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14. ABSTRACT
Space Systems Loral is conducting an IR&D program to determine the feasibility and effectiveness of field effect electron emitters for potential control of geosynchronous altitude spacecraft. This electron emitters will be based on Spindt Cathode Field Emission Array Technologies. The configuration studied here consists of two emitters, each with an area of about 1 cm2 and emitting up to 1 mA of electrons at approximately 50 eV energy.
We show that it appears feasible to use electron emitters to control the surface charge of a satellite. Results concerning the placement and effectiveness of emitters and the spacecraft potential configuration under substorm conditions with and without emitter operations in sunlight, in eclipse, and during eclipse exit. .

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Charge Control of Geosynchronous Spacecraft Using Field Effect Emitters

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***Space Systems Loral (SS/L)* is conducting an IR&D program to determine the feasibility and effectiveness of field effect electron emitters for potential control of geosynchronous altitude spacecraft. This paper presents results concerning the placement and effectiveness of the emitters, and the spacecraft potential configuration under substorm conditions with and without emitter operation in sunlight, in eclipse, and during eclipse exit.**

I. Introduction

The physics of spacecraft charging and engineering techniques to minimize charging hazards have been subjects of research for several decades. However, the threat of electrostatic discharges on solar arrays has not been eliminated,[1,2] and partial array power loss due to such discharges is an industry-wide problem seen often enough to be of significant concern. According to Reference 1, ESD-induced anomalies are occurring at a rate of about one per spacecraft every two years. While effective protective measures have been applied to solar arrays to mitigate consequences of discharges, the possibility of addressing the root cause by preventing significant charge buildup is intriguing. *Space Systems/Loral (SS/L)*, with support from *Science Applications International Corporation (SAIC)*[3] and *SRI International*, is conducting an IR&D program to study the use of field-effect electron emitters ("Spindt Cathodes") to reduce the threat of discharges on solar arrays at geostationary altitudes. In particular, the problem being addressed is that of "inverted gradient" charging in which the back side of an array typically collects an "excess" of electrons with respect to the sunlit, dielectric covered side, thus resulting in damaging electrostatic discharges at relatively low differential voltages. This effect can, therefore, be mitigated by actively ejecting this excess charge from the spacecraft chassis which is, in turn, electrically connected to the back side of the array. It is hoped that this development program will culminate in a flight demonstration of this charge control technique.

Much of the work described here was performed using the *Nascap-2k* computer code[4] for steady-state and dynamic analysis of spacecraft environment interactions. The computer codes leading up to *Nascap-2k* have been under development since 1976, and have been used in support of the SCATHA satellite to study spacecraft charging technology,[5] to study numerous geosynchronous and interplanetary spacecraft,[6,7] for detailed analysis of rocket

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experiments[8,9] and thruster plume interactions,[10] and design and analysis of low-Earth and medium-Earth orbit spacecraft.[11,12,13]

II. Flight Hardware and Diagnostics

SRI is developing the flight hardware and diagnostics for this effort. The electron emitters will be based upon Spindt Cathode Field Emission Array technologies.[14,15] The currently contemplated configuration consists of two emitters, each with area of about 1 cm^2 and emitting up to 1 mA of electrons at approximately 50 eV energy, located on the spacecraft's zenith face. The current and voltage of the emitters will be monitored by an active electronic control system that also serves to maintain the emission current at a pre-selected level. In order to quantify the effectiveness of this technique in reducing the threats associated with inverse gradient charging, an interactions diagnostics package is also being developed. At present this package consists of a dedicated solar array sub-panel (approximately 40 centimeters square) with electric field and current sensors embedded in the front and back faces, along with a discharge-transient detection system. This package is intended to be included on early emitter flights to provide a direct measure of and comparison with the effects on local fields and currents predicted below. Further information regarding this diagnostics package is planned for presentation at a future date.

III. Effects on Spacecraft Charging

A. Current Requirements and Space Charge Effects

A typical value of incident electron current to a geosynchronous spacecraft under substorm conditions is about 1 microampere per square meter, before reduction by secondary electron emission and electron backscatter, and cancellation by photoemission and incident ion current. Multiplying this current by a typical spacecraft area of 200 m^2 suggests that an emission current of 0.2 milliamperes would be adequate to neutralize incident currents to spacecraft. The proposed current requirement for the cold cathode charge control device is one milliampere, which provides the neutralization current with a very large margin while using insignificant power and causing negligible parasitic currents on the spacecraft.

Experiments using thermionic emitters in the ionosphere have failed to neutralize currents to biased spacecraft. However, in those cases the required currents were much higher, and the failure could be attributed to space charge effects. To examine the likelihood of space charge limiting, we write the Child-Langmuir equation:

$$J = \frac{I}{\pi r^2} = 2.33\times10^{-6} V^{3/2} / d^2 \tag{1}$$

where I is the current (A) of electrons at energy V (eV) emitted from a devices with area πr^2, and d is the limiting distance in planar geometry. Solving for d with values of 1 milliampere at 50 eV gives

$$\frac{d^2}{r^2} = \pi\times2.33\times10^{-6} V^{3/2} / I = 2.6\,. \tag{2}$$

As the limiting distance is larger than the device size, classical space charge limiting (in which most of the current returns to the device) does not occur. To see if any substantial space charge effect is to be expected, we calculate the potential at the surface of a 1 cm radius sphere of charge with the charge density of the emitted beam, and find it to be less than ten volts. Since the near-field space charge potentials are too small to have a major effect on the electron trajectories, the electrons spread out over the grounded spacecraft surfaces whose image charges tend to neutralize the space charge. As a result, as long as the spacecraft potential is less than the 50 eV emission energy, most of the emitted current escapes into space and serves to neutralize the charging current to the spacecraft.

With the above justification, the calculations presented here do not explicitly include the effects of space charge. However, we do assume the current to be emitted in a cosine distribution, so that, while the peak current density is forward, half of the current is emitted at angles greater than 45 degrees from the normal to the emitter surface.

B. Spacecraft Models for Nascap-2k

We created Nascap-2k models representative of a generic spacecraft for the purpose of doing the following calculations for substorm conditions:

(1) Predict the spacecraft overall and differential potentials in the absence of the emitters;
(2) Predict spacecraft overall and differential potentials with the emitters operating;
(3) Predict the effect of spacecraft overall and differential potentials on trajectories of emitted electrons;
(4) Evaluate implications of prior calculations for emitter placement;
(5) Predict eclipse exit charging behavior with emitters on;
(6) From the above results, evaluate whether the emitters are effective in reducing the likelihood of electrostatic discharge.

Initially, it was proposed to place a pair of emitters on either the nadir (earth-facing) panel or the zenith (anti-earth facing) panel. Otherwise similar models were made for both of these configurations, as shown in Figure 1 and Figure 2. As the nadir panel is cluttered with antennas and other hardware, placing the emitters on this panel required locating them close to the panel edge and near a radiator panel consisting of weakly conductive OSRs. By contrast, the zenith panel is relatively unobstructed, so that the emitters could be placed farther from the panel edge and away from the radiator panels. In addition to the midnight configuration models shown in Figure 1 and Figure 2, models were made corresponding to 3:00 AM and 6:00 AM (dawn), that differed from the models shown by appropriate rotation of the solar arrays.

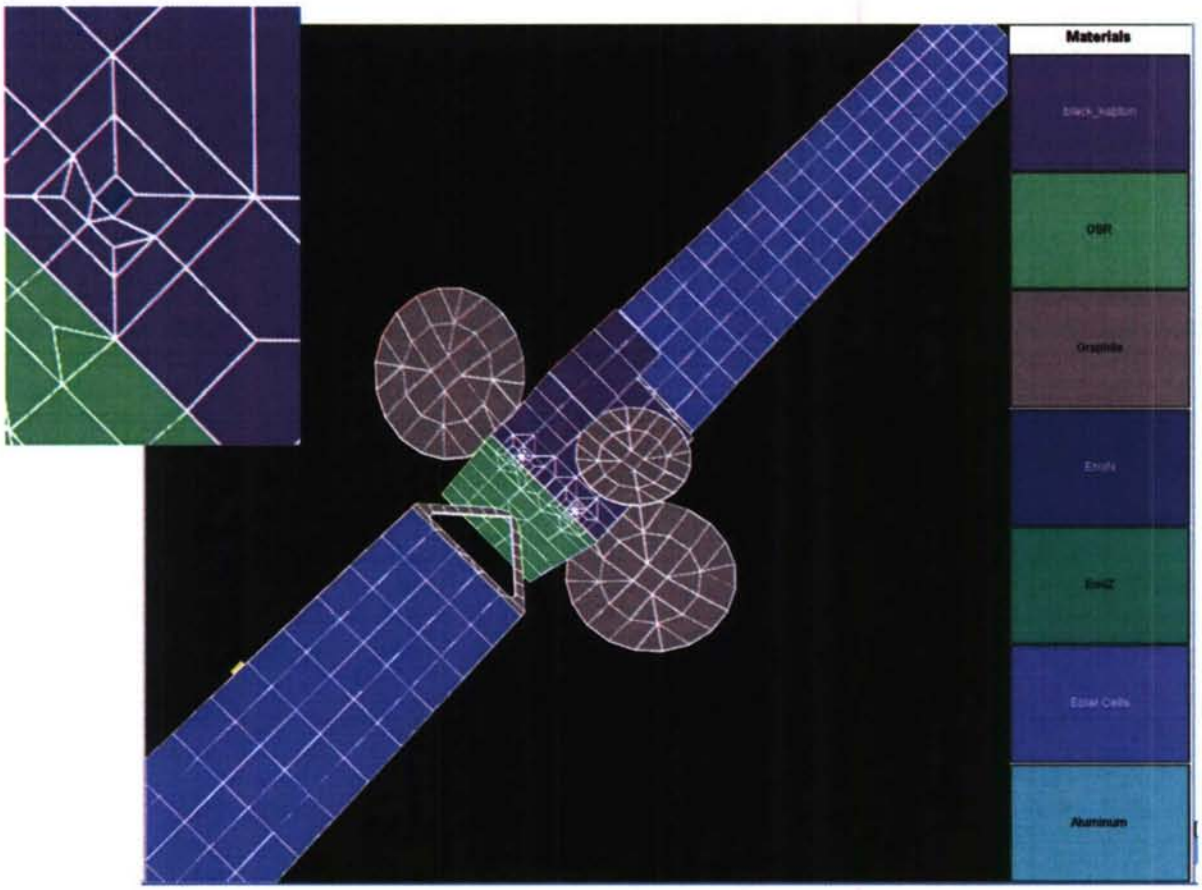

Figure 1. Spacecraft model showing emitters near edge of nadir (earth-facing) panel. Inset shows detail of emitter, with subdivision to achieve good electric field representation in region of emitter and panel edge.

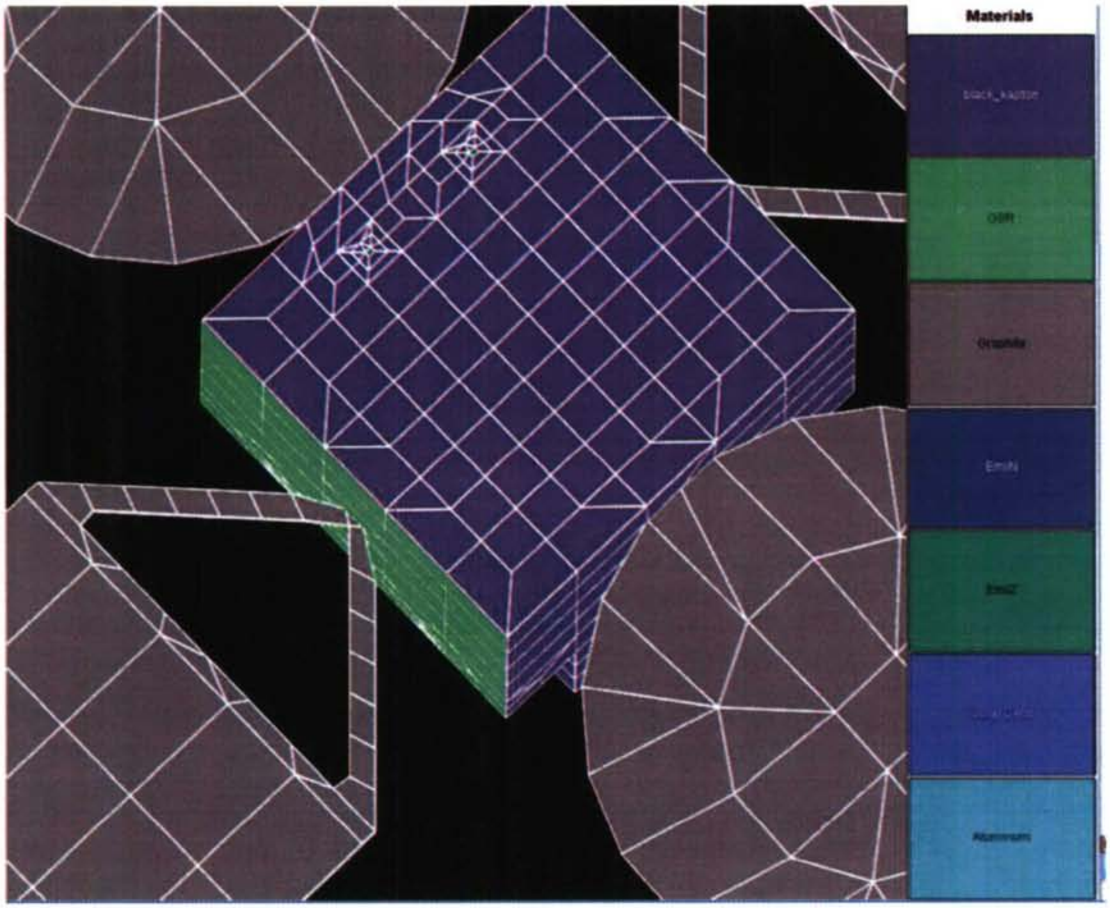

Figure 2. Spacecraft model showing emitters on zenith panel. (Cf. Figure 1.)

C. Substorm Charging Results With No Emitters

All charging calculations were done using the "NASA Worst Case" environment, whose parameters are shown in Table 1. An important parameter determining both the absolute and differential charging is the bulk conductivity of the 5 mil (125 μm) coverglasses. Calculations were done for a "high conductivity" value of $2.6\times10^{-12}\ \Omega^{-1}\mathrm{m}^{-1}$, and a "low conductivity" value of $2.6\times10^{-13}\ \Omega^{-1}\mathrm{m}^{-1}$. The results for charging of the midnight configuration in sunlight are shown pictorially in Figure 3 and Figure 4. These figures are similar in character, although they differ greatly in the magnitude of the charging. In both cases the spacecraft frame charges to negative potential, while photoemission creates a positive differential ("inverted gradient") on the coverglasses, with the largest differential on the outboard panel. The main current path consists of ambient electrons incident on dark surfaces of the spacecraft body and solar array backs being conducted through the coverglasses and photoemitted, with the conductivity process requiring a larger differential potential in the low conductivity case. The shaded OSRs develop a small negative differential in the high conductivity calculation, and a small positive differential in the low conductivity calculation. Table 2 and Table 3 show respectively the frame potential and maximum inverted gradient potential for the two variants of the midnight configuration, as well as the dawn and eclipse cases. The high conductivity eclipse case is omitted from the tables because such a value is unrealistic when the arrays are cold.

Table 1. Parameters (temperature and density) of "NASA Worst Case" environment.

	Temperature	Density
Electrons	12 keV	$1.12\times10^{6}\ \mathrm{m}^{-3}$
Ions	29.5 keV	$0.236\times10^{6}\ \mathrm{m}^{-3}$

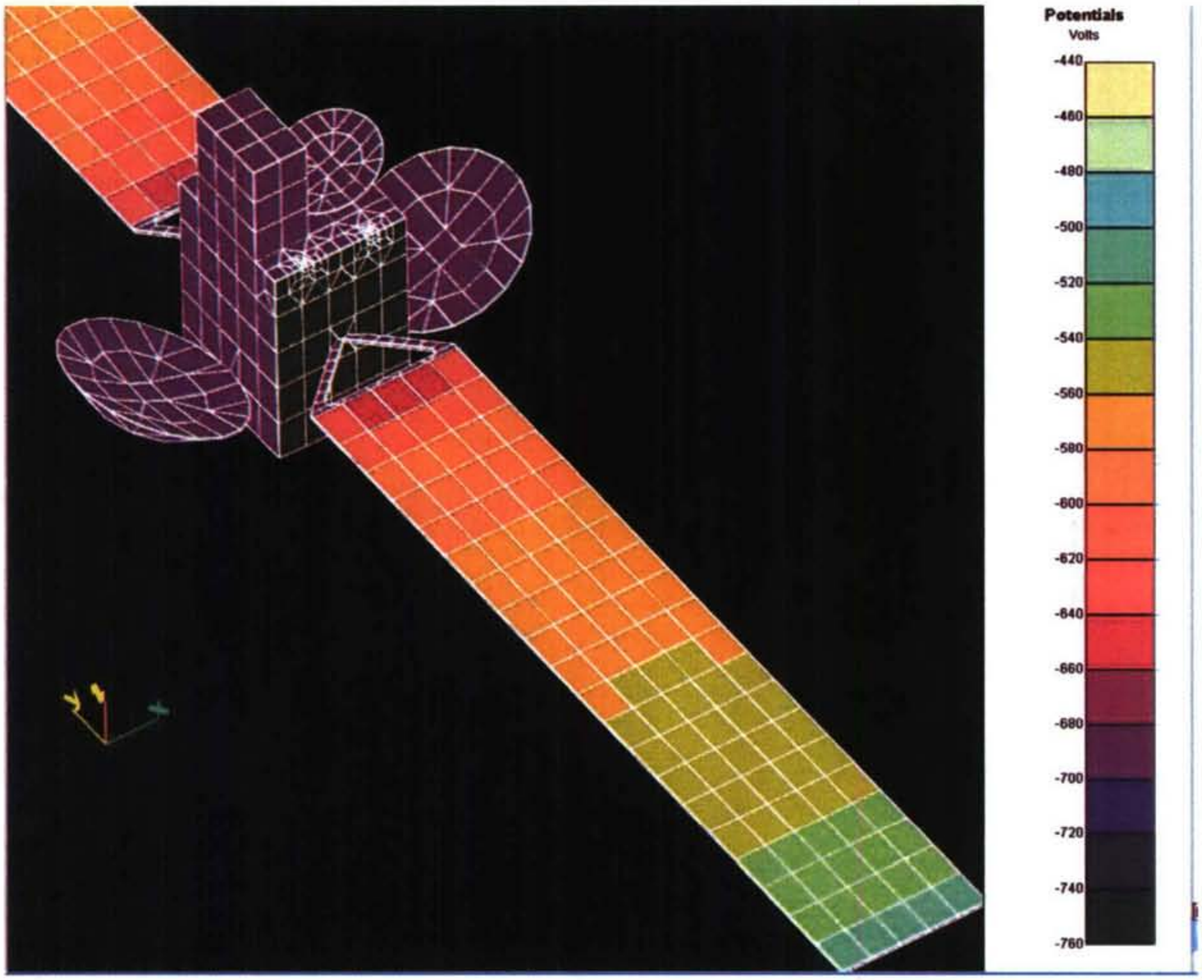

Figure 3. Substorm charging results for the spacecraft in midnight configuration, sunlit, with high conductivity coverglasses.

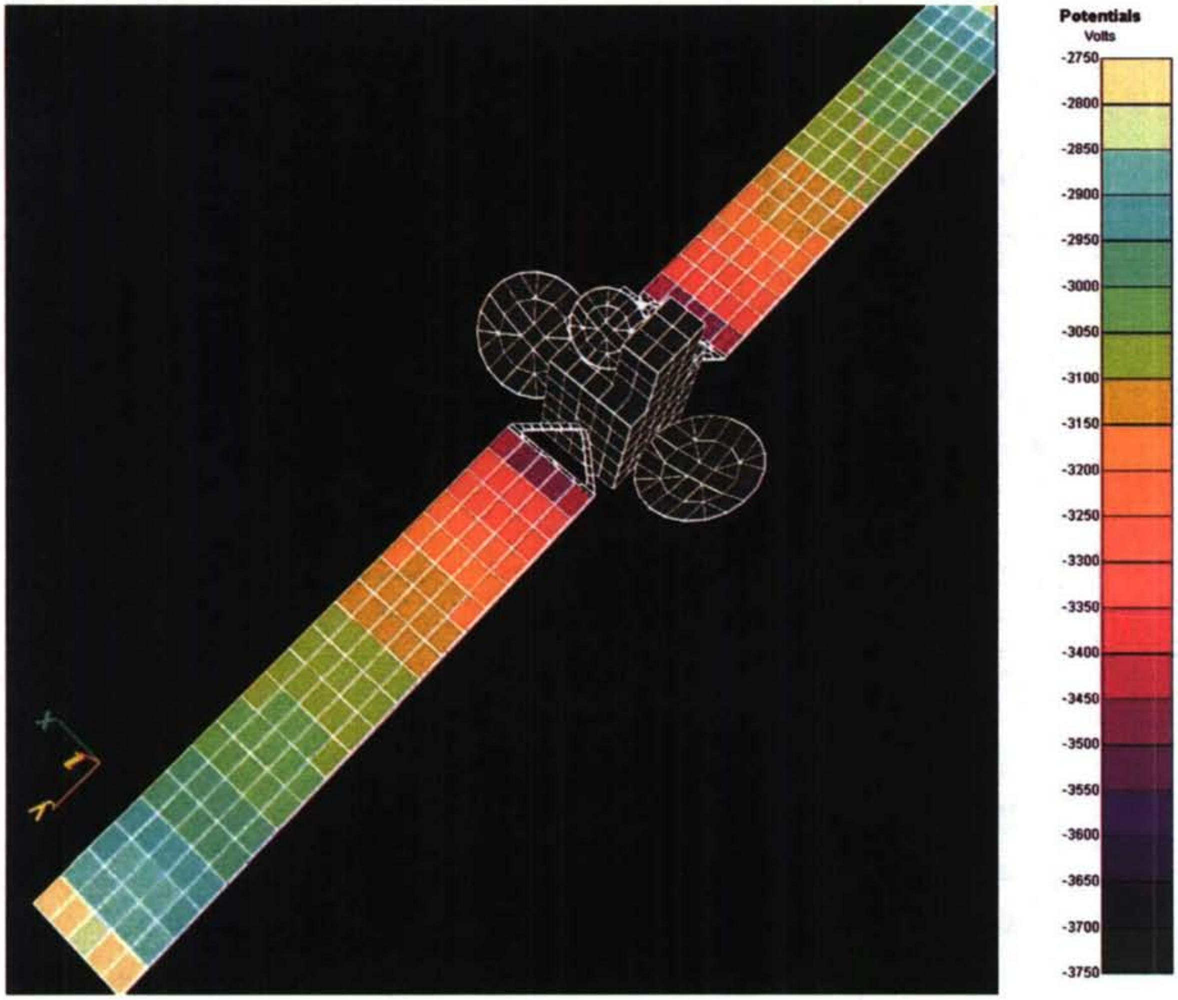

Figure 4. Substorm charging results for the spacecraft in midnight configuration, sunlit, with low conductivity coverglasses.

Table 2. Substorm frame potential for the sunlit midnight and dawn configurations, as well as in eclipse.

Frame Potential			
	Midnight	**6 AM**	**Eclipse**
More Conductive	-680 V	-930 V	
Less Conductive	-3700 V	-5200 V	-6500 V

Table 3. Maximum inverted gradient potential for the sunlit midnight and dawn configurations, as well as in eclipse.

Inverted Gradient Potential			
	Midnight	**6 AM**	**Eclipse**
More Conductive	180 V	250 V	
Less Conductive	1000 V	1400 V	1400 V

D. Charging During Emitter Operation

The effect of the emitter is to hold the spacecraft frame nearly 50 volts positive, at which point the fraction of emitter electrons escaping is a sensitive function of spacecraft potential, so that the potential can self-adjust to maintain balance between the emitted electrons and the incident electron current from the environment. Figure 5 and Figure 6 show trajectories of 50 eV emitted electrons with the spacecraft at + 45 and +55 volts respectively. (These trajectories are calculated neglecting the effects of space charge.) At the lower potential very few electrons return to the spacecraft (although some are deflected), whereas at the higher potential all electrons return, forming a very large cloud of negative charge. This illustrates that if the emitters operate at currents well in excess of the environment charging current the spacecraft frame is maintained at a positive potential close to the emission energy. In the calculations to follow, the spacecraft frame is assumed to be held at a potential of +45 volts.

Figure 7 shows the charging configuration under substorm conditions (sunlit, midnight) with the emitter operating. As stated above, the spacecraft frame is held at +45 volts by the emitters. The solar array coverglasses are discharged by ambient electrons until their photoemission is sufficient to maintain current balance. This occurs at a coverglass surface potential of about +5 volts, leading to a "normal gradient" differential potential of about 40 volts. Because kilovolt potentials are required to cause electrostatic discharges in the "normal gradient" configuration, the coverglass differential potential is not cause for concern. The shaded OSRs are charged to negative potentials by the substorm electrons, limited by the conductivity and secondary emission of the OSRs. As shown in Figure 8, electric fields due to the negatively charged OSRs can have a strong effect on electrons emitted in their vicinity. In the case of Figure 8, 75% of electrons from the nadir-mounted emitters were deflected onto the apparatus mounted on the nadir panel. With the emitters in the proposed zenith panel location (Figure 9) the effect is much smaller (only about 5% return), both because the emitters are further from the charged radiator panel and because there is less clutter to intercept electrons on the zenith panel.

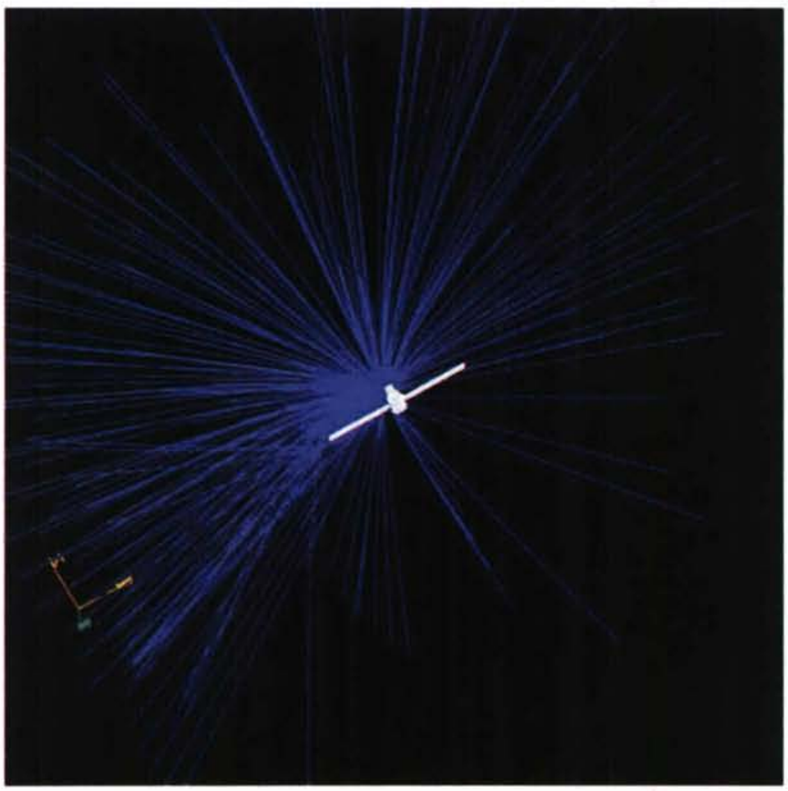

Figure 5. Trajectories of 50 eV emitted electrons with spacecraft at +45 volts.

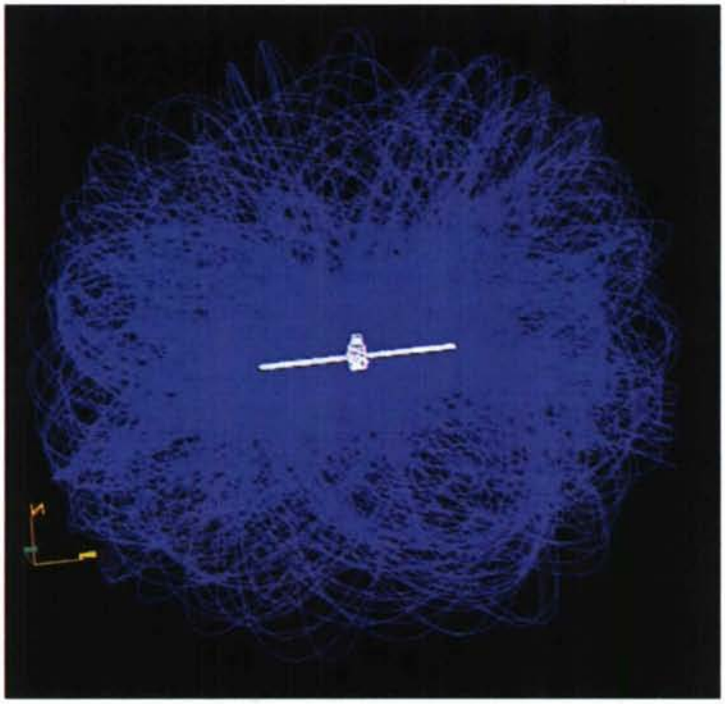

Figure 6. Trajectories of 50 eV emitted electrons with spacecraft at +55 volts.

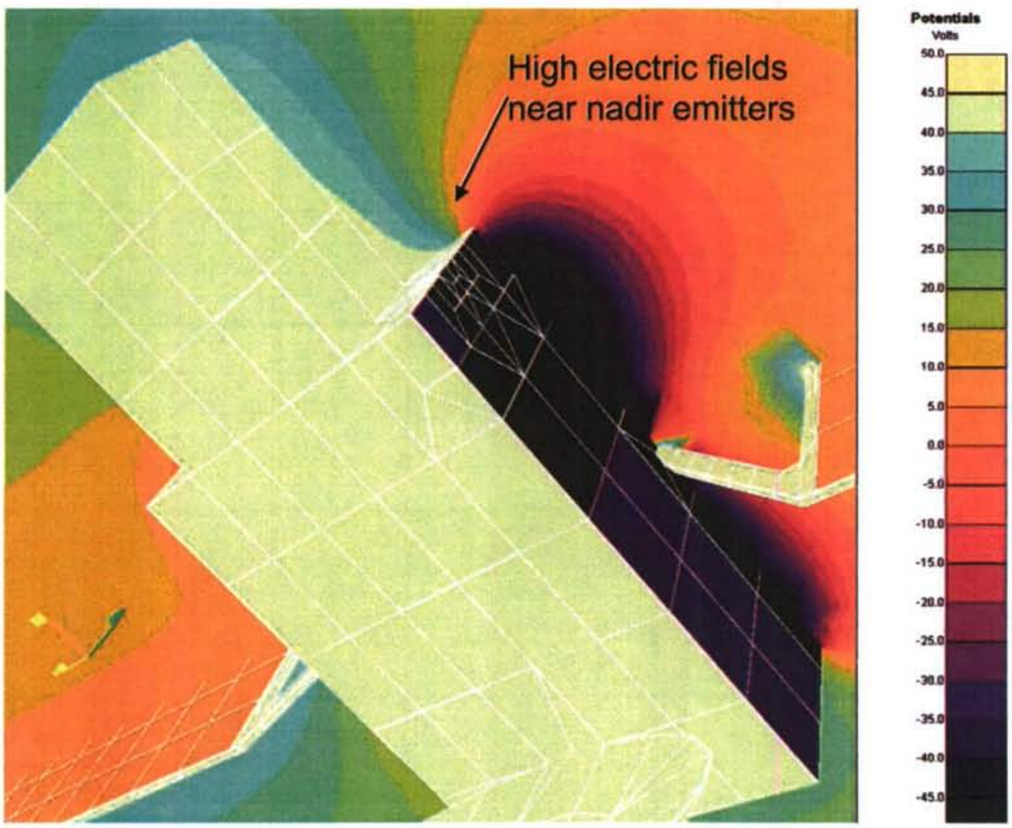

Figure 7. Charging configuration under substorm conditions with emitter operating.

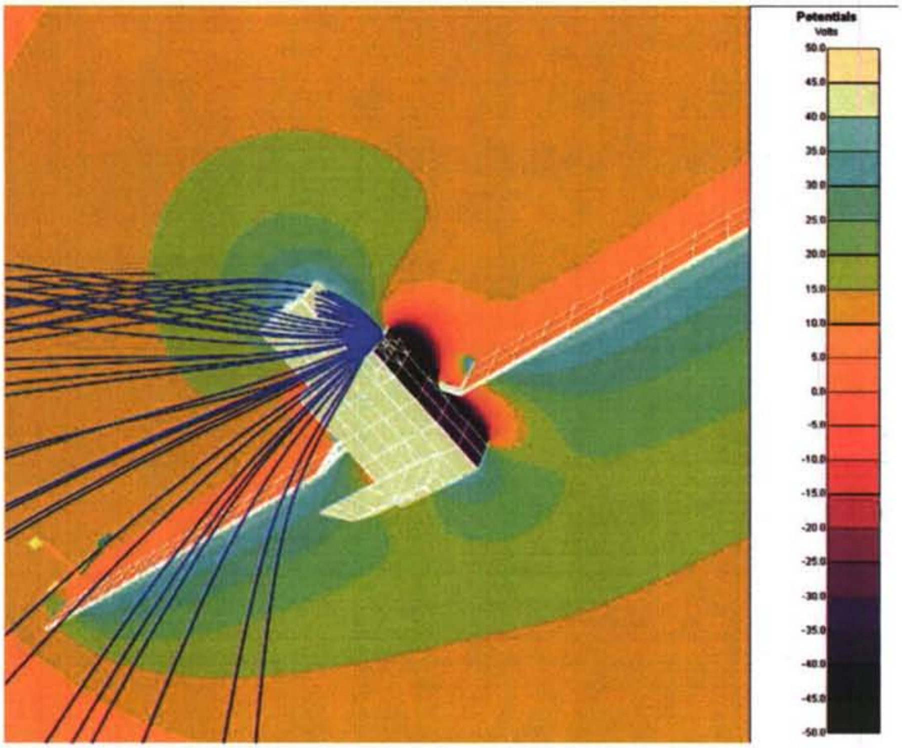

Figure 8. Trajectories of electrons from nadir panel emitters in the potentials of Figure 7.

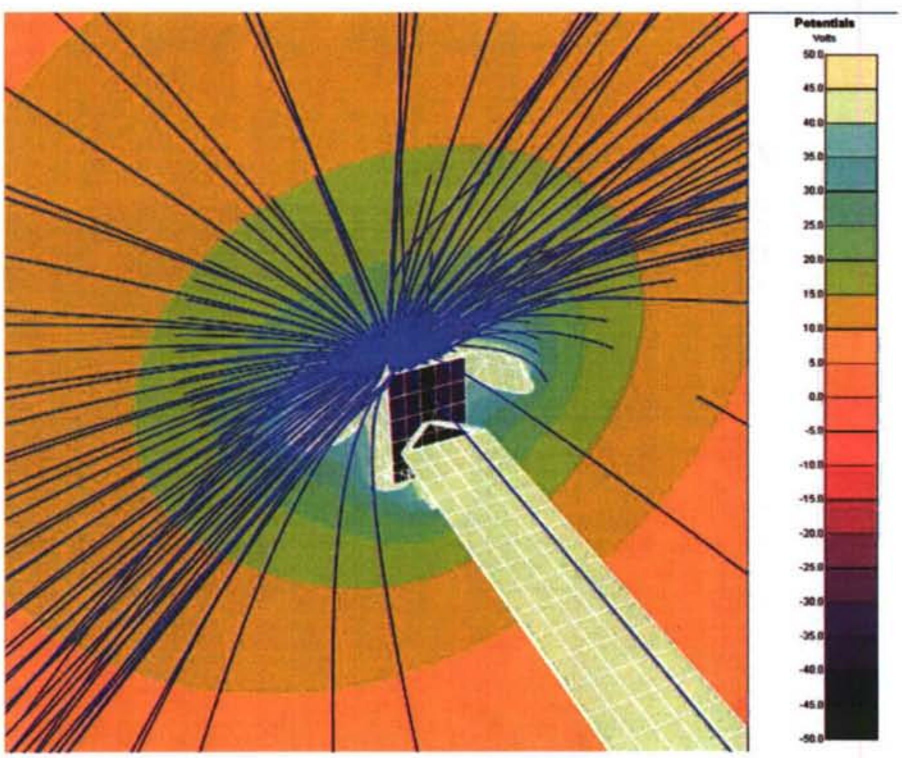

Figure 9. Trajectories of electrons from zenith panel emitters in the potentials of Figure 7.

E. Charging at Eclipse Exit in Substorm with Emitter Operation

Eclipse exit can be a dangerous time for a charged spacecraft. If high inverted gradient potentials have developed during eclipse, the onset of photoemission, the powering of arrays, and the low conductivity of cold coverglasses are likely ingredients leading to ESD. However, the emitter maintains a normal gradient potential on the solar arrays, so the eclipse exit scenario is quite different. In this subsection we deal first with the idealized case of instantaneous exit from eclipse (*i.e.*, sudden onset of photoemission), and then with the more realistic case of gradual onset of photoemission.

1. Instantaneous Onset of Photoemission

Figure 10 shows the changes in total current and coverglass current to a charged spacecraft with operating emitter, beginning at hypothetical sudden emergence from eclipse into full sunlight. In eclipse the net current to the spacecraft was about negative 0.3 milliamperes, balanced by the net emitter current (electron emission less return current of emitted electrons). The emitter held the spacecraft frame at about +50 volts, while the coverglass surfaces experienced "normal gradient" charging to about -300 volts. Sudden onset of photoemission contributed positive current in excess of 1 milliampere (blue trace), causing the entire spacecraft to rise in potential, and rendering the emitter ineffective as all electrons return. After about 0.2 milliseconds the coverglass surfaces reach positive potential, suppressing the photoemission (*i.e.*, low energy photoelectrons now return to the coverglass surface), and the frame potential has increased to about +300 volts. From this point onward the total current to the spacecraft (magenta trace) is nearly zero, with photoemission just sufficient to balance the negative current collected by the frame and other grounded surfaces. These currents discharge the frame potential at a rate of about 20 volts per second until it reaches about +50 volts. The +50 volt frame potential is once again maintained by the emitter.

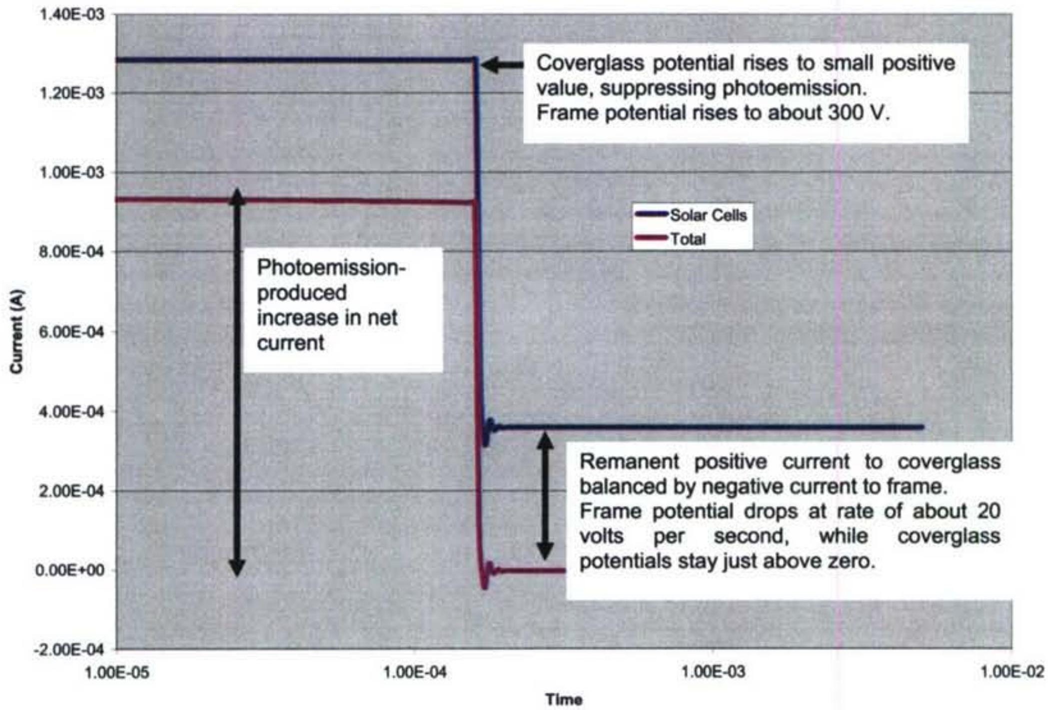

Figure 10. Changes in current to solar cell coverglasses (blue) and total current to spacecraft (magenta) for charged spacecraft with emitter following sudden onset of photoemission.

2. Gradual Onset of Photoemission

Figure 11 illustrates the more realistic case of gradual onset of photoemission while the spacecraft is in penumbra. As in the previous section, the initial conditions (resulting from substorm charging in eclipse) are that the coverglasses are normal-gradient charged to -300 volts (blue curve, right scale in Figure 11), while the frame is held near +50 volts by the emitter, whose current balances the environment current of negative 0.3 milliamperes (yellow curve, left scale in Figure 11). The photoemission current (magenta curve, left scale in Figure 11) is

assumed to increase linearly at a rate such that its full value of about 1.3 milliamperes (blue curve in Figure 10) is reached in 60 seconds. The total current to the spacecraft (yellow curve, left scale in Figure 11) increases at a similar rate and, so long as it remains negative, is balanced by the emitters. Since the reduction in coverglass charging is the integral of the current, the coverglass potential increases quadratically in time, on a scale of tens of seconds (blue curve, right scale in Figure 11). If the coverglass potential reaches zero while the total current is still negative, the eclipse exit event is uninteresting. (For these environment parameters, this condition requires photoemission onset time greater than 100 seconds.) In this case, however, the total current reaches zero while the coverglasses are still negative. Since the emitters cannot compensate for a positive total current, the entire spacecraft rises rapidly in potential (here by about 120 volts) until photoemission is suppressed. The frame potential then drops at a rate of about 20 volts per second until it drops below +50 volts, where it is stabilized by the emitters. We are not aware of any likely hazard due to a few seconds of high positive frame potential.

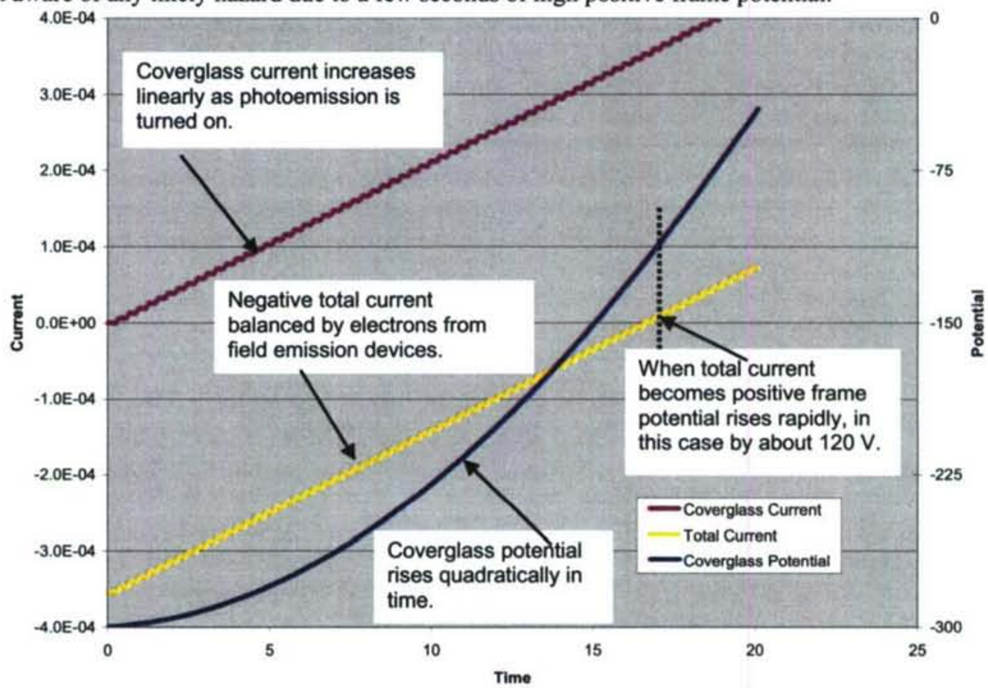

Figure 11. Eclipse exit scenario in which photoemission current (magenta curve, left scale) turns on linearly over the course of one minute, causing the coverglass potential (blue curve, right scale) to rise quadratically until the total current (yellow curve, left scale) becomes positive.

IV. Conclusions

The IRAD has demonstrated the feasibility of using an electron emitter device to prevent surface charging on satellite. The proposed design has the proper electron energy and current range to prevent the satellite charging at various operational conditions. The placement of the emitters has been simulated by *Nascap-2k* to ensure the satellite surface potential will be in an acceptable region with the emitter's operation. The electron emitters and the supporting circuitry are relatively small, so that it will be valuable preventative equipment for use on future GEO satellites.

Acknowledgments

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