Solar Array Arcing Mitigation for Polar Low-Earth Orbit Spacecraft

Grant Bonin[1], Nathan Orr[1], Robert E. Zee[1] and Jeff Cain[2]

[1]UTIAS Space Flight Laboratory
4925 Dufferin St., Toronto, ON. Canada M3H 5T6; +1-416-667-7993
gbonin@utias-sfl.net; norr@utias-sfl.net; rzee@utias-sfl.net

[2]Com Dev Ltd
155 Sheldon Dr., Cambridge, ON. Canada N1R 7H6
jeff.cain@comdev.ca

ABSTRACT

Electrostatic discharge for polar low-Earth-orbit (LEO) spacecraft is a relatively new and unexplored issue. Discharge mechanisms for LEO spacecraft are significantly different from those encountered in high Earth orbits, and seemingly few designers of new, high-voltage small satellites are aware of the differences between the two environments. Polar-LEO spacecraft encounter both plasma-induced arcing risks (at equatorial latitudes) as well as differential surface charging risks (over auroral zones): two different issues that require very different design techniques to address. There do not appear to be any comprehensive guidelines in the open literature that polar-LEO spacecraft designers can use to avoid the potentially catastrophic risk of arcing in high-voltage satellites.

The issue of spacecraft charging and electrostatic discharge (ESD) in the low-Earth orbit environment is discussed, in the context of satellite power system design. Options for controlling spacecraft charging and for preventing trigger and sustained arcs between high-voltage conductors are presented. These guidelines have been used to size solar panels for the upcoming Canadian Maritime Monitoring and Messaging Microsatellite (M3MSat)—a highly capable mission with relatively high power demand—which is used as a design example. It is concluded that ESD issues for polar LEO spacecraft are both challenging and subtle, and demand careful attention from engineers early in the design process.

INTRODUCTION

Until recently, the majority of spacecraft power systems have used photovoltaic / battery (PV-battery) systems to supply a standard 28VDC avionics bus (1).This has generally been driven by the desire to leverage pre-existing standards and practices from the aircraft industry. In particular, micro-spacecraft have employed either standard 28V bus designs or lower, often due to battery and solar array size limitations. As a consequence, solar array and structural degradation due to arcing has been a non-issue except under the most extreme circumstances (i.e. geomagnetic substorms for polar spacecraft).

However, over the past few decades, increasing demands on spacecraft power systems have driven design engineers to ever-higher solar array operating voltages, for ever-decreasing spacecraft sizes. The reason for this is two-fold: first, increased solar array and bus voltages reduce Ohmic (I^2R) losses in harness, or permit reduced harness mass in order to carry a given current load. In many cases, the power or mass savings can be substantial. Second, some spacecraft functions require high voltages, such as Hall and Ion thrusters. It is generally the case that specifying a bus voltage at or near those required for such equipment is a simpler proposition than stepping voltage up at reasonable efficiency (2).

Unfortunately, the use of high-voltage solar arrays in low-Earth orbits (which for the purposes of this paper are classified as 200-1000km altitude) introduce risk associated with solar array and structural arcing. Worse yet, the mechanisms that cause arcing in LEO—in contrast with those in GEO—are directly related to the use of high-voltage systems (2). High-voltage power systems operating in a plasma environment can lead to arcing, power drains, and destruction of spacecraft coatings (2), all of which can be detrimental or catastrophic to the mission. For polar LEO spacecraft, the situation is exacerbated by passage over auroral zones, which can furthermore subject satellites to ovals of electron streams and arcing risks more typically associated with GEO spacecraft.

A standard for protecting polar LEO spacecraft from solar array and structural arcing does not yet appear to exist in open literature, and this paper in no way claims to represent one. Instead, the purposes of this work are two-fold: (1) to make designers of small satellites aware of the risks associated with high-voltage power systems; and (2) to provide a case study of a microsatellite high-voltage power system designed to preemptively address the arcing risk. As small spacecraft continue to increase in capability—and perforce, power requirements—designers of new systems ought to be aware of the risks associated with high voltages in the LEO environment.

THE ORBITAL ENVIRONMENT

When energized conductors are exposed to plasma, positive surfaces collect electrons and negative surfaces collect ions. Potential distributions which drive charge movement are governed by the Poisson equation:

$$\nabla^2\varphi = -4\pi\rho$$

Where φ is the potential and ρ is the charge density (1). When the charge density is low, such as in high-altitude orbits, Poisson's equation reduces to Laplace's equation.

Ionospheric plasma is highly dynamic, and varies significantly over the course of hours to weeks. Variability with latitude is dramatic, and high-latitude regions are particularly difficult to understand or model (1).

When spacecraft interact with a plasma environment, several issues can result (1):

- *Floating Potential Shifts* – Some parts of the spacecraft can be charged to extremely negative voltages relative to the ionosphere. In LEO, the extent of this potential shift is bounded by the system voltage; in GEO, there is no such bound.
- *Parasitic Power Drain* – spacecraft can lose power directly because of current collection from plasma. This loss may constitute several percent of total generated power.
- *Sputtering* – surfaces that charge negatively with respect to the plasma environment will attract ions, which can result in material sputtering.
- *Arcing* – negative surfaces can experience electrostatic discharge, either from surface-to-surface or directly into the plasma, when some critical threshold is exceeded.

Mechanisms for surface charging in different spacecraft orbits are described in this section.

Geosynchronous Orbits

Issues associated with spacecraft surface charging and electrostatic discharge (ESD) were first identified for satellites in geosynchronous orbits (GEO). In general, GEO orbits are characterized by interactions with electrons having energies greater than 1keV (3). During quiet conditions, electron current is less than the photoelectron current, and there are no serious charging risks for spacecraft; however, during storm conditions, electron currents can exceed photoelectron currents and the risk of electrostatic discharge increases for solar arrays (3). If spacecraft surfaces are insulated from each other and electron flux is high, differential surface charging and ESD risks result. Insulator parts, such as solar array coverglass, adhesive, or facesheet can accumulate strongly negative potentials, occasionally on the order of 1 kV or more (3).

Low-Earth Orbits

Spacecraft charging and ESD in LEO is caused by completely different mechanisms than in GEO, and has not been a well-understood mechanism until relatively recently. In contrast to GEO, LEO surface charging and ESD risk is directly associated with high system operating voltages.

Unlike GEO, the LEO environment is characterized by relatively low-energy yet dense plasma, with particle densities on the order of 10^8 to 10^{12} m^2 (3). The issue for spacecraft in LEO arises because surfaces exposed to this plasma will charge to whatever potential is necessary for the net current flow to be zero (i.e. to achieve equilibrium). Thus, a current loop will form that uses the ionosphere as part of the conducting medium. Since electrons have much higher mobility than ions, they are collected more easily by positively charged surfaces—and as a result, high potentials can develop very quickly on spacecraft with high operating voltages. Solar arrays at high potential with exposed interconnects, cell edges, or power traces will collect negative charge, and surfaces that charge more than 100V negative with respect to the plasma environment are subject to arcing, which can be either plasma arcs or arcs to adjacent conductors. These arcs can be momentary discharges or sustained discharges, the latter being the case when current and voltage are above threshold values. Thresholds for so-called trigger arcs are not well understood, but can be as high as -55V.

Polar Low-Earth Orbit

The situation for polar LEO (PLEO) spacecraft is further complex, since PLEO satellites will encounter the

LEO environment in low-latitude portions of the orbit, and a GEO-like environment when passing over the poles. At low latitudes, the PLEO spacecraft encounters the relatively dense LEO plasma. Electrons stream quickly to high voltage surfaces, but less-mobile ions are unable to stream to negative surfaces with comparable flux. Thus, high-voltage surfaces come to the plasma potential and the spacecraft chassis (which is typically connected to system ground) floats negative with respect to the surrounding environment. Then, over polar latitudes, the PLEO spacecraft encounters streams of auroral electrons with energies > 1keV coexisting with the low-energy ionospheric plasma. As in GEO, in some cases the auroral electron current can exceed all other current sources and drive the spacecraft body potential even more negative with respect to the plasma environment, exacerbating the situation. The spacecraft is subsonic with respect to electrons in the ambient plasma, and thus, any surface not in the spacecraft wake can collect negative charge (1). Conversely, the spacecraft is supersonic with respect to ions, and thus, ion collection only occurs in the ram direction. These effects can result in the accumulation of very high potentials.

DIFFERENTIAL CHARGING IN LEO/PLEO

For LEO and PLEO spacecraft with negatively grounded power systems (i.e. conventional systems), the entire vehicle structure can float negative with respect to the ionosphere, with the chassis potential able to go as negative as the system operating voltage—put another way, a spacecraft with a 100V array can see its structure float negative almost 100V with respect to the surrounding plasma environment.

This may appear counterintuitive, but is effectively illustrated by the following example: suppose that two conducting spheres, attached by a suitable conductor, are placed in the LEO plasma environment. Under these circumstances, both spheres will collect charge, and equilibrate to within a few volts of the surrounding medium. Similarly, if these same two conductors are connected to the positive and negative terminals of a floating 100V battery on the ground, they will each see half the battery potential with respect to their surroundings (i.e. one conductor will be at +50V, the other at -50V referenced to their surroundings). However, this situation is very different in the plasma environment: because electrons are collected more readily than ions, the sphere connected to the positive battery terminal will collect electrons easily, whereas the sphere connected to the negative terminal will struggle to collect ions. Experiments show that around 90% of the battery voltage will appear on the negative sphere as a result, with only 10% on the positive sphere with respect to the plasma potential. This has profound implications: on the international space station, for example, the solar array system voltage is approximately 160V on average, and the station structure thus can float more than 140V negative of the ionosphere.

SPACECRAFT ELECTROSTATIC DISCHARGE

The propensity for a high-voltage system to arc will depend on the system breakdown voltage (the voltage required to initiate an arc, which depends on the plasma flux density, system bias voltage, insulation, and arrangement of the solar array). Arcs can be transient or sustained, depending on the characteristics of the arc site and power system.

Solar Array Arcing

Solar array arcs result from strong local electric fields, which can form easily on a spacecraft in the LEO environment. The most common source of a solar array arc is an exposed cell interconnect, which (contingent on location in the string) can be at very high potential indeed. The most serious arc scenarios are those which occur at so-called *triple points* (1), where insulator, conductor, and plasma all meet. For a solar cell in LEO, this is generally at the interconnect, but can also be the edge of a solar cell near the substrate or coverglass. Arcs have been observed at relatively low potentials (~ -75V) between conductors in the presence of plasma.

There are two basic kinds of solar array arcs: fast transient arcs (primary, or *trigger* arcs) and sustained (continuous) arcs. Trigger arcs are characterized by a brief discharge generally on the order of microseconds. The energy that can be discharged is related to available capacitance, which may vary from a single array string to the entire spacecraft, depending on design (1). While EMI can result from such arcs, they are not generally associated with significant permanent damage for small spacecraft (in contrast to larger spacecraft, where the energy available from system capacitance can be very large).

Sustained arcs can cause substantial damage to solar arrays, and can lead to their total destruction. Sustained arcs are precipitated by trigger arcs, and are enabled by the solar array being able to feed sufficient current at the arc site that a sustained discharge occurs. Sustained solar array arcs can be catastrophic. Figure 1 illustrates a sample from the ESA EURECA mission, recovered by the space shuttle, after sustained arcing (1):

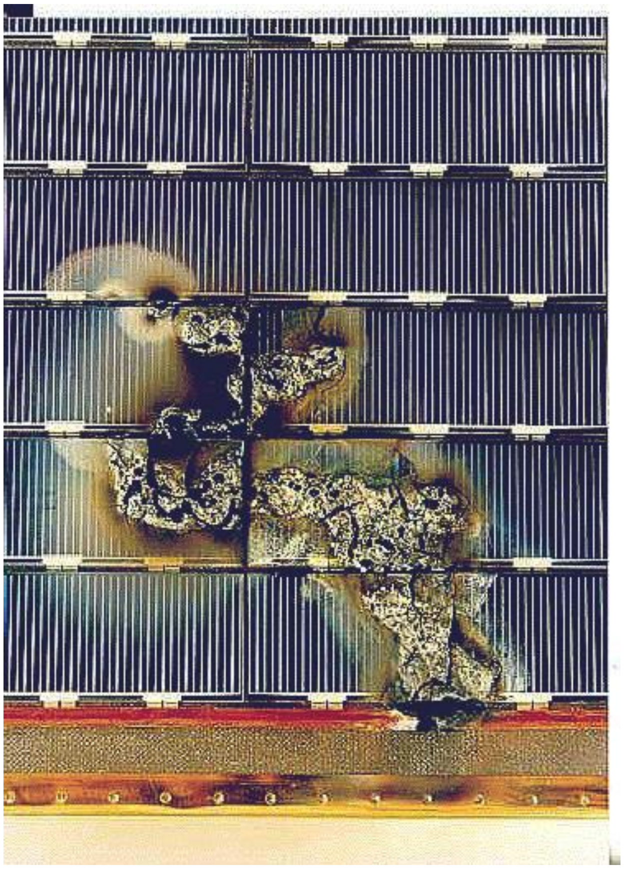

Figure 1: Sustained Arc Damage from the ESA EURECA Mission

Perhaps the most widely recognized event of a sustained arc is that which led to the breakage of the TSS-1R electrodynamic tether, and the loss of the attached satellite (1).

At present, there is no complete explanation of the arcing mechanisms on solar arrays in plasma, though many theories have been proposed (all of which require "fudge factors" to predict observed low arcing voltage thresholds at triple points (1)). The lowest threshold voltage for trigger arcing on solar arrays observed thus far has been 75V, and the current limit for sustained arcing is believed to be on the order of 0.5A (1), (2).

Structural Arcing

There are generally two forms of structural arcing: triple-point arcing (discussed above) and dielectric breakdown. Dielectric breakdown is very different from triple-point arcing, the latter of which occurs at the interface of conductors, insulators, and plasma. Dielectric breakdown, conversely, is the direct discharge through a dielectric that occurs when the applied electric field exceeds the dielectric strength of the material in question.

An insulating surface on a spacecraft that is not in the spacecraft wake will achieve potential equilibrium within a few volts of the ionosphere. If that insulator covers a conductor, the conductor may be at a very different potential, and if the insulator is sufficiently thin, dielectric breakdown can occur. This is of particular interest for anodized aluminum structures, which can have very thin (0.1 – 1 mil) dielectric layers (1), allowing it to breakdown at potentials around 100V or less. Dielectric breakdown has been observed in some cases at voltages as low as -55V (1). Predicting arc thresholds for thin insulating layers is not simply a matter of using published dielectric strengths (2).

Structural arcing can become continuous if the generated arc plasma can contact the solar array or other power source, or if the potential at the arc site can be maintained at sufficiently negative levels by the high-voltage electron-collecting power source (1). This sort of sustained arcing is referred to as a *sizzle arc*.

Structural arcing can be destructive to thermal control coatings in particular. This was identified as an issue for the international space station, the structure of which can float ~140V negative with respect to the ionosphere, and drove designers to include an active plasma contactor in the station's design to forcefully ground its structure to the ionosphere.

MITIGATION STRATEGIES AND TECHNIQUES

Mitigating the risk of solar array arcing for PLEO spacecraft can be particularly challenging, since PLEO subjects a spacecraft to arcing mechanisms characteristic of both LEO and GEO, and design standards for LEO can be at variance with good design practice for GEO (2). The general design approach for GEO spacecraft involves conductively coupling external surfaces to the greatest extent possible, to minimize the prospect of insulators accumulating negative charge. However, in the LEO environment, this design approach will tend to maximize the risk of triple-point arcing, since it brings surfaces at different potentials close to each other and the plasma.

The following mitigation techniques for LEO arcing are from NASA-HDBK-4006 (1) and reference (4), and have been shown to either prevent arcs completely or minimize their damage:

1. If possible, use array string voltages of less than 55 V. No trigger arcs have been seen on LEO arrays of less than about 55 V string voltages even under simulated micrometeoroid bombardment.
2. If solar array cell edges or interconnects are exposed to the LEO plasma and string voltages

are greater than 55 V, the strings should be laid out on the substrate such that no two adjacent cells have a voltage difference of greater than 40 V. Sometimes a leapfrog arrangement will be sufficient. In other high voltage arrays, the strings should be arranged parallel to each other. Serpentine strings can be used to prevent the array width from becoming prohibitive. If the string layout cannot be modified to prevent cells with more than 40 V difference being adjacent to each other (anything less than about 1 cm may be considered adjacent) then the total string voltage must be kept low enough that the initial (trigger) arcs do not take place. The lowest known array trigger arcing has occurred on thin-coverglass cells at about 75 V.

3. For array string voltages greater than about 75 V, trigger arcs in LEO can only be completely prevented by encapsulating the cell or array edges so they do not see the ambient plasma. If encapsulation is not possible, a thorough array bakeout on-orbit (1 week at 100 C or more) may get rid of contaminants and prevent trigger arcing up to about 300 V, or possibly more. Re-contamination may occur on "dirty" spacecraft (spacecraft with excessive venting, cold gas nozzles, etc.). Good encapsulation may prevent arcing up to 1000 V string voltage.
4. Sustained (or continuous) arcs may occur whenever trigger arcs occur and adjacent cells have more than 40V potential differences. However, sustained arcs, in addition to this voltage threshold, have a current threshold, below which they will not occur. It is believed that the current threshold is greater than about 0.5 Amp. If the current produced by each cell is above this threshold, a single string may sustain arcs. If each cell is below this current threshold, then isolating separate strings of solar cells from each other will prevent other strings from "feeding" the arc site, and will prevent sustained arcs. This isolation can be achieved by using blocking diodes in each string (EOS-AM1, now called Terra, e.g. Ref. 6, for instance). Care must be taken that the power bus and/or other components do not have the conditions necessary for sustained arcing. On the Terra arrays, for instance, it was found that diodes used to block interstring currents did not prevent the bus power traces from having sustained arcing events. Covering all exposed bus conductors with Kapton® insulation finally solved the problem. Low-outgassing RTV may be used to cover bare conductors as well.
5. RTV grout between adjacent solar cells and strings that have a high voltage with respect to each other has been shown to effectively block sustained arcs between cells and strings. The degree of coverage, etc., is important in determining the final voltage threshold for sustained arcing.
6. Arrays of 300 V and greater string voltage must be fully encapsulated in order to prevent arcing.
7. Finally, although design and construction are important in preventing trigger arcs and sustained arcs, each new solar array implementation must be tested in a simulated LEO plasma before it can be sure not to arc. This step must not be omitted. The test bias voltage relative to the plasma should include the maximum when the arrays come out of eclipse (or the highest potential expected on the "floating" spacecraft). The interstring voltage should be at least as great as that expected anywhere on the solar array on-orbit. Tests should ideally be conducted at sample temperatures as low as the eclipse-egress temperature.

MICROSATELLITE CASE STUDY: M3MSAT

To a large extent, it appears that designers of small satellites have either been unaware of arcing risks, or more likely have managed to avoid the problem by keeping system voltages low. However, this risk was encountered early in the design of the Maritime Monitoring and Messaging Microsatellite (M3MSat). The process and approach used are summarized in this section.

M3MSat Overview

The Maritime Monitoring and Messaging Micro-Satellite or M3MSat mission will provide maritime surveillance by detecting Automatic Identification System (AIS) messages from space. M3MSat is funded by the Canadian Space Agency (CSA) and Defence Research and Development Canada (DRDC) as a way to augment existing maritime monitoring assets over Canadian areas of interest. M3MSat, depicted in Figure 2, along with several

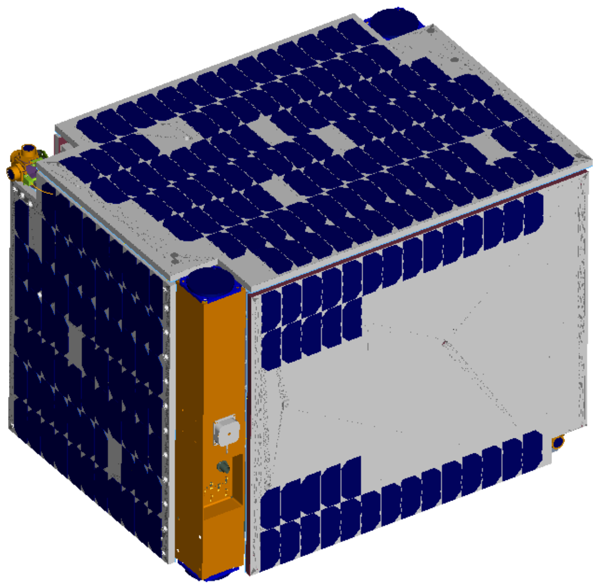

Figure 2: The M3MSat Spacecraft

other Canadian microsatellite missions will also demonstrate the suitability of microsatellites for responsive space applications.

AIS messages were designed for ship-to-ship or ship-to-shore communications for traffic management and collision avoidance. The M3MSat mission goal will be to monitor these AIS signals from orbit and deliver decoded AIS messages and ship tracking information to the clients. AIS broadcasts use a self-organized time division multiple access scheme (SOTDMA) to allow ships in a given cell to broadcast their information without interfering with ships in the same cell. This presents a challenge when monitoring these messages from orbit as multiple self-organized cells will be in the field of view of the spacecraft at any given time. To deal with message collisions caused by this overlap, special purpose signal processing techniques are employed to allow for the extraction of the majority of the original AIS messages.

M3MSat development is centered around the micro-space philosophy, which combines a tightly integrated team and focused design to produce a highly capable spacecraft at a low cost and relatively short lead time. This is achieved by eliminating unnecessary documentation, rapid prototyping and the use of commercial-off-the-shelf (COTS) components. The use of COTS components enables the spacecraft to take advantage of the latest state of the art technologies which are available at a lower cost and shorter lead time compared to traditional space grade components. The design approaches described in the following sections should be taken in the context of the micro-space philosophy.

The M3MSat mission will utilize the Canadian Multi-Mission Microsatellite Bus (MMMB) developed by COM DEV and UTIAS/SFL. This spacecraft bus was designed to support a wide range of missions by providing a large payload area with standard power and data interfaces that can be adapted to accommodate numerous payload configurations. The spacecraft is 60 x 60 x 80cm is size and has a mass of 85kg. The spacecraft has also been designed for compatibility with multiple launch vehicles to maintain launch flexibility

The spacecraft bus features a single-failure tolerant dual string design. The spacecraft structure uses lightweight honeycomb panels and a passive thermal design. The Attitude Determination and Control System (ADCS) provides full 3-axis stabilized control. Attitude determination is provided using a full suite of sun sensors, rate sensors and a magnetometer and can be further enhanced with the addition of a star tracker if necessary. Attitude control is achieved using reaction wheels and magnetorquer coils. The power system can provide 100W average power with a 17Ah Li-Ion battery providing power during eclipse. Fully redundant S-Band receivers and transmitters are used for command and telemetry with a 20Mbps C-Band transmitter available for downloading payload data.

Power System Overview

The M3MSat power system was designed and built by UTIAS/SFL and builds on heritage from numerous successful spacecraft missions. The power system features a series peak power tracking topology which is built around several modular components. The power system provides 28 switched and 4 unswitched outputs and is capable of providing up to 328W of power to the loads at a voltage between 25.5V and 32.4V.

Peak Power Tracking (PPT) of the solar array and battery charge regulation is provided by six Battery Charge Regulators (BCR). Each BCR operates independently of the others and any number of the modular units can be combined in parallel to charge the battery. Each BCR connects to one half of the solar cell strings on two opposite sides of the spacecraft to minimize the impact of failure of any one BCR. This also reduces the input requirements of each BCR as the two panels will never be simultaneously illuminated (with the exception of Albedo).

Power is produced using six body mounted solar arrays consisting of Triple-Junction solar cells with a nominal BOL efficiency of 28%. The battery consists of multiple Li-Ion cells in an 8s3p configuration which allows the battery to operate at the required bus voltage and makes the battery tolerant to the failure of any one cell. In this power system architecture the battery is connected directly to the power bus with no discharge regulation. The low battery impedance contributes to power system stability and suits loads that require intermittently large peak power.

Power is distributed to the loads using two Power Distribution Modules (PDM). To accommodate the dual string bus architecture, loads are distributed between the two PDMs so that any single point failure can be isolated between the two redundant strings.

Solar Array Design Process

At the earliest stages, the M3MSat solar array was designed to maximize the number of cells that could be placed on each panel, with variable string lengths. However, it was realized that arcing could occur with baseline string lengths, which in the initial design (Figure 3) had maximum peak power voltages on the order of 95V. Thus, attention was given early to addressing the arcing risk.

At the microsatellite scale, where budget and schedule are kept aggressive, the issue of arcing is made more serious by the difficulties associated with testing and validating a mitigation approach. Where there is no schedule, budget or facilities for comprehensively testing solar arrays on the ground, system designers must rely almost exclusively on best practice to mitigate the arcing risk.

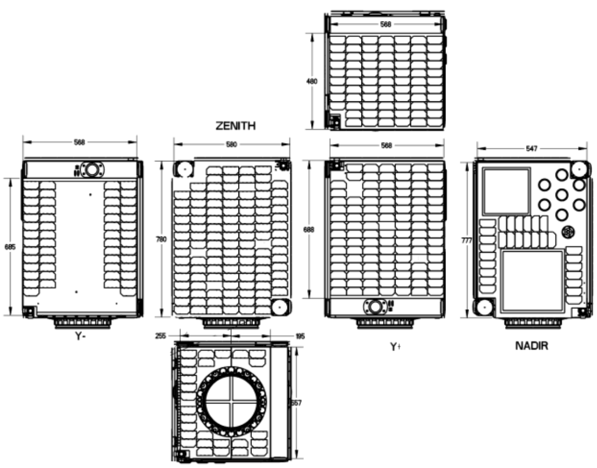

Figure 3: Initial M3MSat Solar Panel Configuration

From the previous section, several mitigation techniques were evaluated. In essence, design options for reducing or marginalizing the risk of arcing could generally be distilled to the following:

1. Place the structure at the most positive potential generated by the power system (positive ground)
2. Force the structure ground to equilibrium with the plasma (plasma contacting)
3. Prevent plasma exposure of high-voltage surfaces (encapsulation).

Most spacecraft power systems are negatively grounded, since most electronics use negative ground polarity. Positive grounding, therefore, was not viewed as a reasonable option for the M3MSat design. An intermediate option using a center-tapped array was briefly considered, which would reduce the maximum structure potential to approximately half the solar array string voltage. For the baseline M3MSat solar array design, this would have been sufficient to mitigate arcing concerns, but would have coupled the design of the solar array into the design of on-board power electronics even further. Since one of the principal advantages of a series-regulated PPT architecture is that array design is largely de-coupled from the main power bus, attempting to change how solar power was received by PPT regulators was viewed as disadvantageous.

Plasma contactors (i.e. devices that are designed to forcefully ground the chassis to the ionosphere) are a brute-force approach, but the complexity and power such systems demand make them untenable for the microsatellite scale, at least in the opinion of power system engineers on M3MSat.

Encapsulation was viewed as the most plausible means of controlling the spacecraft potential. The approach is simple in principle: encapsulate all high-voltage conductors. This would prevent both triple-point arcing as well as parasitic current collection (i.e. power loss). However, there are caveats associated with encapsulation: first, no air can be trapped anywhere. This may appear obvious, but has been an issue in past designs (1). Second, the encapsulant thickness must be able to withstand dielectric breakdown. Finally, encapsulant must not be able to peel away from high-voltage components, or Paschen breakdown can occur in outgassing products at sufficient pressures—this is the phenomenon whereby a neutral gas breaks down and provides an arcing medium (1). Finally, the encapsulant must not degrade in the LEO environment, as it will be subject to many environmental stressors over its lifetime (atomic oxygen and UV/X-ray exposure being notable examples).

It was also noted that, for high-voltage systems, plasma can potentially enter through vents or openings in the spacecraft. To mitigate charging within the spacecraft, vents and openings must be restricted to the plasma

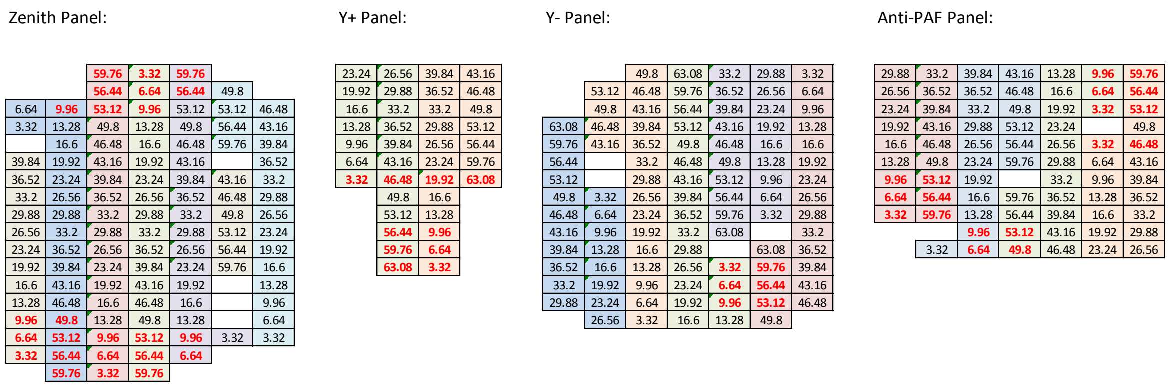

Figure 4: M3MSat Solar Array Voltage Distribution (WCC-BOL)

Debye length (which is a function of plasma density and temperature, on the order of hundreds of millimeters for the candidate M3MSat orbit). High-voltage areas of PCBs can be subject to the same arcing risks as the array and structure.

Lastly, it was unclear programmatically whether sufficient expertise was available to provide quality assurance in the design of an arc-proof solar array. The burden would necessarily fall on the power system contractor to demonstrate that the high-voltage system was immune to the deleterious effects of arcing, which would have been a difficult proposition at best.

It has been said that clever people solve problems, while wise people avoid them. It was decided that the best way to address the arcing risk was to avoid it entirely.

Avoiding the Problem

The M3MSat solar array layout was changed to have maximum string lengths of 19 cells. Voltage and current characteristics for the new array are summarized in Table 1 under worst-case-cold end-of-life conditions, which yield maximum array voltages. Voltage distributions on primary solar panels under worst case cold, beginning of life (WCC-BOL) conditions are shown in Figure 4.

Table 1: Worst-Case Solar Array Characteristics

Solar Array Parameter	Value
Minimum String Temperature	-77 C
Maximum Open-Circuit Voltage, V_{oc}	63.1 V
Maximum Peak Power Voltage, V_{mpp}	56.8 V
Maximum String Current, I_{sc}	506 mA
Maximum Structure Potential	56.7 V

As per NASA-HDBK-4006, the maximum system voltage has been reduced well below 75V. The maximum solar array voltage is approximately 63V, corresponding to the open-circuit voltage of the solar array (i.e. the operating point where no current is available for sustained arcing, in the unlikely event of a trigger arc). The maximum available current for sustained arcing is approximately equal to the maximum value required, but is only available at an array operating voltage of 57V, with all higher voltages having less current available to feed arcs.

As can be seen in Figure 4, the maximum standoff voltage between adjacent cells could not be practically reduced below 40V under WCC conditions in all areas, but this was not deemed necessary with the system voltage being << 75V.

The maximum structure potential is estimated as 90% of the maximum system voltage, as per (1) and (2). This corresponds to approximately 57V, which is only two volts above the recommended 55V structural float. Trigger arcs have some small probability of occurring at this potential, but sizzle arcs are prevented because the array is either open-circuit or has reduced current capacity and will starve the dielectric discharge. For all other array characteristics, the structure potential floats less than 55V, and the risk is mitigated.

CONCLUSIONS

The underlying mechanisms of solar array and structural arcing in the ionospheric plasma environment is still not well-understood, but experiments and experience have shown that such issues can be detrimental or catastrophic to high-voltage spacecraft. For the small satellite designer, many options exist for mitigating the arcing risk, but from a cost and quality assurance standpoint, it is argued here that the best approach is to avoid the issue entirely at the earliest stages of design. This paper has provided an overview of the risks, and presented one approach that has been used to mitigate them. It is hoped that this may serve as a primer, guide, and useful case study for addressing a subtle but dangerous issue in spacecraft power system design.

ACKNOWLEDGEMENTS

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