# SPACECRAFT CHARGING CONSIDERATIONS IN SPACECRAFT DESIGN

## Introduction and Background

Spacecraft charging immunity has been an important spacecraft design consideration since the early 1970s when it was discovered that in-orbit operational anomalies were correlated with activity in the ambient energetic plasma environment. It was discovered that the anomalous triggering of DSCS-II logic states occurred preferentially in the midnight-to-dawn local time sector during magnetic substorm activity. Inquiries to other geosynchronous satellite programs confirmed that numerous electronic anomalies had occurred in the same local time sector[1]. Spacecraft charging was suspected as the source of the false commands since the density of energetic plasma electrons increases significantly during magnetic substorm activity in the midnight-to-dawn local time sector[1]. Further evidence of the spacecraft charging hazard was amassed when the total loss of the payload power on DSCS-II Flight 1 on June 2, 1973 was correlated with unusually violent substorm activity as observed with ground-based magnetometer records (Figure 1) and with the UCSD ATS-5 electron-proton spectrometer data[2].

Prior to the time that the real hazard to spacecraft from environmental charging was recognized, the exterior surface configuration of spacecraft was principally defined by thermal considerations. Thermal blankets, optical surface reflectors (OSRs) and various paints constitute about 75% of the external surface. Together with solar cell cover glasses, used to protect solar cells from degradation by proton bombardment, roughly 95% of the external surface area has been comprised of materials which are very good electrical insulators. The differential voltage built up on these dielectric surfaces relative to nearby or underlying metallic surfaces is the main source of the electrostatic discharges (ESDs) which cause the in-orbit anomalies. Underground metallic surfaces constitute an even greater ESD hazard, because of the larger peak arc discharge currents involved, and it is essential that these be eliminated by proper grounding techniques.

As expectations of future spacecraft performance become more ambitious, as in the consideration of higher power (voltage), larger physical size, longer mission lifetime and different orbits, spacecraft charging interactions with the space plasma are of increasing importance. Knowledge presently being gathered will be applied to produce the design techniques critically needed for operating under the more strenuous projected conditions.

## Technical Basis and Other Considerations

In response to the occurrence of operational in-orbit anomalies, a spacecraft charging technology is being developed, with many conferences and workshops being held[3,4] to disseminate the large amount of resulting technical information. Experimental laboratory studies and associated theoretical studies comprise most of the work to date. Aside from the operational anomaly information only a limited amount of applicable in-orbit engineering data is available, mostly from the P78-2 spacecraft of the Air Force SCATHA program. Instrumentation of operational spacecraft, especially those of a generic series, would be an ideal method of gaining valuable design information; generally, however, this has not been done.

The evolutionary nature of the current state of the spacecraft charging technology must be emphasized[5]. Current studies show that high-voltage (~15 kV) charging, the subject of a great deal of early study, probably is not the cause of the observed anomalies. High-voltage charging was not observed on the P78-2 satellite, and is not predicted by spacecraft charging computer analysis codes. Laboratory studies show that high voltages are required to produce large-area dielectric arcs, originally thought to be the principal mode of arcing. Thus the early work on large area high voltage (~15 kV) arc discharges must be de-emphasized, and other phenomena such as low voltage (~1 kV) reversed polarity (negative metal-positive dielectric) effects[6] must be considered. Furthermore, anomalous commands do occur outside the midnight-to-dawn local time sector, indicating the importance of mechanisms such as penetrating high energy (>50 keV) electron effects,[7] independent of local time sector.

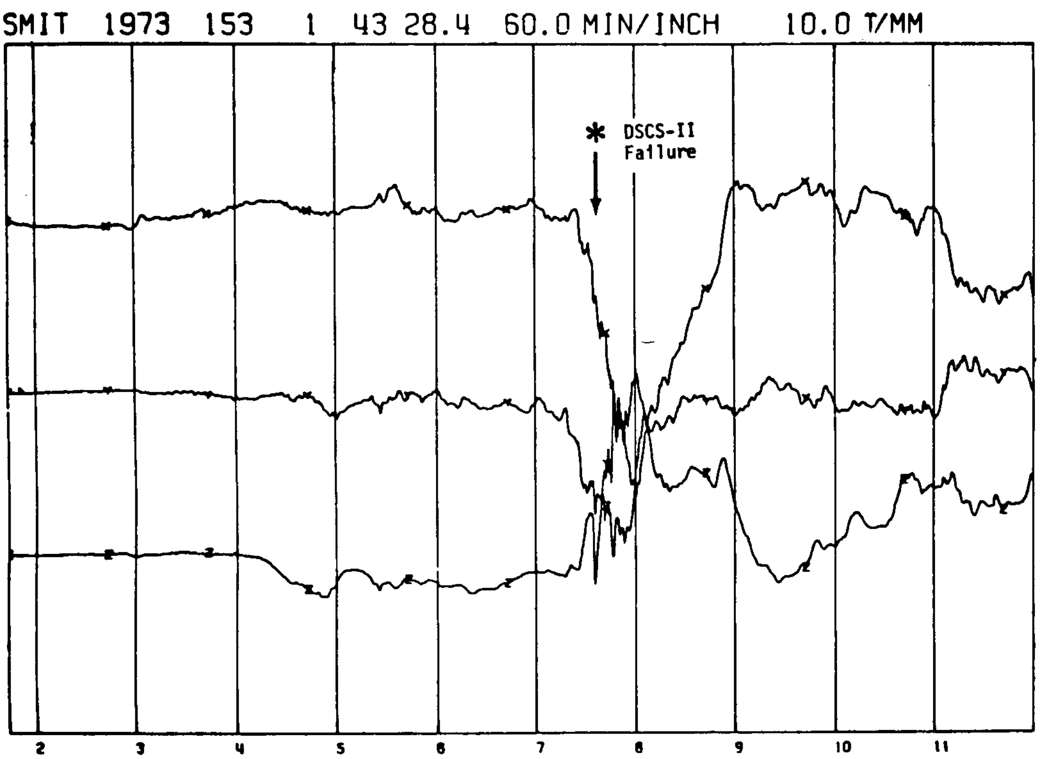


**Figure 1.** All three components of the magnetic field recorded at Fort Smith, N.W.T., Canada, during the first part of the day on which the DSCS-II FLT 1 spacecraft ceased operation, June 2, 1973.

## Design Approaches — Charge Control

Design approaches to reduce the hazards due to spacecraft charging fall into two broad categories: spacecraft charge control, and post discharge hazard reduction. In the first category the amount of charge on spacecraft surfaces is controlled so that discharges do not occur or are significantly reduced in energy, thus eliminating the hazard to the spacecraft. In the second category, the discharge is permitted to take place, but the spacecraft equipment (components, circuits, etc.) are immunized to the effects of the discharge. A cost effective spacecraft charging design approach should consider both of these methods.

Figure 2 summarizes the approaches to these spacecraft charging countermeasures. The first step in spacecraft charging control for a particular satellite design is to determine surface charging using a computer analysis program. Computer codes may be as simple as TSCAT (TRW spacecraft charging analysis technique) or as complex as NASCAP (NASA charging analyzer program). TSCAT is an equivalent electrical circuit analysis program which calculates surface potentials; NASCAP goes beyond this and includes off-surface potentials. The computer codes point to areas of spacecraft charging susceptibility inherent in the design. Control of the spacecraft charge can be performed by active or passive methods. To actively control the charge on the spacecraft, electrons and/or ions are emitted from the spacecraft by means of a powered device such as a thermionic emitter. Experimental versions of active control devices have been flown but have not yet been developed for use on operational spacecraft.

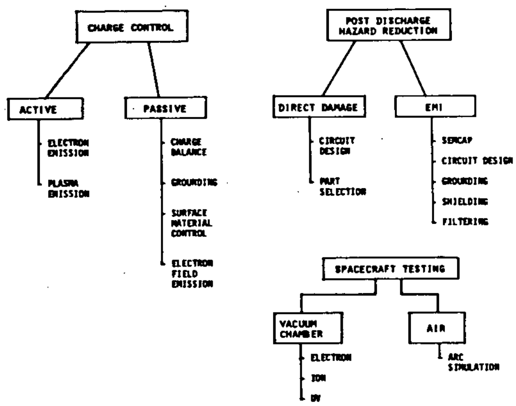


**Figure 2.** Spacecraft Charging Countermeasures.

Present methods of passive surface charge control include a large number of techniques. Some of these techniques, such as grounding, charge balancing, and material selection are used routinely in flight spacecraft programs whereas others, such as the passive field emitter, have been reported in the literature but not yet demonstrated.

Grounding is the one most important countermeasure for charge control and prevention of spacecraft arcs. The avoidance of large isolated conductors exposed to the environment is imperative to prevent high energy metal-to-metal arcs. This includes conductors inside of spacecraft which are exposed through spacecraft apertures. Thus a primary countermeasure is to electrically connect to structure all large (>25 $cm^2$) isolated conductors.

To prevent arcing between the metallic layers of thermal blankets, the various layers are frequently tied to structure. However, grounding of the metallic layers of thermal blankets cannot prevent arcing of the outer (usually Kapton) dielectric surface. The grounding of the metallic film on thermal blankets, usually vacuum deposited aluminum (VDA), has been a recommended practice in spacecraft design for EMC for a long time. However, the need for groundstraps from the metallic film to carry the many amperes associated with the outer layer arc discharges greatly increases the requirements on the durability of these straps. Various techniques have been used to connect groundstraps to the metallic films. Figure 3 shows four different techniques used at TRW. Tests performed at TRW[8] show that the relative durability of the different

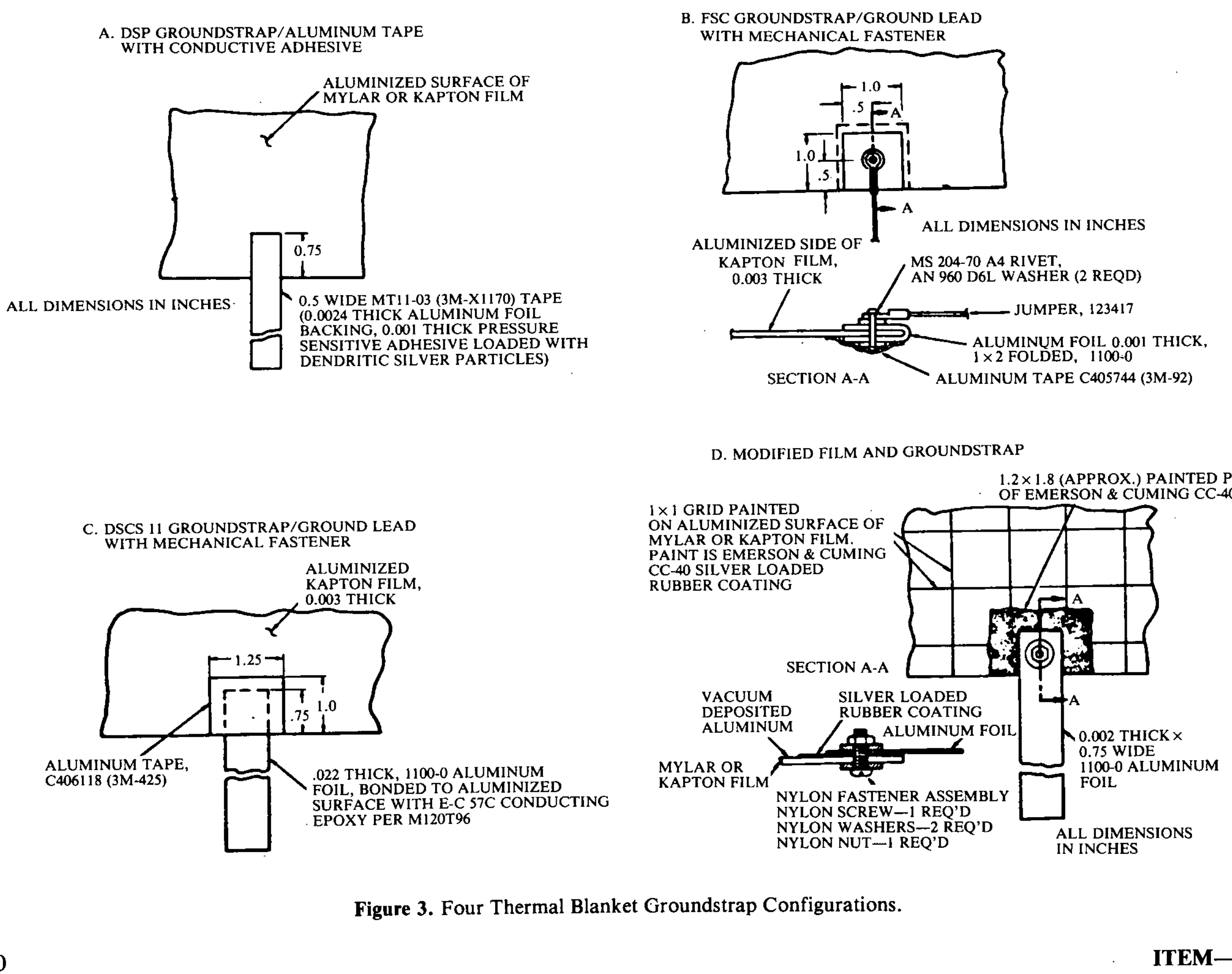


**Figure 3.** Four Thermal Blanket Groundstrap Configurations.

groundstrap configurations to standardized pulses of 100 amperes peak and 1 microsecond decay time constant shows a wide variation, from less than 50 to greater than 10,000 pulses, before burnout.

The test results are summarized in Table 1. The number of pulses required to cause the groundstrap to open-circuit seems to depend on the peripheral length of the contact between the metallizing VDA film and the aluminum foil of the groundstrap itself. These results point out the futility of using the standard EMI grounding techniques for arc discharge prevention without careful consideration of the unique requirements imposed by the spacecraft charging phenomena.

The properties of the surface materials of the spacecraft play a major role in the determination of the floating as well as the differential potential of the spacecraft in the charging environment. The resulting potentials will depend upon the conductivity and the dielectric properties of the surface as well as its secondary emission, photoemission and backscattering properties. Therefore, surface material selection is to some extent part of every charge control technique. The principal problem common to all these techniques is to select materials that have the desirable properties from the point of view of spacecraft charging, which at the same time have satisfactory thermal properties and can withstand the space environment.

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|---|---|
| DSP | 20 to 60 pulses |
| FSC | 40 to 100 pulses |
| DSCS II | 600 to 1200 pulses |
| Modified | Greater than 10,000 pulses |

**Table 1.** Number of Pulses to Burn Out Various Groundstrap Configurations.

A much-discussed method of controlling the charge on the spacecraft surface is to cover the entire surface with conducting material, to eliminate differential charging. This method was used on the Voyager spacecraft. In that case, most of the spacecraft surface was coated with a black Sheldahl conducting paint which has a resistance of about $10^6$ ohms per square corresponding to a resistivity of $10^3$ ohm-cm for a coating thickness of 0.4 mil. With the elimination of solar cells by the use of a radioisotope thermal generator (RTG) power source, only a small portion (2%) of the external surface was dielectric. Even in this case, however, the possibility of dielectric-to-metal arcs at Jupiter was not completely eliminated[9]. Several materials have been developed which can be used as electrically conducting paints for spacecraft which do not seriously compromise the thermal radiative properties of the surface and are spaceworthy. Greater conductivity is required of paints which will be used over insulators than those over conductors. A related but more serious problem with conductive coatings arises if the coating must also be transparent for use on solar cell coverglasses or OSRs. The European Space Agency satellites GEOS and HELIOS used a conductive solar array coating of antimony-doped tin oxide.

## EMC Design

Design measures to reduce the threat of an environmentally induced arc discharge by controlling the chargeup of the spacecraft were discussed in the previous section. In this section we will examine the countermeasures that can be taken to prevent an arc discharge that does occur from interfering with the spacecraft operation. The hazards of the arc discharge are direct damage to components and material by the arc, and electrically induced degradation or interference. In the category of direct damage we have included contamination by the arc by-products, e.g., contamination of optical surfaces by materials expelled in an arc discharge. Very little can be done to reduce the threat of direct damage to components once the arc has occurred, apart from removing the sensitive component from the region where arcing might occur. The probability that an arc to a cable would damage an electronic component can be reduced by shielding the cable and tying the shields to ground at both ends and at intermediate points. Another precaution that should be taken is to design circuits for maximum threshold for burnout or provide circuit protection. It is often easier to remove sensitive components from regions where high stress might occur. This was actually done on the Voyager spacecraft where thermistor wires near the dielectric low gain antenna support cone were removed to prevent arcing to the cables. Similarly, in the case of surface contamination due to arc products, the recommended practice is to reduce the arc intensity using the methods of the previous section.

The hazard due to the arc electromagnetic radiation can frequently be reduced and/or eliminated using standard EMI control techniques. This is most successfully peformed if the arc discharge electromagnetic signal is characterized and if the susceptibility of the various elements (receptors) in the system are identified.

The SEMCAP electromagnetic compatibility analysis program, developed and maintained by TRW, is used for predicting whether circuits will be upset by the arc discharge interference[10]. SEMCAP identifies incompatibilities in the sensitivities of circuits to electromagnetic energy and the onboard source of that energy. SEMCAP was originally designed to identify incompatibilities between various onboard circuit receptors and circuit EMI generators. It has been modified so that it can accept coupled electromagnetic pulses from arc discharges as a generator. Figure 4 shows an overview of SEMCAP.

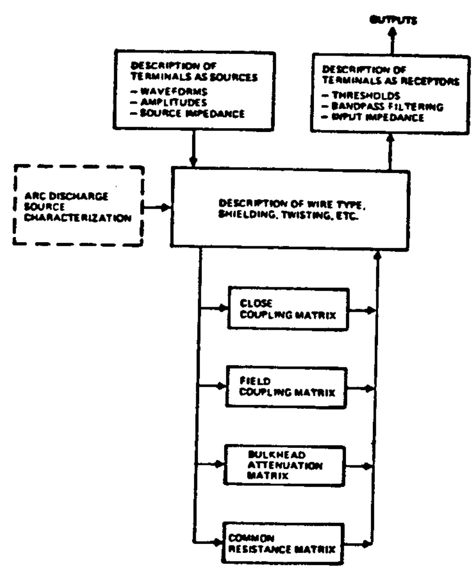


**Figure 4.** SEMCAP Overview.

Once the SEMCAP program has identified circuits which can be disrupted by the electromagnetic radiation associated with the arc discharge, a variety of fixes can be applied to the susceptible circuits. The following are standard EMI/RFI minimization techniques:

1. Place the circuit in an RF-tight assembly.
2. Electrically bond the assembly to the spacecraft structure.
3. Shield the cables to and from the assemblies and ground as frequently as possible.
4. Make the acceptance bandwidth of the circuits only as wide as necessary.

If these precautions have been taken, then special precautions could be taken, such as:

1. Design circuits with the maximum possible trigger threshold. Consider the use of relays rather than solid state switches.
2. Use command and data line interface circuits that provide protection against short high-level transients.
3. Design circuitry for minimum sensitivity in the frequency range up to 400 MHz.
4. Consider the use of differential circuits for common mode rejection.

## Design Validation

Design validation is the process whereby the effectiveness of the features incorporated in the design are verified. The procedure may include a combination of analyses and tests. Tests are preferred in that defects in the manufacturing process are detected in addition to design flaws. Qualification model tests as well as flight model tests are performed. Unit level, subsystem level and integrated system (spacecraft) tests may be required. The conceptual design and actual performance of these tests must be coordinated with the programmatic flow of the production schedule. The spacecraft charging specialist must be as intimately involved in the validation phase as in the design phase. The in-orbit performance of the spacecraft provides the final validation.

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