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# 1. System Aspects of Spacecraft Charging

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Satellites come in a variety of sizes and configurations including spinning satellites and three-axis stabilized satellites. All of these characteristics have a significant effect on spacecraft charging considerations. There are, however, certain fundamentals which can be considered which indicate the nature and extent of the problem.

The global positioning system satellite will serve to illustrate certain characteristics. Each of the two solar panels has the potential for charging and discharging on the front surface which consists of 12 mils of cover glass over the solar cells. The body of the spacecraft has a variety of surfaces. Some areas consist of multiple layer thermal blankets which are typically about 2 mils of dielectric such as Kapton on the outer surface with a layer of vacuum deposited aluminum or silver on the back surface. Another portion of the body has panels of second surface mirrors which are approximately 2 mils of cover glass backed by a thin layer of aluminum or silver. Each of these surfaces becomes a capacitor when electrons are deposited on or near the front dielectric surface.

As others have indicated, under certain conditions these surfaces will charge to the breakdown voltage and then punch through the dielectric and/or discharge around the periphery. The principle effect is to induce spurious signals in the cables between electronic boxes with the possibility of upsetting the electronics or burning them out. With the exception of two possible cases, our problems have been with upset rather than burnout. These problems range a.l the way from the nuisance of having to reset the satellite from the ground to cases where the satellite was almost lost before necessary diagnosis and recommand could be accomplished. The three obvious ways of eliminating this problem are to

(1) eliminate the charging/discharging,

(2) prevent signal coupling into cables by RF shielding of cables and/or satellite housing, and

(3) design electronics which are immune to upset.

Although technology developments may result in materials which eliminate the charge/discharge problem, these are not currently available and we must use other techniques at this time. Utilizing rf shielding may result in a greater weight increase than is practical or necessary. The preference at this time is to design immunity into the electronics. In general, there is no particular weight increase and in the case of military satellites, which are hardened for nuclear effects, the same design fixes may protect the satellite from x-ray ionization, SGEMP, EMP and spacecraft charging.

Such fixes can be based on pulse amplitude or pulse duration. For a variety of reasons, it is preferable to design fixes based on pulse duration. Fortunately, the pulse duration of these effects is similar.

In addition to designing protection into the satellite, it is desirable to conduct tests to verify the fixes. There is a wide range of testing possibilities ranging from testing an entire satellite in a vacuum chamber while subjecting it to a plasma all the way down to moving a Tesla-coil along the spacecraft cables and monitoring for upset.

The first possibility has the disadvantage of relatively high cost and schedule impact on the program. The second possibility is the most convenient and has the smallest impact. However, it is difficult to achieve realistic simulations from the standpoint of signal amplitude and coupling characteristics.

Our current thinking is in terms of testing an entire operating satellite in an ambient environment while subjecting it to simulated discharges and monitoring the satellite operation with the test equipment.

In an actual case, the discharge will be distributed over some portion of a surface and result in both a radiated field which can couple into electric cables and also a current which flows through the structure and grounding paths of the satellite. This latter current also can couple into cables which are near grounding paths such as structural members. Due to difficulties in obtaining a discharge over a large area, our current thinking is in terms of using a point source discharge and grounding to the nearest ground point in an effort to simulate both radiated and conducting conditions. It is further anticipated that the discharge would be located at various points near the center and periphery of capacitive surfaces.

The principle problem in connection with this testing is knowing how large a surface will discharge. Tests conducted by NASA-Lewis on thermal blankets, solar arrays, and second surface mirrors indicate that areas up to at least 1 sq ft may discharge at one time with a pulse duration of approximately 400 nsec. Tests have not been made with larger areas. We have satellites with solar arrays greater than 100 sq. ft in area and with thermal blankets greater than 100 sq. ft. If one considers the capacitance of such surfaces, the breakdown voltage (which is typically around 8000), assumes that the pulse width remains no greater than 400 nsec and that the entire surface discharges, the result is a peak pulse of a approximately 100,000 A. It goes without saying that suggesting such a test to a program office on a multimillion dollar satellite will not result in immediate approval. It is also true that there are reasons to believe that we have not been getting discharges of total capacitor surfaces in current satellites or we would have had much more trouble with burning out electronics. Our interim thinking in this area is to assume discharge of areas no greater than 10 sq. ft and to also assume that if the area increases from 1 sq. ft to 10 sq. ft, the pulse duration increases proportionately. Therefore, a typical test would result in pulses of 1000 A with a 400 nsec to 4 μsec duration and a breakdown voltage of approximately 8000.