ELECTROSTATIC DISCHARGES & SPACECRAFT ANOMALIES

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Abstract

Electrostatics is one of major concerns of spacecraft technology. Space is filled with hot and low-density plasma building up high differential voltages resulting in electrostatic discharges, sometimes causing an anomalous behavior of spacecraft electronics. Typical anomalies will be reviewed, it will be shown how they can be recognized as consequences of environmental induced discharges. The distribution of charges over different parts of the spacecraft is depending of ambient conditions: plasma temperature and density and sun illumination. Surface and internal charging are possible; they result in dielectric or metal arcing discharges. Understanding of charging phenomena allows us to define mitigation techniques by controlling charging, electromagnetic interference propagation and electronics immunity.

1. SPACECRAFT CHARGING CONCERNS

1.1. What is spacecraft charging?

Spacecraft charging is defined as ***"those phenomena associated with the buildup of charge"*** [1]**.** The present document is focusing on detrimental effects of spacecraft charging and resulting environmental induced discharges.

In 1957 when the first satellite, Spoutnik–1, was launched, charging of bodies in space began to be studied [2]. Later on, the measurement of high-voltage charging on the satellite ATS-5 in 1969 has shown that hazardous voltages could buildup on geostationary satellites [3]. The risk was proven by the loss of the DSCS–9431 satellite on June 2, 1973 on a power failure consecutive to an electrostatic discharge [4].

It is helpful to differentiate surface and internal charging. ***Surface charging is defined as charging on areas that can be seen and touched on the outside of the spacecraft*** [5]. It consists in charging at the surface of conductive materials or very close the surface of dielectrics. ***Surface discharges occur near the outer surface and must be coupled to an interior victim*** [5]. ***Internal charging is a consequence of energetic electrons; they can penetrate the spacecraft enclosure and deposit charge very close a victim site*** [5]. They are responsible for deep charging inside dielectrics; they are also cause of surface charging on interior areas. At the altitude of geostationary satellites, energetic-electrons (100 keV to 10 MeV), able to penetrate more than 0.1 µm of matter have a current density of the order of 1 $pA.cm^{-2}$ [5].

All regions of space are not equivalent about charging. Electrostatic discharges are more probable on the synchronous orbit of telecommunication satellites (surface and internal charging), on the mid-altitude orbit of navigation satellites (mainly internal charging) and, to some extent, on the low-altitude, high-inclination orbit of observation satellites (only surface charging).

1.2. In-flight anomalies

1.2.1. What is called "anomaly"?

The word "anomaly" refers to an abnormal configuration of the satellite noticeable by the customer. An example is a temporary outage of the mission caused by a spurious switch-off of a payload. In the worst case, charging is at the origin of a chain of consequences leading to the loss of the mission. In rare cases, electrostatic discharges cause a device failure with loss of function. In other cases, there is no consequence on the mission, for instance a platform working state change by anomalous toggling towards redundant electronics. At the most benign level, a temporary erroneous telemetry status with automatic retrieval is not considered as an anomaly, however it is an event interesting to be studied by design engineers.

In [6] we can find an attempt for an objective definition of the performance during an arc discharge to be used in a Spacecraft Specification. ***"Space vehicle electrical subsystem and system outage shall be permissible during an arc discharge if operation and performance returns to specified levels within a telemetry frame period after onset of the discharge or within some other period as defined by the customer".***

1.2.2. Review of usual problems

It seems impossible to build reliable and exhaustive databases of anomalies [7]. However information released to the public allows us to list generic cases. Exterior units and cable-coupled units to exterior devices are more frequently subjected to charging effects. "Exterior" means in free space or separated from free space by only thin thermal-protection layers. They do not prevent penetration of energetic particles responsible for internal charging. They do not screen either electromagnetic interference.

The attitude and orbit control subsystem is victim of charging at both ends of the chain, sensors and actuators. An infrared sensor allows determination of attitude by scanning the Earth's profile. The active device is highly sensitive and exposed to space. Different modes of susceptibility are experienced. Energetic protons from solar flares generate a shot effect noise current by increase of conductivity when the probe is a high impedance pyroelectric crystal. The high-gain, high impedance amplifier can be sensitive to slowly varying electrostatic field by electrical induction. In case of arcing, the amplifier first stage can become saturated by a voltage pulse. In many cases, the earth-sensor is made of a rotating or oscillating mirror where grounding is impossible. This can be the cause of discharges. Earth-sensor exterior-cables pick-up electromagnetic transients and couple them to the attitude control unit.

The thrusters and conductive pipes also pick-up external discharges and couple them onto interior cables. The multilayer thermal insulation of pipes can be a source of discharge when metal layers are not grounded. At times, we observed discharges triggered by thruster firings; the discharge current flows in pipes and radiates electromagnetic transients onto the power lines and thermistor cables ending to thrusters.

When an increase of solar wind pressure is compressing the magnetosphere, moving the magnetopause below the synchronous orbit, the spacecraft is immerged in the interplanetary magnetic field in place of the terrestrial field with a possible reversal of direction. On several spacecraft, the attitude control subsystem makes use of magneto-torquers. The field reversal causes an opposite and anomalous sense of actuation of the attitude control loop.

On high voltage solar arrays it was reported [8] that under specific circumstances, an electrostatic discharge could trigger an arc supplied by the solar array power, leading to a permanent short-circuit of power lines. The solar array driving equipment can be susceptible to electrostatic discharges resulting in angular errors or mode changes. The umbilical connector is an input path to discharge transients if it is let uncapped after launch.

Numerous anomalies have been pointed out in communication payloads. Repeaters exhibit spurious switch-off of power conditioning units. The starting sequence of a travelling wave tube generally takes several minutes; the resulting outage duration is not acceptable for the customer. Receivers gain changes are experienced which can be as well false commands as erroneous telemetry status. Input stages of EHF receivers are sensitive to electrostatic discharges; some cases of permanent failures or degradation have been reported.

1.3. Analysis methods

1.3.1. General sequence of events

Thirty years after the first evidence of in-flight electrostatic discharges on synchronous spacecraft, they are still a threat. Though physical processes are understood and mitigation techniques known, we are still unable to assess before launch that the satellite will be totally free of charging anomalies. Spacecraft charging is affair of details. In-flight experience and analysis of anomalies will be always necessary for improving design guidelines and standards.

The spacecraft anomaly is at the end of a long chain of causes and consequences. Some regions of space have a radiation and particle content able to build up absolute and differential potentials at the surface or inside the spacecraft up to exceeding the breakdown voltage. Charges are released that induce electromagnetic fields in coupling current and voltage transients to cables. The pulses penetrates into boxes and propagate along printed circuit board tracks, reaching active devices, toggling flip-flops, saturating amplifiers, or fusing lanes inside integrated circuits.

How to know what happens in flight? A Ground Control Center dedicated to a Space System is monitoring for the nominal configuration of the spacecraft. An alarm or warning is triggered when the spacecraft gets out of its nominal working state. An electrostatic discharge is never observed itself but only its permanent consequences. Telemetry data is never designed for surveying unforeseen events, it is only defined for command purpose and good-health diagnosis. Probes are exceptionally implemented on commercial spacecraft to determine the state of environment at the location of the spacecraft at the time of the anomaly.

Spacecraft event understanding is the conclusion of three convergent ways of analysis: environmental data, vacuum charging tests, electromagnetic immunity tests.

1.3.2. Environmental data

When there is no on-board sensors, we rely on Space Weather plots issued by the Space Environment Center of Boulder, CO, USA (Figure 1). It is only informative since measurements are not made at the same location of the orbit as the spacecraft of concern. A proton analyzer provides proton flux data from solar flares in three spectral bands: E>10 MeV, E>50 MeV and E>100 MeV. By experience, only the upper band is related to upsets (which are not charging effects). Electron fluxes are plotted for two bands: E>600 keV and E>2 MeV. The collapse of high-energy electrons is an effect of substorm mid-energy electron fluxes. On the third panel, the H_p magnetic component at GOES location is plotted. The plot would be a clean sine curve in absence of geomagnetic activity. Magnetic noise on this curve witnesses to substorm fluxes of precipitating mid-energy particles. On the lowest panel, planetary K-indices bar-plots confirm occurrence of substorms (K_p above 3).

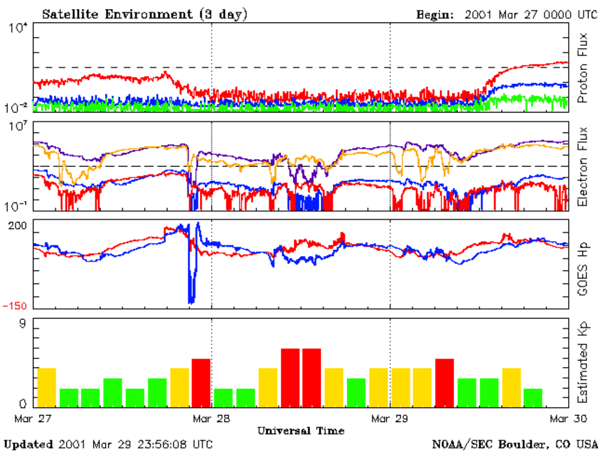

Figure 1. Example of data used for determination of solar and geomagnetic environment. Information from the Space Environment Center, Boulder, CO, National Oceanic and Atmospheric Administration (NOAA), US Dept. of Commerce.

1.3.3 Vacuum tests

Charging properties of rough materials are more or less known. Bulk and surface resistivity, secondary emission yield to electrons and ions, photoemission efficiency have to be documented during the spacecraft development for the validation of charging performance. Actually, materials are used in complex assemblies; possible interactions between different materials have an influence on the charging equilibrium potentials and discharge processes. Testing flight-representative items in a vacuum chamber in the right charging ambiance is necessary for a reliable forecasting of in-flight potentials and assessment of the discharge risk.

A set of electron guns generates an electron beam in a wide range of energy from 10 keV to 200 keV or more (Figure 2). The item under test is polarized with respect to the chamber walls for simulating the absolute voltage of the spacecraft. UV-light or a proton source simulates locally the positive charging effect of sunlight.

1.3.4. Electrical effects

The plasma created by the electrostatic discharge has primary effects, for example, triggering a cold arc discharge between cells of the solar generator [8]. However, in most cases we have only to deal with electromagnetic effects of electrostatic discharges. Two types of electromagnetic sources have been identified.

The discharge process is a transition from a charged state to a discharged one, it is seen as an electric field step transient. During the transition, a replacement current is flowing in the spacecraft frame; this is a first source for the electromagnetic pulse.

At the same time, electrons are blown off from the site of discharge, repelled by the negative potential of the surface. A typical value of the charge present at a given time in space before being recollected (or lost in space) is 30 nC. The space charge generates an electric field pulse of several tens of kilovolts per meter in the immediate vicinity of the site of discharge, inducing a common mode current in close bundles and common mode voltage pulses on individual wires.

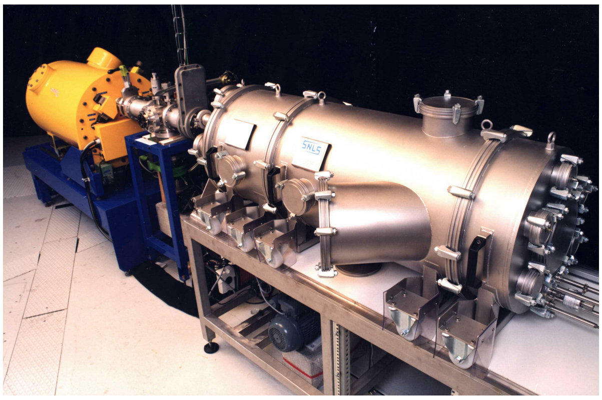

Figure 2. Wide-spectrum charging environment facility SIRENE (photo ONERA)

As electromagnetic tests performed inside a vacuum chamber are expensive and not really practical, we prefer reproducing the equivalent electromagnetic field pulse in air or equivalent voltage and current pulses on cables. Specific instrumentation and methods have been defined for this purpose.

2. CHARGING MODEL

2.1. Environment

2.1.1. Mid-energy electron fluxes

In earlier times, 90% of anomalies of spacecraft were seen in the morning quadrant of the synchronous orbit with a maximum probability around 3am in local time [9], (Figure 3). A typical case is MARECS-A experience, the first European telecommunication satellite launched in December 1981. In this region of space, which rather extends from 9pm to 9am we find the two conditions for surface charging of a spacecraft: low density of cold ions ($n<10$ cm^{-3}) and large electron fluxes of 10's keV electrons (up to 1 nA.cm^{-2} expressed in current density). They are usual substorm conditions.

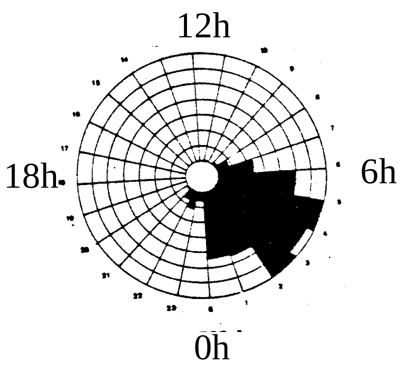

Figure 3. Distribution in local time of MARECS-A anomalies.

All surface areas do not charge with the same time-voltage slope and have not the same equilibrium limit. Differential voltages of hundred volts, maybe kilovolts, are possible, which exceeds arcing threshold. These conditions are frequently encountered and have been observed with a K_p-index as low as 3^+.

The buildup of a negative absolute voltage of the spacecraft frame is possible in this environment providing a specific potential distribution favorable to arcing as it will be exposed hereafter.

2.1.2. Electronic radiation belt

Energetic electrons with energies up to a few megaelectronvolts constitute the most important component of the electron radiation belt. In the equatorial plane the belt extends to about 60 000 km, the maximum flux occurring by 25 000 km. This is a severe internal-charging threat for navigation satellites cruising in this region of space.

At the altitude of geostationary satellites, the flux exhibits fluctuations over several decades as it can be seen on the second panel of Figure 1. In hours following an injection event an energization process occurs, increasing the flux of energetic electrons at the geostationary altitude leading to internal charging. From flight experience, the hazard threshold has been settled at 10^3 s^{-1}cm^{-2}sr^{-1} for E>2 MeV electrons, the dotted line on the second panel of Figure 1.

2.1.3. Sunlight effects

Sunlight tends to keep all illuminated surfaces near plasma potential, whereas shaded insulated surfaces may charge strongly negative [1]. Breakdowns are believed to occur at interfaces between illuminated and shaded surfaces. The analysis is performed on seen-from-the-sun spacecraft drawings (Figure 4). Some repeaters of the French telecommunication satellite TELECOM1-A, launched in August 1984, experienced frequent arcing-induced switch-off. Plotting their distribution in local time allowed us to determine they were correlated with the position of the sun direction in spacecraft axes.

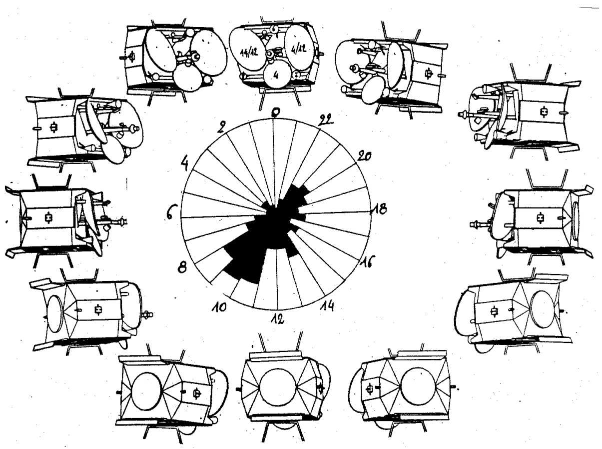

Figure 4. Telecom-1 seen from the sun (solar array not drawn), distribution of anomalies in local time.

It is suspected that charges are stored inside dielectrics or on floating conductors during substorms in the morning quadrant and released later when the electrostatic configuration makes breakdowns possible. The surface of a satellite comprises a lot of appendices, (Figure 5), allowing countless possibilities of high-voltage interfaces by differences of incident flux or illumination and shadowing.

For a synchronous spacecraft, stabilized in a geocentric referential, the diurnal and seasonal effects are consequences of the displacement of the sun axis in the satellite axes. During the 24h-rotation the earth and anti-earth faces, the east and west walls are successively illuminated. The north and south walls see the sun alternatively six months a year, at a grazing angle at equinoxes. Often anomalies are correlated with the time of the day or the day in the year. This is a presumption of surface charging.

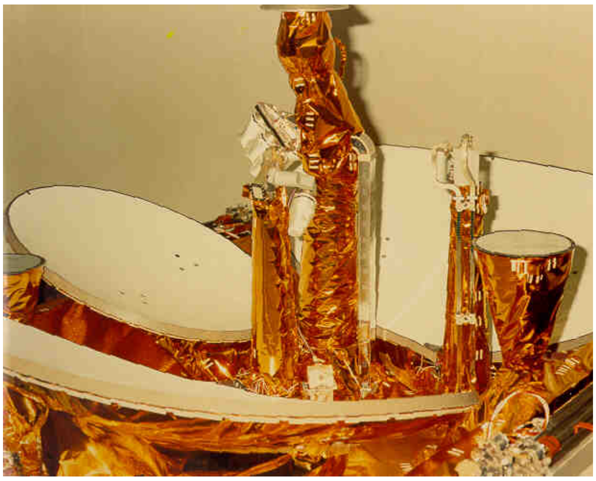

Figure 5. Shadowing permits situations of differential voltage; this view of the Earth face of Telecom-1 is showing the complexity of actual geometrical configurations.

2.2. Subsurface charging

Incident energetic (E>10keV) electrons become embedded in dielectrics. On the synchronous orbit, most of the electron flux of substorms is inside a 100 keV range, with an incident current density up to 1 nA.cm^{-2}. Charges are buried in few micrometers layer from the surface. Several possibilities are opened. They can drift towards backside, attracted by the positive electric field at the interface with the conductive substrate. They can drift towards the front side (mutual repulsion of same-sign bulk charges) where they are neutralized by the environmental particles. They can stay trapped permanently. In-flight surface voltage measurement have shown a secular increase of potential over years, reaching several (negative) kilovolts on a Teflon® sample [10]. They can also be released by electrostatic discharges after staying stored several minutes or hours.

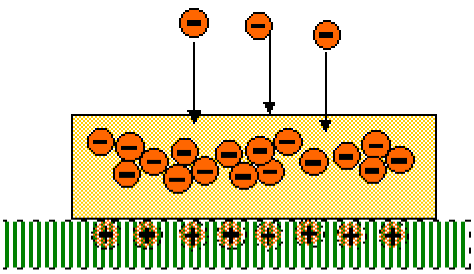

Figure 6. Illustration of dielectric bulk charging: negative charge of dielectrics and positive charge of underlying conductor at the interface

2.3. Surface charging

While energetic electrons negatively charge dielectrics by burying their negative charge below the surface, a positive charge can build up in a thin submicrometer layer nearby the surface. When impinging electrons are in a medium range of energy (Figure 7), there is more than one extracted electron for one incident electron, the corresponding ratio is the yield d. It leaves a positive charge by electron depletion. The yield of dielectrics is currently larger than 2 when the maximum yield of metal is near 1. It is what for a floating metallic component is often negative with respect to surrounding dielectrics seeing the same incident flux.

A similar phenomenon occurs with impinging positive ions, also a cause of secondary emission of electrons.

Short wavelength photons (with energy larger than 10-15 eV), X-rays or Lyman-α sun lighting, have enough photoemission efficiency to create a positive current onto the surface; near the Earth orbit the solar photoemission current density is about 1 nA.cm^{-2}.

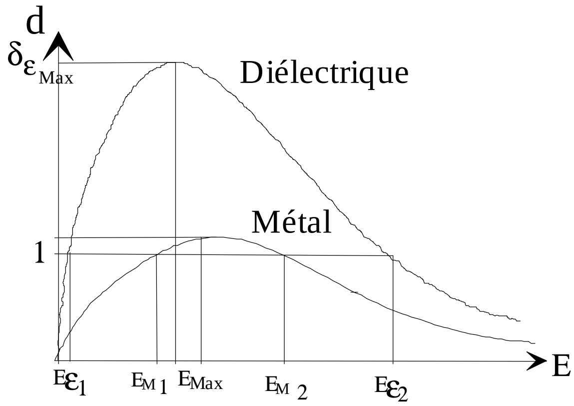

Figure 7. Secondary emission yield versus energy of impinging electrons.
Teflon®: d_{max}=3.2; E(d_{max})=300eV; E2=1850 eV

These three processes are cause of positive surface charging (Figure 8). This positive charge generates a positive differential voltage with respect to surrounding uncharged components. Depending on the global electrostatic equilibrium the local absolute voltage of each area can be positive or negative.

The combination of negative internal charging by penetrating electrons and positive surface charging by extraction of electrons generates a bilayer configuration of charges at the surface of dielectric parts, which can play a role for triggering discharges. As a matter of fact the thickness of the bilayer zone is very small, so the electric field near the surface can reach the breakdown threshold at relatively low voltages. On the contrary, the electric field can induce dielectric conductivity preventing the buildup of high voltages.

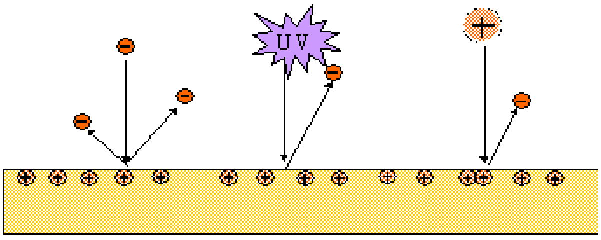

Figure 8. Three causes of positive surface charging: secondary emission to electrons, photoemission, secondary emission to ions.

2.4. Buildup of potential by induction

Consider the case of a floating metal separated from environment by a non-conductive material. A typical example of this configuration is a wire with its dielectric coating (Figure 9) or a thermal insulator sheet when the conductive rear-side is let ungrounded.

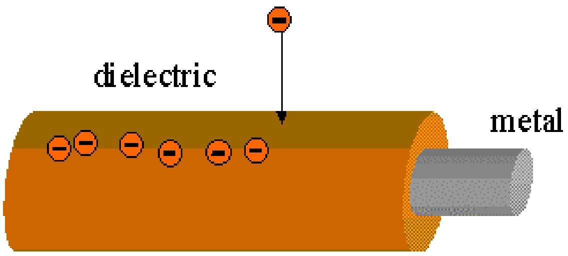

Figure 9. Charging a wire with electrons. Electrons deposit on the dielectric part. The metallic part stays globally uncharged.

Even if no electron reaches the conductor, it takes the dielectric mean voltage. Generally the insulator is not uniformly charged, one section can be outside, submitted to space environment, another one in interior side, screened by spacecraft walls. In consequence the dielectric potential is not uniform. As the metal is at the same potential everywhere, locally the voltage difference between the insulator and the conductor is either positive or negative.

3. BREAKDOWNS AND ARCING

3.1. Dielectric discharge from buried charges

The exact breakdown strength is not known, but it is considered that ***most common good quality spacecraft dielectrics may break down when their internal electric fields exceed 2.10^5 Vcm^{-1}*** [5]. This figure can be reached if the flux is exceeding 2.10^{10} e.cm^{-2} in less than 10 hours. A partial discharge is initiated and electronic stored charge is ejected to space (Figure 10).

The internal electric field is the result of equilibrium between the internal charge deposition rate and the leakage rate. From ground and flight experience, we consider there is no problem when the current density is smaller than 0.1 pA.cm^{-2}.

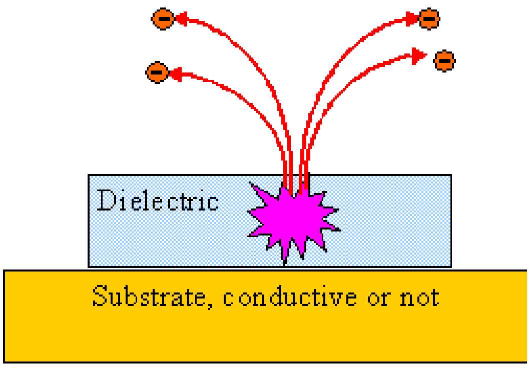

Figure 10. Illustration of buried charge breakdown

Best candidates for this mode of breakdown are dielectric composites, for example epoxyglass pieces used for structural stiffeners or electrical circuit boards.

3.2. Dielectric discharge: flash-over

Flashover discharges are currently observed in laboratory experiments on dielectric sheets with grounded conductive rear face. This dielectric discharge is propagating at a constant speed in the range 10^5 to 10^6 m.s^{-1} in every direction from the site of ignition (Figure 11). A sustained discharge needs a surface voltage of several kilovolts to get the conditions of propagation. We have to point out that there is no evidence these conditions have ever been reached in flight on large areas.

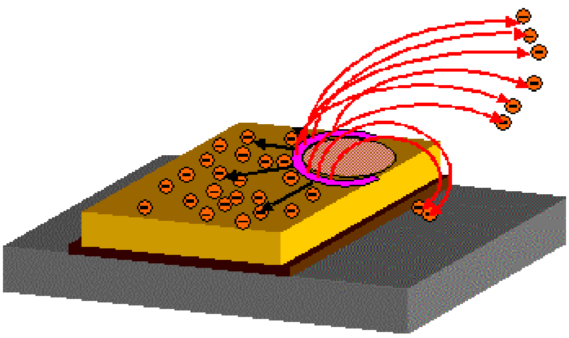

Figure 11. Illustration of the propagated flash-over discharge with blow-off

A discharge occurs if ***the interface between a dielectric and an exposed conductor has an electric field greater than 10^5 V.cm^{-1}*** [1]. The discharge over a charged polymer film can also be initiated by a punch-through breakdown from buried charges. It gives the boundary conditions: a tangential electric field and dense plasma.

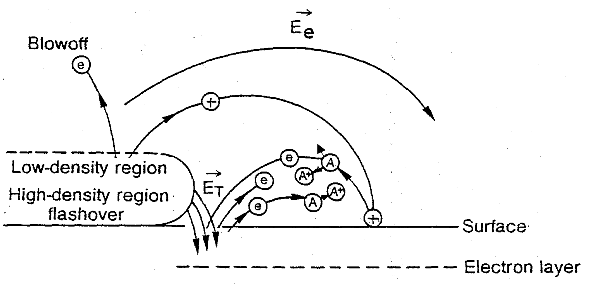

Figure 12. Tip of the flashover discharge, boundary between the charged and discharged zones (after Marque, [11])

A physical model was developed by Marque [11]. The drawing on Figure 12 represents the boundary between the charged and the discharged region. The tip of the discharge is made of plasma and neutrals. The electric field has a dipolar configuration, lines are perpendicular near the charged surface, but curved to tangential direction at the boundary between the charged zone and the previously neutralized region. Along lines of the electric field, electrons are repelled to space, to near-by surfaces or to the initiation site in the surface conductive gas layer created by the discharge. Meanwhile, positive ions are driven in the opposite direction and impinge the charged surface, freeing neutrals by a desorption process. If the conditions of density and field are adequate, previously freed electrons ionize atoms in the manner of a Townsend discharge, providing free ions and electrons. The basis of the electrical model is the existence of a low-pressure self-maintained glow discharge in a thin layer above the surface.

The discharge current is sized by the width of the front of discharge. Depending on materials, it is in the range 100 to 1000 $A.m^{-1}$, which is considerable. Half is a blown-off current component ejected to space.

3.3. Metallic discharge: grounded conductor

A criterion of metallic discharges is: ***discharge can occur if dielectric surface voltage are greater than 500 V positive relative to an adjacent exposed conductor*** [1]. If a conductor is at the frame potential, this condition can only be attained when the spacecraft ground is negative with respect to space.

When the voltage of a conductor exceeds some hundred of volts, if the curvature radius is small enough, the electric field on the tip is sufficient to generate field emission (also called *cold emission* or *Fowler-Nordheim emission*). The electron is driven towards the positive dielectric area with energy near the maximum secondary emission yield (Figure 13). If the general configuration of the electric field provides lines to infinite, secondary electrons are blown off. For a yield larger than 1, the dielectric region is let more positively charged than before, which increases the electric field and field emission capability. So, the process avalanches only limited by the fusion of the tip, when heated by the increasing current density.

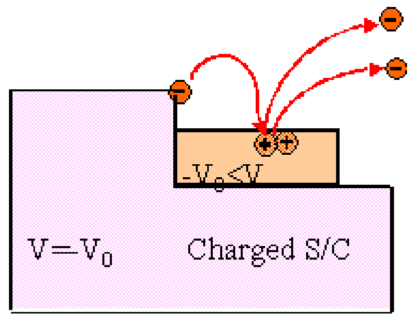

Figure 13. Metallic discharge from a grounded conductor

The discharge interrupts itself when there is no more available charge on the conductor. In the case of a grounded conductor, the charge released is the total absolute spacecraft charge.

A typical current measured at laboratory is in the 10 mA-1 A range. The discharge of a spacecraft frame can takes several microsecondes.

3.4. Metallic discharge: floating conductor

The field configuration is identical but the spacecraft potential may be zero. Every time a metallic component is let floating, differential secondary emission yields render it negative with respect to surrounding dielectrics. If the differential voltage reaches the hazard threshold of about 500 V, arcing becomes possible (Figure 14) with the same process as for grounded conductors.

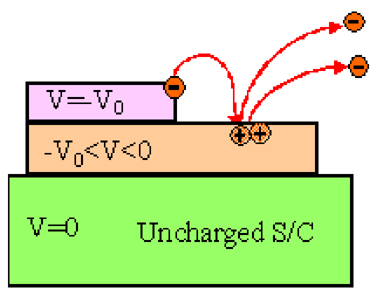

Figure 14. Metallic discharge from a floating conductor

This process is known as the "inverted gradient discharge" since the metal is negative with respect to dielectric in opposition with the "normal" gradient where the dielectric surface is negative with respect to the spacecraft frame.

3.5. Typical cases

3.5.1. Second surface mirror (SSM)

Flexible second surface mirror are rather used for thermal control purposes than rigid optical solar reflectors made of quartz or glass on curved surfaces or on small areas. The polymer film is made of Teflon® with vacuum deposited aluminum (VDA) or silver on the backside (Figure 15). Pinholes to avoid air bubbles trapping. The hole is pierced from the backside causing the presence of sharp transverse tips.

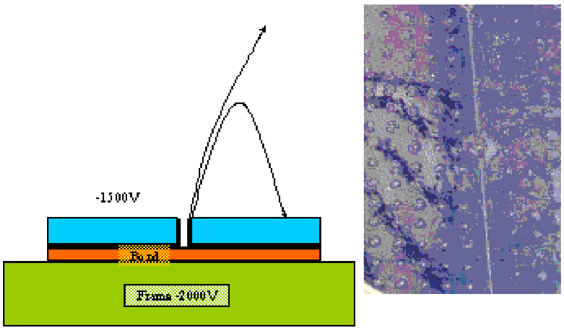

Figure 15. Pin-holed Second Surface Mirror (SSM)

When the positive voltage difference reaches about 500 V, metallic discharges occur. This material is also a source of dielectric discharges when the negative surface voltage is larger than 10 kilovolts.

3.5.2. Optical solar reflector (OSR)

Optical solar reflectors are widely used on North and South walls of geostationary spacecraft. The back side is made of silver and Inconel®, a nickel alloy (Figure 16). It is bonded with non-conductive glue on the spacecraft structure letting the conductive backside floating.

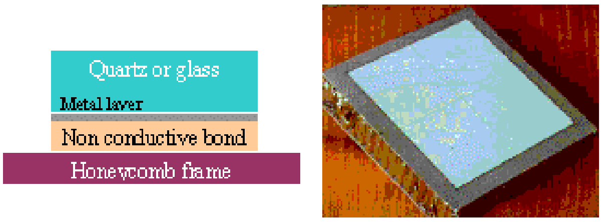

Figure 16. Quartz or glass OSR

In laboratory tests we never observed metallic discharges, at any positive voltage. In the negative voltage configuration flashover discharges are only possible with monoenergetic electrons in the range 10-50 keV. When using a space representative broadband spectrum for electrons the induced conductivity of glass or quartz holds the surface voltage under 1 kV, less than the voltage required for a dielectric discharge.

3.5.3. Solar array

The electrical and geometrical configuration is quite similar to OSR. The difference is the presence of leads between cells in the gaps (Figure 17). In laboratory, metal discharges from sharp edges of the leads occur when the spacecraft ground is about 1 kV more negative than the coverglass surface.

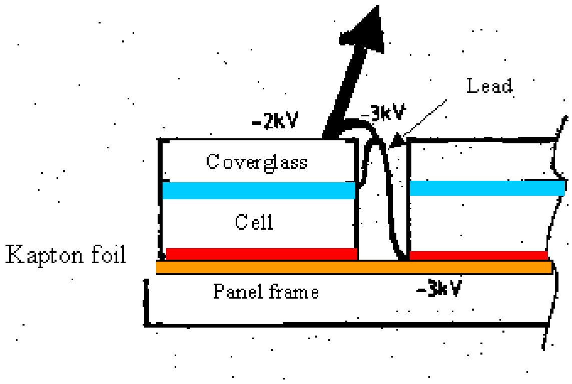

Figure 17. Solar cell discharge from an interconnection lead.

3.5.4. Floating wire discharge

Floating wire discharges are a main threat for spacecraft immunity since wires are tightly coupled. Consider the case of an unused wire let ungrounded, partly outside, partly inside the spacecraft frame. This configuration is not unusual and may be encountered on EED (electro-explosive device) firing lines after use (Figure 18).

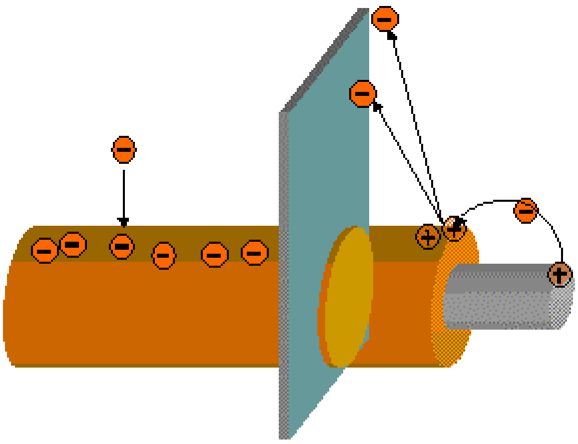

Figure 18. Floating wire discharge

Outside, the insulator is trapping electrons, metal voltage becomes negative but remains uncharged. Inside, the insulator is not charged. As metal is negative the inverted gradient condition is fulfilled, arcing can occur as soon a threshold is reached.

3.5.5. Multilayer thermal blankets

Multilayer blankets are made of several layers of both-sided VDA Mylar®, and one outside layer of one-sided VDA Kapton®. The edge of the blanket is covered with a bonded VDA Kapton® tape to prevent solar rays from penetrating between the layers. An epoxy glass stiffener is sometimes added to hold the blanket (Figure 19).

All aluminum layers have to be grounded, including the edge tape. If a thermal insulator VDA film is used to cover the stiffener, it must be grounded notwithstanding its small size. As a result of flight experience we consider there is no lower limit for letting VDA-films ungrounded.

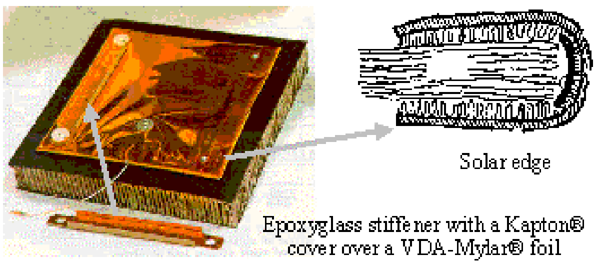

Figure 19. Edges of multilayer blanket and stiffener

4. MITIGATION TECHNIQUES

A reasonable and achievable goal would be one electrostatic event per satellite, per year. An event does not necessarily mean loss, degradation or break of the mission. An event is any deviation from the nominal configuration or nominal telemetry.

Mitigation techniques lay on following three methods by increasing order of importance.

4.1. Controlling build-up of charge

4.1.1. Numerical simulation versus tests

From material properties and a model of environment, it is possible to calculate the electrostatic equilibrium of the spacecraft and the voltage reached by any surface or inside dielectrics. In these areas, well-known programs are respectively NASCAP (for **NAS**A **C**harging **A**nalyzer **P**rogram) and ESADDC (**ESA** **D**eep **D**ielectric **C**harging).

NASCAP provides a good understanding of the charge of the spacecraft as a whole. At the needed level of detail for assessing the charging risk and, finally, for selecting safe materials, the actual charge voltage, the actual breakdown threshold and discharge figures can only be obtained from tests.

4.1.2. Proper choice of dielectric materials

Surface materials are first chosen for their thermo-optical properties and their stability against degradation in space conditions. An upper limit of resistivity may be specified, figures of 10^9 Ω.cm from the surface to the structure [1] or 10^{12} Ω per square for interior dielectrics [5] are mentioned.

Leaky dielectrics are helpful in two manners. In the volume, they prevent buried charges to accumulate and reach dielectric breakdown threshold. On the surface they limit the negative absolute voltage of the spacecraft in the flux of substorms. This allows to keep the positive dielectric-to-metal voltage at a low level, below the arcing threshold.

Practically, we prefer defining a list of forbidden materials to specifying a hard-to-measure bulk or surface conductivity.

4.1.3. Grounding

The rule is simply expressed: ***All conducting elements, surface and interior, should be tied to a common electrical ground, either directly or through a charge bleed-off resistor*** [1]**.**

The DC resistance between any two points should be less than 0.1 Ω [6] and 10 Ω for all thin conducting surfaces on dielectric materials To prevent fault currents flowing in the spacecraft frame and shorts between the power terminals, we sometimes need insulation of structural panels. It is actually achieved for solar panels. Letting them floating is forbidden, so a bleed-off resistor between the solar array and the spacecraft frame is inserted, a 50 kΩ value is suggested. Short-circuits or electrostatic discharges can induce voltage spikes across this ground separation resistor, an EMC analysis is needed.

The question arises about the minimum size of the grounded conductive item. There is no absolute low limit, values as low as 0.3 cm^2 are mentioned [5]. The key parameter is the distance between the floating item and the susceptible circuit electromagnetic input point. A preferred method consists in measuring discharge properties in a vacuum chamber and reproducing the electromagnetic environment for an immunity test.

No wires will be let floating. An electrostatic discharge from a wire may have severe consequences since the discharge wave is tightly coupled to other cables of the same bundle. This is a list of wires let floating by accident:

- modification of cabling during the development and unused wires are let in place,
- a section of cabling between two open contacts,
- EED lines after ignition,
- wires for ground testing.

Sometimes grounding is impossible, for instance grounding of rotating devices. Letting some items floating is workable if qualified through tests.

Grounding of conducting elements does not prevent metallic discharges but significantly minimizes their number. The inverted voltage gradient configuration remains possible when the spacecraft is absolutely charged.

4.2. Electromagnetic shielding

As expressed in [1]: ***The primary spacecraft structure, electronic component enclosures, and electrical cable shields shall provide a physically and electrically continuous surface around all electronics and wiring***.

4.2.1. Faraday cage

The purpose of this Faraday cage is to shield electronics from the radiated noise of discharges, and to keep space plasma outside, preventing the creation of sites of discharges in the immediate vicinity of circuits and their cabling.

The multilayer blankets used for thermal control are too thin for preventing entry of energetic electrons in the interior of the spacecraft.

The aluminum film deposited on polymer films is thinner than the skin depth and does not provide screening effectiveness. A 0.1 mm thick foil of aluminum has shielding effectiveness beginning at 1 MHz, it provides electromagnetic screening.

The total thickness of matter of the multilayer insulation is between 0.1 and 0.2 mm, added to the radiofrequency shield and seems enough on synchronous orbits to limit the high-energy electron flux liable for internal discharges.

4.2.2. Bundle shielding

Exterior cables, outside the enclosure of the spacecraft and outside the radiofrequency shield cannot be avoided, to the solar array, to antennas, or to external electroexplosive devices. They have to be shield. The shield has to be grounded on the frame at the entry point because the shield itself can pick-up the radiofrequency noise from the discharge and propagate it in the interior along internal bundles. If bonding at this place is not practicable external cables will be overshielded with a screen grounded circumferentially at the entry point.

4.3. Tests

4.3.1. Immunity tests at unit level

Because we are never sure to totally suppress breakdowns, a verification of immunity to their effects is at the first level of good engineering practices. It should be required by test for all units comprising devices or cables unscreened with respect to space.

It can be executed by a coupled method using a wire adjacent to the harness (Figure 20). This test has been designed to replace the radiated field test from a sparking device and the conducted test by discharging an arc onto the structure, tests defined by the old MIL-STD-1541 standard. It is described in [12].

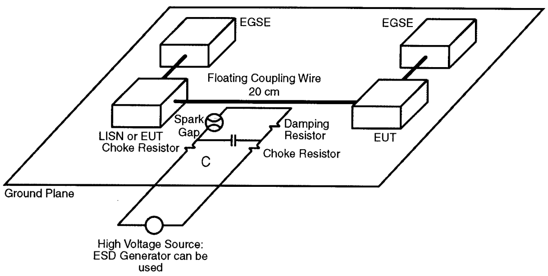

Figure 20. Recommended spacecraft charging ESD immunity test at unit level

The following list contains specified current parameters:

a) Spark gap: typical value is 6 kilovolts. Hermetically sealed, pressurized envelope overvoltage spark gap with fast breakdown time is preferred. An air gap should not be used, as discharge characteristics would be dependent on atmospheric conditions.

b) C (capacitance): typical value is 100 pF, high-voltage capacitor with low inductance.

c) Damping resistor: typical value is 47 Ω, may be adjusted at critical damping depending on value of capacitance C and self-inductance of the discharge circuit.

d) Choke resistor: used to prevent high-frequency component of discharge from flowing in uncontrolled paths. The minimum value is 10 kΩ. With this precaution, the discharge parameters are not dependent on length and position of high-voltage source wires.

e) High-voltage source: could be a dc source, in this case a choke resistor of more than 10 MΩ is used. However, for safety reasons, an ESD generator as in IEC 61000-4-2 is preferred. It will be used in air discharge mode but with permanent connection of the discharge tip to one of the choke resistors and the discharge return connection being connected to the second choke resistor.

f) Discharge circuit: floating and tightly coupled 20 cm along the harness of the EUT (Equipment under test).

g) Transient current pulse: a goal is 30 A peak, 30 ns duration at mid-height (Figure 21).

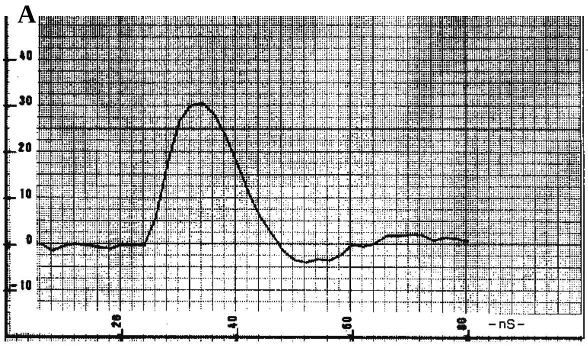

Figure 21. Pulse shape of the primary current, X-axis 0 to 100 ns, mid-height duration 16 ns Y-axis -10 to 50 A, pulse 35 A_{pp}

As well as for in-flight discharges, ESD testing can cause catastrophic failure (more insidiously, latent failures) of test article, for that reason verification is generally performed on engineering or prototype models, not the flight article.

4.3.2. Immunity tests at system level

The rationale for doing immunity test at system level would be a best confidence in test results when tests are made on a flight model with actual units and cabling. The earlier system-level test was defined by the MIL-STD 1541 standard. It consists in injection of a current pulse onto the frame from a high-voltage sparker. Frequently, spacecraft "qualified" with this method have experienced in-flight ESD anomalies so we feel the need for a new-generation immunity test at system level.

Electrostatic discharges have a short influence radius rapidly decreasing with distance, but with very high amplitude in the sphere of influence. Applying test pulses at system level with qualifying amplitude should be done from hundred of injection points, which is not practicable. Performing the test with a limited number of injection points needs a large amplitude, which means overtesting and an enlarged risk of failure. At this day, a confident system-level test has still to be defined.

4.3.3. Inspection procedures

All ground ties shall be inspected, dc resistances shall be tested before the delivery of the spacecraft. A visual inspection will ensure that no conductive part has been forgotten and let floating and that no forbidden material has been used.

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