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National Aeronautics and
Space Administration

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AVOIDING PROBLEMS CAUSED BY SPACECRAFT ON-ORBIT INTERNAL CHARGING EFFECTS

NASA TECHNICAL HANDBOOK

FOREWORD

This handbook is approved for use by NASA Headquarters and all NASA Centers and is intended to provide a common framework for consistent practices across NASA programs. Its contents are equally applicable to any spacecraft. The handbook was developed to address the growing concerns associated with the in-flight buildup of charge on internal spacecraft components due to space plasmas with high energy electrons.

Spacecraft charging, defined as the buildup of charge in and on spacecraft materials, is a significant phenomenon for spacecraft in certain Earth and other planetary environments. Design for control and mitigation of surface charging, the buildup of charge on the exterior surfaces of a spacecraft due to space plasmas, is treated in detail in NASA TP2361, “Design Guidelines for Assessing and Controlling Spacecraft Charging Effects” (September 1984). This handbook details some methods and techniques to mitigate internal charging. It is written as a companion document to NASA TP2361 and should be used in that context.

Although many of the ideas presented here have a long heritage, this document collects them in one convenient place and quantifies and illustrates the design guidelines necessary to reduce the effects of internal charging.

This handbook is intended to be an engineering tool, written for use by engineers. Much of the environmental data and material response information has been adapted from published and unpublished scientific literature for the purpose of this document. The authors would like to acknowledge the long history of scientific research from which this material has been obtained. The authors emphasize that the contents are a best effort at this time and may be subject to revision as more research is done.

Requests for information, corrections, or additions to this handbook should be directed to the Reliability Engineering Office, Mail Code 301-466, Jet Propulsion Laboratory, 4800 Oak Grove Dr., Pasadena, CA 91109. Requests for additional copies of this handbook should be sent to NASA Engineering Standards, EL01, MSFC, AL, 35812 (telephone 256-544-2448). This and other NASA standards may be viewed and downloaded, free-of-charge, from our NASA Standards Homepage: http://standards.nasa.gov.

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Chief Engineer

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AVOIDING PROBLEMS CAUSED BY SPACECRAFT ON-ORBIT INTERNAL CHARGING EFFECTS

1. SCOPE

1.1 Scope. This document establishes guidelines and design practices to be used by NASA and other spacecraft designers to minimize the detrimental effects of spacecraft internal charging. These effects are due to interactions between the in-flight plasma environment and spacecraft electronic systems. The detrimental effects are disruption or damage from electrostatic discharges (ESD's) at interior regions of the spacecraft that may be charged by space plasmas. The handbook applies to any spacecraft whose circular Earth orbit is described by the labeled regions of Figure 1, as well as other energetic plasma environments such as at Jupiter. Designs for spacecraft with orbits in these regions should be examined for the threat from internal charging. This document does not train the reader to do the necessary design work; program managers must use knowledgeable ESD experts for design evaluations.

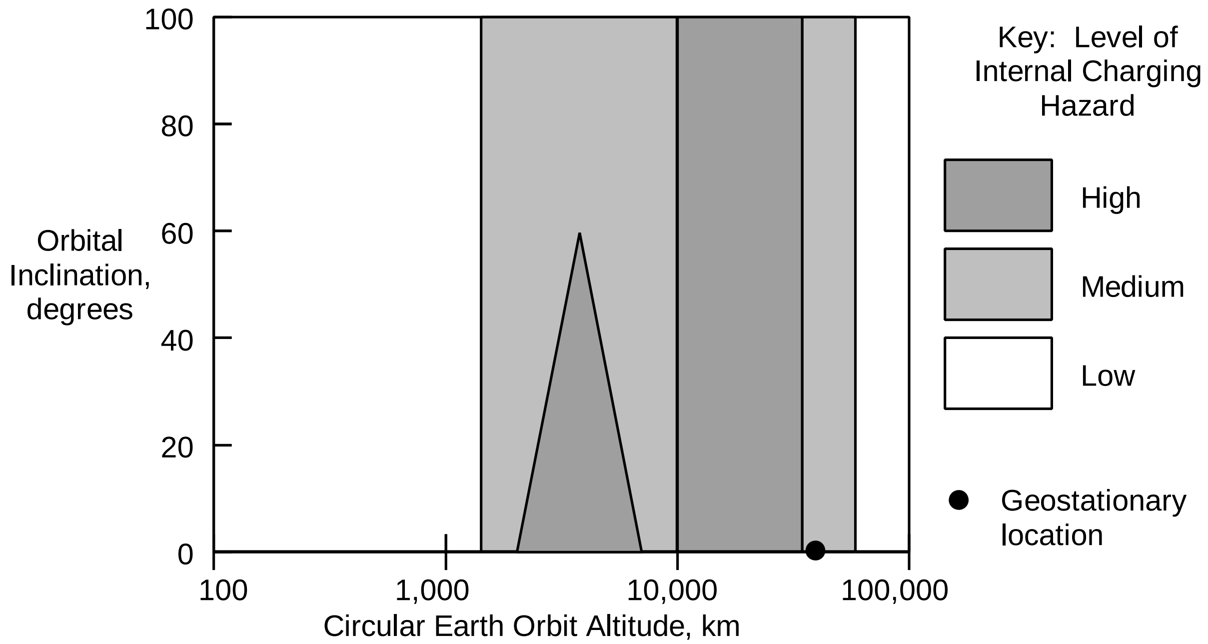


FIGURE 1. Earth Regimes of Concern for On-Orbit Internal Charging Hazards, for Spacecraft With Circular Orbits (simple chart, but relatively accurate)

In this handbook, the distinction between "surface charging" and "internal charging" is that internal charging is caused by energetic penetrating particles that can penetrate and deposit charge very close to a victim site. Surface charging is on areas that can be seen and touched on the outside of a spacecraft. Surface discharges occur on or near the outer surface of a spacecraft and discharges must be coupled to an interior victim. Discharge energy from surface arcs is attenuated by the coupling factors, and therefore is less threat to internal electronics. External wiring and antenna feeds of course are susceptible to this threat. Internal charging, by contrast, may cause a discharge directly to a victim pin or wire with very little attenuation. Geosynchronous/geostationary orbit (a circular orbit in the equatorial plane of the Earth at ~35,063 km altitude) is perhaps the most common example of a region where spacecraft are affected by internal ESD's, but the same problem can occur at lower Earth altitudes, polar orbits, and at Jupiter.

This handbook details the methods and designs necessary to mitigate on-orbit internal charging problems. Control and mitigation of surface charging is treated in NASA TP2361, "Design Guidelines for Assessing and Controlling Spacecraft Charging Effects" [Purvis et al., 1984].

1.2 Purpose. The purpose of this handbook is two-fold. First, it is intended to serve as a single reference source that contains suggested detailed spacecraft design requirements and procedures to minimize the effects of spacecraft internal charging and to limit the effects of the resulting ESD. Secondly, it contains supplementary material and references to aid understanding and assessing the magnitude of the phenomenon.

1.3 Applicability. This handbook recommends engineering practices for NASA and other space programs and projects. Determining the suitability of this handbook and its provisions is the responsibility of program or project management and the performing organization. Individual provisions of this handbook may be tailored to meet specific program/project needs and constraints.

2. APPLICABLE DOCUMENTS

2.1 General. The applicable documents cited in this handbook are for reference only.

2.2 Government documents.

2.2.1 Specifications, standards and handbooks. The following specifications, standards and handbooks form a part of this document to the extent specified herein. Unless otherwise specified, the issuances in effect on date of invitation for bids or request for proposals shall apply.

DEPARTMENT OF DEFENSE

MIL-STD-461	-	*Requirements for the Control of Electromagnetic Interference Emissions and Susceptibility*. Generally, protection for ESD depends on a good EMC foundation built into the spacecraft.
MIL-STD-883	-	*Test Methods and Procedures for Microelectronics - Method 3015, Electrostatic Discharge Sensitivity Classification*. V-zap tests for measuring electrostatic discharge response of electronic parts to human body ESD source model.
MIL-STD-1686	-	*Electrostatic Discharge Control Program for Protection of Electrical and Electronic Parts, Assemblies, and Equipment (Excluding Electrically Initiated Explosive Devices)*

2.2.2 Technical References and Bibliography. See Appendix G.

2.3 Order of precedence. Not applicable to this handbook.

3. ACRONYMS

Acronyms used in this handbook.

Ω	ohm
A	Ampere
AE8	NASA Space Radiation Model for Trapped Electrons
AP8	NASA Space Radiation Model for Trapped Protons
CRRES	Combined Release and Radiation Effects Satellite
dc	direct current (zero frequency)
e	electron
EMC	electromagnetic compatibility
EMI	electromagnetic interference
ESD	electrostatic discharge
eV	electron volt (unit of energy)
F	farad (measure of electrical capacitance)
GEO	geosynchronous Earth orbit (about 35,800 km altitude, 24-hr. period)
GOES	Geosynchronous Operational Environmental Satellites
H-field	magnetic field (casual usage)
i	electrons or current per square centimeter to a flat surface
IC	integrated circuit
IESD	Internal Electrostatic Discharge
j	normalized electron flux, e/cm^2-s-sr (see "J")
J	omnidirectional electron flux, e/cm^2-s (J = 4pi*j)
JPL	Jet Propulsion Laboratory
keV	kilo electron Volt
LNA	low noise amplifier
LEO	low Earth orbit (about 200-2,000 km altitude)
kΩ	kilohm
m	meter
MΩ	megohm
mA	milliampere
MeV	million electron Volt
MEO	medium Earth orbit (about 2,000-25,000 km altitude)
MHz	megahertz
MIL	military
Molniya	an elliptical orbit with apogee ~~40,000 km, 12-hr period, and ~63 degree inclination
N/A	not applicable
nA	nano Ampere
NOAA	National Oceanic and Atmospheric Administration
ns	nanosecond

PEO	polar Earth orbit (roughly 80+ degree inclination, 700-1000 km altitude)
pA	pico Ampere
pF	picofarad
RF	radio frequency
s	siemens (reciprocal of conductivity)
SEU	single event upset
SOPA	Synchronous Orbit Particle Analyzer (Los Alamos)
STD	standard
sr	steradian
T	temperature
v	velocity
V	volt
W	watt
WDC	World Data Center (of NOAA)
<	less than
>	greater than
µF	microfarad

4. INTRODUCTION AND BACKGROUND TO SPACE PLASMA CHARGING

(Persons wanting design guidelines only, go directly to sections 5.2.)

4.1 Physical concepts. The fundamental physical concepts that account for space charging are described in this section.

4.2 Plasma. A plasma is an ionized gas - the atoms and molecules that make up the gas have some or all their electrons stripped off leaving a mixture of ions and electrons. The simplest ion, a proton (corresponding to ionized hydrogen), is generally the most abundant ion in the environments considered here. The energy of the plasma, of its electrons and ions, is often described in units of electron volts (eV). This is the kinetic energy that is given to the electron or ion if it is accelerated by an electric potential of that many volts in the Earth's rest frame. Temperature, T, is another descriptor used by plasma physicists to describe the energy of a group of particles. For electrons, Te = eV*11,604 (4,300 eV is equivalent to 50 million degrees Kelvin).

The mechanical energy of a particle is given by the equation $E = 1/2 \times mv^2$, where E is the energy, m is the mass of the particle, and v is the velocity of the particle. Because of the difference in mass (~1:1836 for electrons to protons), electrons in an equilibrium plasma generally have a larger velocity than the protons (an electron will have a velocity ~43 times that of a proton of the same energy). This translates into a net flux or current of electrons onto a spacecraft that is much higher than that of the ions (typically near-nA/cm^2 for electrons versus pA/cm^2 for protons at geosynchronous orbits). This difference in flux is one reason for the observed charging effects (a surplus of negative charge at affected regions).

Although a plasma may be described by an average energy, there is actually a distribution of energies. The rate of charging in the interior of the spacecraft is a function of the flux versus energy, or spectrum, of the plasma at energies well in excess of the mean plasma energies (for GEO, the plasma mean energy may reach a few 10's of keV). Surface charging is usually correlated with electrons in the 0 to ~50 keV energy range while internal charging is associated with the high energy electrons (100 keV to 10 MeV).

A simple plasma is illustrated in Figure 2. The electrons (e^-) and ions (represented as hydrogen - H^+) are moving in random directions (omnidirectional) and with different speeds (a spectrum of energies). To estimate internal charging, the electron spectrum must be known, usually over the energy range from about 100 keV to 10 MeV. Although fluxes may be directed, omnidirectional fluxes are assumed in this document because spacecraft orientation usually can't be known.

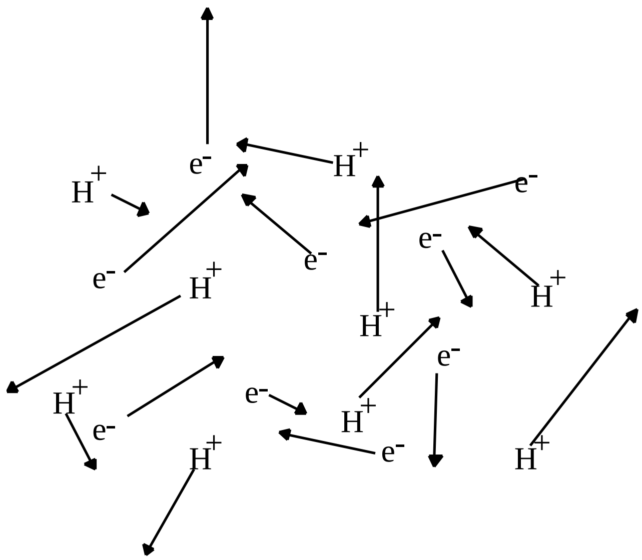


FIGURE 2. <u>Schematic Illustration of a Simple Plasma</u>

4.3 <u>Penetration</u>. Electrons and ions will penetrate matter. The depth of penetration of a given species (electron, proton, or other ion) depends on its energy, its atomic mass, and the composition of the target material. Figure 3 shows the depth of penetration versus energy of electrons and protons into aluminum and presents the approximate deepest penetration into a slab of aluminum. Therefore, only particles with an energy that corresponds to a range greater than the spacecraft shield thickness can penetrate into the spacecraft interior. If the material is not aluminum, an equivalent penetration depth may be approximated by substituting an equivalent number of grams per square centimeter of thickness.

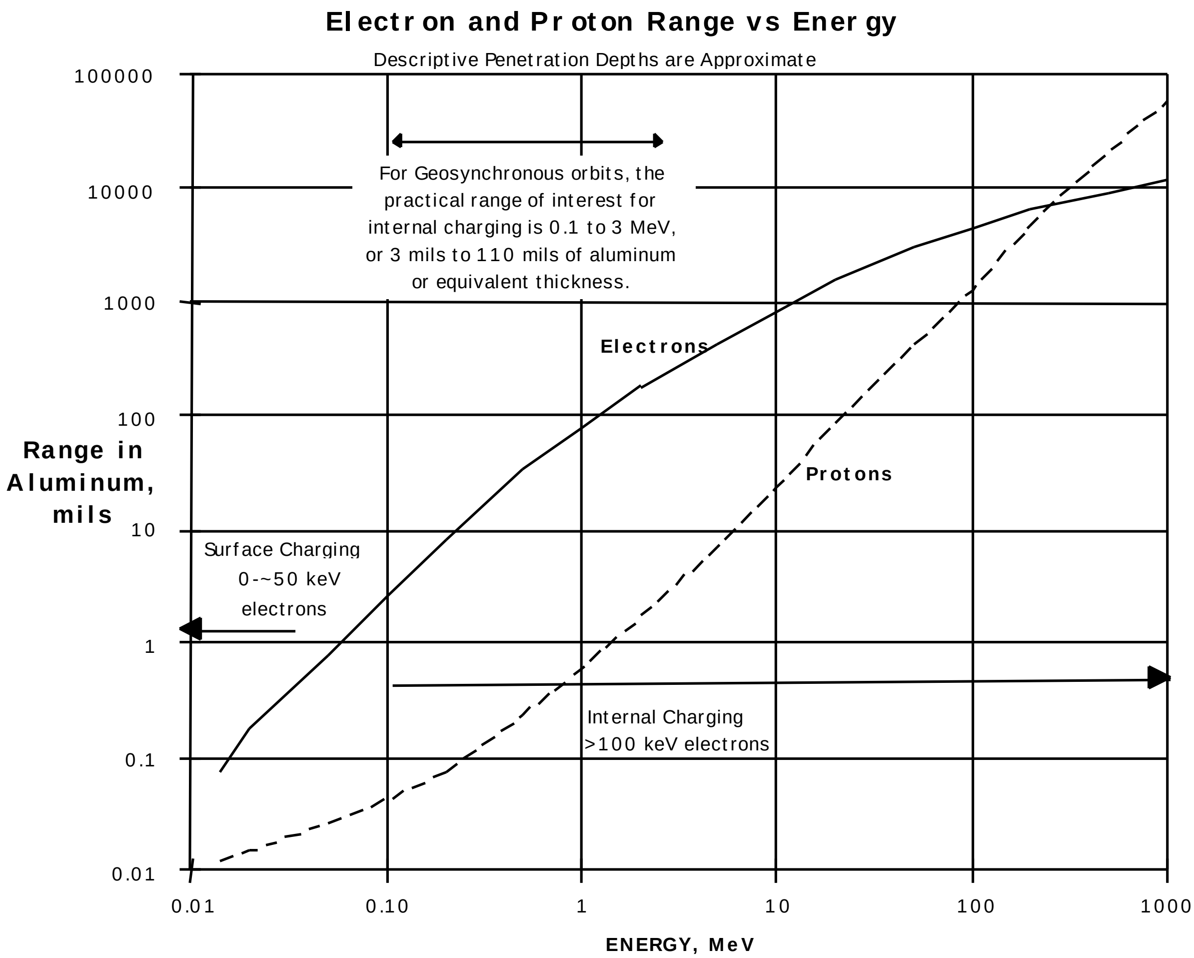


FIGURE 3. Electron/Proton Penetration Depths in Aluminum

This document uses the terminology “surface” and “internal” charging. The literature also uses “buried dielectric charge” or “deep dielectric charge”. It is believed that these other terms can be misleading because they focus attention on dielectrics only. Although dielectrics can accumulate charge and discharge to cause damage, ungrounded conductors can also accumulate charge and must also be considered an internal charging threat. In fact, ungrounded conductors can discharge with higher peak current and higher rate of change of current, and can be a greater threat.

Because electrons may stop at a depth less than their maximum penetration depth, and because the electron spectrum is continuous, the penetration-depth/charging-region will be continuous, ranging from the charges deposited on the exterior surface to those deposited deep in the interior. Internal charging as used here often is equivalent to “inside the Faraday Cage”. For a spacecraft that is built with a Faraday Cage thickness of 30 or more mils of aluminum equivalent, this would mean that internal effects deal with the electron spectrum for energies above 500 keV and protons above 10 MeV. At GEO orbits, the practical range of energy is 100 keV to about 3 MeV, bounded on the lower end by the fact that most spacecraft have at least 3 mils of shielding and, on the upper end, by the fact that, as will be shown later, environments above 3 MeV don’t have enough plasma flux to cause internal charging problems.

Figure 4 illustrates the concept that energetic electrons (100 keV to 10 MeV) will penetrate into interior portions of a spacecraft. Having penetrated, the electrons may be stopped in dielectrics or on ungrounded conductors. If too many electrons accumulate, the resultant high electric fields inside the spacecraft may cause an ESD to a nearby victim circuit.

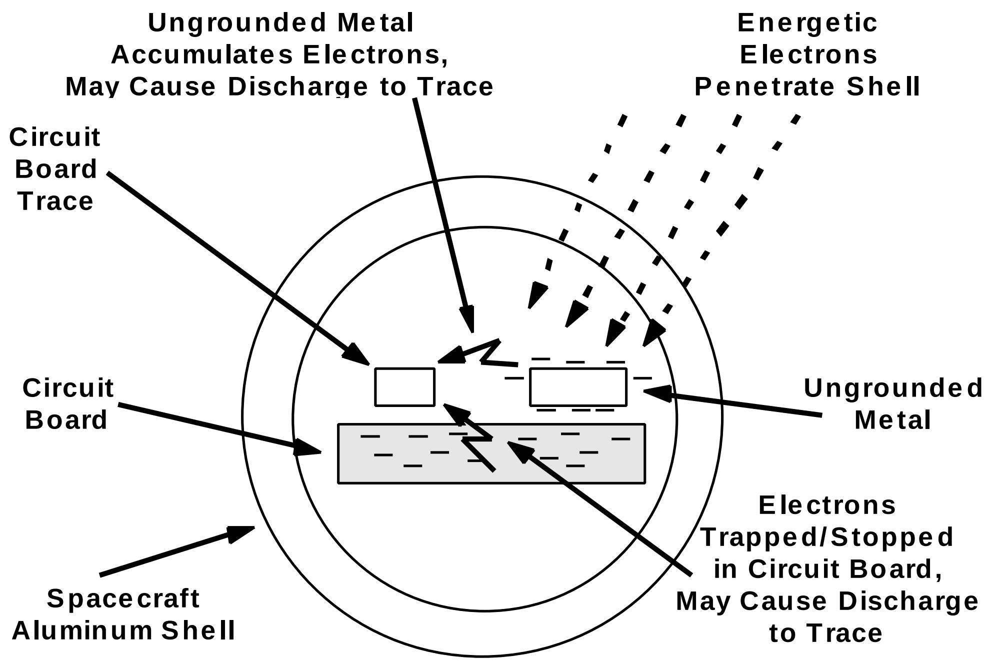


FIGURE 4. Illustration of the Internal Charging Process

4.4 Charge deposition. The first step in analyzing a design for the internal charging threat is to determine the charge deposition inside the spacecraft. It is important to know the amount of charge deposited in or on a given material as well as the deposition rate as these determine the distribution of the charge and hence the local electric fields. An electrical breakdown (discharge) will occur when the local electric field exceeds the dielectric strength of the material or, between dissimilar surfaces, a critical potential. The amplitude and duration of the resulting pulse are dependent on the charge deposited. These values in turn determine how much damage may be done to spacecraft circuitry.

Charge deposition is not only a function of the spacecraft configuration but the external electron spectrum. Given an electron spectrum and an estimate of the exterior shielding, the penetration depth versus the energy chart (Figure 3) permits an estimate of electron deposition as a function of depth for any given equivalent thickness of aluminum, from which the likelihood of a discharge can be predicted. However, because of complexities including hardware geometries, it is normally necessary to run an electron penetration or radiation shielding code to more accurately determine the charge deposited at a given material element within a spacecraft. See Appendix A for a listing of some environment and penetration codes.

4.5 Conductivity. Material conductivity plays an important role in determining the likelihood of a breakdown. The actual threat posed by internal charging depends on accumulating charge until the resultant electric field stress causes an electrostatic discharge. Charge accumulation depends on retaining the charge after deposition. Since internal charging fluxes at GEO are on the order of 1 pA/cm^2 (1 pA = 10^{-12} A), resistivity on the order of 10^{12} ohm-cm will conduct charge away so that high local electric field stress (10^5 to 10^6 V/cm) conditions cannot occur and initiate an arc. Modern spacecraft dielectric materials such as Teflon and Kapton, FR4 circuit boards, and conformal coatings may, however, have high enough resistivities to cause problems (see 6.2, Material List). If the internal charge deposition rate exceeds the leakage rate, these excellent dielectrics can accumulate charge to the point that discharges to nearby conductors are possible. If the conductor leads to or is close to a sensitive victim, there could be disruption or damage to the victim circuitry.

Metals, although conductive, may be a problem if they are electrically isolated by more than 10^{12} ohms. Examples of metals that may be isolated are radiation spot shields, structures that are deliberately insulated, capacitor cans, IC and hybrid cans, transformer cores, relay coil cans, wires that may be isolated by design or by switches, etc. Each and every one of these isolated items could be an internal charging threat and should be scrutinized for its contribution to the internal charging hazards.

4.6 Breakdown voltage. The breakdown voltage is that voltage at which the dielectric strength of a particular sample (or air gap) cannot sustain the voltage stress and a breakdown (arc) is likely to occur. The breakdown voltage depends on the basic dielectric strength of the material (V/mil is one measure of the dielectric strength) and on the thickness of the material. Even though the dielectric strength is implicitly linear, the thicker materials usually have less strength per unit thickness. Manufacturing blemishes or handling damage can all contribute to the variations in breakdown voltages that might be observed in actual tests. As a rule of thumb, if the exact breakdown strength is not known, most common good quality spacecraft dielectrics may break down when their internal electric fields exceed $2\text{x}10^5$ V/cm.

4.7 Dielectric constant. The dielectric constant of a material, or its permittivity, is a measure of the electric field inside the material compared to the electric field in a vacuum. It is commonly used in the description of dielectric materials. The dielectric constant of a material, ε, is generally factored into the product of the permittivity of free space, (epsilon zero, $\varepsilon_o = 8.85 \text{ x } 10^{-12}$ F/m) and the relative permittivity (epsilon sub r, ε_r) of the material in question. Relative dielectric constants of insulating materials used in spacecraft construction generally range from 2.1 to about 4, so assuming a dielectric constant of $\sim 2.7 \bullet 10^{-11}$ F/m (between Teflon and Mylar) is an adequate approximation if the exact dielectric constant is not known. Appendix E, Paragraph E.7, provides examples of the use of the dielectric constant for calculating time constants.

4.8 Shielding density. The density of a material is important in determining its shielding properties. The penetration depth of an electron of a given energy, and therefore its ability to contribute to internal charging, depends on the thickness and density of the material through which it passes. Since aluminum is a typical material for spacecraft outer surfaces, the penetration depth is commonly based on the “aluminum equivalent”. To the first order, the penetration depth in materials depends on the shielding mass. That is, if a material is one-half the density of aluminum, then it takes twice the thickness to achieve the same shielding as aluminum.

4.9 Electron fluxes (fluences) at breakdown. Figure 5 compares spacecraft disruptions as functions of environmental flux at the victim location. Experience and observation of these and other satellites have shown that if the normally incident internal flux is less than 0.1 pA/cm^2, there have been few if any internal charging problems (2x10^{10} e/cm^2 in 10 hours appears to be the threshold). For geostationary orbits, the flux above 3 MeV is less than 0.1 pA/cm^2, and a suitable level of protection can be provided by 110 mils of aluminum equivalent (see Figure 3). Modern spacecraft are being built with thinner walls or only thermal blankets (less mass), so the simple solution to the internal charging problem (adding shielding everywhere) is generally not practical. However, adding spot shielding mass (grounded!) can help in many cases.

Figure 5 also allows a direct comparison between common units as used in the literature and other places in this document (i.e., 10^6 e/cm^2-s is about 2x10^{-1} pA/cm^2).

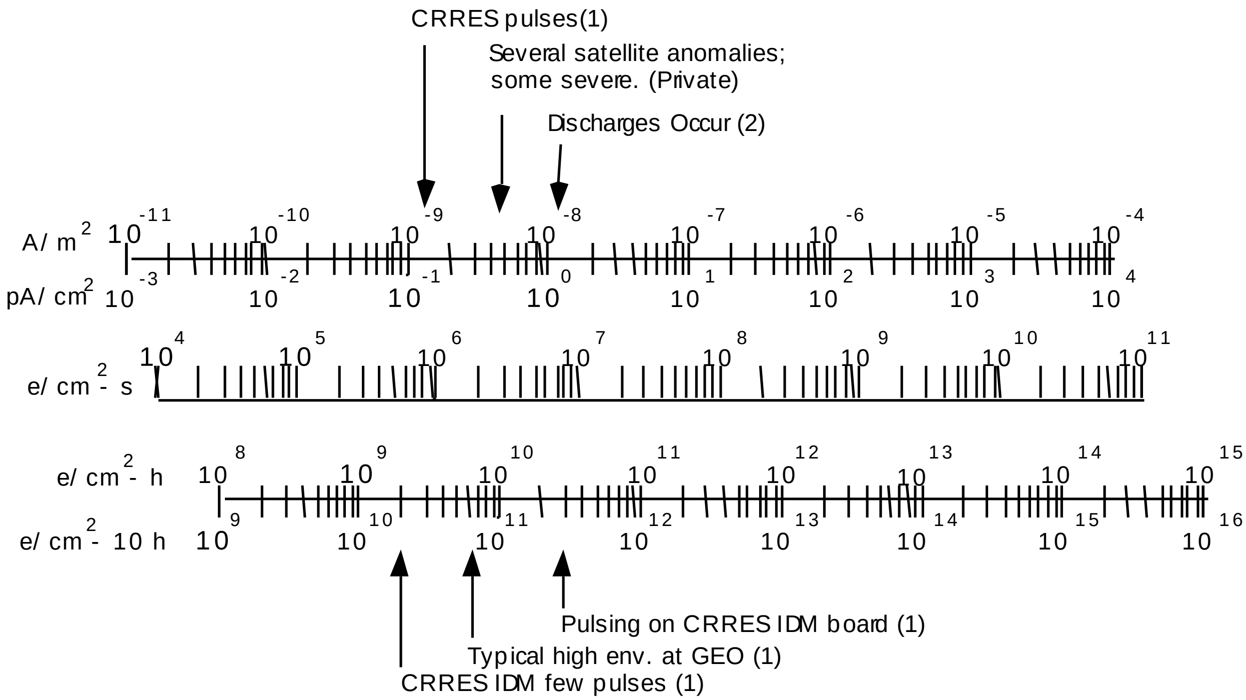


FIGURE 5. IESD Hazard Levels versus Electron Flux (Various Units)
(1) Frederickson [1992], (2) Robinson [1989], and others.
See Appendix F, Section F.3 for information about CRRES.

4.10 Electron Environment. In order to assess the magnitude of the IESD concern for a given orbit, it is necessary to know the electron charging environment along that orbit (as noted before, the protons generally do not have enough penetrating flux to cause a significant internal charge). The electron orbital environments of primary interest are geosynchronous Earth Orbits (GEO), medium Earth orbits (MEO), and polar Earth orbits (PEO). Other orbital regimes that are also known to be of interest are Molniya orbits and Jupiter and Saturn (see Appendix C).

The 11-year variation between the most severe electron environments and the least severe can vary over a 100:1 range and shows correlation with the solar cycle (Appendix B, Figure B-1). A project manager might consider "tuning" the protection to the anticipated service period, but even in “quiet” years, the worst flux sometimes will be as high as the worst flux of “noisy” years. The environment presented in this document represents a worst-case level for GEO.

A worst-case GEO spectrum has been generated by selecting dates when the GOES E > 2 MeV electron data values were elevated to extremely high levels, and then using worst-case electron spectrum data from the geosynchronous SOPA satellite for the same days. The result is shown in Figure 6. It is approximately a 99.9th percentile event (one day in three years, roughly). See Appendix F for descriptions of the GOES and SOPA satellites.

The GEO integral electron spectrum varies with time in both shape and amplitude. Figure 6 also plots the corresponding long-term “nominal” electron spectrum as estimated by the NASA AE8min code for the same energy range. The large difference between the nominal time-averaged (AE8) and shorter-term “worst-case” conditions is characteristic of the radiation environment at Earth. Higher environments are less frequent, but can also occur.

The GEO environment varies with longitude. 200 degrees E is a maximum (see Fig. B-4). The NASA AE8min code presents flux in units of /cm^2-d. That has been converted to per second by dividing by 86400 s/d. It has then been divided by 4 pi, to present the data in the same units reported by the SOPA instruments (e/cm^2-s-sr).

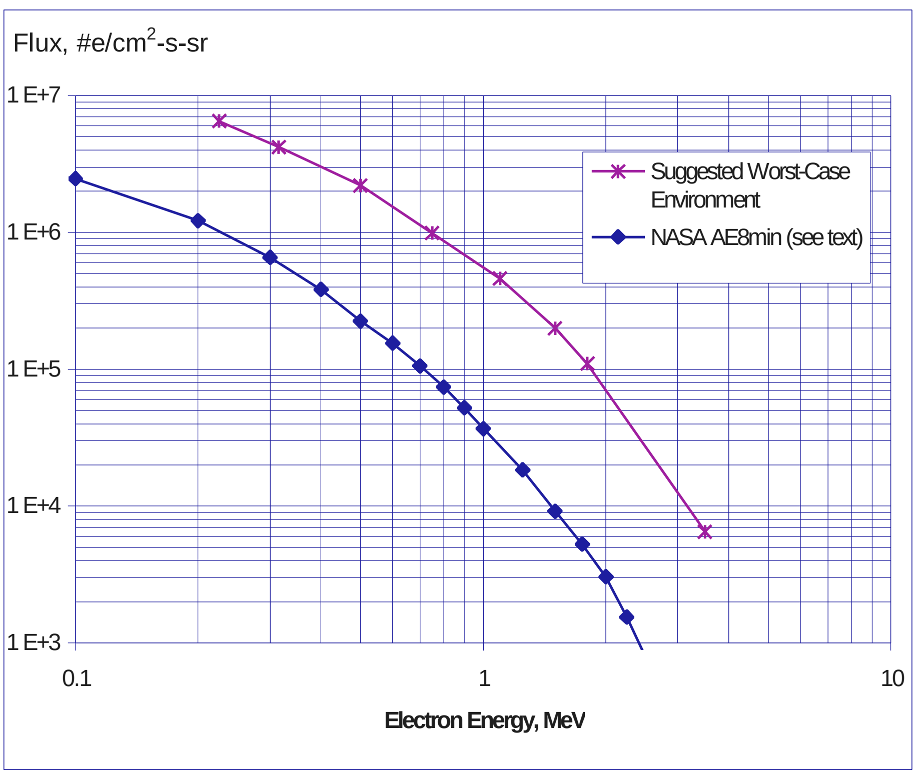


FIGURE 6. GEO Electron Flux Greater than Specified Energy

Upper: Worst-Case Short-Term GEO Environment (May 11 1992, 197 degrees East Longitude peak daily environment over several hour period, with no added margin.)
Lower: NASA AE8min Long-Term Average Environment (200 degrees East Longitude) Daily fluence converted to flux per second and from omnidirectional to “per sr” units

4.11 Units. The primary units that describe the electron environment are flux and fluence. In this handbook, flux corresponds to the rate at which electrons pass through or into a surface element. Units will generally be the number of electrons per square centimeter per steradian (j), but the AE8 code uses electrons per square centimeter (J = 4pi*j). The time unit (per day or per second, for example) should be explicitly present. Some reports present a fluence (a total quantity such as number per square centimeter), but additionally describe the accumulation period (a day or 10 hours, for example), which then could be compared to a flux. Electron fluxes may also be expressed as amperes (A) or picoamperes (pA). Figure 5 interrelates some flux and fluence units. See also paragraph B.7.

The flux can be described as an integral (electrons with energy exceeding a specified value as shown in Figure 6), or differential (flux in a range of energy).

ESD damage potential is related to the stored energy, which is related to fluence.

4.12 Additional Information. For more detailed explanations of the radiation environment, see Hastings and Garrett (1996), Roederer (1970), or other texts on space physics.

5. PROCESSES AND DESIGN GUIDELINES

5.1 Processes. The system developer should demonstrate through design practices, test, and analysis that internal charging effects will not cause a failure to meet mission objectives. This section briefly discusses those processes.

5.1.1 Introduction. The classic approach to avoiding or eliminating electromagnetic problems is to look at the source of the problem, the victim, and the coupling between them. Internal charging is the result of electrons which have penetrated directly to victim circuit areas (perhaps the charge is buried in a circuit board immediately adjacent to the victim), and the three (source, coupling, and victim) are not always clearly distinguishable. For that reason, this handbook has disregarded these categories. However, this approach may sometimes be fruitful and is described below for completeness.

5.1.2 Source. The basic source of internal charging problems is the high energy plasma environment. If the environment cannot be avoided, the next sources of IESD threats are items which can store the internally delivered charge. Ungrounded (isolated) metals are very important because they can accumulate charge and energy. Similarly, dielectrics store charge as well. Limiting the charge storing material or charging capacity is a useful method for reducing the internal charging threat.

5.1.3 Coupling. Coupling energy from a source via a spark (ESD) is very configuration dependent and a function of the radiated and directly coupled signals. An ESD can occur in a variety of ways: from metal to metal, metal to space, metal to dielectric, dielectric to dielectric, dielectric breakdown, etc. The configuration of the charges determines the type of breakdown and hence the form of coupling. An isolated conductor can discharge directly into an IC lead causing serious physical damage at the site or the arc can induce a highly attenuated signal into a nearby wire causing little damage but inducing a spurious signal. As these examples illustrate, the coupling must thus be estimated uniquely for each situation. Eliminating coupling paths in the spacecraft interior, just as is done on the exterior, will significantly lower the IESD threat.

5.1.4 Victim. A victim is any part, component, subsystem, or element of a spacecraft that can be adversely effected by an arc discharge caused by internal charging. Given the different effects of IESD's, the types and forms of victims can be highly variable. ESD and EMC-induced parts failures, while the major problems, are not the only ones. Effects can range from the so-called soft errors (e.g., a memory element may be reset) to actual mechanical damage where an arc physically destroys material. Thus the victims can range from individual parts to whole systems, from electronic components to optical parts (IESDs have long been known to cause fracturing and damage to optical windows or dielectrics). Here, however, the major victims will be either individual electronic components or cables that can couple the transient voltage into a subsystem. Shielding or filtering the victim will limit the effects of IESD's.

5.1.5 Design. The designer should be aware of design guidelines to avoid internal charging problems (see 5.2). All guidelines should be considered in the spacecraft design and applied appropriately to the given mission.

5.1.6 Analysis. Analysis should be used to evaluate a design for internal charging in the specified orbital environment. There are three possible approaches to this analysis: a simple analysis, a detailed analysis, and a coupling evaluation. A simple analysis is illustrated in Appendix D.

5.1.7 Test and measurement. Testing usually ranks high among the choices to verify and validate the survivability of spacecraft hardware in a given environment. For the spacecraft charging environment, and especially the internal charging environment, it is difficult to replicate the actual energetic plasma in all respects. The real electron environment can envelop the whole of the spacecraft and has a spectrum of energies. There is no test facility that can replicate all the features of the actual space environment. As a consequence, verification and validation of internal charging is done with lower level hardware tests and with less realistic test environments. This does not reduce the value of the tests, but additional analyses must be done to provide design validation where testing alone is inadequate. All tests and analyses must be done by persons who are knowledgeable in this area. Tests that can be performed to validate some aspects of internal charging problems are briefly described below.

5.1.8 Material testing. Material electrical properties should be known before they are used. The key material properties needed are (1) their ability to accumulate charge (that is, resistivity or conductivity), and (2) their pulse threat (e.g., voltage, energy). Knowledge of these parameters can be obtained from reference texts or by electron beam tests or conventional electrical tests. See 6.2, sample material list. Some test methods are described in Appendix E.

5.1.9 Circuit/component testing. The susceptibility threshold of components (transistors, integrated circuits, etc.) is useful in understanding the threat from ESD events. The susceptibility can be a disruption threshold or a damage threshold. A "Vzap" test (Appendix E, paragraph E.8) can be used to determine an electronic device's capability to withstand the effects of an electrical transient. Test levels must be determined by a knowledgeable person.

5.1.10 Assembly testing. Potentially susceptible assemblies should be tested for sensitivity to IESD. The assembly to be tested is to be mounted on a baseplate and functioning. Pulses are to be injected into the assembly and the performance of the device monitored for upsets. The pulses used are to cover the expected range of current amplitudes, voltages, and pulse durations. It is very important that the pulse device be electrically isolated from the assembly being tested and the monitoring equipment. It is also important to ensure the support equipment is not being disturbed by the transient.

5.1.11 System testing. System level testing is often the final proof that a system can survive a given environment. For IESD environments, testing is not feasible. Materials, circuit, and assembly testing, together with analysis by knowledgeable persons, must provide the system level verification for internal charging concerns.

5.1.12 Inspection. Inspection is an important means for recognizing and minimizing spacecraft charging discharge-induced anomalies. This inspection should be conducted as the spacecraft is being assembled by a person experienced in recognizing likely areas of concern from environmentally induced interactions. A list of acceptable values of resistance for joints and connections within the spacecraft should be generated ahead of the inspection, but the inspection should take a broader view and look for other possible areas of concern (hence the need for experience).

5.2 Design guidelines. This section contains general guidelines and quantitative guidelines.

5.2.1 General design guidelines

5.2.1.1 Orbit avoidance. **If possible, avoid orbits and altitudes where internal charging is an issue**. See Figure 6 for GEO environments. Usually this is not an option.

5.2.1.2 Shielding. **Shield all electronic elements with sufficient aluminum equivalent thickness so that the internal charging rate is benign**. Experience has shown that for GEO orbits and today's hardware, the proper shielding level is on the order of 110 mils of aluminum equivalent shielding. For some hardware, the amount needed may be less; it almost certainly will exceed 33 mils but may be as low as 70 mils. This is the total shielding, accounting for geometry. A more accurate determination can be done by ray tracing using radiation shielding codes capable of handling detailed geometric and spacecraft material descriptions, and comparing results to the sensitivity of possible victims.

5.2.1.3 Grounding. **Ground all structural elements**. Identify isolated conducting elements and provide grounds for those areas. Make a separate ground strap for conductive items mounted at the end of dielectric booms. Every conductive internal part should be connected by a leakage impedance to each other as measured with an ohmmeter (10^{12} ohms in a vacuum is adequate). Conductive fittings on a dielectric structural parts should comply also.

5.2.1.4 Conductive path. **Have a conductive path to the structure for all circuitry**. A simple and direct ground path is preferred. Note areas where circuits or wires may be isolated for any reason. Place bleed resistors on all circuit elements which may become isolated.

5.2.1.5 Material selection. **Limit usage of excellent dielectrics**. Metals are conductive and protecting them from internal charging is a matter of ensuring a charge leakage path. Therefore, the materials of concern in controlling internal charging are dielectrics. Prominent dielectrics in modern satellites include, but are not limited to, Teflon, Kapton, and FR4 circuit boards. These are excellent charge-storing materials. Their use should be avoided if possible, especially in large blocks. Usages such as wire insulation or thin films (5 mils, for example) seem to contribute less or no problems on the interior of spacecraft. Circuit board materials may be a problem, but densely populated boards are less of a problem (short paths through the dielectric to nearby circuit traces permit easy electron bleed-off). Validate materials with electron beam tests per Appendix E, paragraph E.1.

5.2.1.6 Leaky dielectrics. **Make all interior dielectrics electrically leaky**. Internal dielectrics should be "ESD-conductive" or “leaky.” This applies specifically to circuit boards, but would be desirable for all dielectrics, including cable wiring and conformal coatings. The degree of "leakiness" or conductivity does not need to be great enough to interfere with circuit performance. It can be on the order of 10^{12} Ω-cm or 10^{12} Ω per square and still provide a bleed path to electrons for internal charging purposes. Verify that the conductivity remains adequate over the mission life. Today, there are no materials that meet this requirement and still provide the other necessary properties (mechanical, workable, etc.).

5.2.1.7 Spot shields. **Ground radiation spot shields**. This can be done in a number of ways. If a Solithane or other conformal coating has adequate resistivity (on the order of 10^{10} or less ohm-cm), a separate ground wire is unnecessary. It must be determined that any solution, such as partially conductive Solithane, will not degrade in the expected radiation and long-term vacuum environments. At the time of writing, a suitable leaky conformal coating is only a prototype [Frederickson, 1992]; it needs validation to prove that it meets all other necessary space flight properties.

5.2.1.8 Filter circuits. **Use low pass filters on interface circuits**. Use low speed, noise-immune logic if possible. Use CMOS circuits which have higher interface noise immunity. However, beware of the latch-up sensitivity of CMOS. For IESD purposes, the filter or protection network must be applied so that it is physically at the device terminals.

5.2.1.9 Isolate windings. **Isolate the primary and secondary windings of all transformers**. Reduce primary to secondary winding capacitance to reduce common mode noise coupling. This is an EMC solution to reduce coupling of ESD-induced noise.

5.2.1.10 Bleed Paths. **Provide a conductive bleed path for sometimes forgotten conductors (including structural elements)**, including but not to be limited to the following items.

- Signal and power transformer cores
- Capacitor cans
- Metallic IC and hybrid device cases
- Unused connector pins
- Relay cans

These items may be protected by stray leakage through their conformal coating, normal bleed paths, or small charge/energy storage areas. Ensure that the presumed bleed path really works before depending on it.

5.2.1.11 Interior paints and conformal coatings. **Most paints and conformal coatings are dielectrics and can be charged by energetic particles**. This must be considered in evaluating the interior charging likelihood of a design. If conductive coatings are used, then these must be grounded to the structure to allow charge to bleed off. For conductive coatings, conductive primers must be used. If non-conductive primers are used, then the conductive coating will be isolated from ground and will charge. Other grounding means must be provided if the primer or substrate is non-conductive.

5.2.1.12 Antenna feeds and parabolas. **Coverings on antenna feeds and parabolas must be considered**. Isolated dielectric materials on an antenna system, especially near feed lines, can store excess charge or energy. For example, if there is an isolated dielectric mounted on top of a fiberglass separator that is adjacent to the feed electrical path, then there can be discharges directly into the receiver. These dielectrics are special problems because they are on the outside of the spacecraft and have less shielding. Assess each of the region's hazards and compare to the receiver or LNA ESD sensitivity.

5.2.1.13 Cable harnesses. **Route cable harness away from apertures**. Care should be taken in the layout of the internal electrical harnesses to minimize exposure to the environment's energetic particles. The harness should not be close to the edges of apertures.

5.2.1.14 External wiring. **Provide additional protection for external cabling.** Cables external to the spacecraft structure should be given adequate protection. The dielectric coatings can charge to a point where discharge can occur. At present, there are no simple design rules for the degree of shielding needed. Cables should be tightly wrapped to minimize gaps where discharges can propagate.

5.2.1.15 Thermal blankets. **Thermal blankets provide very little protection against electron penetration**. Don't consider thermal blankets to be more shielding than their (small) mass density per square centimeter. Ground all electrically conductive layers of thermal blankets.

5.2.1.16 Slip ring grounding. **Carry grounds across all articulated and rotating joints.** For a rotating joint with slip rings (such as in a solar array), the chassis or frame ground (bond) must be carried through the slip ring also and then grounded. Note that for the case of the solar array and other situations that may involve transfer of ESD current, a series resistance in the path from spacecraft frame to solar array frame will limit the amount of current that can carry this ESD noise current into the satellite. A resistance value of 50 k ohms is suggested; the GEO basis is surface charging with 1 nA/cm^2 charging current, times the area being grounded, and a 10-volt limit.

5.2.1.17 Wire separation. **Segregate cabling from outside the spacecraft after it enters the Faraday Cage.** Wires coming from outside the spacecraft should be filtered and then separated from the filtered wires going to the telemetry or other electronic elements. This is based on an assumption of external ESD noises, and is to prevent coupling to the interior. It is a poor design practice to route the filtered and unfiltered wires together in the same bundle because noise can be coupled between them.

5.2.1.18 ESD sensitive parts. **Pay special attention to ESD-sensitive parts**. In the parts list, flag all parts that are Class 1 ESD-sensitive per MIL-STD-883 Method 3015. Do a charging analysis after completion of the spacecraft design. Evaluate the charging rates with respect to paragraph 5.2.2 parameters. Protect the devices if they fail to meet the criteria of paragraph 5.2.2.

5.2.2 Quantitative guidelines. The following quantitative guidelines are recommended:

5.2.2.1 Grounding conductive elements. **Spacecraft cables, circuit traces, and other conductive elements greater than 3 cm^2 in surface area (0.3 cm^2 for conductive elements on circuit boards) or longer than 25 cm in length shall be ground referenced** (provide a bleed path for all radiation spot shields; for other ungrounded wires and metal, being in a circuit is usually adequate). (It is best not to have any deliberate ungrounded metals, including unused connector pins as an example.) Exceptions are allowed in situations in which one of the following conditions is true: (1) discharges will not occur in the expected charging environment, or (2) the discharges expected to occur will not damage or disrupt the most sensitive circuits in the vicinity nor cause electromagnetic interference EMI which exceeds the electromagnetic compatibility EMC requirements (assuming separate EMC requirements exist). These historic quantitative guidelines may need reconsideration for newer spacecraft.

5.2.2.2 Shielding. **Determine electron fluxes at all part locations using a worst-case electron spectrum (Figure 6 for GEO), and shield all electronic circuitry to the following levels (Figure 5 basis with no margin; projects may wish to consider margins):**

a. If there are 110 mils of aluminum equivalent, there is no need to shield further and there is no need to do an electron transport analysis, unless there is a desire to save weight (GEO orbit rule only).

b. If the computed flux is less than 0.1 pA-cm^{-2}, the circuit needs no additional shielding (any electron environment). (Basis: less than 10^{10} e/cm^2 in 10 hrs.)

c. If the incident flux is between 0.1 pA-cm^{-2} and 0.3 pA-cm^{-2}, shield to a level of 0.1 pA-cm^{-2} if the circuitry is Class 1 ESD-sensitive (MIL-STD-883, Method 3015), or if this type of circuitry has had a known on-orbit anomaly.

d. If the incident flux is between 0.3 pA-cm^{-2} and 1 pA-cm^{-2} and Class 2 ESD-sensitive circuitry is present, then shield to < 0.3 pA-cm^{-2}.

e. Having circuit boards exposed to > 1 pA-cm^{-2} may lead to IESD problems.

5.2.2.3 Filter circuits. **For wiring protected less than the levels of paragraph 5.2.2.2, protect attached circuits by filtering**. To protect the interior sensing circuit for temperature transducers that are located outside the main box of the spacecraft, RC filters can be used to suppress any ESD effects. Another reason for filtering is that the shielding levels of 5.2.2.2 cannot be achieved. The filter should anticipate a pulse on the order of 20 ns wide. As a rough example, filtering should protect against a 20 pF capacitance charged with 4 nanocoulombs (about 5 kV stress, 250 microJoule). The real estimated threat as determined by an knowledgeable person should be used if possible.

5.2.2.4 Voltage stress. **Keep the electric field stress in dielectrics below 100 volts/mil.** When designing high voltage systems, keep the electric field below 100 volts/mil. This voltage stress could be in dielectrics (circuit boards) being charged by the incident electron flux while the adjacent lands remain at a low voltage. Other sites of concern are ungrounded metal radiation shields on insulating surfaces charged by the electron flux while the adjacent surfaces remain at low voltages, or insulated surfaces being charged while internal wires remain at low voltage. Power supplies can sustain a discharge after an arc has been initiated, so power lines should never be uninsulated (exposed). All such possible sources must be eliminated where possible.

6. NOTES

(This section and the Appendices which follow contain items of a general or explanatory nature that may be helpful but are not mandatory.)

6.1 General comments. There are no clean rules to be followed in determining whether or how much to test. General guidance dictates that an engineering version of the hardware be tested in lieu of the flight hardware, and that this testing have margins (be more severe than the expected environment). The trade-offs are common to other environmental testing; the main difference is that the IESD-specific threat is more difficult to replicate in practical tests than for other environmental disciplines.

The danger in not testing is that serious problems due to internal charging will go undetected, and that these problems will affect the survivability of the spacecraft. The best that can be done in the absence of testing is good design supplemented by analysis. Good IESD design techniques are always appropriate no matter what the overt environmental threat is, and should be followed as a necessary precaution in all cases.

6.2 Sample material list. The partial list of dielectric material properties in Table I is provided for illustration only. Data were taken from the Fifth Edition of the Radio Engineers Handbook [Westman, 1968], Handbook of Electrical and Electronic Insulation Materials, 2nd Ed., [W. Tillar Shugg, 1995] and other sources. Some of the data may only be specified minimum or maximum limits and not typical values; actual resistivity values may differ by many orders of magnitude (e.g., FR4) Each project doing its assessment of internal charging hazards must compile its own list based on the most current and appropriate data.

TABLE I. List of Representative Material Characteristics for Internal Charging Studies

Parameter/ Material (Units)	Relative Dielectric Constant	Dielectric Strength* (V/mil @ mils)	DC Volume Resistivity (Ω-cm)	Density (gm/cm^3)	Time Constant (as noted)
DIELECTRICS					
Ceramic (Al_2O_3)	8.8	340 @125	$>10^{12}$	2.2	>0.78 s
Delrin	3.5	380 @ 125	10^{15}	1.42	310 s (5.2 min)
FR4	4+	420 @ 62	$>4x10^{14}$	1.78	> 141 s
Kapton	3.4	7000 @ 1	10^{18}	1.4	3.5 days
Kapton	--	580 @ 125	10^{18}	1.4	3.5 days
Mylar	3	7000 @ 1	10^{18}	1.4	3.1 days
Polystrene	2.5	5000 @ 1	10^{16}	1.05	37 min
Quartz, fused	3.78	410 @ 250	$>10^{19}$	>2.6	>38 days
Teflon**	2.1	2-5k @ 1	10^{18}	2.1	2.1 days
Teflon	--	500 @ 125	10^{18}	2.1	2.1 days
CONDUCTORS					
Aluminum	N/A	N/A	$2.62x10^{-6}$	2.7	N/A
Al Honeycomb	N/A	N/A	Variable	~0.049	N/A
Brass (~70-30)	N/A	N/A	$3.9x10^{-6}$	8.5	N/A
Copper	N/A	N/A	$1.8x10^{-6}$	8.9	N/A
Graphite Epoxy	N/A	N/A	Variable	1.5	N/A
Invar	N/A	N/A	$81x10^{-6}$	8.1	N/A
Iron-steel	N/A	N/A	$9\text{-}90x10^{-6}$	7.87	N/A
Kovar A	N/A	N/A	$284x10^{-6}$	~7.8	N/A
Magnesium	N/A	N/A	$4.46x10^{-6}$	1.74	N/A
Tantalum	N/A	N/A	$13.9x10^{-6}$	16.6	N/A
Titanium	N/A	N/A	$48x10^{-6}$	4.51	N/A
Tungsten	N/A	N/A	$5.6x10^{-6}$	18.8	N/A

* Dielectric strength is often specified as a function of thickness.
** Generic numbers for Teflon. PTFE and FEP are common forms in use for spacecraft.

NOTES:

1. Time constant (s) = permittivity (Farad/m) x resistivity (Ω-m)
2. Permittivity (dielectric constant) = relative dielectric constant x 8.85×10^{-12} F/m
3. Resistivity (Ω-m) = resistivity (Ω-cm)/100
4. N/A = not applicable
5. > is “greater than”; >> is “much greater than”; if the numbers in the table are “greater than”, then the actual time constants could be much larger than shown (calculated) in this table.

APPENDIX A

ENVIRONMENT AND ELECTRON TRANSPORT COMPUTER CODES

Environment Codes

NASA AE8 and AP8. These are traditional and long-used electron and proton models of the Earth environment. In these codes the fluxes are long-term average, and do not have the peak electron fluxes necessary for the internal charging calculations recommended in this handbook.

CRRES. Combined Release and Radiation Satellite, that monitored Earth's radiation belts in an eccentric orbit for 14 months starting July 1990. Its actual data is in the form of electron and proton flux and dose-depth as a function of time and altitude. Environment codes from CRRES include: CRRESRAD (dose versus depth); CRRESPRO (proton flux energy spectrum); CRRESELE (electron flux energy spectrum). They are available from AFRL, Attn: Dr. G. Ginet, Space Physics Division, Hanscom AFB, MA 01731-3010.

Alternate source of space radiation data are: Severn Communications Corporation, 1023 Benfield Boulevard, Millersville MD 21108 (including AP8 and AE8); and Al Vampola 23452 Kathryn Ave., Torrance CA 90505 (CRRES data).

MIL-STD-1809 (USAF), 15 February 1991, Space Environment for USAF Space Vehicles. This includes some electron spectra that can be used in the electron transport codes. It has good information that supplements the Earth environmental information in this handbook.

Transport Codes

ITS (Integrated Tiger Series) has been published for electron flux and deposition and validated by experiment. It would be the first choice for the electron deposition calculations suggested in this handbook. Some packages have been simplified to handle simple geometries such as cylinders and slabs. Contact: Radiation Shielding Information Computational Center (RSICC), Oak Ridge National Laboratory, Bldg. 6025, MS 6362, P. O. Box 2008, Oak Ridge, TN 37831-6362 (ITS CCC-467).

MCNP/MCNPE. Can be used to determine electron flux inside complex spacecraft geometries. MCNP does detailed 3-D Monte Carlo modeling of neutron and photon transport. MCNPE does 3-D modeling of neutron, photon, and electron transport. They have a powerful geometric capability; however, transport to very deep depths can take extremely long computer runs with a large uncertainty in the results. At shallow depths (up to 0.6 inch of aluminum thickness) codes like MCNPE and ITS are preferred. At deep depths, adjoint codes are preferred. Contact: Radiation Shielding Information Computational Center (RSICC), Oak Ridge National Laboratory, Bldg 6025, MS 6362, P. O. Box 2008, Oak Ridge, Tennessee 37831-6362.

NOVICE. A charged-particle radiation transport code. It uses an adjoint Monte-Carlo technique to model particle fluxes inside a user-specified 3-D shield geometry. Contact: Thomas Jordan, Experimental and Mathematical Physics Consultants, PO Box 3191, Gaithersburg, MD 20885, phone 301-869-2317. This source may also have codes for electron deposition calculations.

SHIELDOSE. A charged-particle radiation transport code that calculates the dose inside slab and spherical shield geometries. Contact: Stephen M. Seltzer, Ionizing Radiation Division, National Institute of Standards and Technology, Gaithersburg, MD 20899, phone 301-975-5552.

APPENDIX B

GEOSTATIONARY ELECTRON ENVIRONMENTS

The geostationary environment (GEO) is the most well characterized of the Earth orbits because of its dominant role for communications satellites. Quantitative data for GEO are more readily available than for other orbits. There are, however, a number of characteristics of the environment that need to be considered. These range from variations with longitude to time-dependent variations in the electron spectra.

B.1 Variation with solar cycle. The electron population of the plasma at GEO has a long-term variation with the solar or, more commonly, the sunspot cycle (about 11 years). The E > 2 MeV electron population as measured by the geosynchronous GOES satellites is roughly anticorrelated with the sunspot cycle; when the solar sunspot number is low, the GOES E > 2 MeV electron flux is high. This is demonstrated in Figures B-1 and B-2.

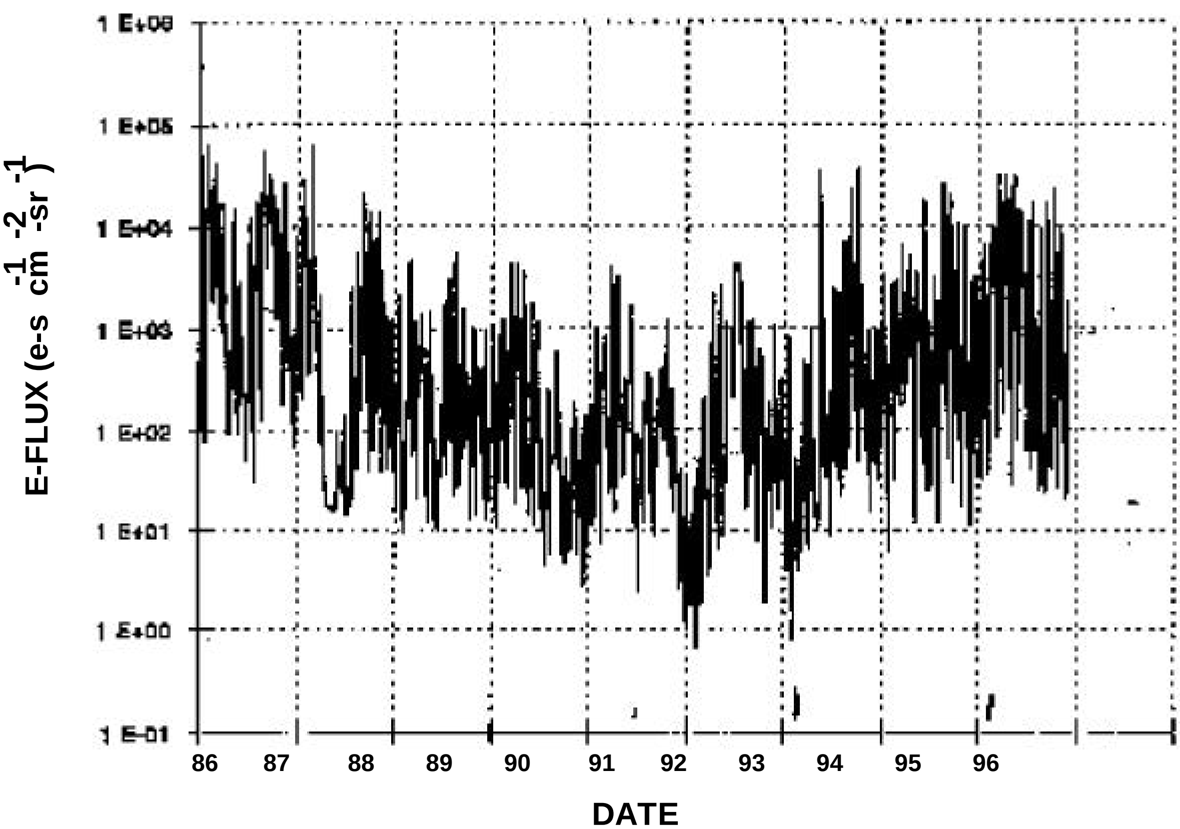


FIGURE B-1. Average Flux at Geosynchronous Orbit for E > 2 MeV Electrons as Measured by the GOES Spacecraft [Sauer, 1996]

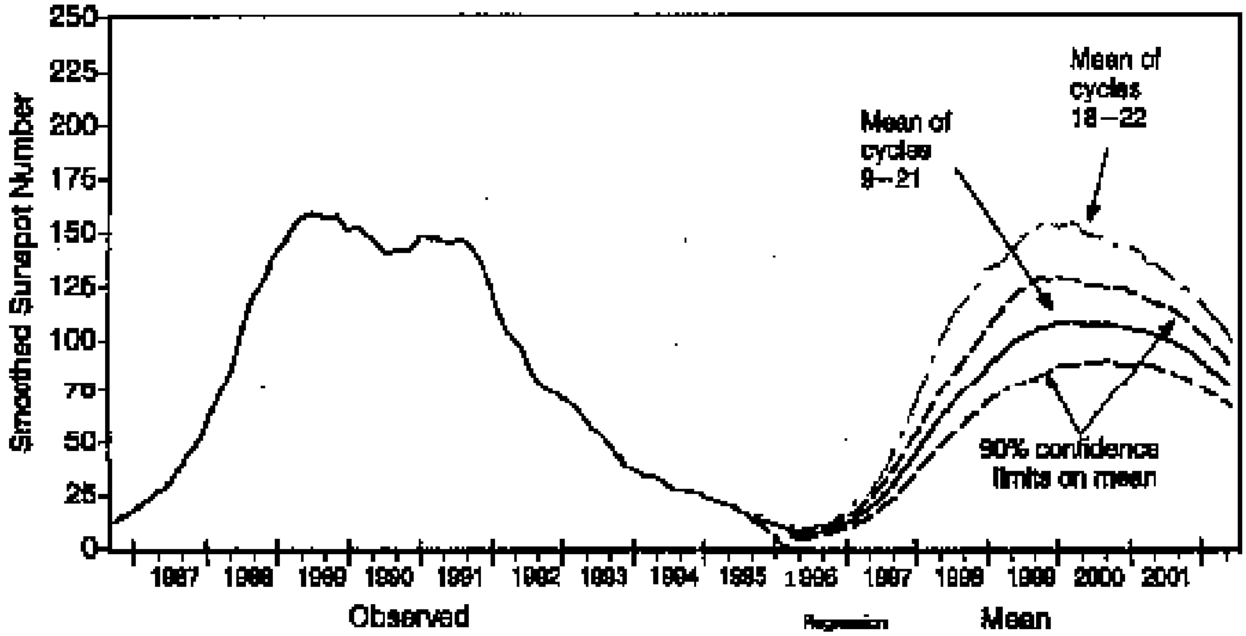


FIGURE B-2. Observed and Predicted Smoothed Sunspot Numbers for 1986 - 2002

For projects with an unknown launch date, the satellite must be designed to withstand the worst of these periods. The range between the worst-case conditions and the least stressing is more than 100:1 in energetic electron flux. The worst event observed in modern times occurred in 1972 during “solar minimum.” However, the Sun, which drives these environments, does not strictly obey “averages” and even during times when the >2 MeV electron fluxes are usually low, the energetic electron fluxes can be extremely high. The project manager knowing the mission schedule may wish to assume some risk in order to save project resources, but it is not recommended because of the environmental variation.

B.2 Variation with longitude. The plasma/radiation environment is linked to the Earth’s magnetic field lines. Magnetic field lines are described in terms of “L-values,” the distance that a given magnetic field line crosses the magnetic equator in Earth radii. Following the path of a field line around the Earth traces out a so-called L-shell. As charged particles (electrons, protons, etc.) are trapped along the magnetic field lines, the radiation environment can be described in terms of the magnetic field strength and the L-shell (the “B-L” coordinate system is often used in the literature on radiation belts). Because the Earth’s magnetic dipole is tilted with respect to the rotational axis, the L-shell at the rotational equator is not constant over all longitudes (see Figure B-3). Because the radiation environment is approximately constant on a particular L-shell at the magnetic equator, there is a change in the plasma/radiation environment at different longitudes as different L-shells are encountered at GEO altitudes. The fluence and dose variance at GEO are shown in Figure B-4.

The resultant GEO electron fluence per the AE8 model and the dose from electrons per the CRRESRAD model are shown in Figure B-4. This is shown only to illustrate the longitudinal variation on the average. The maximum electron environment should be used for all satellites, even if their longitudinal location is known.

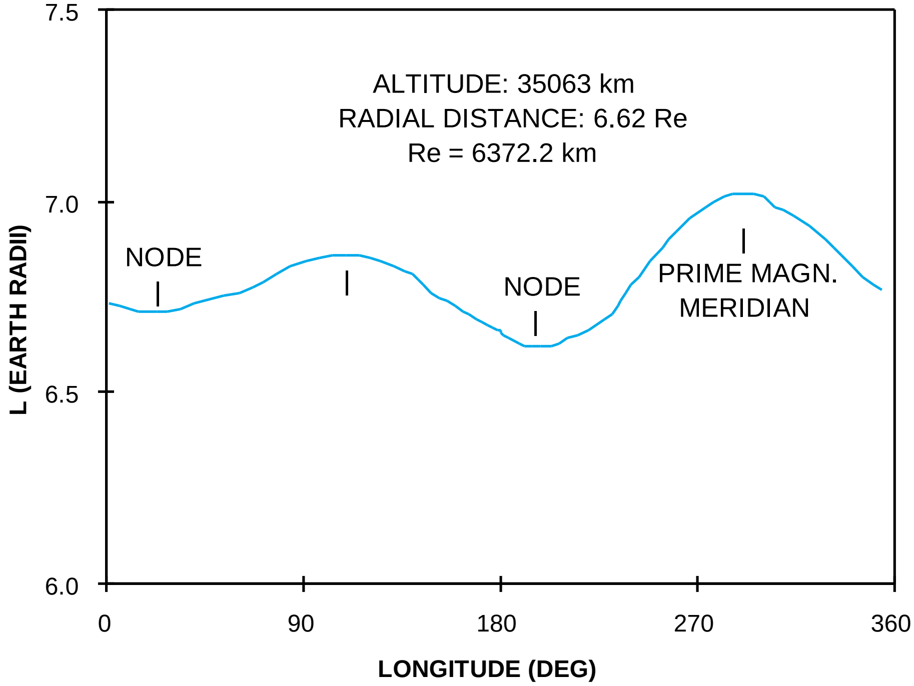


Figure B-3. L-Shell Values Around the Earth's Equator (Corresponding to Geostationary Orbit) versus East Longitude [Stassinopoulos, 1980]

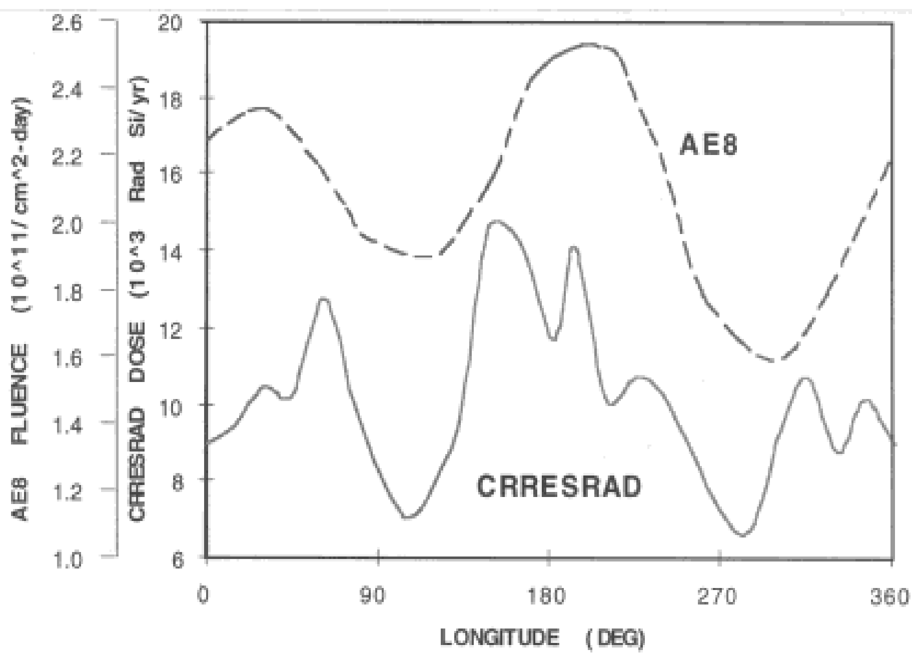


Figure B-4. AE8 >0.5 MeV Daily Electron Fluence, and CRRESRAD Annual Dose Due to > 1 MeV Electrons Plotted as Functions of Satellite East Longitude at 6.6 R_e for the AE8 (>0.5 MeV) and CRRESRAD (>1 MeV) Models [Wrenn, 1995]

B.3 Variation with averaging interval. In addition to long term-solar cycle variations, there are short term temporal variations associated with geomagnetic activity and changes in the Earth's magnetosphere. As a consequence, the average electron flux varies with the time interval over which the averaging is carried out. This can be seen when a large data set, gathered with a high time resolution, is averaged over increasingly longer integration times. The GOES E > 2 MeV electrons are returned with a 5 minute resolution. The variation between the daily peak flux determined in a 5 minute interval to the peak flux averaged over a 24 hour period is about 3 to 4 (the 24 hour average peak is, as would be expected, lower). This issue of averaging interval must be kept in mind when comparing different data sets.

B.4 Variation with local time. The high energy electrons at a given geosynchronous longitude position vary daily with the local time. On active days, the flux variation is about 10:1 from local noon to local midnight, with the highest flux near local noon. This can be seen on the NOAA web site, http://solar_sec.noaa.gov/today.html, which shows three-day electron and other data. The normal daily average of the GOES E > 2 MeV electron flux ($e\text{-}cm^{-2}\text{-}s^{-1}\text{-}sr^{-1}$) is about 1/3 of the peak daily flux (the highest flux in a five minute period.

B.5 Spectrum. The integral electron spectrum varies with time in both shape and amplitude. Figure 6 shows a "worst-case" high amplitude energy spectrum from the SOPA detectors, and compares that spectrum to the AE8, which is a long-term average. Data from AE8 "average" days shows a different spectral shape as well as lower amplitudes. That is, the ratio of integral electron flux at 2 MeV to that at 600 keV is generally not the same from day to day. It can be seen that whereas at low energies (E < 100 keV), the curves approach each other, above 1 MeV the spectra rapidly diverge, with the "worst case" spectrum approximately 2 orders of magnitude higher than the AE8 spectrum. This large difference between nominal, time-averaged and short term, worst case conditions is characteristic of the radiation environment at the Earth. The AE8 model is inappropriate for internal charging calculations.

B.6 Amplitude Statistics. An excellent set of data for the statistical analysis of the long term variations in the total electron flux at geosynchronous orbit is that from the NOAA GOES-7. The data are only available for electrons for E > 2 MeV, but the measurements are from one detector and available for nearly one complete solar cycle (Figure B-1, [Sauer, 1996]). This data set is unique in its completeness and has not been duplicated as yet for any other satellite over the energy ranges of interest. The SOPA satellites have similar data and include other energy ranges for geosynchronous, but the data have not yet been analyzed and made available to the degree that the GOES-7 data have been. Therefore, much of the statistical analyses for geosynchronous orbit are based on GOES-7 E > 2 MeV electron data.

Figure B-5 plots the cumulative probability of occurrence of GOES-7 electron fluxes. The time span was an eight-year period encompassing the largest energetic fluxes in that solar cycle. Figure B-5 shows amplitude statistics for three statistics from that data set: a) for the worst 25 months, the day's highest 5 minute average flux; b) for the worst 25 months, the daily average flux; and c) for the whole 8 years, the daily average flux.

The reader is cautioned about trying to use these probabilities for use in a design sense. Use the worst case energy spectrum of Figure 6.

B.7. Advanced units. See also par 4.11. Some publications, including NASA's AE8 code, use the term "omnidirectional flux", which implies an isotropic (uniform in all directions) electron flux. We will call this "J", having the units #/cm^2-s (number of electrons per square centimeter per second). Other publications report a flux on a per steradian basis ("j", with units of #/cm^2-s-sr). Assuming an isotropic plasma, the two are related by

J = 4pi*j (units for J, #/cm^2-s; j, #/cm^2-s-sr)

The net current impinging on a flat surface will be

i = (1/2)(1/2)*J = pi*j (electrons per square centimeter per unit time)

The first 1/2 is present because the current only comes from one side of a flat surface, and the second is the average value of the cosine for non-normal incident angles. If the flux is not isotropic, these simple calculations must be redone for the actual distributions.

For the simple penetration calculations presented in Appendix D, the multiplier would be smaller than pi (the non-normally incident electrons cannot penetrate as deep because of the longer path length), but the difference depends on the depth, and spectrum of the electrons.

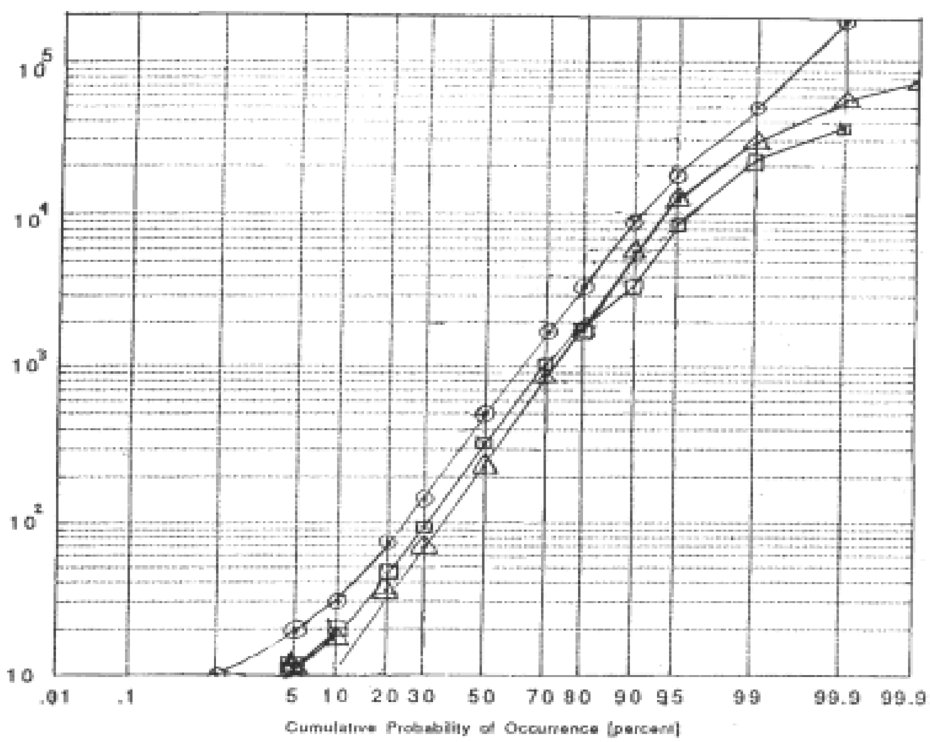


FIGURE B-5. Cumulative Probability of Occurrence of GOES-7 E > 2 MeV Electron Fluxes for Several Different Assumptions. (Data Are From Sauer [1996])

Vertical Scale: E/(cm^2-s-sr)

KEY:
- O : 5-minute peak GOES flux (largest 5-minute amplitude each day, 25 continuous months at high level in solar cycle, 1/1/92-1/31/94
- Δ : daily GOES average fluxes over the 8-year time span 1/1/86-11/30/94
- □ : GOES daily averages in the 25-month time span 1/1/92-1/31/94

The open circles are the peak GOES electron flux data (largest amplitude 5 minute value in the day) for times of higher flux (January 1, 1992, through January 31, 1994). The open triangles correspond to the cumulative probability for the daily GOES average fluxes over the 8-year span from 1986 to 1994. The open squares correspond to the GOES data for all daily averages from January 1, 1992, through January 31, 1994. All data are from Sauer [1996]. The key feature to be noted here is that a Gaussian probability distribution implied by a straight line fit from about 10 per cent to about 95 percent does not explain the data above the 95th percentile. This makes it difficult to extrapolate with any confidence to a 99.99 percentile environment. The fall-off at the higher percentiles is believed to be real. Thus, the largest environments, although real, are less frequent than a simple Gaussian distribution would imply.

APPENDIX C

OTHER EARTH PLASMA ENVIRONMENTS

C.1 Medium Earth orbit. Medium Earth Orbit (MEO, roughly 2,000 to 25,000 km altitude, but worst at 20,000 km altitude) is perhaps the most stressing of the environments that need to be considered for internal charging. As several of the new multi-spacecraft communications systems plan to fly in this orbit also, it could potentially become a major driver in the study of internal charging phenomena. Figure C-1 is a meridional schematic of the Earth's radiation belts showing the AE8 and AP8 predictions of the electron (E > 1 MeV) and protons (E > 10 MeV) fluxes. This plot clearly shows the "two belt" structure of the electron belts and the "horns" that extend down to lower altitudes near the poles. This plot is in generalized magnetic coordinates and does not reflect the Earth's magnetic field tilt and offset. Even so, it gives a clear picture of the MEO environment and how it is related to orbital characteristics. Each region has a unique spectrum associated with it, which would affect internal charging calculations.

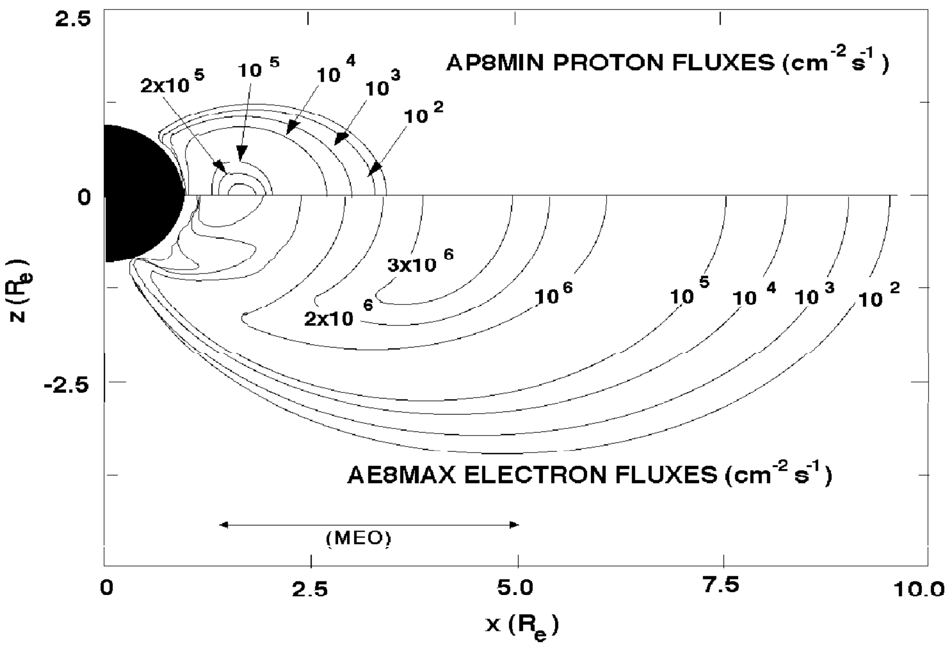


Figure C-1. Schematic of the Earth's Radiation Belts as Estimated by the AE8 and AP8 Models Contours are for E > 1 MeV Electrons and E > 10 MeV Protons.

NOTE: Figure C-1 shows both electron and proton fluxes as referenced to the Earth's generalized magnetic coordinates, combined onto one chart. The left-hand vertical axis is the magnetic pole axis with vertical units of Earth radii. The horizontal scale is magnetic equatorial distance from the axis in Earth radii. The upper chart represents protons; the southern hemisphere proton flux is a mirror image. The electrons (lower chart) also are symmetric above and below the magnetic equator.

C.2 Polar Earth orbit. A second important orbital regime is that associated with highly inclined polar orbits. As seen in Figure C-1, a polar orbit at low altitudes can pass through the horns of the electron belts and experience a significant, if short duration, flux of high energy electrons. Many military spacecraft, most imaging spacecraft, and low altitude communications fleets are in polar orbits. For low altitude orbits (<1000 km), the risk of internal charging is present but generally much lower than at GEO or MEO. At higher altitudes, the interaction is dependent on the details of the orbit and can be minimized with a proper choice of eccentricity and inclination. Even so, any high inclination orbit should be evaluated for potential internal charging issues early in the mission design.

C.3 Molniya orbit. Satellites in a Molniya orbit follow an elliptical track with a perigee of 500 km and an apogee of 36,000 km. This orbit is inclined at 63 degrees and the period is on the order of 12 hours. In this orbit, satellites traverse a full range of space environments from the higher-density, low-energy plasma at low Earth orbit through the radiation belts to interplanetary environments. With the Earth's rotation, the orbit is exposed to night and day so that the satellite is subjected to all environmental variations. Again, the high-energy electron environment should be evaluated for possible internal charging issues for Molniya missions.

APPENDIX D

CHARGING ANALYSES

D.1 Simple analysis. The following is an example of a simple and probably conservative analysis that will be used to estimate the current flux into a dielectric of a spacecraft at GEO. If the simple analysis indicates that the flux is close to the design limit, then a complete analysis should be used to determine if the criteria is exceeded.

The example is to determine the flux of electrons in a 10 mil thick layer of Teflon under a 10 mil thick sheet of aluminum. Refer to Figure 3 for penetration depth/energies. Refer to Figure 6 for fluxes versus energy. Refer to Table I for material densities.

Electron Flux	Penetration Energy	Exiting Integral Flux
Into 10 mils of aluminum	~250 keV	$6x10^6$ e/cm^2-s-sr
Through 10 mils of Teflon (equivalent to 7.8 mils of aluminum, total 17.8 mils)	~300 keV	$4.5x10^6$ e/cm^2-s-sr

The net electron flux in the Teflon is

$$j1 = 6\text{-}4.5x10^6 \text{ e/cm}^2\text{-s-sr} = 1.5x10^6 \text{ e/cm}^2\text{-s-sr.}$$

Converting to normal incidence (in this case using a factor of 3, but see par. B-7),

$$j2 = \sim j1*3 = 1.5x3x10^6 \text{ e/cm}^2\text{-s} = 4.5x10^6 \text{ e/cm}^2\text{-s}$$

Converting to current in the Teflon,

$$I = 1.602x10^{-19}x4.5x10^6 = 0.72 \text{ pA/cm}^2$$

FIGURE D-1. Simple Charging Example

The assumed electron environment is worst-case GEO, as shown in Figure 6. In Figure 3, 10 mils of aluminum require 250 keV energy to penetrate the aluminum and enter the Teflon. Teflon density is 78% of aluminum (Table I, paragraph. 6.2); therefore, 10 mils of Teflon is equivalent to 7.8 mils of aluminum. Electrons with greater than 300 keV can penetrate through the 17.8 mils aluminum equivalent and exit the “sandwich.” Referring to Figure 6, worst-case flux entering the Teflon is about $6x10^6$ e/cm^2-s-sr, and exiting flux is about $4.5x10^6$ e/cm^2-s-sr, leaving a net flux of $1.5x10^6$ e/cm^2-s-sr. Equivalent normally incident flux is more than the omnidirectional flux for equivalent charge deposition. For this simple example under 10 mils of aluminum, it is taken to be a factor of three times the omnidirectional flux. Converting to current requires multiplying by $1.602x10^{-19}$ A/e-s. The net (approximate) result is that the charging rate in the 10-mil layer of Teflon is 0.72 pA/cm^2.

Per Figure 5 and paragraph 4.9, the charging rate in this Teflon sample exceeds the “safe level” of 0.1 pA/cm^2. Therefore, this sample is threatened by occasional discharges. More than 10 mils of shielding are required over this sample.

D.2 Detailed Analysis. A proper analysis can be done by a knowledgeable person, using the models and tools listed in Appendix A, to determine charge deposition rates (fluxes and fluences). Then the analysis must determine if sufficient charge exists for breakdown (ESD's).

Detailed formulations have been developed for determining the development of electric fields in irradiated insulators. In the end, for good insulators at high flux, the electric field builds up to and stabilizes at $1x10^5$ (rarely to $1x10^6$) V/cm.

The conductivity of the material is a critical parameter and generally is not known well enough to provide meaningful calculations. For proper answers, one must know the conductivity under irradiation and extended vacuum in order to perform a meaningful detailed analysis. Even then, predicting pulse amplitudes and rates is only a guess.

As a matter of comparison, a computer code was used to repeat the simple example of Paragraph D-1. The results were that the electron flux in the Teflon was computed to be about 40 percent of the result from the simple analysis. This shows that for this test case the simple analysis was conservative by a factor of 2.5. Although believed to be conservative and calculated for this case, the simple analysis should always be treated with some suspicion.

D.3 Spacecraft Level Analysis. This level of analysis is used to predict the current density within the spacecraft interior. It can use radiation analysis tools modified as required to accomplish this task. Conventional radiation analysis inside a spacecraft is carried out using transport codes that carry out three dimensional tracking of energetic particles through the spacecraft walls to a specific target. The output of these codes is the radiation dose as a function of a detector material (usually silicon). Several computer codes which use electron spectra and spacecraft geometric features as inputs can be used to determine internal fluxes or radiation dose at specific sites (see Appendix A). This is first done with only the walls and shelves in place. Once the isoflux contours are determined, then the sensitivity of the components is compared to the contours. If the predicted radiation levels exceed the component sensitivity, then a box level analysis is conducted. If the radiation level in the box still exceeds the component capabilities, then spot shielding is considered.

The criteria to be used is that the current deposited must be less than 0.1 pA/cm^2. If it is, there should be no problems with internal charging.

NOTE: To determine an approximate electron flux/fluence from a radiation transport code, a simple equivalence from dose (Rads-Si) to electron fluence has been used. Dose and fluence are related by the equation (Wenaas et al. 1979, and SD 71-770 1971):

$$\text{Fluence } (e/cm^2) = 2.4x10^7 * \text{Dose (Rads-Si)}$$

The results from this factor are almost certainly conservative (it predicts greater electron fluence than actually exists), so its use would lead to conservative design and greater costs.

APPENDIX E

TEST METHODS

Tests that can be performed to validate some aspects of internal charging problems are conceptually described below. Details of test levels, test conditions, instrumentation ranges, bakeout time, pass/fail criteria, etc., must be addressed by the knowledgeable individuals responsible for doing the tests. Vacuum bakeout/aging of materials before tests is important because resistivities increase with aging in space.

E.1 Electron beam tests. Electron beam test facilities are to be used to test smaller elements of the spacecraft. This test can be used to determine whether a material sample will arc in a given electron environment, and can measure the size of the resultant ESD, if any. Electron beam tests have the advantage that they are real: The electrons can be accelerated to energies that will penetrate and deposit more or less to the depth desired by the experimenter. They have the disadvantage that the beam is usually monoenergetic rather than a spectrum - the electrons initially will be deposited in a layer, rather than distributed throughout the exposed material. Usually the illuminated area is less than 10^3 cm^2 in size. The real area may not be testable, in which case scaling must be applied to the measured results to estimate the real threat. A typical test configuration in a vacuum chamber is shown in Figure E-1.

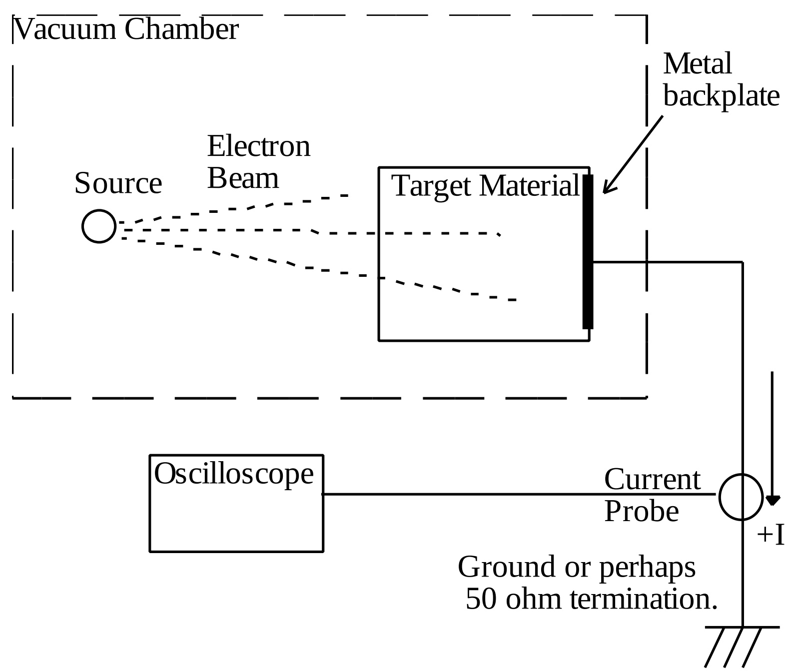


FIGURE E-1. Typical Electron Beam Test Facility Setup

The electron source must have both the requisite energy (usually expressed in keV or MeV) and the requisite flux (usually expressed as pA/cm^2, but sometimes as e/cm^2-s) [Note: 1 pA/cm^2 = 6.242×10^6 e/cm^2-s]. The target material in Figure E-1 shows a grounded back plate. Some tests may involve a front metal plate, grounded or ungrounded, to more closely simulate the in-flight hardware. In this example, the electrons, after depositing in the target material, may leak off to the back plate or they may remain in the material if its resistivity is high. If they do not leak off to the back plate (harmlessly), they continue accumulating until the internal electric field exceeds the dielectric strength of the material and an electrostatic discharge occurs.

The current probe and oscilloscope are used to determine the current waveform of the electrostatic discharge from the material. If a simple breakdown between the material and the metal backing plate occurs, the current probe can measure the discharge directly. From the waveform, the peak current, the pulse width, and the charge are calculated. If there is a 50 Ω termination, the voltage waveform can be measured and the power and energy in the discharge estimated.

The best way to test a dielectric is to use an electron beam that penetrates to the middle of the thickness. First, dry the sample in vacuum (for a month is best), then irradiate at 1 to 10 nA/cm^2 for several hours and monitor all wires. A sample not arcing after this test will be excellent in space.

Other diagnostics can be included, including a Rogowski coil to measure electrons “blown off” the front surface of the material to “space” (the chamber walls) or RF field sensors (EMC antennas and receivers) to measure the spectrum of the radiated noise.

E.2 Dielectric Strength. This number can be used for ESD analyses to determine the magnitude of the ESD. Usually, the dielectric strength (breakdown voltage) of a (dielectric) material is determined from published tables. If necessary, a test can be performed as illustrated conceptually in Figure E-2. ASTM D-3755 is a standard test method for breakdown voltage. Normal precautions are to use mechanically sound and clean samples of the material under test. Generally, for any materials involved in internal charging studies, it is appropriate to have a vacuum bakeout to remove the water. The test is intended to measure the applied voltage until breakdown. The result is the dielectric strength which is often reported as “V/mil of thickness.” The result should also report the tested thickness: ”V/mil @ thickness d.” Note that this is not the usual cause of breakdown for space ESD events, which are usually caused at interfaces.

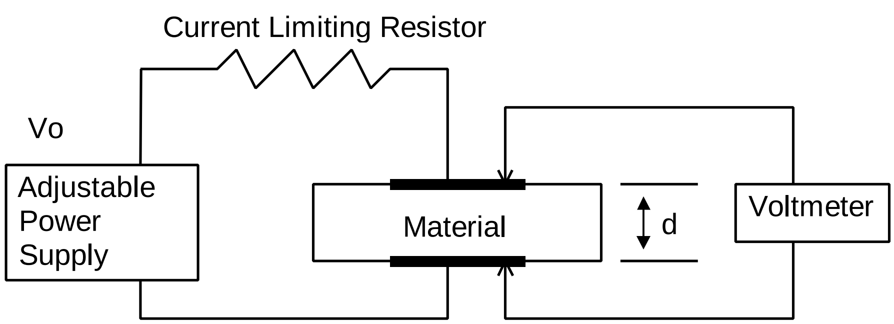


FIGURE E-2. Testing for Breakdown Voltage

E.3 Conductivity determination. Conductivity and resistivity are reciprocals of each other. Rho (ρ, Ω-m) = 1/sigma (sigma, σ, siemens: mho/m or 1/ohm-m). The volume resistivity of a material is a useful parameter for internal charging assessments. Volume resistivity refers to the bulk resistance of a volume of material. Volume resistivity is determined in terms of the equations supporting Figure E-3. If the material’s volume resistivity is not found in existing tables or the manufacturer’s data, it can be measured in one of several ways as described in the following paragraphs. ASTM D-257-91 is a standard test method for dc resistance or conductance. Paragraph 5.2.1.6 gives acceptance criteria for avoiding IESD events.

There is another resistivity, surface resistivity, that is applicable to thin layers of material or surface coatings. Surface resistivity, sometimes called ρ_s (rho sub s), is the resistance of a flat two-dimensional square material as measured from one edge to an opposite edge. It may also refer to a surface layer of conductivity on an insulator, which (if the surface has been contaminated by handling or processing) may differ significantly from the bulk resistivity. The resistance measured in this manner will be:

$$R = \rho_s \times l/w$$

where:

R = resistance (Ω)
ρ_s = surface resistivity (ohms; Ω per square)
l = length of sample
w = width of sample

For a square sample (l = w) it can be seen that the resistance from edge to edge will be the same value regardless of the size, so surface resistivity is sometimes called "Ω per square," although the proper units are simply ohms.

E.4 Simple resistivity measurement. Figure E-3 shows the concept of resistivity. The resistance from end to end of the material is:

$$R = \rho \times l / (h \times w)$$

where:

R = resistance (Ω)
ρ = volume resistivity (Ω-m in SI units); sometimes called ρ_v (rho sub v)
l = length of sample (m)
w = width of sample (m)
h = height of sample (m)

Therefore:

$$\rho = R \times (h \times w) / l$$

Conductivity (s or σ, sigma) is the reciprocal of resistivity:

$$s = 1/\rho$$

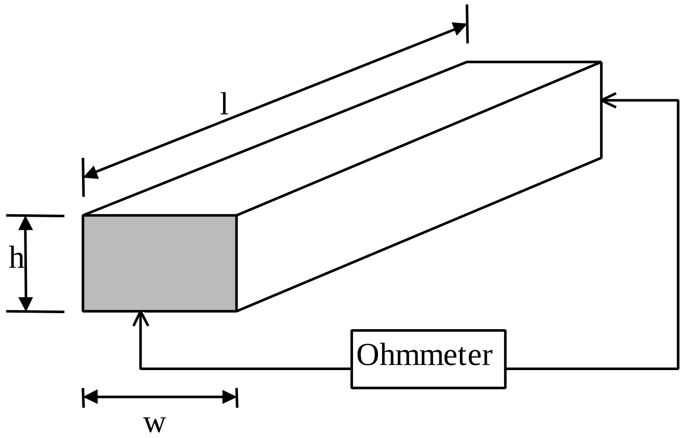


FIGURE E-3. Testing for Volume Resistivity

Various difficulties occur when measuring high resistivities, such as higher resistance than can be measured by the ohmmeter, resistivity as a function of voltage stress, resistivity as a function of temperature, resistivity modifications due to presence of absorbed moisture, and surface resistivity leakage rather than current flow through the bulk of the material. Test devices, such as the HP Model 4329A high resistance meter when used in conjunction with a Model 16008A Resistivity Cell, can account for some of these problems [Operating and Service Manual, 1983]. That instrument combination can measure very high resistances, has several user-defined test voltages, and has “guard rings” to prevent surface leakage effects from contaminating the results. The person doing the test must still “bake out” the test sample to get rid of moisture-caused conductivity. Testing versus temperature is important for cold situations (on the outside of the spacecraft) because resistance is significantly higher at cold space temperatures. For resistances above 10^{11} ohms, moisture bakeout and vacuum tests are appropriate, because moisture adsorption increases conductivity.

Exposure to radiation may increase conductivity (radiation-induced conductivity). That is, materials may have more conductivity than measured in a ground environment. The quantitative details of this phenomenon are too involved for this document but in general should not be assumed to be significant help in the IESD situation.

E.5 Electron beam resistivity test method. This method has the advantage in that it measures the material in a vacuum and in response to an electron beam applying the voltage stress. With a metal front and back plate or plated contacts (or none at all), an electron beam is directed onto the front surface of a flat sample of the material as is Figure E-4. A non-contacting voltage probe is used to measure the potential on the front surface of the material. A picoammeter then measures the current flowing from the back surface to ground. The resistivity is calculated in the manner of Figure E-3. Shielding is needed to avoid stray electron false data.

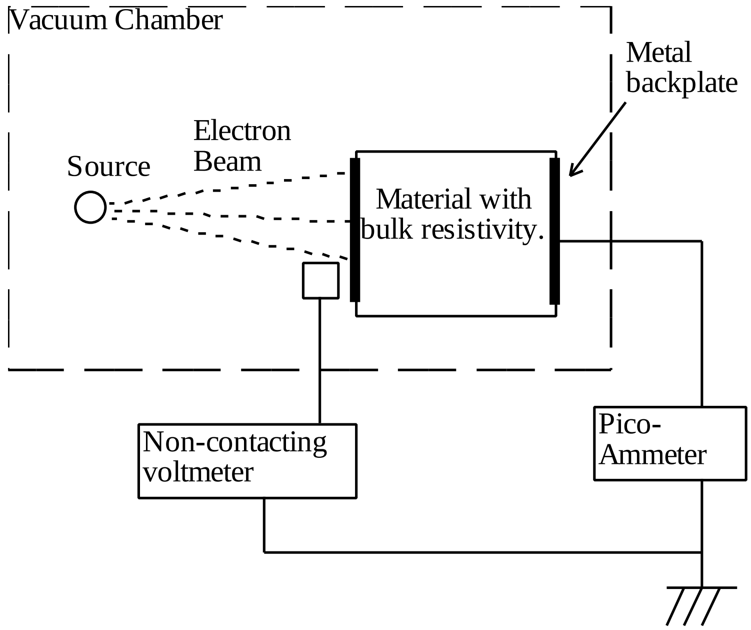


FIGURE E-4. Electron-beam Test for Resistivity

E.6 Non-contacting voltmeter resistivity test method. This method, illustrated schematically in Figure E-5, assumes that the resistivity is a constant with respect to applied voltage stress. The method requires plating the upper and lower surfaces of the material being tested to create a capacitor. The capacitance is determined and the capacitor charged. The power supply is disconnected. The voltage decay is monitored as a function of time as measured by a non-contacting voltmeter. The non-contacting voltmeter is necessary because most voltmeters have lower resistance than the test sample and would lead to incorrect measurements. The resistivity is determined by the equations given earlier and by making use of the voltage-decay versus time-curve given by the equations:

$$V = V_o \times e^{-(t/\tau)}$$

where

- t = time, seconds
- τ = R x C time constant, seconds
- R = resistance from top to bottom of the sample, ohms
- C = capacitance of the sample, farads

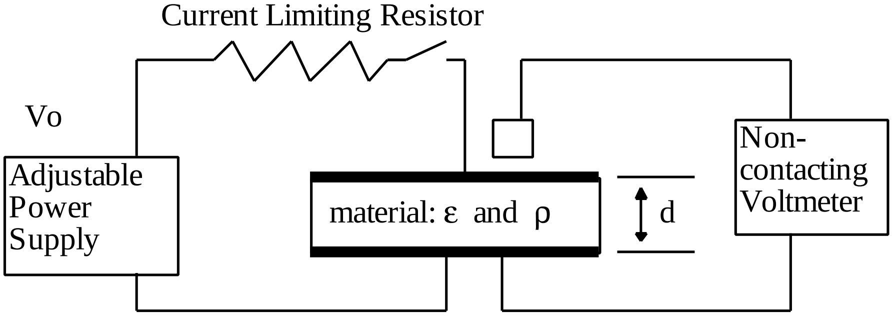


FIGURE E-5. Non-contacting Voltage Decay Resistivity Test

Problems with this method include the sample preparation (cleanliness, absorbed water, and temperature) and surface leakage around the edge; all must be properly considered. The test could be done in a vacuum chamber to reduce water absorption contamination of the sample. An electron beam, as shown in Figure E-4, can be used to charge the sample. Then turn off the electron beam and monitor the voltage decay rate.

E.7 Dielectric constant, time constant. The dielectric constant, ε, of a material can be determined experimentally, but it almost always can and should be obtained from the manufacturer. From knowledge of ε and resistivity, the material's relaxation time constant can be determined.

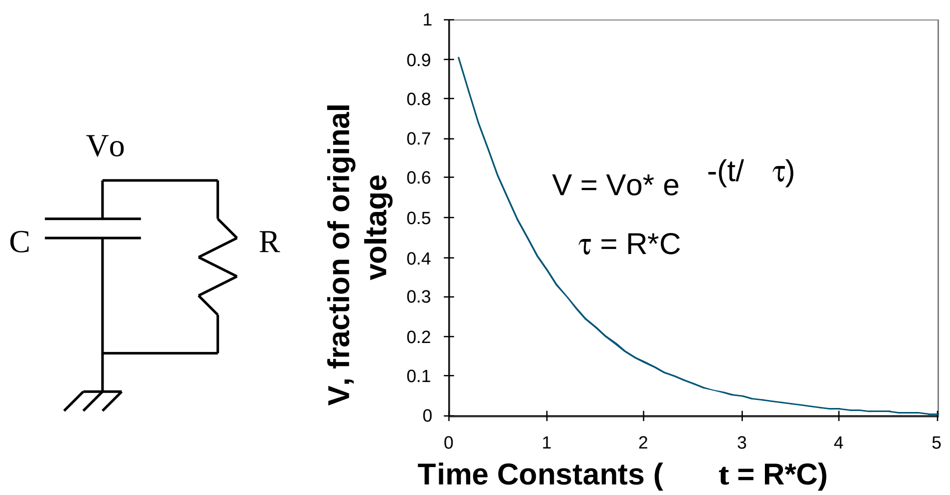


FIGURE E-6. RC Time Constants

Figure E-6 illustrates the physical arrangement for illustrating the concept of material time constants. If a rectangular slab of material as shown in Figure E-7 has metal electrodes on the top and bottom surfaces, it forms a capacitor, whose value is given by:

$$C = \varepsilon \times A/d,$$

where:

- ε = permittivity of the material = $\varepsilon_o \times \varepsilon_r$
- ε_o = permittivity of free space = 8.85×10^{-12} F/m,
- ε_r = relative dielectric constant of the material, usually between 2 and 4
- A = area of the sample = l x w (length x width)
- d = thickness, top to bottom

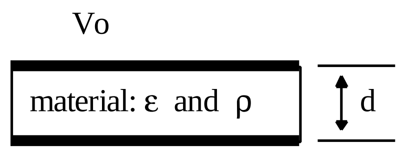


FIGURE E-7. Determining Material Time Constant (see text)

If the units are SI, the capacitance will be expressed in farads. Usually capacitance related to space charging is expressed in pF because typical values for space charging are in this range.

The leakage resistance from top to bottom of the same rectangular slab is given by:

$$R = \rho \times d / A,$$

where:

ρ = material resistivity, often given in Ω-cm

If the units are consistent, the answer will be in ohms. For the geometry in Figure E-7, it can be seen that the leakage time constant, τ, is:

$$\tau = \rho \times \varepsilon$$

The time constant τ is the time during which the charge falls by 1/e (to 37 percent of original voltage). At five time constants, there is less than 1 percent of the original voltage; at 0.01 time constant, the voltage is still 99 percent of the original. A material time constant of 1 hour or less is desirable.

Materials can thus be characterized by their time constants if both the dielectric constant and the resistivity are known. This is a theoretical description. Many high resistivity materials behave non-linearly with applied voltage or applied radiation. Thus, these concepts are introductory and approximate. For example, electron beam tests have found that the discharge time obtained when the beam is turned off (with vacuum maintained) can be hundreds of hours.

E.8 Vzap. A Vzap test is a test of an electronic device's capability to withstand the effects of an electrical transient simulating fabrication handling. It is useful when attempting to decide whether a device can withstand an ESD transient. Figure E-8 shows a typical test configuration (MIL-STD-883, Method 3015). The parameters are intended to represent the threat from a human body.

The capacitor in this layout (100 pF) is charged and the power supply disconnected (switch S1). The capacitor is then discharged (through R2 = 1500 Ω) to the device under test. Hardware is classified as Class 1, 2, or 3, depending on its damage threshold (0-1,999V, 2,000-3,999V, or >4,000V, respectively).

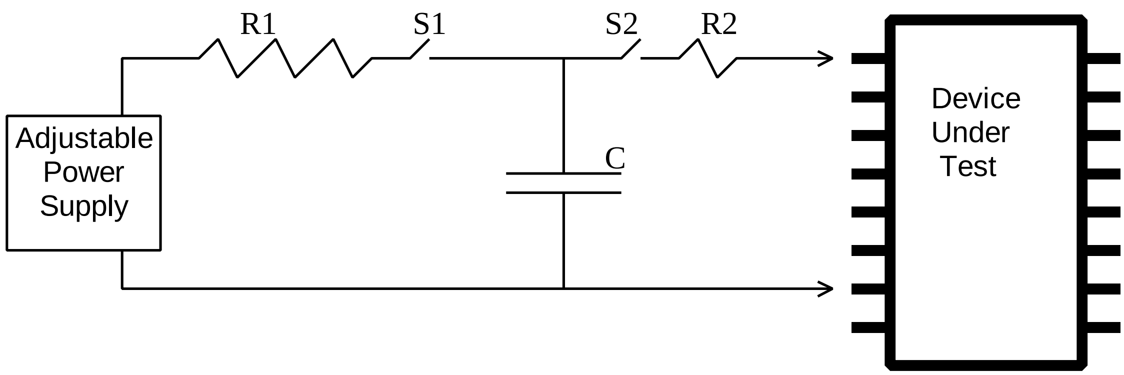


FIGURE E-8. Vzap Test Configuration

The results are only reported in broad ranges of damage threshold. Although providing some idea of the ESD sensitivity of the part, the manufacturer's data are often not as quantitative as desired. This test is mentioned because device sensitivity information may exist from the manufacturer. For actual space discharge events, the value of R2 is in the range of 10 to 100 ohms, and more likely 10 to 50 ohms.

Results obtained by Trigonis (1981) for various parts, capacitor sizes, and series resistors (R2) are graphed in Figure E-9. It illustrates how the damage threshold varies with each of the test parameters. Each point represents a different sample for the same part type subjected to a Vzap capacitor discharge at different voltages for various size capacitors. Both polarities are tested and are applied to the weakest pin pairs. The plotted lines envelope the least energy that damages any part under any combination of the variables. One feature of the plot is the existence of a minimum damage threshold for each device. This can be as low as 5 V for some newer devices. The second feature is a constant energy region at low capacitances (not obvious in this chart). The third feature is that the energy appears to go up for the lowest capacitor sizes; this may be an artifact of stray capacitance in the test fixture. It is appropriate to choose the lowest energy as the victim's sensitivity for analyses. It can be seen that for these parts, the weakest component was damaged by 0.5 µJ. Therefore, based on these test results, an ESD needs to deliver at least 0.5 µJ to damage a part. Of course, having data for the actual parts in question is more desirable.

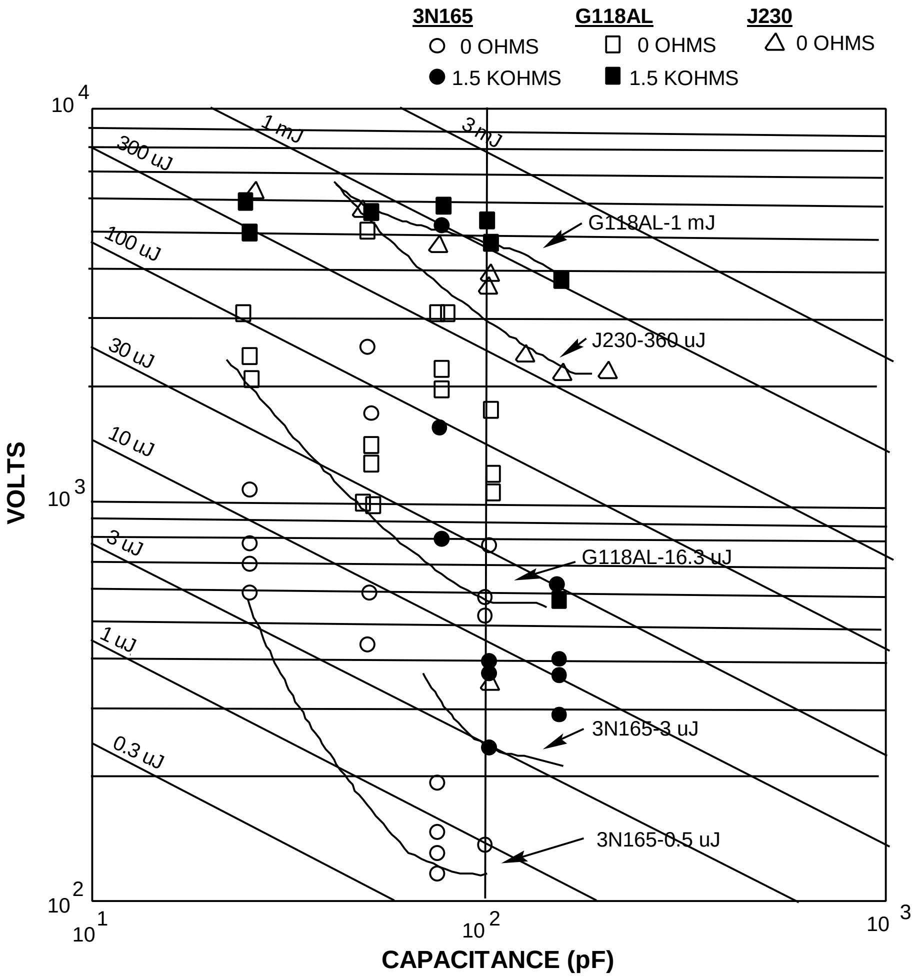
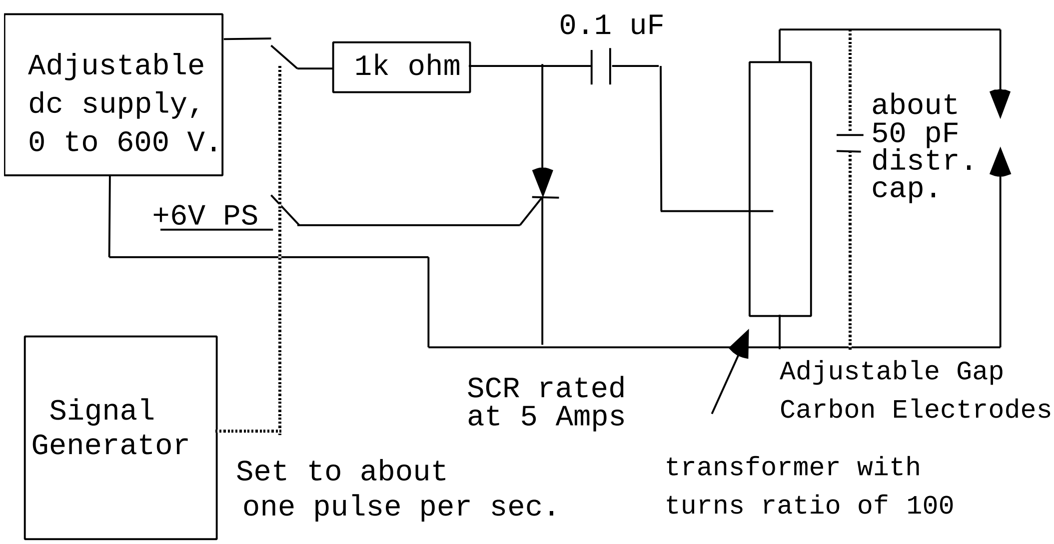


FIGURE E-9. Typical Results for Vzap Test Showing Lines of Minimum Damage Threshold for Given Parameters. Diagonal lines are for constant energy: $E = 1/2 \times C \times V^2$.

E.9 Transient susceptibility tests. Transient susceptibility tests are very common in the EMC community. EMC MIL-STD-462 shows test details. Transient injection is done by inductive or capacitive coupling per MIL-STD-462. The difference between EMC and ESD is the width of the transient pulses - the EMC pulse is typically 10 µs wide while an ESD pulse is on the order of 10 to 100 ns. A thorough and comprehensive test of a victim device would include varying the pulse width and then determining the voltage threshold of susceptibility. The test should include all pins on the victim device and both polarities of the transient. It should be applied when the input signal is in the high state, the low state, and/or transitioning states. Such a comprehensive characterization would involve more work than is usually done, but the analyst should understand that anything less will not be complete. Test details should be tailored by a knowledgeable engineer.

E.9.1 Transient pulse sources. There are two common sources for generating transient pulses for susceptibility testing. The first is the MIL-STD-1541A pulse source shown in Figure E-10. This source provides a capacitive discharge with the amplitude set by the voltage used to charge the capacitor and also the electrode separation gap.



Simplified Schematic, MIL-STD-1541 ESD Arc Source

NOTE: Typical gap spacing, voltage, and energy levels

Gap	Vb	Energy
1 mm	1.5 kV	56 uJ
2.5	3.5	305
5.0	6.0	900
7.5	9.0	2000

FIGURE E-10. MIL-STD-1541 Pulse Source for Transient Testing

The second source is a commercial human body discharge source (Schaeffner, among others, supplies one such test device). These sources can be battery operated and also provide a capacitive discharge pulse. The charging voltage is variable so that the amplitude can be controlled. Transients from this source are fast (on the order of 150 ns) and the signal is very clean as opposed to the MIL-STD-1541 noisy source.

The state-of-the-art is such that ESD test simulators should be improved, to more properly simulate on-orbit ESD pulses. The reader may research for better sources.

E.9.2 Component/assembly testing. Potentially susceptible components/assemblies should be tested for sensitivity to IESD. The component to be tested is to be mounted on a base plate and functioning. Pulses are to be injected into the component and the performance of the device monitored for upsets. The pulses used are to cover the expected range of current amplitudes, voltages, and pulse durations. It is very important that the pulse device be electrically isolated from the component being tested and the monitoring equipment.

E.9.3 System testing. There is no way to do a system level internal ESD test.

APPENDIX F

DATA SOURCES

An important issue in evaluating the likelihood of an IESD event being the cause of an anomaly is the state of the environment during the event. The following subsections contain a brief listing of satellites and sources from which energetic electron data can be obtained. Note that the names, addresses, and web sites, although current at the time of writing, are subject to change.

There are problems in attempting to obtain calibrated electron data from space, even though it seems that published data are exact. Electron detector data are sometimes affected by the presence of energetic protons that generate secondary electrons during their passage through the detector. Detectors degrade and become less efficient over time or may not be precisely calibrated to start with over all energy ranges. View factors and orientation relative to the magnetic field also contribute to uncertainties in the count rate to flux conversion. In spite of these concerns the errors are usually small enough to permit the data to be used in estimating internal charging, at least for engineering purposes.

F.1 GOES. The most readily available data set is that from the NOAA GOES series of spacecraft at geosynchronous orbit. These data consist primarily of E > 2 MeV electron fluxes expressed in $e\text{-}cm^{-2}\text{-}s^{-1}\text{-}sr^{-1}$. Starting with GOES 8, data are also available for the E > 600 keV electron environment. Data from at least early 1986 to the present are readily available. GOES satellites are generally at United States East and West Coasts, but have varied over the years. Contact Dan Wilkinson, phone 303-497-6137. Data are available in near real time over the Web at: http://www.sel.noaa.gov/today.html and http://web.ngdc.noaa.gov/stp.

F.2 Los Alamos SOPA detectors. Detectors on board various DoD geosynchronous spacecraft provided by the Los Alamos National Laboratory are referred to as the SOPA experiments. The data cover the energy range from E > 30 eV to E > 5 MeV electrons. Current data from the SOPA instrument on DoD satellite 1990-095 (37° W) and satellite 1989-046 (163.7° W) are now available. Older data are being processed and may be available in the future. The data are well calibrated and provide a more detailed snapshot of the environment than the GOES data but have not been as readily available. Contact Richard Belian, phone 505-667-9714, or Geoff Reeves, phone 505-665-3877. Their web site can be accessed at: http://leadbelly.lanl.gov/lan_ep_data/lanl_ep.html.

F.3 CRRES. Launched in 1990, the CRRES spacecraft provided by far the most accurate and detailed measurements of the Earth's radiation belts in many decades. A landmark in internal charging (it carried the first experiment specifically designed to study internal charging), it provided extensive data on the location and occurrence of IESD's throughout the magnetosphere. CRRES was launched into an eccentric, 18 degree inclination orbit that took it from below the Van Allen belts out to Geosynchronous. It had an orbital period of 10 hours and measured from a few eV to 10 MeV electrons. The primary data are from July 25, 1990, to October 1991 and include extensive measurements of internal arcing rates in addition to the radiation data. Contact G. Mullen, Air Force Phillips Laboratory, 617-377-3214 (e-mail is preferred: mullen@plh.af.mil), or Dr. G. Ginet, 617-377-3974.

F.4 SAMPEX. Launched in 1992, the Solar, Anomalous, and Magnetospheric Particle Explorer (SAMPEX) has returned a wealth of data on the low altitude radiation environment. The satellite is in a high inclination (82 degrees) polar orbit with an altitude of 520 by 670 km. Its orbit passes through many L-shells, and its data, although not from a high altitude, contains information from those L-shells. The SAMPEX Proton/Electron Telescope (PET) can provide measurements on precipitating electrons from 0.4 to ~30 MeV over the polar regions. Contact Dr. Dan Baker, phone 303-492-0591.

F.5 Other Sources. The NASA International Solar-Terrestrial Physics (ISTP) program has several satellites in orbit that can be useful, but the data availability is uncertain. A web site is: http://www-istp.gsfc.nasa.gov.

Often when carrying out after-the-fact anomaly investigations, it is desirable to quickly determine what the state of the electron environment was during the event. No appropriate electron flux data may be available for either the period or the particular spacecraft orbit. In that case, possible secondary sources are the geomagnetic indices or anomaly data from other spacecraft in orbit at the same time. These data are also of value as support material in carrying out anomaly investigations, as they may allow a separate determination of other causes such as surface charging or SEUs as potential causes. The National Oceanic and Atmospheric Administration's World Data Center (WDC) at Boulder, Colorado, provides geomagnetic indices on a near real-time basis and maintains a spacecraft anomaly data base. This material can be addressed through the web at: http://www.ngdc.noaa.gov/wdc/wdca/wdca_solar.html.

Increasing interest is being shown in the development of a simple "universal" electron detector for flight on commercial spacecraft to monitor the internal charging fluxes. As of this date, one such device is being flown by INTELSAT, and it is hoped that more opportunities will become available. If a net of such sensors should become available, it might be possible to provide real-time measurements of the state of the Earth's radiation environment for monitoring internal discharging effects.

APPENDIX G

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