by using the preservation of energy and angular momentum, which is called the orbital restriction theory. When the radius of a spacecraft can be considered as a sphere, the current of the charged particles attracted to the spacecraft is given by the following formula:

$$I = 4\pi R^2 j_{so}\left(1 - \frac{q_s V}{\kappa T_e}\right) \qquad \text{(I.3-7)}$$

where, j_{so} stands for the density of the current that crosses the sheath boundary represented by Formula (I.3-1).

I.3.1.5 General cases

When the thickness of a sheath is equivalent to the curvature radius of the surface of a spacecraft, the current attracted to a spacecraft cannot be expresses as a simple formula as shown in Formula (I.3-2) or Formula (I.3-3). Instead, the current should be calculated in view of both the three-dimensional shape of the spacecraft and, at the same time, the motion of the charged particles in the sheath. Since this type of calculation is difficult to be performed by hand, it is desirable to use a Particle-in-Cell-based or Particle Tracking-based three-dimensional computer simulation program for particle tracking. Available computer simulation programs include MUSCAT [[10]], NASCAP [[11]], [[12]], and SPIS [[13]].

I.3.2 Basic formula

There is a potential difference between the structure of a spacecraft and the surrounding plasma because the spacecraft is literally “floating” in the plasma. The potential of a spacecraft (structure) in a steady state is determined so that no current is collected by the exposed metal surface from the surrounding plasma. Figure I.3-1 shows an example of such current that flows on the surface of the spacecraft. Each current component is expressed as a function of the spacecraft electrical potential ϕ_{sc}. In a steady state, the current satisfies the following formula:

$$I_e(\phi_{sc}) - \{I_i(\phi_{sc}) + I_{se}(\phi_{sc}) + I_{si}(\phi_{sc}) + I_{be}(\phi_{sc}) + I_{ph}(\phi_{sc}) + I_a(\phi_{sc}) + I_s(\phi_{sc})\} = 0 \qquad \text{(I.3-8)}$$

where, I_e stands for external electron current, I_i for external ion current, I_{se} for electron-impact secondary electron current, I_{si} for ion-impact secondary electron current, I_{be} for backscattered electron current, I_{ph} for photoelectron current, I_a for dynamic emission current, and I_s for leakage current from the surface of an insulator. The first term of the formula refers to the inflow of negative electric charge, whereas the second and the following terms refer to the inflow of positive electric charge or the emission of negative electric charge. The external electron current and the external ion current can be evaluated according to the approach described in Item I.3-1.

Note: In addition, the surface current effect [[14]] resulting from the electron-impact secondary electrons moving along the electrical filed along the surface material. However this effect is not described in this document because of the difficulty of the quantification.

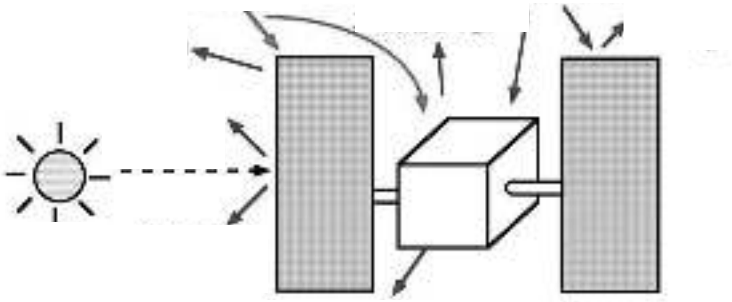


Fig. I.3-1 Current inflowing in and outflowing from a spacecraft

I.3.3 Photoelectron

Photoelectrons are emitted when photons contained in sunlight and having an energy level larger than the work function of an object fall on the surface of the object. The emitted photoelectrons can be approximated by the Maxwell distribution with temperature T_{ph}, which is normally about 2 eV. The photoelectron current is proportional to the irradiation intensity of sunlight, and the proportionality constant depends on the surface material of the object. Table I.3-2 shows photoelectric current density of typical materials in AM0.

Table I.3-2 Photo-electric current density [6] of typical materials in AM0

Material	Photoelectric current density (μA/m^2)
Aluminium Oxide	42
Indium Oxide	30
Gold	29
Stainless steel	20
Graphite	4

I.3.4 Secondary electron and backscattered electron

Secondary electrons are emitted from the interior of an object when high-energy electrons coming out of a sheath collide with the surface of the object. In addition, some incident electrons themselves are scattered outside the object. These emitted or scattered electrons are comprehensively called "backscattered electrons." The secondary electrons can also be approximated by the Maxwell distribution with temperature T_{se}, which s normally about 2 eV. The number of the secondary electrons emitted by a single incident electron is called the secondary electron emission coefficient: There are various approximation formulae including the following typical formula [6]:

$$\delta_{ee}(E_i,\theta) = \delta_{e\max}\frac{E}{E_{\max}}\exp\left(2-2\sqrt{\frac{E}{E_{\max}}}\right)\exp\left[2(1-\cos\theta)\right]$$

(I. 3-9)

where, E stands for the incidence energy of a primary electron, θ for the incident angle (the angle from the vertical direction), $\delta_{e\max}$ for the maximum secondary electron emission coefficient, E_{max} for the maximum incident energy. Figure I.3-2 shows the calculation results of Formula (I.3-9) on the assumption that $\delta_{e\max}$ = 3 and E_{max} = 300 eV. Furthermore Table I.3-3 shows $\delta_{e\max}$ and

E_{max} of typical spacecraft surface materials.

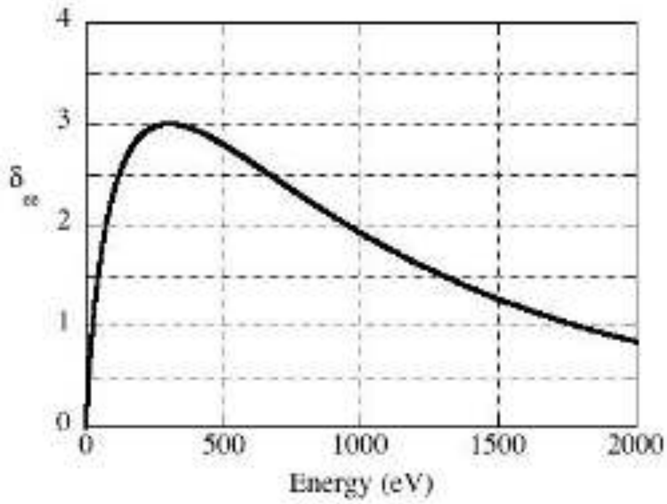


Fig. I.3-2 Dependency of the secondary electron emission coefficient on incident energy

Secondary electron emission is also caused by incident ions. Compared with electron-impact secondary electrons, however, the secondary electron emission can be neglected because the incident energy presents the maximum value at a few hundred keV and is not larger than 1.0 in the incident energy range less than 10 keV.

The amount of backscattered electrons depends on incident energy and incident angle. Emitted energy has a wide dispersion area comparable to that of incident energy.

Table I.3-3 The maximum secondary electron emission coefficients and the maximum energy of typical materials [6]

Material	$\delta_{e\max}$	E_{max}
Aluminum	0.97	300
Aluminum oxide	1.5-1.9	350-1300
Magnesium oxide	4.0	400
Silicon dioxide	2.4	400
Teflon	3	300
Kapton	2.1	150
Magnesium	0.92	250

I.3.5 Input parameter examples for charging analysis software

Input parameter(Secondary Electron Emission, Photon , Volume Resistivity,Dielectric Constant) examples for charging analysis software are shown in table I.3-4.

Table I.3-4 Input parameter examples for charging analysis software

Metal												
Measured(Domestic)						Reference(Foreign)						
Material	Secondary Electron		Photoelectron	Dielectric Constant	Volume Resistivity	Material	Secondary Electron		Photoelectron	Dielectric Constant	Volume Resistivity	Surface Resistivity
	Emax, keV	δ max	$A \cdot m^{-2}$				Emax, keV	δ max	$A \cdot m^{-2}$			
Aluminum Oxide	＜0.60	1.93	Not Taken	N/A	N/A	Aluminum	0.3	0.97	4.00E-05	N/A	N/A	N/A
Gold	＜0.60	1.93	72	N/A	N/A	Gold	0.8	0.88	2.90E-05	N/A	N/A	N/A
		Not Taken				INDOX	0.8	1.4	3.20E-05	N/A	N/A	N/A
		Not Taken				Magnesium	0.25	0.92	4.00E-05	N/A	N/A	N/A
Silver	0.7	1.67	Not Taken	N/A	N/A	Silver	0.8	1	2.90E-05	N/A	N/A	N/A

Insulator												
Measured(Domestic)						Reference(Foreign)						
Material	Secondary Electron		Photoelectron	Dielectric Constant	Volume Resistivity	Material	Secondary Electron		Photoelectron	Dielectric Constant	Volume Resistivity	Surface Resistivity
	Emax, keV	δ max	$A \cdot m^{-2}$				Emax, keV	δ max	$A \cdot m^{-2}$			
Aluminized Kapton (146446)	＜0.60	1.38	Not Taken	Not Taken	8 X 10E14	Kapton	0.15	2.1	2.00E-05	3.5	1.00E-16	1.00E+16
	Not Taken					SiO2	0.4	2.4	2.00E-05	4	1.00E-14	1.00E+19
CMG100AR	1	7.66	10	6.4	5 X 10E13	Cover Glass	0.41	2.05	2.00E-05	3.8	1.00E-17	1.00E+19
	Not Taken					Teflon	0.3	3	2.00E-05	2	1.00E-16	1.00E+16

Paint												
Measured(Domestic)						Reference(Foreign)						
Material	Secondary Electron		Photoelectron	Dielectric Constant	Volume Resistivity	Material	Secondary Electron		Photoelectron	Dielectric Constant	Volume Resistivity	Surface Resistivity
	Emax, keV	δ max	$A \cdot m^{-2}$				Emax, keV	δ max	$A \cdot m^{-2}$			
D13N	0.8	1.36				AQUADG	0.3	1	2.10E-05	N/A	N/A	N/A
	□					Non conducting paint	0.15	2.1	2.00E-05	3.5	5.90E-14	1.00E+13
S13GP	＜0.60	2.03	7	2.3	3 X 10E11	Conducting paint	0.15	2.1	2.00E-05	3.5	N/A	N/A

Note:N/A means proper value to be enterd for charging analysis software.(For example: '-1' for NASCAP)

I.3.6 Active emission current

Active emission current is caused by the charged particle beams or plasma emitted from the main body of a spacecraft. They are emitted for electrical propulsion, scientific experiments, or spacecraft electrical potential control.

When ions are discharged from the electrical propulsion system of a spacecraft, the amount of the ion beam current can be approximated by the following formula:

$$I_{beam} \approx \frac{eF}{m_{ion} g I_{sp}} \qquad \text{(I.3-10)}$$

where, $e = 1.6 \times 10^{-19}$ (C), g = 9.8 (m^2/s), F stands for propulsion force (N), I_{sp} for specific propulsion force (s), and m_{ion} for mass of an ion (kg). In ion engines or hall thrusters, the amount of the ion beam current often exceeds 100 mA, causing the beam current to be much greater than the other current values described in Formula (I.3-8) when these propulsion systems are used in GEO or interplanetary pathways. Thus, electrical propulsion systems are forced to simultaneously discharge the electron current comparable to the ion beam current by using neutralization devices so as to prevent the reduction of the spacecraft electrical potential to a negative value equivalent to the acceleration voltage.

Some neutralization devices for electrical propulsion are used as an active charging controller (plasma contactor) as used in the ISS because they are capable of discharging an electron current of more than 100 mA. A plasma contactor is a device that produces a high-density plasma in a discharge chamber with an electrical potential equivalent to the potential of a spacecraft and discharges the plasma outward. When the spacecraft electrical potential becomes negative to a large degree compared to that of the surrounding plasma, the plasma contactor provides electrons to the surrounding plasma so that the electrical potential can be automatically controlled. When the spacecraft electrical potential becomes positive, the contactor serves to collect electrons if the surrounding plasma has a sufficient density or otherwise discharges ions and begins to control the electrical potential. Furthermore, the contactor can contribute to mitigate the local charging on the surface of a spacecraft because of its bipolar diffusion effect, which makes electrons or ions to spread around the spacecraft, thus electrically grounding the insulators on the surface of the spacecraft to the grounding positions around the insulators even if a potential difference is generated in spacecraft surface insulators.

The electrical potential of a spacecraft is almost equivalent to that of the surrounding plasma as long as the electrical propulsion systems are operated in GEO or in an interplanetary pathway and the neutralization devices are in normal operation with a sufficient amount of current capacity. Possible major problems include the contamination due to backscattered low-speed ions produced in ion beams, power leakage and reduced neutralization functions due to interference between high-voltage solar panels and low-speed electrons. These problems are detailed in Item I.5.

I.3.7 Leakage current

The leakage current from an insulator is the current of the electric charge accumulated on the surface or the interior of the insulator and flowing through a resistance in the surface or bulk direction. The leakage current is detailed in Item I.4.

I.4 Surface charging

I.4.1 Basic formula

The surface electrical potential of the insulators placed on the surface of a spacecraft ϕ_d is determined so that the inflowing and outflowing electric charge at each point on the surface of the insulators are cancelled out with each other and often differs from the electrical potential of a spacecraft. If the difference from the spacecraft electrical potential exceeds a certain value, a discharge is generated. In addition, the surface charging of the insulators can lead to contamination of optical sensors when they attract charged particles around the spacecraft or can affect measured values of particle, electrical field, or magnetic field sensors. By substituting the current in Formula (I.3-8) with current density, the surface electrical potential ϕ_d at each point can be calculated by the following formula:

$$j_e(\phi_d) - \{j_i(\phi_d) + j_{se}(\phi_d) + j_{si}(\phi_d) + j_{be}(\phi_d) + j_{ph}(\phi_d) + j_a(\phi_d) + j_s(\phi_d)\} = 0 \qquad \text{(I.4-1)}$$

I.4.2 Local charging

The electrical potential of the structure of a spacecraft ϕ_{sc} varies according to the following formula:

$$\frac{d\phi_{sc}}{dt} = \frac{1}{C_{sat}}\left(-I_e(\phi_{sc}) + I_i(\phi_{sc}) + I_{se}(\phi_{sc}) + I_{si}(\phi_{sc}) + I_{be}(\phi_{sc}) + I_{ph}(\phi_{sc}) + I_a(\phi_{sc}) + I_s(\phi_{sc})\right) \qquad \text{(I.4-2)}$$

where, C_{sat} stands for the capacitance between the spacecraft structure and the reference point of potential (surrounding plasma) at an infinite distance. The capacitance can be approximated by that of a sphere with equivalent radius R as follows:

$$C_{sat} = 4\pi\varepsilon_o R \qquad \text{(I.4-3)}$$

Assuming J as the total current density inflowing into and outflowing out of the spacecraft and by substituting the current in the right side of Formula (I.4-2) with $4\pi R^2 J$, the charging time for the spacecraft structure to reach the electrical potential ϕ_{sc}^* can be approximated by the following formula:

$$\tau_{sat} = \frac{C_{sat}\phi_{sc}^*}{4\pi R^2 J} = \frac{\varepsilon_o \phi_{sc}^*}{RJ} \qquad \text{(I.4-4)}$$

Considering the fact that J = 10 μA/m^2 as a typical value in GEO, it takes only about 10^{-3} second for the structure to be charged to the order of kV. In LEO, the spacecraft structure will be charged in shorter time because the current density J is greater than the potential value.

On the other hand, the capacitance of an insulator of dielectric constant ε_d , area *A*, and thickness *t* can be calculated by the following formula:

$$C_d = \frac{\varepsilon_d A}{t} \qquad \text{(I.4-5)}$$

Thus, the time constant required for generating a potential difference Δ*V* across the insulator is given by the following formula:

$$\tau_d = \frac{C_d \Delta V}{AJ} = \frac{\varepsilon_d \Delta V}{tJ} \qquad \text{(I.4-6)}$$

By using the above formula and assuming that *J* = 10 μA/m^2, the time τ_d required for an insulator of thickness 1 mm to be charged to a potential of 1 kV is calculated to be about 1 second.

A high-level charging environment generally continues for about several hours in GEO and several minutes in PEO. When a spacecraft encounters such environment, the spacecraft structure is rapidly charged to a negative potential. Within such time-scale, almost no electric charge is accumulated in the insulators that cover the surface of the spacecraft. Thus, the surface electrical potential of the insulators is almost the same as that of the spacecraft structure. The charging of the insulators begins only after the electrical potential of the spacecraft structure has reached a steady state, producing a potential difference between the insulator potential (Dielectric Potential) and the spacecraft electrical potential (Spacecraft Potential). This potential state in which there is a potential difference between them is called “local charging.” Furthermore, the potential difference between the insulator potential and the spacecraft electrical potential, which is shown in the following formula, is called Differential Voltage.

$$\Delta V = \phi_d - \phi_{sc} \qquad \text{(I.4-7)}$$

The state in which the differential voltage is positive, namely, $\phi_d > \phi_{sc}$ is called the normal potential gradient, whereas the state in which the differential voltage is negative, namely, $\phi_d < \phi_{sc}$, is called the normal potential gradient. During a local charging process, the time-variation of the differential voltage can be represented by the following formula:

$$\frac{d\Delta V}{dt} = \frac{1}{C_d}\left(-j_e + j_i + j_{se} + j_{si} + j_{be} + j_{ph} + j_a + j_s\right)$$

(I.4-8)

Fig I.4-1 shows a schematic diagram of the formation process for the inverted potential gradient predicted when a substorm occurs in GEO.

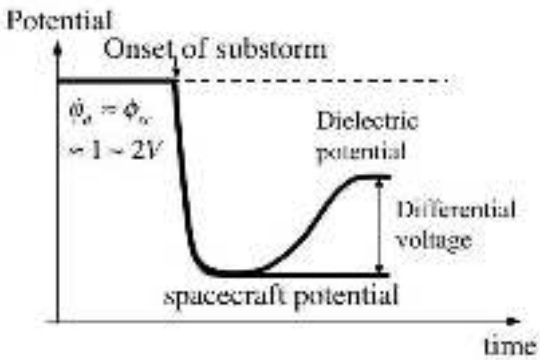


Fig. I.4-1 A schematic diagram during local charging progression when a substorm occurs in GEO. In the figure, the coverglass is assumed as an insulator.

I.4.3 Local charging in GEO

If the current with a high-energy electron density of tens of keV drastically increases because of the occurrence of a substorm around a spacecraft, the external electron current may become larger than the photoelectron current even in the daytime as shown in Fig. I.4-1, causing the spacecraft electrical potential to fall down to a large negative value depending on the attitude of the spacecraft. In this case, the governing electric current that determines the spacecraft electrical potential is classified into four types: external electron current density j_e, electron–impact secondary electron current density j_{es}, photoelectron current density j_{ph}, and material conducting current j_s.

$$\frac{d\Delta V}{dt} = \frac{1}{C_d}\left(-j_e + j_{es} + j_{ph} + j_s\right) \qquad \text{(I.4-9)}$$

Both the external electron current density j_e and the electron-impact secondary electron current density j_{es} are the functions of the insulator surface potential ϕ_d. The relationship between j_e and j_{es} can be described by the following formula:

$$j_{es} = \delta_{ee}\left(E_e, \theta\right) j_e \qquad \text{(I.4-10)}$$

The material conducting current j_s can also be described in the following formula:

$$j_s = -\frac{\Delta V}{\rho t} + \frac{1}{R_s}\left(\frac{\partial^2 \Delta V}{\partial x^2} + \frac{\partial^2 \Delta V}{\partial y^2}\right) \qquad \text{(I.4-11)}$$

where, ρ stands for the volume resistivity (expressed in Ωm) of an insulator of thickness t, and R_s for surface resistivity (expressed in Ω/□) of a rectangular insulator.

When the bulk conductivity in the thickness direction through a volume resistance is dominant, that is, when the second term of the right side of Formula (I.4-11) is zero (refer to Fig I.4-2 (A)), the differential voltage ΔV^* in the quasi-steady state can be given by the following formula because the bulk conducting current j_s is proportional to the differential voltage:

$$\Delta V^* = \rho t J \qquad \text{(I.4-12)}$$

where, J stands for the density of the current exchanged with outside and the sum of all the items except for the bulk conducting current. Assuming that A (= wd) is the surface area of an insulator, R can be expressed by $R = \rho t/A$. Thus, Formula (I.4-12) can be given as follows:

$$\Delta V^* = RAJ \qquad \text{(I.4-13)}$$

When the distance d between the insulator and a grounding point is short, or when the surface resistance is much lower than the volume resistance, the component of the bulk conducting current in Formula (I.4-11) becomes dominant in the surface direction (refer to Fig. I.4-2 (B)), and j_s can be expressed by the following formula:

$$j_s \approx \frac{\Delta V}{R_s d^2} \qquad \text{(I.4-14)}$$

In the steady state, the differential voltage can be expressed as follows:

$$\Delta V^* = R_s d^2 J \qquad \text{(I.4-15)}$$

Note: Pay attention to the fact that the calculation formula for the differential voltage varies depending on the shape of the collection electrode. When the collection electrode is installed along a side of a high-resistance material as shown in Fig. I.4-2 (B) and by considering the fact that J inflows into the surface uniformly, Formula (I.4-15) can be formulated into $\Delta V^* = R_s dw/2$. Assuming that I is the total current inflowing into the surface, the formula can be rewritten as $\Delta V^* = R_s I/2$.

Note: When the collective electrode is installed around the circular high-resistance material (outer diameter *Do* and thickness *t*) as shown in Fig. I.4-2 (C) and by considering the fact that *J* inflows into the surface uniformly, Formula (I.4-15) can be formulated into $\Delta V^* = R_s J(Do/2)^2$. Assuming that I (= $J\pi(Do/2)^2$) is the total current inflowing into the surface, the formula can be rewritten as $\Delta V^* = R_s I/\pi$.

Note: When the collective electrode is installed in the center of the circular high-resistance material (outer diameter *Do*, inner diameter *Do*, and thickness *t*) as shown in Fig. I.4-2 (D) and by considering the fact that *J* inflows into the surface uniformly, Formula (I.4-15) can be formulated into the following formula:

$$\Delta V^* = \frac{R_s J}{2}\left(\frac{D_o}{2}\right)^2\left[ln\left(\frac{D_o}{D_i}\right)-\frac{1}{2}\left\{1-\left(\frac{D_i}{D_o}\right)^2\right\}\right]$$

The formula similar to Formula (I.4-13) and Formula (I.4-15) is also true with a floating conductor with area *A* exposed to outer space and connected to a ground point of a spacecraft through resistance *R*. The electrical potential of the floating metal conductor to the ground point in the quasi-steady state is given by the following formula:

$$V^* = ARJ \tag{I.4-16}$$

where, *J* refers to the density of the current exchanged with outside through *A*.

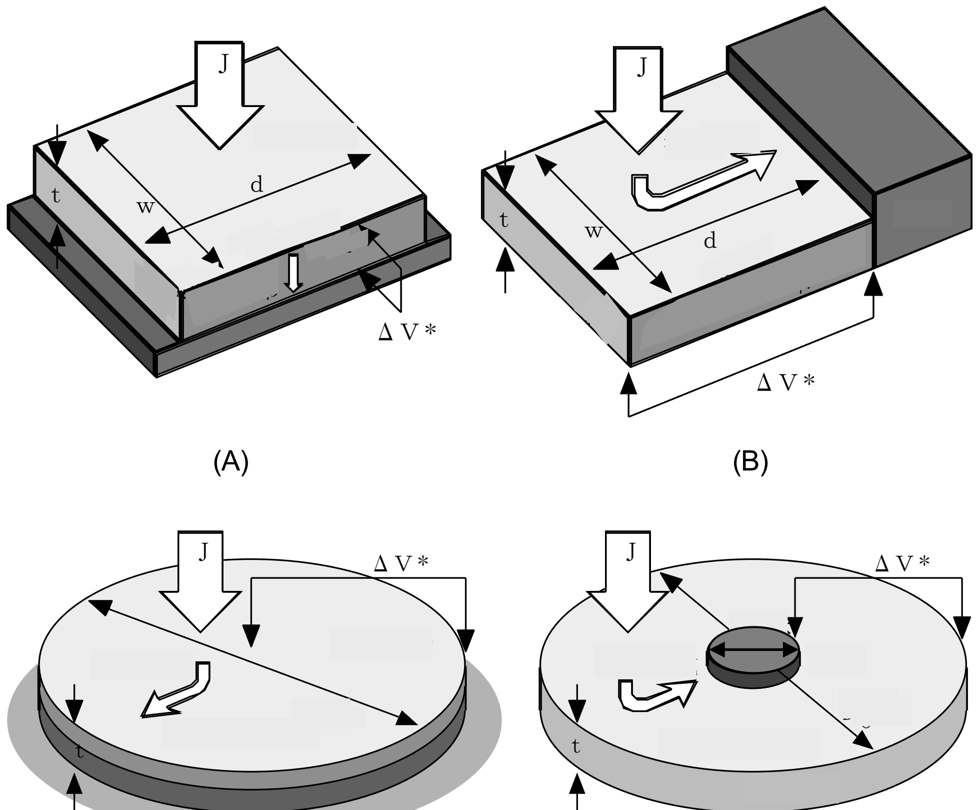


Fig. I.4-2 Surface current and bulk current that flow through a high-resistance material

There is an upper limit to the current of the charged particles coming from external plasma beyond a sheath (10-100 μA/m^2 [9] for GEO), and the requirements related to AR, ρt, and $R_s d^2$ are determined by the requirements related to the minimum differential voltage values of insulators or floating metals.

It is important to give attention to the fact that the volume resistivity varies depending on radiation, temperature, light, and charging levels (refer to Item 3.1.6 and Item I.6). Furthermore, the surface resistivity is affected by the contaminated surface condition.

I. 4.4 Local charging in LEO

The plasma density in LEO is larger than that in GEO by four to six orders of magnitude. Thus, the governing current items to the structure of a spacecraft are external electrons and ions, and they have a temperature of only 0.1-0.2 eV. The spacecraft electrical potential is determined so that the inflow of electrons and ions from the sheath boundary can be balanced.

If the interconnector of a solar panel and the electrode sections of solar cells are exposed to outer space, both the positive and the negative terminals of the solar panel circuit are inevitably exposed to plasma. This state is equivalent to the state in which a double electrostatic probe is placed in the plasma. The steady-state electrical potential of a spacecraft is determined so that the current of electrons collected by the side of the positive pole side can balance the current of ions collected by the side of the negative pole (Fig. I.4-3). When the negative terminal of the solar panel circuit is grounded to the spacecraft structure, the grounded surface of the main body of the spacecraft can also participated in the collection of ions. Thus, the total area that is negatively charged to plasma becomes very large. Most solar panels, however, generate negative electrical potential to that of plasma because electron mobility is much higher than ion mobility. The spacecraft electrical potential ϕ_{sc}^* can be approximated by the following formula as the first-order approximation:

$$\phi_{sc}^* \approx -V_a \qquad \text{(I.4-17)}$$

where, V_a stands for the generation voltage.

The electrical potential of an insulator on the spacecraft surface is also determined so that the zero-current condition can be satisfied. The potential is a negative value about several times the electron temperature or even if the potential is positive, the value is about 5 V, which is equivalent to the orbital motion energy of the ions. The positive potential occurs because the negative sheath that spreads outward from a neighboring exposed conductor inhibits the inflow of Ionospheric electrons [15]. No matter whether the insulator is positively or negatively charged, the electrical potential is quite smaller than the generation voltage by the solar panel and may be considered to be zero as the first-order approximation. Thus, it can be considered that the inverted potential gradient in which the electrical potential of a spacecraft insulator is higher than that of the spacecraft structure is the nominal state. The differential voltage in the steady-state is given by the following formula:

$$\Delta V^* \approx V_a \qquad \text{(I.4-18)}$$

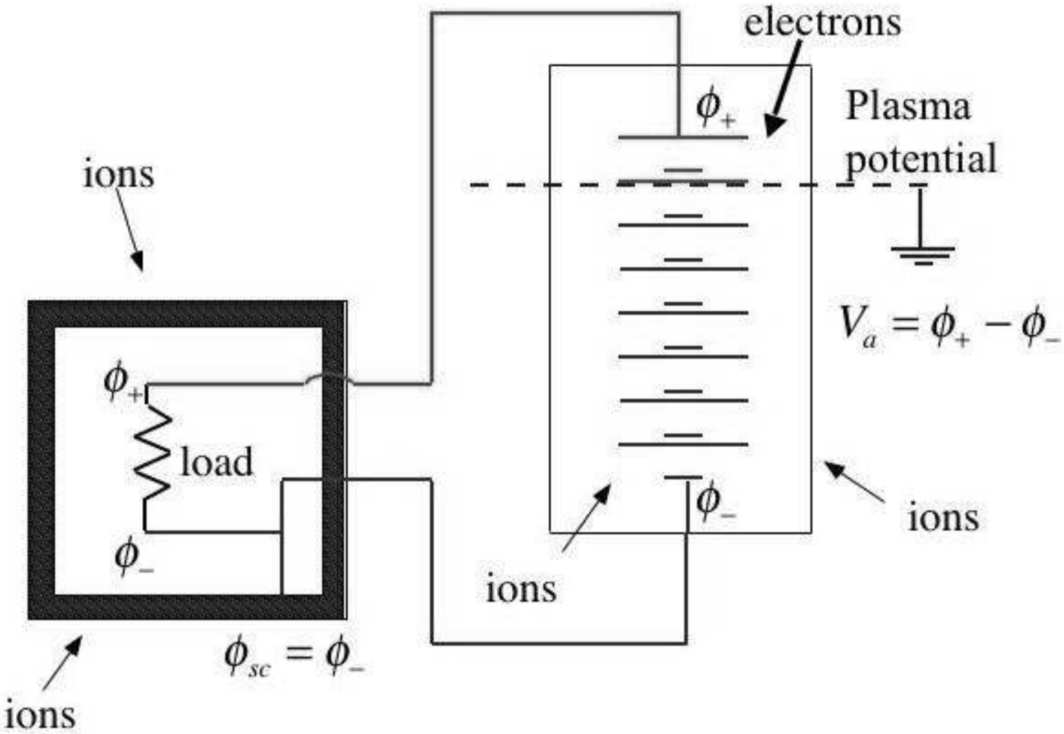


Fig. I.4-3 A schematic diagram of the current collection by a spacecraft in LEO

I.4.5 Local charging in PEO

While a spacecraft is passing the aurora zone, some dominant charging current items can turn into high-energy aurora electrons, not to ionosphere electrons. It was observed through Freja and DMSP [16] [17] that the spacecraft electrical potential of PEO satellites became negative when the background ionospheric ion density was 10^9 m^{-3} or less. Thus, for the spacecraft electrical potential to be negative in PEO, the altitude of the spacecraft should be quite high, or aurora electron flux should be extremely high.

Even though the spacecraft electrical potential is not negative, local charging can occur on the wake side of a spacecraft. Over the North and South Poles, the solar panel are facing the travelling direction. If a sufficient amount of ions can be collected on the side of the travelling direction (ram side), the spacecraft electrical potential can be maintained at about $-V_a$. Ions, however, cannot easily come around behind the back side of the solar panel because they inflow into the solar panel at supersonic speed [18] [19]. Thus, mounting insulators on the wake side results in producing a state in which only aurora electrons are collected. In particular, no photoelectron current can also be expected on the backside of the solar panel because the surface is not irradiated with sunlight. When the energy of aurora electrons is several keV or higher, the secondary electron emission coefficient, in most cases, falls below 1.0, and consequently, negative current exerts a dominant effect on the wakes side insulators, continuously causing negative electric charge to be accumulated on the insulators and thus, achieving the normal potential gradients state in which $\Delta V^* < 0$. Two-dimensional numerical calculation analysis [20] [21] predicts that a differential voltage larger than -1 kV can be generated.

If aurora electron flux is so high as to cause the spacecraft electrical potential to be negative and if ionospheric ion density decreases, the resulting situation becomes similar to the state during the occurrence of a substorm in GEO. At this time, whether the differential voltage becomes positive or negative depends on the energy spectrum of the aurora electrons. To calculate the charged state of an actual spacecraft in a PEO environment, it is necessary to define the sheath boundary around the spacecraft, which is generated by ionospheric plasma, as well as to use a three-dimensional charge simulation program such as MUSCAT [10] or NASCAP/2K [11][12].

I.5 Interference with electrical propulsion thruster

This section deals with engines, hall thrusters (SPT), and electrical field emission thrusters as the main electric thrusters. The plasma produced by electrical propulsion thrusters include high-energy primary ion plasma beams (300 eV for SPT, 1,000 for ion engine, and 8,000 eV for FEEP), low-energy charge exchange ion aggregates (up to several tens of eV), neutral particles, and low-temperature plasma emitted from neutralization devices

Emissions from thrusters can be classified as follows [3] [22]:

I.5.1 Primary beam ion

Most of these ions are concentrated in a space with a half-solid angle of 10º-40 º. Most thrusters have a high ionization efficiency of 95% or more. The primary ions normally consist of singly ionized ions: Doubly ionized ions account for only several percent (they account for more than this value for hall thrusters). Note that the current distribution in a typical ion beam can be approximated by a Gaussian distribution within a half-solid angle

I.5.2 Neutral particle

Since usage efficiency of propellants is not 100%, a neutral environment is also produced around a thruster. In addition, there are a small amount of metal particles sputtered from grid materials or acceleration channels. Use of metal propellants can cause serious contamination problems, but they do not affect the charging of a spacecraft until they are ionized. Table I.5-1 summarizes major plasma characteristics at the exit surface of typical electrical propulsion thrusters.

Table I.5-1 Plasma state at the exit surface of several electrical propulsion thrusters [23]

	Hall thruster	Ion thruster	FEEP thruster
Application	Telecommunication satellite and primary propulsion	Telecommunication satellite and primary propulsion	Scientific satellites
Exit plasma density	10^{17} m^{-3}	10^{16} m^{-3}	10^{15} m^{-3}
Exit neutral density	10^{17} m^{-3}	10^{18} m^{-3}	10^{18} m^{-3}
Ion energies	300 eV	1 keV	8 keV

I.5.3 Electric charge exchange ion

Electric charge exchange collisions may occur when high-energy primary ions collide with neutral particles of the propellants that are left unionized or collide with sputtered recoil particles. These collisions results in producing high-speed neutral particles and low-speed ions with heat speed. In the case of Xe ions, the collision is exemplified as follows:

$$Xe_{fast}^{+} + Xe_{slow} \rightarrow Xe_{fast} + Xe_{slow}^{+} \qquad \text{(I.5-1)}$$

When the electric charge exchange reaction occurs between similar atoms, the cross-section is particularly large and exceeds 10^{-19}m^{-3}.

Since primary ion beams exhibit a Gaussian distribution, the electrical potential of the beams also exhibit a similar Gaussian distribution, which is a mountain-shaped distribution of positive potential with the peak at its center line. Low-speed ions move away from the beams like a ball rolling on the mountain of positive potential. The solid angle of low-speed ions is much larger than that of primary ions, and the low-ions can partially flow toward the surface, solar panels, or other sections of a spacecraft. These backflow ions cause sputtering when they collide with negative high-voltage sections at the exit surface of the thrusters. In addition, some metal particles arising from sputtered recoil particles can be ionized through the electric charge exchange and can cause contamination when they adhere to optical sensors on the surface of the spacecraft.

To evaluate the backflow ion current, the orbit of these particles are generally calculated by using the Particle-in-Cell method or the Particle-tracking method [24][25][26][27]. At that time, it is important to precisely estimate the incidence rate of the electric charge exchange collisions and the electrical distribution of the primary ion beams. When a spacecraft is equipped with high-voltage solar panels and the electrical field due to the solar panel is affecting the electrical distribution around the ion beams, the only method is to use three-dimensional programs, but the actual calculation is quite difficult because of the restriction of the computational capacity of a computer.

I.5.4 Neutralization electron

Electrons emitted from neutralization devices are attracted to the electrical potential of primary ion beams through the low-temperature plasma emitted at the same time, thus neutralizing the ion beams. Furthermore, they are attracted to the positive electric charge of the electric charge exchange ions moving off the beams and flowing backward to the spacecraft side, thus diffusing in a bipolar pattern. The amount of electronic current for the latter is normally lower by several orders of magnitude than the amount of the current required for the neutralization of the ion beams. When a spacecraft has high-voltage solar panels and when the panels and the electrical propulsion thrusters are closely located, the electrical field by the solar panel can reach neutralization device electrons. Since these electrons are at the same electrical potential as that of the spacecraft potential, the positive terminal of the solar panel is almost at the same potential as the generation voltage to neutralization device electrons. This means that a closed circuit is established through electrons between the solar panel and the neutralization devices and the current flowing through the circuit is a leakage of power generated by the solar panel because the current is not supplied to spacecraft loads.

If excessive neutralization device electrons flow into the spacecraft side, the electrons to be used for beam neutralization are also involved; the beams are not neutralized and the spacecraft electrical potential reaches a large negative value [5]. Thus, there is a fear that failure in the beam neutralization during the operation of thrusters can lead to massive sputtering due to the backflow of the beams into the exit of thrusters.

A phenomenon called a “snap-over” is confirmed when during the mutual interaction between positively charged solar panels and electrons. It is also known that in this phenomenon, electrons drastically increase in number under a potential difference of about 200 V. Furthermore, the increase in the neutral gas density close to the surface of the solar panel drastically increases the amount of current leakage due to the occurrence of glow discharge. Fig. I.5-1 shows the relationship, which was confirmed by ground tests, between gas density and glow discharge inception voltage. Great care should be given to the increase in the gas density around the solar panel during the operation of electrical propulsion thrusters.

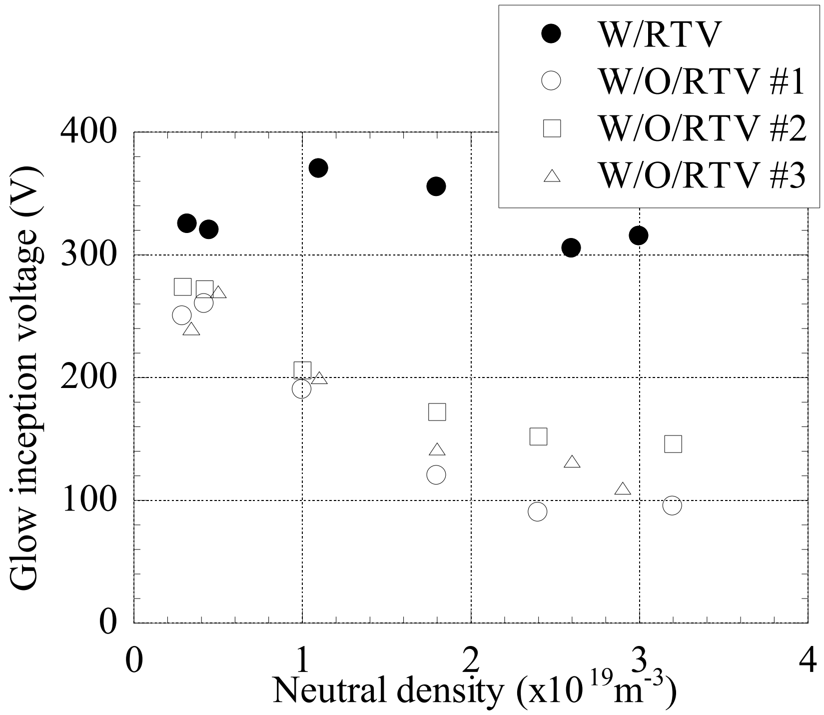


Fig. I.5-1 Glow discharge inception voltage on a positively charged solar panel surface. W/RTV refers to solar panels wrapped with RTV silicon except for interconnectors, whereas W/O/RTV refers to fully exposed solar panels [28].

I.6 Internal charging

Internal charging (e.g., refer to [29]) stands for the accumulation of electric charge in any section inside the structure of a spacecraft (excepting for its surface) because of the particles coming from an external space environment. Internal charging often occurs in the interior of dielectrics and often called “deep dielectric charging.” It, however, can happen in the interior of electrically floating conductors in a spacecraft.

I.6.1 Relationship with surface charging

Although electric charge can accumulate in thick dielectrics on the surface of a spacecraft over a certain depth range, surface and deep dielectric charging are not always distinguished clearly. General analysis considers the accumulation of electric charge in the materials near the surface and in the interior of materials as a whole. Internal and surface charging, however, can be considered separately because of the following practical reasons:

Surface charging is related to a large amount of current caused by low-energy plasma aggregates (up to 10 keV) that normally vary on a minute time scale. The current is mainly due to secondary electron emission and photoelectron emission, which can serve as dominant current items. Charging time scale varies: spacecraft structure can be charged in one or less than one second, whereas local charging requires one second to one minute. It is important to consider the balance between the primary and secondary current, as well as current exchange on the surface of a spacecraft. In general, these time scale values are too short as the time for the electric charge significant to the bulk conduction in electric materals to pass through the insulators.

Internal charging (bulk charging) is related to a small amount current caused by high-energy charged particles (larger than 0.5 MeV) that normally vary on a time scale of several hours to several days. It is not related to photoelectron emission. Secondary electron emission is generally also not important because the secondary electron emission coefficient for the primary energy is very small. In general, the time scale required for charging to occur is determined by the capacitive time constant of a whole body of material because the time scale is often a day or a longer period of time. Note that bulk conduction current plays an important role in the overall balance of electric current.

I.6.2 Insulation breakdown

Internal charging becomes an item of focus when stored electric charge suddenly begins to discharge, or insulation breakdown occurs. The most direct adverse effect of internal charging is the direct inflow of a large amount of transient current into electrical circuits or generation of indirect transient current through electromagnetic coupling. Furthermore, insulation breakdown can cause a permanent change in material properties, thus reducing the insulating capacity of the materials.

Insulation breakdown is considered to be caused by the electron avalanche effect due to high electrical field. Note that the discharge due to insulation breakdown can for tree-shaped discharge pathways called a Lichtenberg pattern.

According to published data, dielectric have an insulation breakdown electrical field of about 10^8 V/m. In laboratory experiments, however, partial discharge pulses were often observed at weaker macroscopic electrical field probably because of the concentration of local electrical fields. Thus, a threshold of 10^6 V/m should be used as the critical “danger level” at which discharge can occur.

I.6.3 Electric charge accumulation

Internal charging occurs when high-energy electrons penetrate the surface skin of a spacecraft. In general, it occurs at an electric potential is 0.5 MeV or higher. This phenomenon is observed both in the interior and exterior of the radiation belt. However, the electrical potential reaches the highest level in the exterior (especially in the case of high energy). In effect, only the external radiation belt is related to harmful internal charging. Satellites in GEO are likely to be affected by internal charging because although they are far away form the top of the external radiation belt, they are continuously located at the edge of the external belt.

The external radiation belt is very dynamic. In the external belt, an electron flux of more than 2 MeV can increase by two or thee orders of magnitude within several hours. Furthermore, such high-level of electron flux can continue over several days.

Electrons in the radiation belt penetrate the surface skin of a spacecraft and accumulate in the internal materials of the spacecraft. The penetration depth depends on electron energy and material properties (especially density). Note that the depth can be calculated with high accuracy by using the Monte Carlo transport code [[30]]. Furthermore, there are several empirically derived formulae [[31]].

I.6.4 Conductivity of materials

In determining the internal charging level in dielectrics , the electrical conductivity plays an extremely important role. When a dielectric is continuously irradiated with charged particles, they increase the internal electrical field of the dielectric , eventually causing the accumulated current to balance with the bulk conduction current, thus achieving an equilibrium state. In an equilibrium state, the maximum electrical field in a single layer of a material with depth d can be calculated by applying Ohm’s law.

$$\mathbf{V=IR \quad i.e. \qquad Emax = V/d = (I/A).R\ A/d = j/\sigma} \tag{I.6-1}$$

where,

- V = Electrical potential ［V］
- I = Electrical current ［A］
- R = Resistance ［Ω］
- j = Electrical current density ［A/m^2］
- σ = Volume conductivity (=1/ρ) ［S/m］

$$\sigma = \frac{d}{RA} \quad \text{and} \quad j = I / A \quad \text{(A: area of a body)}$$

The conductivity of a dielectric significantly depends on temperature, electrical field, and radiation. Thus, the calculation of internal charging usually becomes more complex. At present, a standard electrical conductivity measuring method is established [32]. However, there are doubts about whether the measuring method can apply to the cases that continue over a long time scale of several days [33].

I.6.4.1 Temperature dependency

The conductivity of dielectric materials is significantly affected by temperature. In other words, the higher the temperature, the higher the conductivity because captured electrons can be easier to be excited to conduction bands. Thus, the resistivity of a conductor is inversely proportional to the temperature. Note that the temperature dependency of conductivity is generally expressed in the following formula:

$$\sigma(T) = \sigma_{\infty} \exp\left(-\frac{E_a}{kT}\right) \qquad \text{(I.6-2)}$$

where,

E_a = Activation energy (material-dependent)
k = Boltzmann constant (J/K)
T = Temperature (K)
σ_{∞} = Maximum conductivity as T tends to infinity [S/m]

It is important to note that E_a is not equal to a bandgap voltage related to exciting electrons from a valence band to a higher energy conduction band (E_a is about 1 eV for polyethylene although its band gap is about 8.8 eV). The activation energy has been found through experimental work; it is normally about 1 eV for most dielectrics .

I.6.4.2 Electrical field dependency

Electrical conductivity can be increased by electrical field because high electrical field improves the mobility of carriers and further activates the carriers. A recent relevant study was conducted by Adamec and Calderwood (A&C) in 1975 [34]. In their study, the relationship between electrical field and electrical conductivity is given by the following formula:

$$\sigma(E,T) = \sigma(T)\left(\frac{2+\cosh(\beta_F E^{1/2} / 2kT)}{3}\right)\left(\frac{2kT}{eE\delta}\sinh\left(\frac{eE\delta}{2kT}\right)\right) \qquad \text{(I.6-3)}$$

where,

E = electrical field [V/m]

$$\beta_F = \sqrt{\frac{e^3}{\pi\varepsilon}}$$

δ = Jump length [m]
e = Electric charge of an electron (= 1.609 x 10^{-19} C)

The above formula is basically a theoretical formula (however, δ is used to so as to apply experimental data to the formula). In addition, $\sigma(T)$ refers to the effect of temperature on the above mentioned bulk conductivity.

I.6.4.3 Effect of radiation

Polymers are proved to increase the electrical conductivity when they are irradiated with radiation, and many empirical studies have been reported on the effect of radiation to the conductivity. Most of these studies, however, were conducted assuming much higher dosage-rates than those at which a spacecraft will actually encounter during its operation. In theory, irradiating electrons in a polymer with radiation serves to excite the electrons and cause them to move into conduction bands, producing the electric charge carriers that are directly proportional to the energy absorption rate (that is, dosage-rate) in the polymer. The dosage-rate depends on high-energy electrons, ions, or gamma-rays.

In 1956, Fowler showed the following basic formula related to the electrical conductivity σ of an irradiated polymer—the formula is widely used now [35]:

$$\sigma = \sigma_o + k_p D^{\Delta} \qquad \text{(I.6-4)}$$

where,

σ_o = Dark conductivity [Ω^{-1} m^{-1}]

k_p = Coefficient dependent on material conductivity induced by incident radiation [Ω^{-1} m^{-1} $rad^{-\Delta} s^{\Delta}$]

Δ = Dimensionless index dependent on materials (Δ<1)

After the end of irradiation, the radiation induced conductivity RIC gradually decreases [36]. The time constant for the decrease tends to be longer with increasing amount of irradiation. Note that slow decrease in RIC is often called "delayed" RIC.

I.6.5 Time dependency

With respect to laminar dielectrics , the electrical field in the bulk after irradiation exponentially reaches an equilibrium state with time.

$$E = \frac{J}{\sigma}\left(1 - \exp\frac{-t}{\tau}\right) \qquad \text{(I.6-5)}$$

where, τ denotes time constant. The above formula is the same as that for planar capacitors. When the electrical properties of a dielectric are uniform across the material, the time constant $\tau = \varepsilon/\sigma$ (ε: dielectric constant). Furthermore, when the electrical conductivity varies across a test sample of thickness x, the time constant τ is given by the following formula:

$$\tau = \varepsilon \int \frac{1}{\sigma(x)} dx \qquad \text{(I.6-6)}$$

Internal charging of dangerous levels is likely to occur only when the materials have a large time constant. In general, internal charging is likely to occur with materials having a time constant of one day or longer. The time constant τ is, in effect, an integral time of electron flux that charges a material. Thus, considering the average electron flux per day will give a precise time-resolution for defining a harmful internal charging environment.

I.6.6 Consideration items related to geometric shapes

A flat sheet structure is the easiest for the calculation of internal charging. Electrical conductivity, however, varies across a whole body even if it is the simplest structure because of ever-varying electrical field and bulk conductivity induced by radiation. Furthermore, the electrical field varies across a whole dielectric , and in an equilibrium state, the electrical field exhibits its maximum value on the boundary between the dielectric and the underlying conductor.

Cable insulators are normally a one-dimensional, symmetrical cylinder. The current in a cable with a single core conductor in the center concentrates on the cylindrical core conductor because the current in the surrounding dielectric flows toward the central part of the cable. As a result, the cable has a stronger electrical field than equivalent plane cables.

The calculation of the maximum electrical field of an insulator with a complex three-dimensional structure requires a three-dimensional electric charge accumulation model and a special three-dimensional model related to current and electrical field. In these cases, however, the electrical field exhibits its maximum value on the boundary of the dielectric and the conductor.

I.6.7 Internal floating conductor

A floating conductor in a spacecraft is usually grounded through an insulator. The electrical field between the floating conductor and the ground position continues to increase until a sufficient amount of current flowing through the insulator balance the charge current or it causes insulation breakdown before reaching the equilibrium state if exceeds the breakdown strength of the insulator.

I.6.8 Electrical field-sensitive system

Internal charging can affect systems consisting of semiconductors and electrically floating conductors. Floating conductors especially sensitive internal charging include three-axis acceleration meters and the gravity meters. The MEMS can also be affected by radiation-induced electrical field

I.7 Spacecraft-induced phenomena

I.7.1 Wake

A wake is formed when a spacecraft is moving through ionospheric plasma at a speed of 8 km/s, which is a supersonic speed for ions (heat speed: about 1 km/s) but a subsonic speed for electrons (heat speed: about 100 km/s). Thus, electrons can come around behind a spacecraft moving through ionospheric plasma although ions cannot come around easily. Even if some particles with negative high voltage are on the wake side, they remain negative because ions cannot reach the wake side. Aurora electrons can cause insulators on the wake side to be negatively charged to the degree of its electron temperature because they have an energy of several keV or higher.

I.7.2 vxB induced voltage

According to Fleming's right-hand rule, a conductor crossing a magnetic field can generate an electromotive force. In other words, when a conductor of length L is moving at a velocity vector of *v* through the magnetic flux density *B*, the electromotive force V_{emf} is given by the following formula:

$$V_{emf} = \left(\vec{v} \times \vec{B}\right) \cdot \vec{L} \qquad \text{(I.7-1)}$$

Since the terrestrial magnetism in LEO is about 30-50 μT, a potential difference of about 20-30 V will be generated between the both ends of a large-scale spacecraft like the ISS in LEO. If a tether of length 20 km is deployed in LEO, a potential difference of 5 kV will be generated across the tether.

If this voltage is used for power generation, electric current flows through the tether; the tether receives a resistance opposite to the orbital velocity, converting the kinetic energy of a spacecraft into electrical energy. In contrast, if an electric current is flown through the tether with a power-supply, a propulsion force (Lorentz force) is generated [37] [38]. In the inverse TSS-1R experiment [39]., a tether of length 19.7 km, which was deployed in outer space, successfully generated a potential difference of about 4 kV, thus verifying the theory of electricity generation, but it was unfortunately burn out by the discharge occurred at the base section on the side of the space shuttle [40].

When electric current flows through a tether, the circuit is closed when LEO plasma inflows into the both ends of the circuit. The current flowing through the tether can be collected by installing such devices as plasma contactors for emitting or collecting electrons and ions at the both ends of the tether or by directly collecting current from the exposed conductive part of the tether. In electric current collection from a tether, it is necessary to consider both plasma flow and the magnetic field at right angles to the flow—general particle simulation methods for the consideration are based on the Particle-in-Cell method. Furthermore, consideration should be given to the collision between charged and neutral particles because of the increase in the neutral gas density of plasma contactors.

I.7.3 Snap-over

When the electrical potential of the exposed conductor section near the positive pole of the solar panel is positive and higher than the plasma by more than 100 V, the exposed conductor collects a large amount of electrons from the surrounding plasma. Thus, the collected electrons and the ions collected on the negative potential side establish a loop of the current that flows through the plasma. These current results in the loss of the output power from the solar panel. The amount of collected electron current is determined by the flux that leaks from the interconnector of the solar panel and inflows into the sheath and leaking. However, when the voltage increases, a large amount of electron current begins to inflow into the sheath because the solar panel are covered as a whole with the sheath. This phenomenon is called a snap-over because external electron current inflows in the coverglass with the emission of large amounts of electron-impact secondary electrons [41] [42].

The snap-over itself is not affected by the surrounding gas pressure. However, when the gas density reaches about 10^{18} m^{-3}, a glow discharge begins to occur at the exposed metal sections of the solar panel even at a voltage of about 100 Vas mentioned in Item 5.5.4 [28] [43]. The glow discharge, which repeats itself to produce continual pulses, is considered to be caused by the electron-collecting sheath that are being spread outward by the ionized secondary ions that remain in the sheath for a long period of time [44] [45]. A gas pressure of 10^{18} m^{-3} is higher than the normal gas pressure in outer space by several orders of magnitude, but it may well be possible for such a high gas pressure to be produced in the vicinity of an altitude and orbit-control thruster while the thruster is in operation. Thus, care should be given to the increase in the gas pressure in the vicinity of the solar panel when the electrical potential of a high-voltage spacecraft is controlled by a plasma contractor.

I.8 Discharge

Spacecraft discharge occurs when the potential difference between the electrical potential of an insulator (or a floating conductor) and that of the structure of a spacecraft—the electrical potential is caused by surface charging or internal charging—exceeds a certain threshold value. The threshold value depends on many parameters including insulating materials, shapes, adjacent conductor materials, and external environments. Even if all the parameters were fixed, no unique and precise threshold value could be determined. Thus, the threshold value has been generally treated by stochastic approaches.

Electrostatic discharge (ESD) is classified into two types: (1) the discharge that occurs when the electrical field in an insulator exceeds its insulation breakdown potential or the discharge triggered by a partial discharge occurring in a void or a flaw in an insulator, and (2) the discharge resulting from what is called the Triple Junction (TJ), that is, the contact of an insulator with an externally exposed electric conductor. In both cases, the electrostatic discharge is the release of the electrostatic energy accumulated as a result of charging in various capacitors in a spacecraft.

I.8.1 Name of discharge current

Punch-through is the flow of current due to the recombination of the electric charge accumulated between both surfaces of an insulator (or the electric charge in a conductor and that induced in a conductor in contact with the insulator). The energy is supplied by the capacitance of the insulator.

Flash-over is the flow of current during the process through which high-density plasma spreads along the surface of an insulator from a discharge inception point, thus neutralizing the electric charge accumulated on the surface of the insulator. The energy is supplied by the capacitance C_d of the insulator.

Blow-off is the flow of current during the process through which negative electric charge is emitted from a discharge inception point to outer space. When charging occurs in GEO or LEO, the spacecraft is negatively charged, and when high-energy density plasma is generated at an originating point, electrons begins to flow outward along the electrical field in the sheath. This energy is supplied by the capacitance C_{sat} between the spacecraft structure and outer space.

I.8.2 Triple junction

The position at which three different materials come in contact is called a triple junction (TJ). Triple junctions require the most attention in the insulation design of electric equipment because electrical field tends to concentrate on TJs. A triple junction, which is important in the discharge on the surface of a spacecraft, consists of an insulator exposed on the surface, a conductor, and the surrounding plasma (or vacuum). There are a wide variety of TJ examples that can become an originating point of charging, such as solar panel coverglass, solar panel facesheet, wire coating, adhesive, thermal control material, and OSR. As long as the boundary with an electric conductor is exposed to outer space, there is a possibility of electrostatic discharge. Furthermore, even if the conductor is fully covered with a thin insulating film, TJs can be formed when the underlying conductor is exposed to outer space because of the collision of fine particles (debris) in orbit. In macroscopic perspective, the average electrical field of a TJ can be given by the following formula:

$$E = \frac{\phi_d - \phi_{sc}}{t} = \frac{\Delta V}{t} \qquad \text{(I.8-1)}$$

The Electrostatic discharge occurs when the differential voltage ΔV between the electrical potential of an insulator and that of the structure of a spacecraft exceeds a certain threshold value. In general, the threshold value in the inverted potential gradient stage when the differential voltage is positive is lower than that in the normal potential gradient state. This is because the initial electrons necessary for the occurrence of electrostatic discharge are easier to be supplied by electrical field emission as electric conductors serve as a negative pole in the inverted potential gradient state.

Fig. I.8-1and Fig.I.8-2 show example pictures of ESD on a solar panel test-coupon, which were taken in an experimental laboratory. In Fig. I.8-1, the solar cell circuit of the test-coupon was grounded and irradiated with electron beams of 15 keV: The surface of the coverglass was in the normal potential gradient state (ΔV = -7 kV) because the discharge on the surface of the glass occurred at an electrical potential of -7 kV. In Fig. I. 8-2, the solar cell circuit of the test-coupon was irradiated with the electron beams having an energy level of 2-8 keV distributed over the test-coupon by aluminum foil scattering while a direct voltage of -5 kV was applied to the solar cell circuit. Thus, the surface of the coverglass was in the inverted potential gradient state in which the electrical potential is kept at -4 kV (ΔV = 1 kV). Flash-over associated with discharging can be observed both in the normal and the inverted potential gradient states, but they have their differences [[46]].

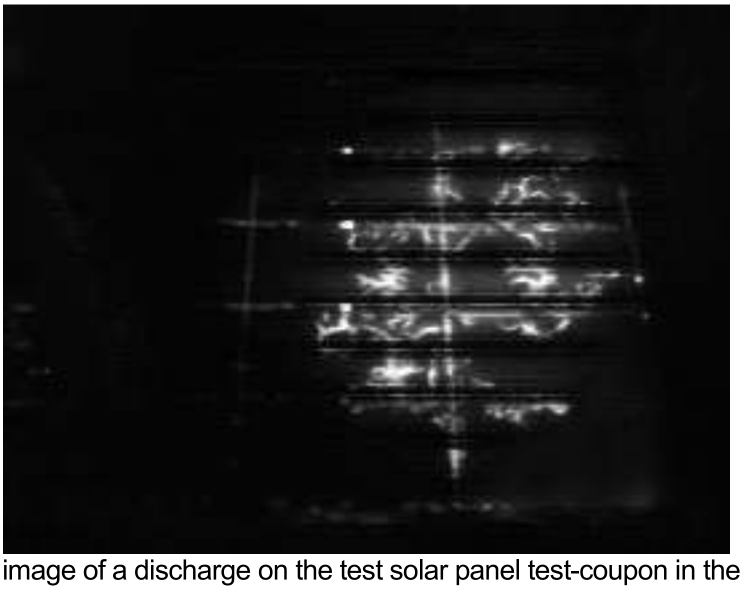


Fig. I. 8-1 Video image of a discharge on the test solar panel test-coupon in the normal potential gradient

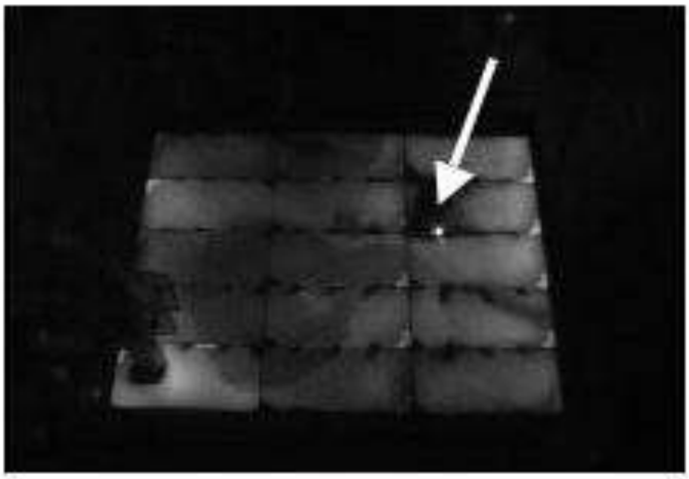


Fig. I. 8-2　Video image of a discharge on the solar panel test-coupon in the inverted potential gradient. The arrow shows a discharge inception point.

Research on the normal potential gradient has been conducted since 1970s, including ground experiments in which grounded specimens were irradiated by electron breams (e.g., [[47]]). Research on the inverted potential gradient in an LEO environment has also been conducted from 1970s (refer to [[48]] for details), but almost no research in a GEO environment has been conducted except for several studies conducted in the 1980s [[49]] [[50]] [[51]] [[52]]. However, the research on the inverted potential gradient in the GEO environment has been very active in many countries since the end of 1990s when spacecraft power was increased to the level of 10 kW and the inverted potential gradient was considered to be the cause of failures in the solar panel as shown later in this document [[53]].

I.8.3　Discharge generating mechanism under inverted potential gradient

The electrostatic discharge in the inverted potential gradient has been often called the “arc.” This is because the electrostatic discharge is quite similar to the arc occurring in a vacuum in which metal vapor is ionized. In this section, a solar panel is used to explain the discharge-generating mechanism. This mechanism, however, is applicable to the discharge that occurs at the TJs on the surface of a spacecraft in the inverted potential gradient state.

In a solar panel, each solar cell is connected in series to generate a predetermined voltage. In Fig. I.8-3, the structure of the solar panel— consisting of a CFRP sheet and an aluminum honeycomb substrate stacked one above the other—has the same electrical potential as that of the structure of a spacecraft. The electrical potential of the solar cells and the interconnectors are determined by the order in which they are positioned from the negative terminal of the solar panel circuit. In other words, the electrical potential at an arbitrary position in the solar panel is the potential of the spacecraft structure plus the sum of the voltage generated by individual solar cells located between the arbitrary position and the negative terminal. A normally designed solar cell has a facesheet adhered to the stacked CFRP-aluminum structure. Cove glass plates are mounted on the top of each solar cell. These solar cells are finally connected to the positive and negative terminals of the solar panel through the interconnectors. Most interconnectors are exposed to outer space in order to absorb the stress due to thermal cycle. Each solar cell is separated by less than 1 mm. The insulator consisting of the coverglass and the transparent adhesive mounted on each solar cell have a thickness of 100- 200 μm in total.

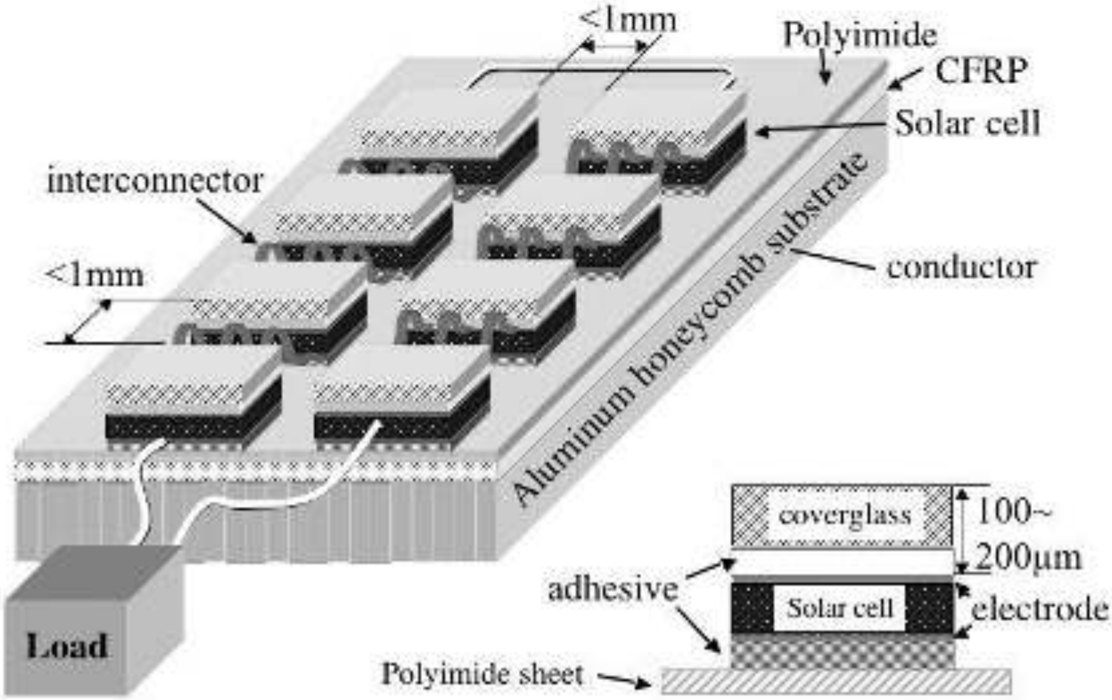


Fig. I.8-3 Structure and cross-section of a typically designed rigid solar panel

It has been known since 1970s that a discharge occurs when solar panels are in the inverted potential gradient state, and its generation mechanism was actively studied from the late 1980s to the early 1090s. This discharge is an isolated phenomenon observed by ground experiments as a current pulse having a width of 100 ns to several μs—it is sometimes called a primary arc (trigger arc or primary ESD) in order to distinguish the discharge due to a secondary arc (sustained arc, continuous arc), as described later in this document.

Fig. I.8-4 explains how a primary arc begins. This schematic diagram was proposed by [54] [55] and [56]. In Fig. I.8-4, the electric conductor corresponds to interconnectors, solar cell electrodes, or bus bars of solar panels, whereas the insulator corresponds to coverglass plates, adhesive, or polymer sheets. Fig. I.8-5 shows the occurrence sites at which the presence of primary arc was confirmed by laboratory experiments. As shown in the figure, the occurrence sites of primary arc match the positions of triple junctions.

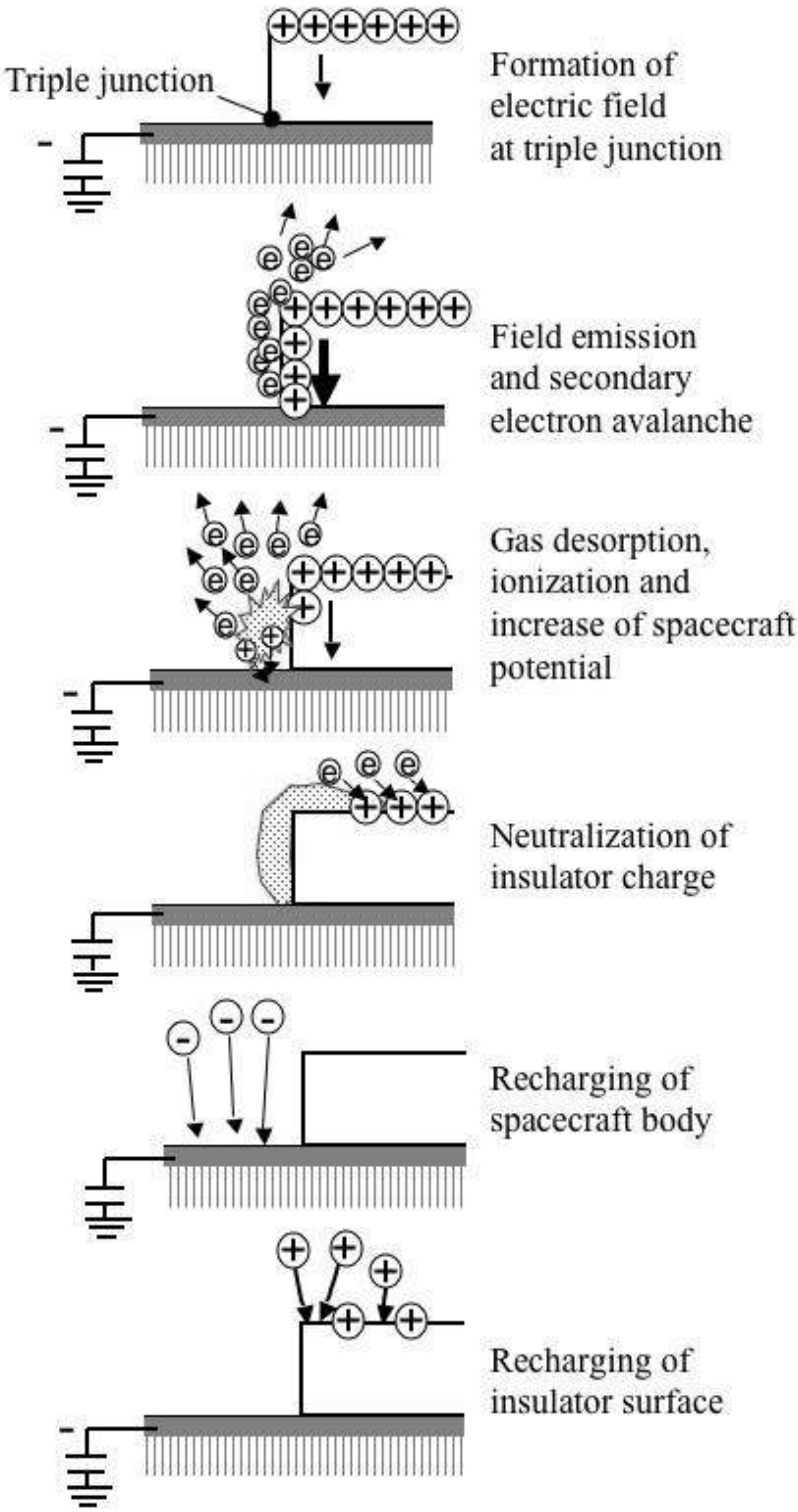


Fig. 8-4 A schematic diagram for primary arc generation mechanism in the inverted potential gradient

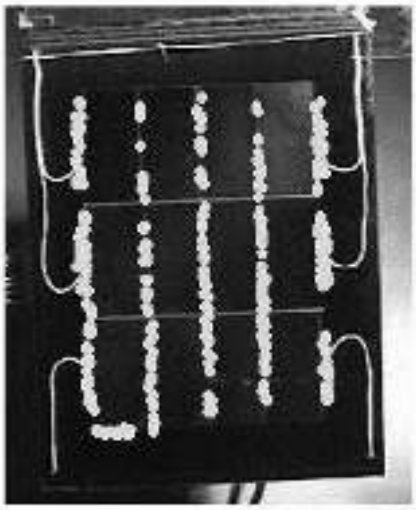


Fig. I.8-5 Occurrence sites of primary arc (white circle) in the inverted potential gradient confirmed by ground experiments [57]. The solar panel test-coupon was biased to a negative potential of -500 V in an LEO plasma-simulating environment. The gap between solar panel rows are filled with RTV silicon.

(1) The inverted potential gradient occurs when the insulator surfaces are positively charged. The arrows in Fig. I.8-4 indicate that the electrical field are focusing on the vicinity of TJs. In LEO, the surfaces are positively charged by external ions, whereas in GEO, they are charged by secondary electrons or photoelectrons.

(2) Electrons are emitted from the surface of a conductor through electrical field emission when an electrical field in the order of about 1.0 MV/m is applied to the vicinity of TJs by the inverted potential gradient. These elections, in turn, collide with the side surface of an insulator because the electrical potential in the vicinity of TJs is formed so that it can attract the electrons to a side of the insulator. Thus, electron-impact secondary electrons are emitted from the insulator. If the electron-impact secondary electron coefficient is 1.0 or higher, the insulator is positively charged because of the remaining positive electric charge on its side. This charge serves to create a positive feedback circuit because it intensifies the electrical field in the vicinity of TJs. Thus, the electrical field emission current increases explosively. Note that an electrical filed of 1.0 MV/m is weaker than the value necessary to cause electrical field emission by about two orders of magnitude. Thus, the above explanation is based on the assumption that there are some microspikes or insulating impurities that can locally intensify the electrical field in the vicinity of TJs.

(3) As the electrical field emission current explosively increases, the number of electrons colliding with the side surface of insulators also increase. At the time, the gas attached to the side surface is detached and turns into a thin layer. Then ionization occurs in the gas layer, leading to the generation of discharge. This discharge, in turn, causes positive electric charge to flow into conductors, thus generating a discharge current. On the other hand, the positive charge accumulated in the surrounding insulators is neutralized by the discharged plasma electrons. The mechanism described in the above items (2) and (3) is similar to the electron-impact secondary electron avalanche [58] [59] proposed with respect to vacuum flash-over but different in that no positive pole is located in the immediate vicinity.

(4) With the ionization of the detached gas and the neutralization of the electric charge of insulators, a discharge current flows on the surface of a conductor. When the discharge current concentrates on a particular part of the surface, it causes the part to evaporate and produces a metal vapor, forming a state similar to ionizable vacuum arc. This fact is demonstrated by the light-emission spectra from cratered surfaces [60] on a conductor, gas particles [61], and metal particles [62].

(5) Along with the emission of electrons into outer space, there is an emission of electric charge accumulated in the capacitance between the structure of a spacecraft and outer space. Thus, the electrical potential of a spacecraft rapidly increased up to the external plasma potential.

(6) The discharged plasma begins to spread in the form of flash-over, neutralizing the electric charge on insulator surfaces. When the flash-over stops, the supply of energy and electric charge is also stopped; thus the primary arc is stopped.

(7) The exposed conductor surface of the spacecraft begins to collect negative electric charge, and the spacecraft is again negatively charged.

(8) The insulator surface begins to collect positive electric charge from external plasma, thus causing the inverted potential gradient to occur and returning the state of item (1).

The discharge generating threshold, which is confirmed in laboratory experiments for solar panels exposed to the plasma equivalent to that in LEO ΔV , is 100 V [63][64]. The plasma equivalent to that in LEO refers to a state in which it has a density of 10^{10}-10^{13} m^{-3} and is created by a low energy of about 1.0 eV under a degree of vacuum of 10^{-3}-10^{-2} Pa. A discharge can occur when not only solar panels but also TJs are exposed to outer space—CFRP surfaces, holes in thermal control materials and flaws in power cables. In particular, a discharge is likely to occur on CFRP surfaces because carbon fibers are intimately entangled with resin, forming a number of TJs when the generation voltage on the surface exceeds -70 V [65][66]. Thus, the spacecrafts having a generation voltage of 70 V or higher should be defined as high-voltage systems.

It is confirmed that in a laboratory environment that can simulate GEO (an electron beam with an energy level of several keV and with a current density of 10 μA/m^2 to 10 mA/m^2 can be irradiated under a degree of vacuum of 10^{-4} Pa to 10^{-3} Pa), a primary arc occurs on solar panels when $\Delta V \geqq$ 400 V [28]. Laboratory experiments with the same solar panel test-coupons have also shown that the threshold value in the GEO environment is generally higher than that in the LEO environment. It is, however, still unknown whether the difference in the threshold vale is due to the difference in back pressure or that in charging method.

I.8.4 Paschen discharge

Increase in pressure during the use of high-voltage will increase the risk of causing a discharge based on the Paschen's law. In particular, if air is used as a residual gas, the minimum direct voltage for Paschen discharge is as much as 330 V, as shown in Fig. I.8-6 and Table I.8-1.

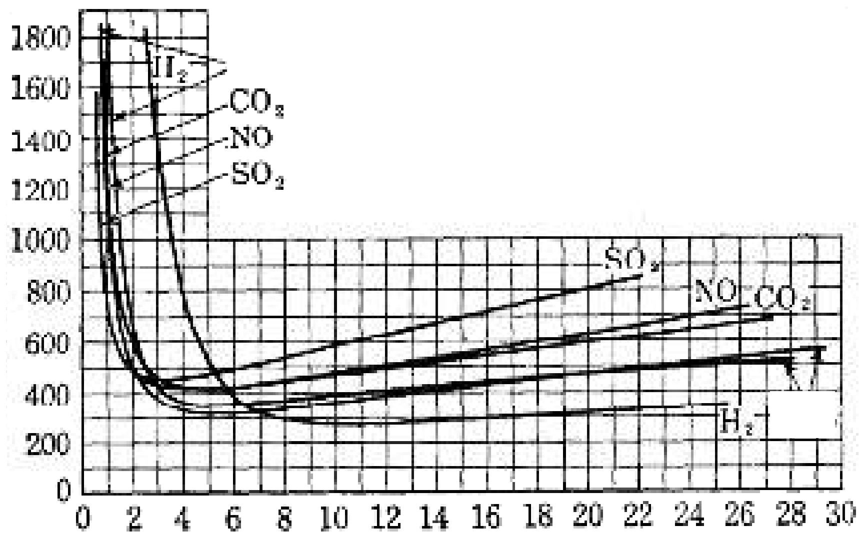


Fig. I.8-6 Paschen curves (example)

Table 8-1 Minimum sparking voltage $(V_s)_{min}$ and corresponding pd (pd)

Gas	$(V_s)_{min}$ (V)	$(pd)_c$ (mm・Torr)
Air	330	5.67
H_2	270	11.5
O_2	450	7.0
N_2	250	6.7
He	approx. 156	approx. 40
Ar	233	7.6
Ne	186	3.0
Navaho	335	0.4
CO_2	420	5.4

Discharge inception voltage based on Paschen's law was the value measured on the premise of a closed electrode-edge system. However, most electrode-edges in an actual high-voltage system for a spacecraft are partly or totally an open system. In this case, a long discharge distance can be ensured even at low pressure provided that the electrode-edge is an open system because, according to Pashen's law, the discharge voltage is determined by the product of pressure *p* and the discharge distance *d*. Thus, the minimum Paschen voltage is maintained even at low pressure. Fig. I.8-7 shows the pressure dependency of flash-over voltage with an electrode spacing of 1 mm [[67]]. Comparison with the Paschen curves in Fig. I.8-6 when *d* = 1 mm shows that the region of the minimum discharge voltage extends to the low pressure side. Thus, it is desirable not turn on power-supply voltage before sufficiently high degree of vacuum is reached.

The above Paschen discharge voltage characteristics are the ones measured by applying direct voltage. The minimum Paschen discharge, as shown in Fig. I.8-8, decreases with increasing power-supply frequency [68]. Thus, care should be given to high-frequency devices such as TWTs. They should be considered sufficiently by experiments.

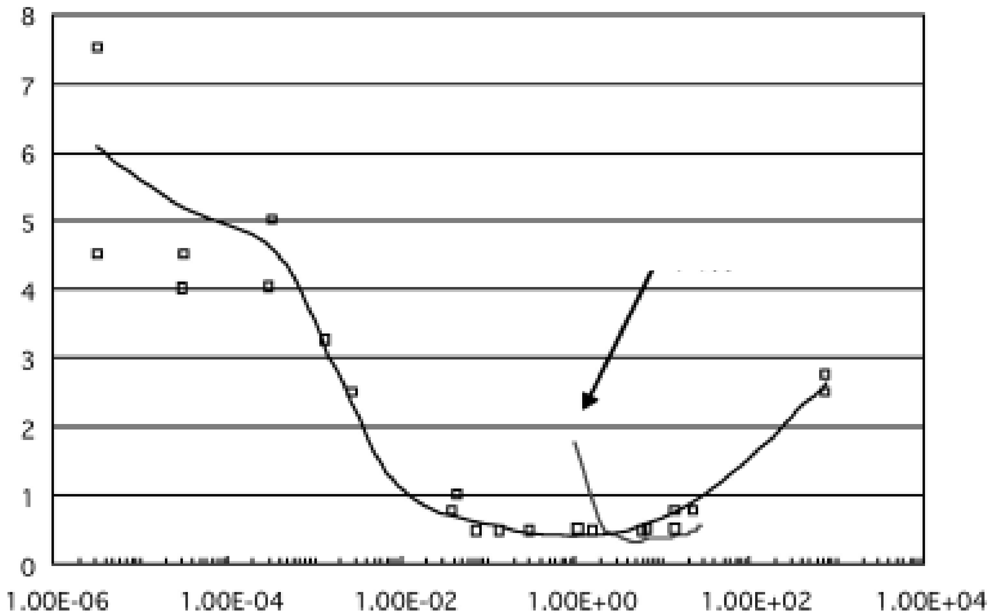


Fig. I.8-7 Pressure-dependency of flash-over voltage (printed board: FR-4, electrode spacing: 1 mm) [67]

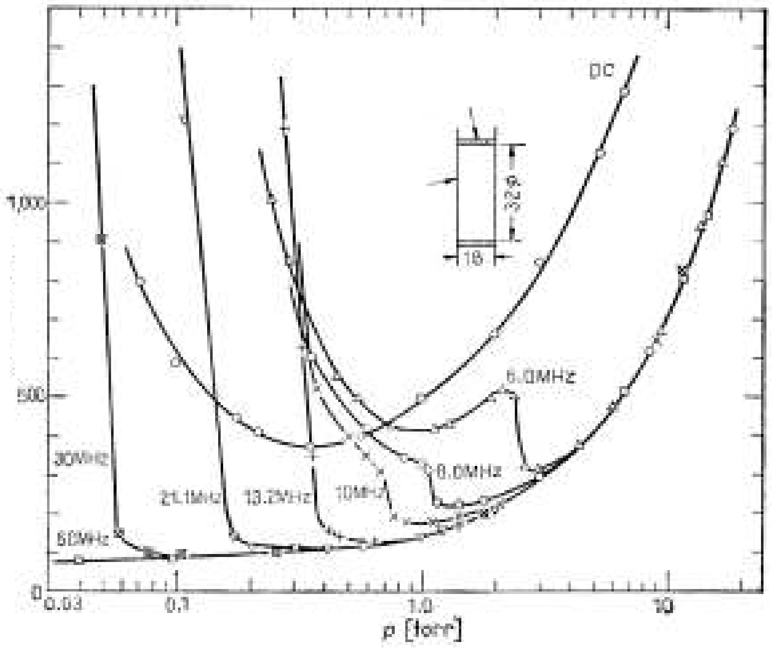


Fig. I.8-8 Dependency of power-supply frequency on Paschen curves [68]

I.8.5 Multipactor discharge

In dealing with high-frequency devices such as travelling wave tubes and output multiplexers, sufficient consideration should be given to multipactor discharge.

This section deals with the case when high-frequency voltage is applied to electrodes in a vacuum. When a high-frequency voltage is applied to two electrodes with electrons between them, and when the electrons have spent just half the period of the high-frequency to reach the positive pole, the positive pole that has accelerated the electrons as a positive pole turns into a cathode and begins to accelerate the electrons in the opposite direction. The electrons cause secondary electron emission on reaching the opposite electrode provided that their energy is at a certain level or higher. If the secondary electron emission coefficient is 1.0 or higher, a greater number of secondary electrons than incident electrons are emitted toward the opposite direction. Furthermore, if the polarity of the opposite electrode is changed as soon as these electrons have reached the electrode, a greater number of new secondary electrons are emitted again; these emitted electrons return to the previous electrode. In this way, the number of electrons increases innumerably while they are moving back and forth between these electrodes. In practice, this state comes to equilibrium when these electrons have increased to reach a certain level of electron density: They sometimes collide with residual neutral particles and are ionized, thus resulting in a steady discharge state. This is what is called the multipactor discharge phenomenon, which is the discharge caused by the secondary electron resonance multiplication effect in a vacuum [68]. Thus, multipactor discharge can be summarized as follows:

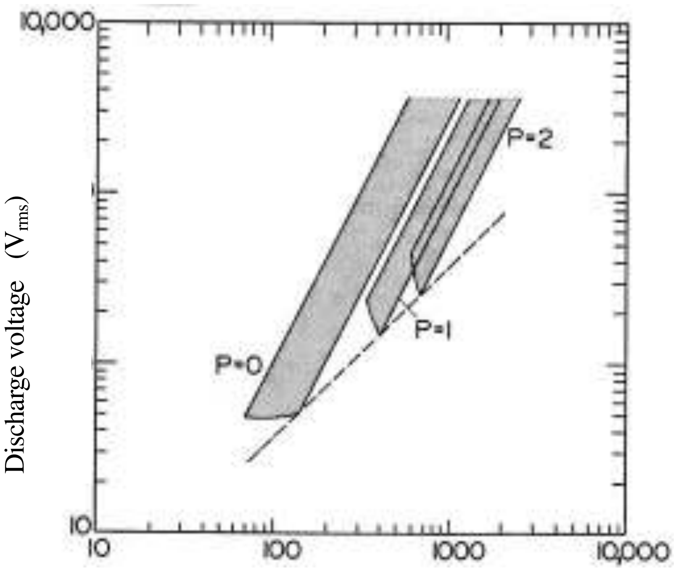
(1) Discharge inception depends on only electrode materials irrespective of gas pressure or gaseous species.

(2) Conditions for discharge inception

(a) Electron transit time between electrodes is equal to half the period of high-frequency voltage

(b) The secondary electron emission coefficient of electrode materials is 1.0 or higher.

Multipactor discharge has been theoretically studied and reported. Fig. I.8-9 shows a research example showing a relationship between discharge voltage and frequency.



Frequency (Hz)

Fig. I.8-9 Generation of multipactor discharge

I.8.6 Flash-over in the interior of spacecraft

The power-supply for high-voltage systems is not affected by the external plasma outside a spacecraft because it is normally installed inside the spacecraft. On the other hand, high-energy charged particles (normally electron beams with an energy level of 500 keV or higher) in a space environment penetrate the surface skin of a spacecraft, lose their energy, and are accumulated in the insulating materials inside the spacecraft. The effect of the accumulated electric charge on the power-supply for the high-voltage systems inside a spacecraft, particularly on the flash-over voltage, should be considered to verify that no problems will occur through experimental simulation.

Fig. I.8-10 shows the effect of electron beam irradiation on the flash-over voltage between the electrodes mounted on a printed board (material: FR-4) [67]. The figure shows that the electron beam irradiation serves to lower the flash-over voltage: the lower the electron energy, the lower the flash-over voltage. Thus, it is necessary to sufficiently understand the effect of electron beam irradiation on flash-over characteristics and to consider design problems.

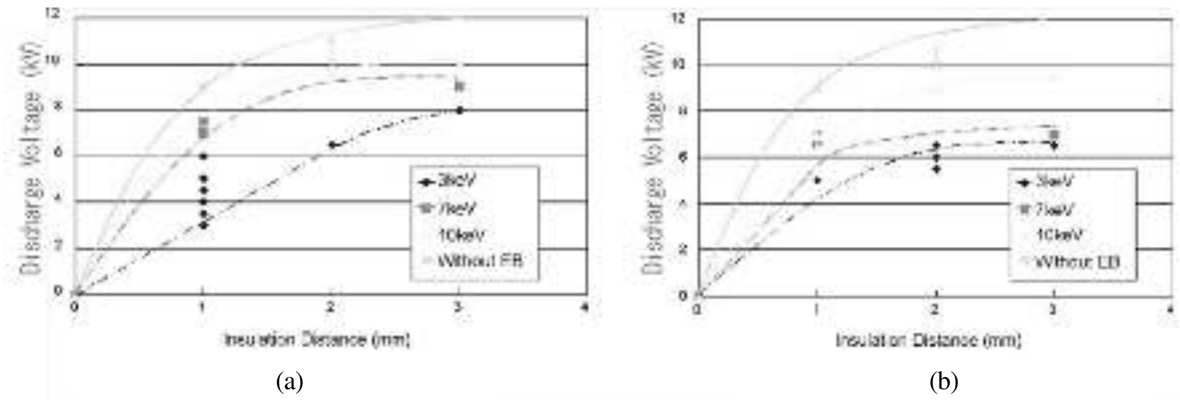


Fig. I.8-10 Electrode spacing dependency of flash-over voltage in a vacuum under electron beam irradiation (printed board: FR-4) [67]

I.8.7 Primary discharge development

If a primary discharge occurs at a position directly connected to the power-supply circuit, the degree of the adverse effect depends on the amount of the primary discharge current or energy. Primary discharge begins as a local ionization phenomenon, but it can develop into a significant phenomenon that involves a whole solar panel or a whole spacecraft. The energy of primary discharge immediately after its occurrence is supplied by the electrostatic energy between the spacecraft and outer space, but the energy is as small as about 1 μJ because of the small capacitance is usually less than 1 nF. After the energy has been used up, the development of the primary discharge depends on the electrostatic energy stored in the capacitance of insulators. Along with the sharp increase in spacecraft electrical potential to about zero potential, the surface electrical potential of insulators increases sharply with the insulator surfaces attracting electrons in the vicinity of discharge inception points. A current circuit is then established between a discharge inception point and an insulator surface through the flash-over plasma and a spacecraft circuit, and the electric charge stored in the capacitance of insulators is released through it like an RC discharge. The insulator capacitance on the spacecraft surface is often larger than 10 μF. Thus, an amount of total suppliable energy can exceed 10 J.

I.8.8 Secondary arc in the electric power system of a spacecraft

Consideration should be given to adverse cumulative effects due to such as the reduction in power generation capacity of solar panels due to primary discharge or progression of contamination. However, a single primary discharge is unlikely to cause a fatal damage to the operation of a spacecraft: primary discharge serves to trigger secondary arc.

Fig. I.8-11 shows a schematic diagram of a typical spacecraft power system. If a primary discharge occurs at the starred positions in the figure, the power system may suffer a short-circuit failure.

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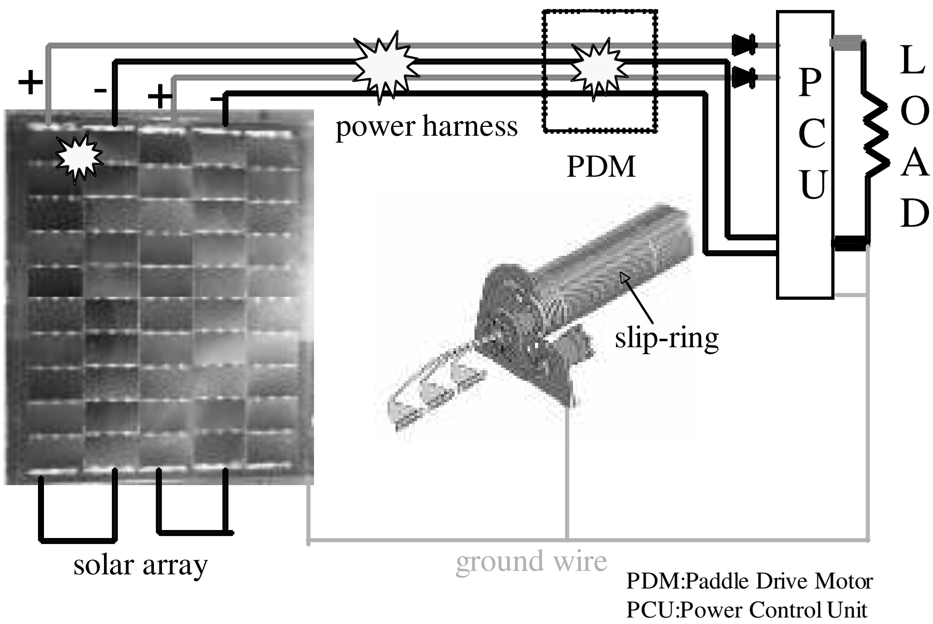

Fig. I.8-11 Typical spacecraft power system and dangerous positions in which secondary arc can occur.

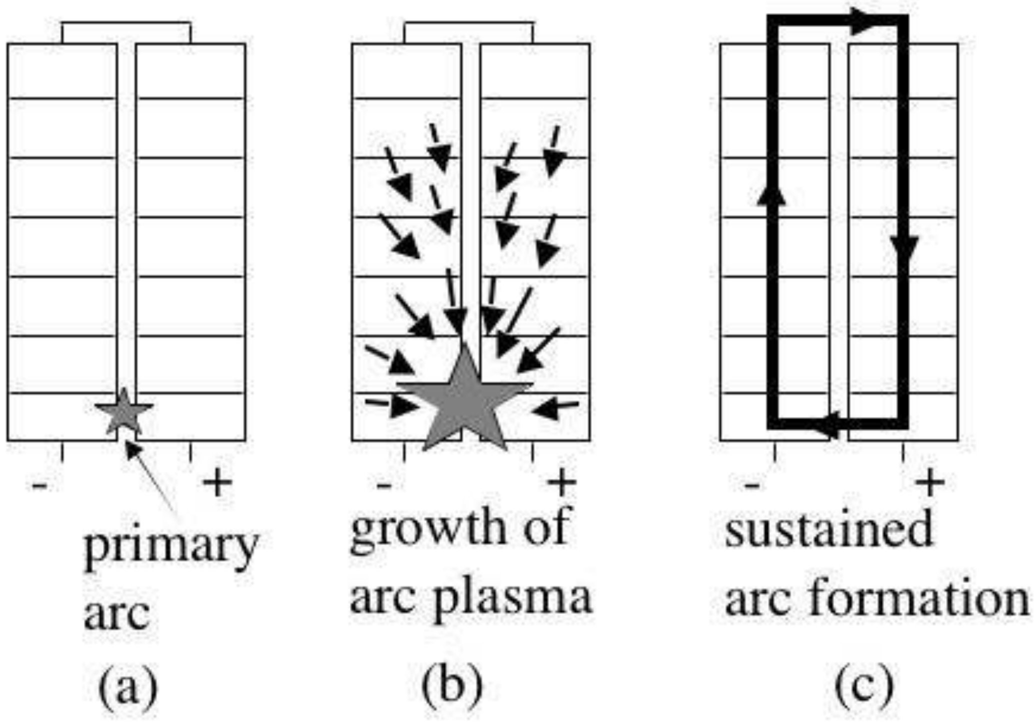

Fig. I.8-12 Schematic diagrams showing primary discharge development and a secondary arc formation process

Fig. I.8-12 shows schematic diagrams for primary discharge development and a secondary arc formation process. If a primary discharge occurs between two points with a high potential difference, the two points are short-circuited by the discharge plasma. The short-circuit current can continue to flow depending on the maximum current suppliable by the solar panel circuit. This state is called the secondary arc.

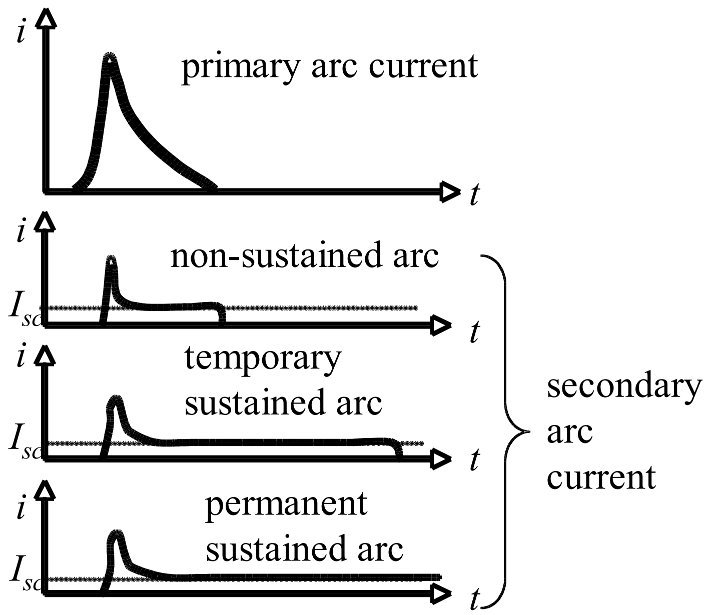

Fig. I.8-13 The name of each level in the secondary arcphenomenon . The current I_{sc} refers to the suppliable short-circuit current from the solar panel

Fig. I.8-13 shows technical terms related to secondary arc. When a primary discharge or a primary arc occurs and when the current from a solar panel circuit is added to the current caused by the primary discharge, the resultant discharge is generally called the secondary arc. When the short-circuit current flows only when the primary discharge is flowing, the discharge is called the non-sustained arc. When the current is continuously supplied by the solar panel circuit even after the termination of the primary discharge but the current stops flowing spontaneously after a certain period of time, the discharge is called the temporary sustained arc, or TSA. When the supply of current from the solar panel continues, the discharge is called the permanent-sustained arc or PSA. In laboratory experiments, once a permanent-sustained arc occurs, it causes the insulating materials near the discharge inception point to be carbonized by the generated heat in several to several tens of seconds, thus establishing a grounding pathway to a CFRP and aluminum honeycomb substrates. Then, the current from the solar panel circuit begins to flow through the grounding pathway, not through the arc. Thus, as shown in Fig. I.8-14, the word “permanent” refers to a time length of several tens of seconds for the insulation near a discharge inception point to be broken. In the state shown in Fig. I.8-14, electric power can hardly be supplied to the load side.

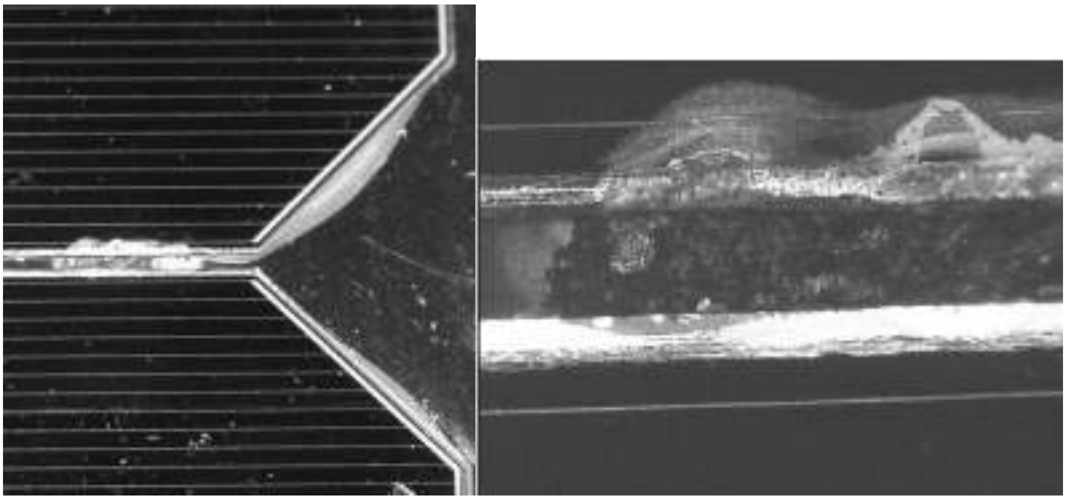

Fig. I.8-14 A part of a solar panel damaged by the permanent-sustained arc (left) and a close-up picture of the damage portion (60 magnifications). The distance between solar cells is about 600 μm.

Appendix-II Charging analysis program

The following shows computer programs used for the analysis of surface charging. They are designed so that they can deal with the interaction between spacecraft surfaces and chaged particles as well as three-dimensional shapes of a spacecraft. The most popular charging analysis program widely used in the world in the early 2000s was NASCAP/GEO jointly developed by NASA and the United States Air Force. Its commercial version is now available. The program was updated to NASCAP-2K. However, it is difficult to use this program outside the U.S.A. Thus, MUSCAT and SPIS were developed in Japan and in Europe, respectively as a next-generation chafing analysis program.

Spacecraft surface charging analysis is based on the calculation of the current that inflows into each part of the surface of a spacecraft by tracing the trajectory of each charged particles in consideration of the interaction between particles and the surface, such as emission of secondary electrons or photoelectrons from the surface. The electric charge distribution on the surface is updated on the basis of the current density on each part of the surface and the resistance between the surface and the main body of the spacecraft. The surface electrical potential is also updated according to Poisson equation and Laplace equation. Then, particle trajectory is calculated in consideration of the updated electrical potential distribution. Then a series of procedure is repeated for a prescribed length of time or until a predetermined steady state is reached. The key points in charging analysis programs are as follows:

・They can simulate the three-dimensional shape of a spacecraft.

・They can precisely calculate surface physical processes such as secondary electron emission, photoelectron emission, and leakage current flow from insulators.

・They have a database related to the physical properties of various space surface materials

The following shows the outline of typical charging analysis programs:

NASCAP-GEO [[69]]:

This program is used to calculate the charging on the surface of a satellite due to magnetospheric plasma. In this program, the primary arc inception mechanism in the inverted potential gradient state is simulated on the assumption that ions and electrons are distributed according to the double Maxwellian distribution. The total current can be calculated as the sum of each current component that flows into each part of the three-dimensionally modeled satellite. The change in electrical potential on each surface is calculated by these current components. The calculation of the current and electrons are continued until an equilibrium state is obtained.

NASCAP-LEO [[70]]:

This program was developed to clarify the interaction of substances having a low-temperature, high-density plasma and high-voltage, which is characteristic of the LEO environment. Just like NASCAP/GEO, the program analytically deals with the total current as the sum of each component current. The difference is that NASCAP-LEO can consider the effect of the sheath covering the target satellite in its process of current collection. The program is mainly used for the analysis of the power leakage from the exposed electrode sections of a high-voltage solar panel. This program provides more complex satellite simulation models than NASCAP/GEO: It uses the finite element method for electrical potential calculation.

NASCAP-2K [[71]]:

This program was developed as a successor of NASCAP. It is extremely difficult to use the program outside the U.S.A. This program provides drastically improved satellite simulation models than the previous version and can deal with LEO, PEO, and GEO.

SPIS [[72]]:

This program is based on the three-dimensional Particle-in-Cell (PIC) method, which can deal with the current collected by a precisely simulated spacecraft model and the structure of the surrounding sheath. It can also deal with surface interaction such as secondary electron emission and photoelectron emission. The source code can be downloaded free of charge from www.spis.org. However, it is supported in a limited way only by authorized members on the mailing list.

MUSCAT [73]:

This program is three-dimensional particle calculation code applicable to LEO, PEO, and GEO. The algorism is based on a combination of the PIC method and the particle tracing method. To accelerate the speed of calculation, this program is prepared for parallel calculation. It includes a GUI programs based on JAVA3D so that a satellite can be simulated into a three-dimensional model and the calculation results cane visualized. This program can consider surface physical processes such as secondary electron emission, photoelectron emission, and leakage current flow; it has a database related to physical properties of materials. The commercial version of this program is now available.

Appendix-III List of ECSS-related references

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