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Charging of mirror surfaces in space

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[1] Spacecraft often charge to negative potentials of several kilovolts in eclipse at geosynchronous altitudes. We suggest that optical mirrors at geosynchronous altitudes will charge in sunlight as if in eclipse. Modern mirrors can attain very high reflectance, the reflected light being nearly as intense as the incoming light. With high reflectance, the sunlight photon energy imparted to mirror surfaces is greatly reduced, resulting in little or no photoemission. As a result, mirrors will charge as if they would in eclipse, the equilibrium potential being governed by the balance of currents without photoelectrons. When the plasma electron temperature is high, the equilibrium potential may reach several kilovolts negative, despite sunlight. This occurs often in the morning hours and in severe space weather. We stress that in general, the finite reflectance and Sun angle should be included in calculations of spacecraft charging in sunlight. As an important application for mirror charging, we bring to attention recent news, the Boeing 702 model geosynchronous satellite fleet, featuring two long solar panels on each side. Each solar panel is equipped with two mirrors flanking both sides for sunlight enhancement on the solar cells. The entire satellite fleet has suffered a similar fate, namely, gradual, permanent, and sometimes stepwise degradation. While the true cause of the Boeing solar panels deterioration may never be known, we suggest that sudden development of differential charging between the solar panels and the mirrors on their sides could be the culprit. Differential charging of mirrored solar panels may develop rapidly when the satellite is coming out of eclipse. Indeed, the sudden 25% degradation of PanAmSat PAS-7, a Boeing 702 model satellite, did occur shortly after eclipse exit in the morning of 6 September 2001. Finally, we suggest a simple mitigation method for solving the problem.

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1. Introduction

[2] Spacecraft charging can detrimentally affect electrical operations on space systems. Most communication and surveillance spacecraft are at geosynchronous altitudes and many more such spacecraft will be deployed in the new millennium. The plasma density in the geosynchronous environment varies from over 100 cm^{-3} to 0.1 cm^{-3} and the energy varies from a few eV to tens of keV, depending on local time and geomagnetic conditions. Spacecraft surface charging occurs at high plasma energies. While surface material properties and spacecraft geometry are defined by spacecraft design, spacecraft charging is controlled by the dynamic plasma condition, which varies in time.

[3] Of all the space environmental factors, sunlight and electron temperature are most important in controlling spacecraft charging in the geosynchronous environment. Sunlight photoemission tends to suppress negative charging. The photoemission electron flux of typical surface materials is about 2×10^{-9} A/cm^2 [*Lide*, 2002; *Stannard et al.*, 1981], whereas the average ambient electron flux experienced at geosynchronous altitudes by the SCATHA satellite was 0.115×10^{-9} A/cm^2 [*Purvis et al.*, 1984]. Therefore the outgoing photoemission flux exceeds the incoming ambient electron flux. As a result, charging to positive potentials is expected for spacecraft surfaces. Indeed, the SC10 booms of SCATHA normally charged to a few volts positive in sunlight [*Lai et al.*, 1986] except when encountering periods of hot electrons in severe space weather [*Lai*, 1991a, 1991b].

[4] Electron temperature is the single most important factor controlling surface charging at the geosynchronous environment in eclipse [*Lai and Della-Rose*, 2001]. Other space environment parameters, such as electron density, ion temperature, and ion density, are much less important in affecting the spacecraft potential.

[5] Spacecraft charging is governed by current balance. The incoming ambient electron current is certainly the most important one. The outgoing electrons comprise of secondary emission, backscattered electrons, and photoelectrons emitted from surfaces. Secondary electron current is important and photoemission current is often dominating. In the

early decades of spacecraft charging research, little or no attention was paid to the effects of surface condition on secondary emission. Coefficients of secondary emissions were routinely used as if they were physical constants. It has been realized in recent years that secondary emission coefficients vary very much with surface conditions [e.g., *Davis and Dennisson*, 2000]. Yet little or no attention has been paid to the effect of surface conditions on photoemissions. Much previous research has been done on photoemissions on surfaces in space conditions [e.g., *Grard*, 1973; *Pedersen*, 1995; *Grard and Tunaley*, 1971; *Szita et al.*, 2001; *Nakagawa*, 2000; *Anderegg et al.*, 1973]. These studies considered surfaces in space with ordinary surface smoothness only, but none has considered the highly smooth conditions as on the highly reflective modern mirrors in space.

[6] In section 2, we use a specific model, the Mott-Langmuir formulation, to contend that mirrors can charge to negative voltages in sunlight as they would in eclipse. In section 6, we apply this theory to a practical problem by examining the degradation of the solar panels of the Boeing 702 model satellite fleet. In section 4, we estimate the sputtering rate of the charged mirror surfaces. Finally, in section 10, we suggest a simple method to solve the problem.

2. Charging of Mirrors in Space

[7] The Mott-Langmuir equation is often a good approximation for describing the current balance of a spacecraft in the geosynchronous environment.

$$I_e(0)[1 - \langle \delta + \eta \rangle] \exp\left(-\frac{e_e \phi}{kT_e}\right) - I_i(0)\left(1 - \frac{e_i \phi}{kT_i}\right) = I_{ph} \quad (1)$$

where the notations are standard [see, e.g., *Lai and Della-Rose*, 2001]. If an uncharged ($\phi = 0$) spacecraft were put in the geosynchronous environment, it would intercept an ambient electron current $I_e(0)$ that is larger than the ambient ion current $I_i(0)$ by two orders of magnitude [*Reagan et al.*, 1983; *Lai and Della-Rose*, 2001]. This difference explains why geosynchronous spacecraft often charge to hundreds or thousands of volts negative during eclipses. In this example, the solution ϕ of equation (1) with $I_{ph} = 0$ represents the (negative volts) spacecraft potential. With a negative potential ($\phi < 0$), the spacecraft repels the ambient electrons as described by the exponential factor and attracts the ambient ions as described by the factor in parenthesis following the $I_i(0)$ factor. The secondary electron coefficient δ [*Sternglass*, 1954a; *Sanders and Inouye*, 1978] and backscattered electron coefficient η [*Sternglass*, 1954b; *Prokopenko and Laframboise*, 1980] account for the outgoing electron currents [e.g., *Lai and Della-Rose*, 2001]. At the equilibrium potential ($\phi < 0$), the electron current collected is reduced while the ion current is enhanced so that a current balance is achieved.

[8] With photoemission, the current balance is significantly affected. Since the flux of photoemission (I_{ph} per unit area) exceeds the ambient electron flux by a factor of 20 to100, there exists no solution ϕ (<0) to satisfy equation (1). Indeed, charging to hundreds or thousands of volts negative rarely occurs in sunlight. An exception is when the ambient electron current is unusually high, as in geomagnetic storms or when the photoelectron current is significantly blocked by potential barriers [*Olsen et al.*, 1981; *Besse and Rubin*, 1980], which depend on the geometry and material properties of the spacecraft.

[9] Normally, spacecraft surfaces often charge to a few volts positive in sunlight [*Lai et al.*, 1986]. The dominant line in the solar spectrum in the magnetosphere is the Ly α (121.6 nm) which has an energy $h\nu$ of about 10.2 eV. Typical values of work functions W_f of metals are about 4 to 5 eV. If the photon energy is transferred to an electron with full efficiency, the energy E of the photoelectron emitted is

$$E = h\nu - W_f \quad (2)$$

which is only about 5 to 6 eV. Therefore if the charging is beyond a few volts positive, the photoelectron would return, which explains why positive voltage charging is up to a few volts only, unless the satellite is near the sun where far UV lines become intense.

[10] It has been common to neglect the reflectance of surfaces in calculations of photoelectron currents emitted from spacecraft. To be careful, however,one should include reflectance [*Lai et al.*, 1986]. The photoemissivity J_{ph} per unit incident photon is called the photoelectron yield function Y, which is related to γ, the photoelectron yield per absorbed photon [*Hughes and Dubridge*, 1932; *Samson*, 1967], by

$$Y(\theta) = \gamma(\theta)[1 - R(\theta)] \quad (3)$$

where R is the reflectance of the surface material and θ is the Sun angle. In the limit of zero reflectance, $R = 0$ and $Y = \gamma$. In the limit of perfect reflectance, $R = 1$ and $Y = 0$. Since modern mirrors can achieve very high reflectance, $R \approx 1$, it is logical to infer that their photoemissivity is nearly zero, implying that they generate nearly zero photoelectron current. To prove this conjecture, we suggest that laboratory experiments on mirror charging in simulated exoatmospheric sunlight be conducted.

3. Remark on Reflection Depth

[11] Before we proceed, we address a fairly common question, "Does a mirror reflect at a deeper depth than photoemission?" If it does, photoemission would be unaffected, no matter how high the reflection is. We answer that the reflection depth μ can not be deeper than the photoemission depth x. For, if μ > x, the photon loses its energy hν to attentuation (i.e., excitation, photoionization, and transferring energy to the photoelectron as kinetic energy) at x before the photon reaches μ. As a result, the intensity of the photon will be much attenuated at μ and therefore high-efficiency reflection can not occur. Conversely, if $\mu < x$ and high-relectivity R occurs first, the reflection would shut off the attenuation process.

[12] In reality, the depths are not delta functions but have finite spreads. Nevertheless, the average relection depth $\langle \mu \rangle$ is shallower than the attenuation depth as argued above. The light intensity $I(x)$ at x is given in standard texts as follows [e.g., *Spicer*, 1972]:

$$I(x) = I(0)[1 - R(\nu)] \exp(-\alpha(\nu)x) \quad (4)$$

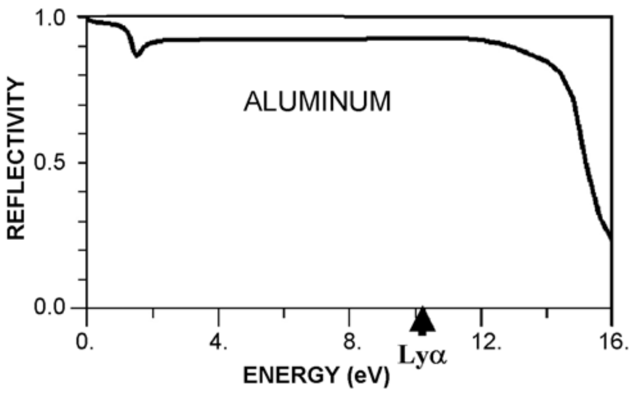

Figure 1. Reflectance of aluminum surface as a function of incoming photon energy (plotted by using data taken from *CRC Handbook of Physics and Chemistry*, CRC Press, 2001).

where R is the reflection coefficient, ν is the frequency of the photon, and α is the absorption coefficient which accounts for the attenuation losses due to excitation, photoionization, and kinetic energy transfer. If R is high (near unity) in equation (4), the intensity $I(x)$ at x would be shut off (near zero).

4. Remark on High Reflectivity

[13] High reflectivity of mirrors can be achieved in the UV region. The most common surface material for mirrors used in space is aluminum. For example, the mirror surface material of the Hubble telescopes is aluminum coated with magnesium fluoride [*Keski-Kuha et al.*, 1999]. Aluminum is highly reflective (nearly 90%) at the solar UV region [*Lide*, 2002] (Figure 1). Aluminum is subject to oxygen erosion and therefore is not suitable for low-altitude use. At geosynchronous altitudes, where the Boeing PAS-7 is, however, oxygen is much less abundant than at ionospheric altitudes. Coatings are used to protect aluminum from oxygen erosion at space shuttle altitudes (ionosphere) and to make the mirrors reflective for UV light (http://hubblesite.org/sci.d.tech/nuts_.and._bolts/optics/). Aluminum mirror coatings with reflectivity at 86% [*Keski-Kuha et al.*, 1999] at Ly α, the main solar UV spectral line responsible for photoemission, is used on Hubble.

5. Remark on Photoemission as a Function of Reflectivity

[14] It is a fact that all standard textbooks on photoemission states that photoemission is proportional to $(1 - R)$, where R is the reflection coefficient, or reflectance. The equation of photoemission yield is given in equation (3) and that of photoemission intensity is given in equation (4). That photoemission is greatly reduced by high surface reflectance is well accepted, with no known exception, in theory and in experiments by the top authorities in the physics of photoemission (M. L. Cohen, personal communication, 2002; Y. Petroff, personal communication, 2002; J. A. R. Samson, personal communication, 2002; F. Zimmermann, personal communication, 2002). There exists, however, one experiment [*Samson and Cairns*, 1965] which reported otherwise. In that experiment, an UV beam was aimed at a polished surface placed inside a vacuum chamber and an electron detector positioned nearby measured the photoelectron flux collected. A rough surface, with deep clevices generated by sand blasting, was then put in place of the polished surface. That paper reported the surprising result that less electron flux was measured in the sand-blasted case. J. A. R. Samson (personal communication, 2002) was not sure of the reason of this ‘‘extreme case’’ result but suggested that the deep clevice walls may reabsorb some of photoelectrons. We suggest that one needs to be careful by taking into account vacuum chamber wall effects for photoemission measurements conducted inside vacuum chambers, pipes, or closed compartments. It is highly possible that the UV light reflected efficiently by polished surface samples can generate photoelectrons from the walls of the vacuum chambers, pipes, or any closed compartments, resulting in high photoelectron fluxes collected by the electron detectors. The positve voltage charging of an isolated and floating sample emitting photoelectrons [*Lai et al.*, 1986] needs also be considered in such experiments.

6. Applications

[15] Having called attention to the possibility of mirror charging in sunlight, we turn next to applications in space. Mirrors have been used on spacecraft for optical and infrared communication relays, for example. They will probably be used in space more extensively in the new millennium. Recently, the Boeing company has used mirrors to enhance the solar radiation intensity on their solar panels in a fleet of satellites (Figures 2 and 3). This fleet, the Boeing 702 model satellites, includes communication satellites such as Telesat Anik F1 and Anik F2, PanAmSat’s Galaxy 11, and PAS-1, and PAS-7 (http://sat-nd.com/failures/702arrays.html). The solar panels of the entire 702 model fleet have suffered from sudden and permanent degradations, apparently of the same type but at different times, according to *Space News*, October 2000 (see also http://www.spaceandtech.com/digest/flash2001/flash2001-082.shtml). Degradation means that the solar cell becomes less efficient than expected and/or the mirrors become foggy.

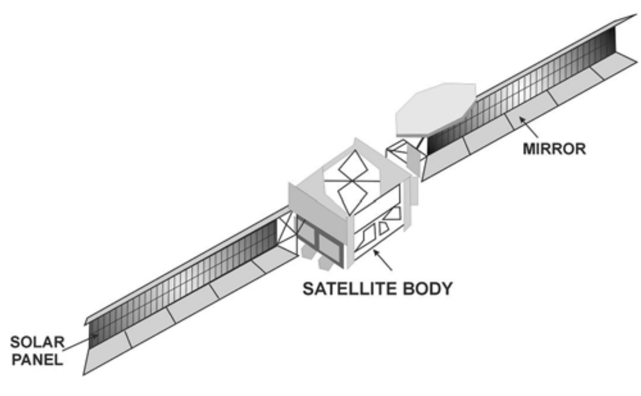

Figure 2. A schematic diagram of the PAS-7 satellite. The solar panels are flanked by mirrors for enhancing sunlight on the solar panel.

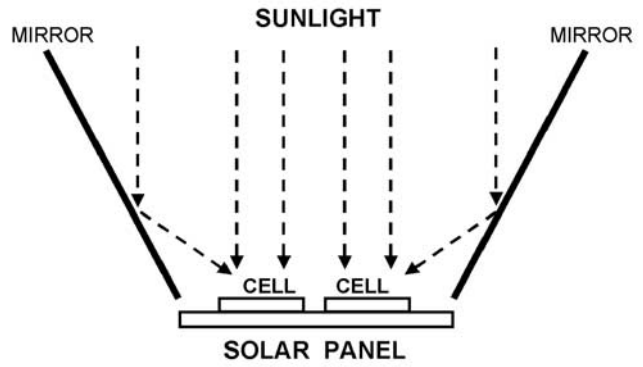

Figure 3. A schematic diagram of sunlight enhancement on the solar cells located on the solar panel.

[16] In particular, the PAS-7 panel suffered from a 25% sudden and permanent degradation after it came out of eclipse in the morning of 6 September 2001 (*Space News*, 1 October 2001; see also http://www.spaceandtech.com/digest/flash2001/flash2001-081.shtml). No cause has been officially announced. The true cause may never be known, but we suggest that the mirrors and the solar panel may have experienced differential potentials during the morning hours upon the eclipse exit. Differential charging is a potential space hazard. It may have been the cause, leading to sudden discharges and a resulting degradations. Degradation due to occasional discharges and prolonged sputtering by acclerated ions can reduce the smoothness and therefore the reflectance of the mirrors (see sections 8 and 9).

[17] We remark that it is not the purpose of this paper to pin down the exact cause of the PAS-7 and Boeing mirror solar panel stepwise and gradual degradations. Lacking engineering details and diagnostic data, it is impossible to pin down the exact cause. The main point in this paper, as emphasized in the beginning, is to point out the important property that highly reflective mirrors reflect sunlight efficiently resulting in lowering of photoemission. If the reflectivity is sufficiently high and therefore the photoemission is sufficently lowered, the surface can charge to high negative potentials in sunlight as if in eclispe. This property can have significant consequence.

7. Space Environment of the PAS-7 Satellite

[18] The PAS-7 satellite was at 68.5° (east) longitude in a geosynchronous orbit when the solar panel degradation occurred in the early morning of 6 September 2001. The nearest satellite that measured spacecraft charging was the geosynchronous satellite LANL-97A at 69.4°. Figure 4 (lower) shows the LANL-97A charging level (about −3 kV) as revealed by the ion flux spectrum measured on the satellite. The ion peak was due to the shift of the ion energies by the spacecraft (negative) potential which attracted the (positive) ions during eclipse. The amount of the shift, given by the energy location of the ion peak, indicates directly the spacecraft potential (readers unfamiliar with this measurement technique may consult *Lai* [1998]).

[19] As Figure 4 (bottom) shows, the jumps in spacecraft potential (to and from about −3 kV) were abrupt; the negative potential appeared upon eclipse entrance and disappeared upon eclipse exit. Figure 4 (top) shows the electron flux spectrum measured on LANL-97A in the same period. During eclipse, the low-energy electrons (up to about 3 keV) were absent, partly because they were repelled by the −3 kV potential of the spacecraft. The absence of low-energy electrons was also partly due to the lack of ambient photoelectrons in space during eclipse (M. F. Thomsen, personal communication, 2001).

8. Differential Charging as a Space Hazard

[20] Since PAS-7 was near LANL-97A, it may have experienced similar charging during its eclipse passage. If charging occurred, the satellite ground, the solar panels, and the mirrors would charge negatively to perhaps thousands of volts. When the satellite exited from the eclipse, the charging of the satellite ground and the solar panels would cease, and they would reach practically zero potential. The time of potential change depends on the capacitances of the surfaces and the coupling of the surfaces involved. However, according to our conjecture, the mirrors, being nearly perfect reflectors, would continue to remain at thousands of volts negative as if they were still in eclipse. Therefore the potential differences between the mirrors and the solar panels and between the mirrors and the satellite body could reach thousands of volts, but the distance separating the panels and the mirror could be small.

[21] Differential charging does not necessarily imply that a sudden discharge must follow. Yet it poses a potential space hazard. A small perturbation may trigger a discharge, either a small transient one or a sustaining avalanche. Such a triggering perturbation may come from impacts of energetic ions that generate cascades of ionization charges or from the hypervelocity impacts of debris or meteorites [*Lai*, 2001a; *Lai et al.*, 2002]. Whether an avalanche ionization condition is satisfied for sustaining a discharge depends on the electric field supplying kinetic energy (Paschen discharge) to the electrons that perform impact ionization, the ambient plasma density, the cross sections of ionization, and the loss mechanisms. The situation is reminiscent of the critical ionization velocity (CIV) discharge conditions [*Lai and Murad*, 1992; *Lai*, 2001b], the main difference being that in a CIV discharge the energy is supplied by the relative velocity between a plasma and a neutral cloud, whereas in a Paschen discharge the energy is furnished by an external electric field. In the PAS-7 case, the electric field could be significant if the flanking mirrors were very near the solar panels and the difference in potentials were thousands of volts. The magnitude of the discharge current depends not only on the capacitances of the surfaces involved but also on the avalanche ionization generated from the ambient neutrals and the vapor coming off the discharge points. While the details of PAS-7 solar panel and spacecraft structures are unavailable, we can only refer to a basic theory of Paschen discharge on differentially charged spacecraft [*Lai et al.*, 2002].

[22] For completeness, we mention the criteria in section 2.3.1 of *Purvis et al.* [1984]: "If either of the following criteria are exceeded, discharges can occur: (1) Dielectric surface voltages are greater than 500V positive relative to an adjacent exposed conductor, (2) The interface betweeen a dielectric and an exposed conductor has an electric field

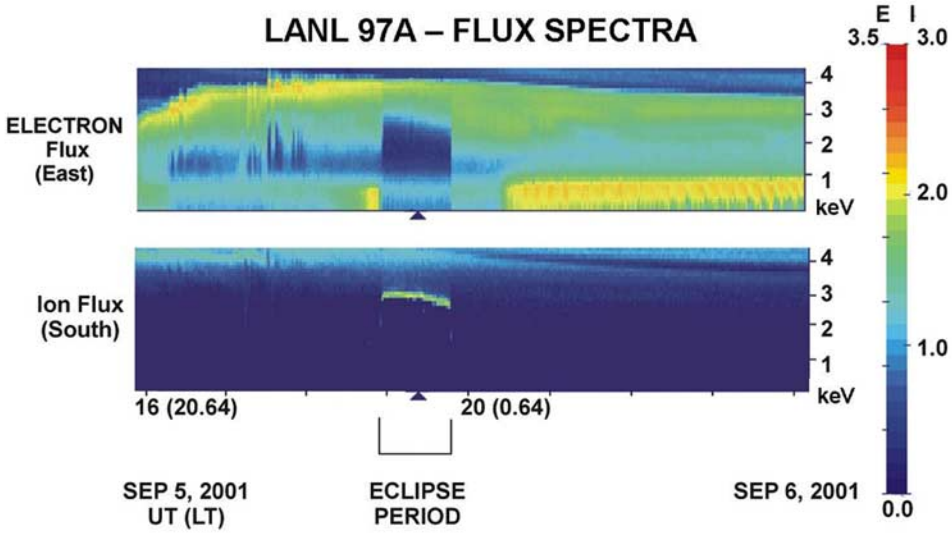

Figure 4. (top) The electron spectrum measured on LANL in the eclipse period, 5–6 September 2001. (bottom) Charging measurement on LANL-97A. The bright ion line of the ion flux spectrum jumps to about −3 kV during the eclipse period, 5–6 September 2001.

greater than 1×10^5 V/cm." Without knowing the ionization cross sections of the desorbed molecular species involved, we can not comment on criterion 1. Without knowing the distances of separation between the solar panel and the mirrors, edges, interconnects, etc., we can not comment on criterion 2. We merely point out that if, upon eclipse exit, the mirrors continue to charge at the eclipse charging level (−3 kV in Figure 4) while the rest of the satellite body charges to near zero potential in sunlight, then the critierion 1 is easily satisfied because the differential charging ($\Delta\phi \approx 3$ kV) is greater than 500 V. If the separation Δs is small, say 0.01 cm, then an electric field $E = \Delta\phi/\Delta\Gamma \approx 3 \times 10^5$ V/cm can be reached. We do not attempt to pin down the exact numbers here, since it is not the purpose of this paper. Much of the engineering details are unavailable.

9. Mirror Degradation by Sputtering

[23] The degradation in performance may be caused by damage to the mirror surface by sputtering. We now provide a rough estimate of this degradation effect. If mirrors are negatively charged to several kilovolts in eclipse and in sunlight, prolonged bombardment by positive ions attracted toward the mirrors may cause physical and optical degradation of the mirror surfaces. A standard formula [*Thomas*, 1985] to calculate the coefficient of sputtering is of the form

$$S = 6.4 \times 10^{-3} M \left[\frac{4mM}{(m+M)^2}\right]^{5/3} E'^{1/4}\left(1 - \frac{1}{E'}\right)^{7/2} \quad (5)$$

where S is the sputtering coefficient, defined as the number of target atoms ejected per incoming ion; m and M are the projectile ion and target atom masses (in amu), respectively; and E' is the ratio of the projectile kinetic energy E and the critical energy E_o for the onset of sputtering.

$$E' = E/E_o \quad (6)$$

[24] The value of the coefficient S(E) of aluminum is about 10^{-2} for H^+ and about 2×10^{-1} for He^+ at $E = 1$ keV [*Thomas*, 1985]. The average ion flux J_i at geosynchronous altitudes is about $J_i = 2 \times 10^{-7}$ ions cm^{-2} s^{-1} [*Lai and Della-Rose*, 2001]. The average sputtering rate R of Al by H^+ is given by $R = J_i S = 2 \times 10^7 \times 10^{-2} = 2 \times 10^5$ cm^{-2} s^{-1}. Taking the number density n of aluminum to be $n \approx 6 \times 10^{22}$ cm^{-3}, we estimate the average rate α of depth removal of Al by H^+ sputtering to be $\alpha = R/n = 2 \times 10^5/6 \times 10^{22}$ or approximately 3.3×10^{-18} cm s^{-1} = 10^{-11} cm/month. In geomagnetic storms, the density [O^+] of oxygen ions increases significantly [*Daglis*, 2001]. The sputtering yield with O^+ as the projectile is higher than with H^+, while the flux is also higher in storms. Therefore the mirror degradation rate during severe space weathers greatly exceeds the estimate given above.

[25] Sputtering is nonuniform because some atomic species are more likely to be removed from materials made of various atomic species. As a result, prolonged sputtering may cause roughness on mirror surfaces and therefore diminish the reflectance. Unlike electrostatic discharges between the solar panels and their adjacent mirrors, sputtering degradation of mirror surfaces causes lower current production but not actual solar cell damage.

10. A Proposed Method for Mitigating Mirrors Discharge

[26] Most mirrors in space are coated with metals such as aluminum or titanium dioxide. We propose connecting the conductive mirror coating to a nonreflective sunlit satellite ground via a charge storage device such as a rechargeable battery (Figure 5). The connection would channel the electrons collected on the mirror surfaces to the storage battery. Removing the electrons would lower the voltage difference between the mirror surfaces and the nonreflective sunlit satellite ground. Reducing the voltage difference would eliminate the potential hazard, sudden discharges, and degradations caused by the discharges.

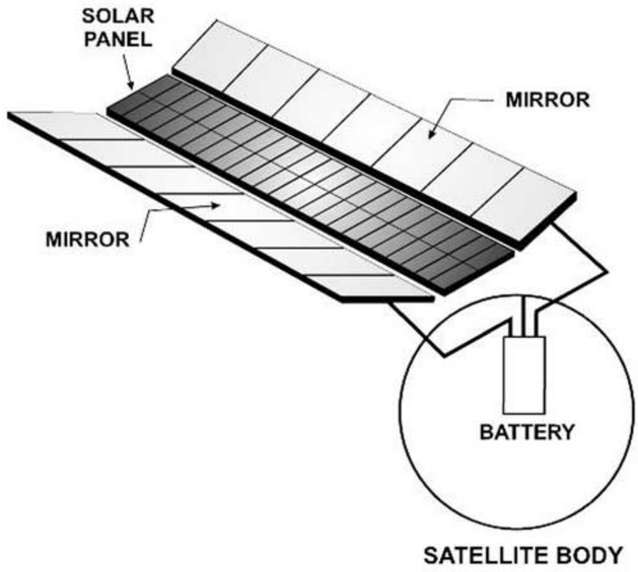

Figure 5. A suggested mitigation method to solve the differential charging problem. A conducting wire connects the conducting mirror surface to a non-reflective ground of the spacecraft via a rechargeable battery.

[27] The charge stored in the battery over months and years may be used in times of need or to supplement the solar batteries when necessary. As an estimate, we take the average electron flux J in the geosynchronous environment to be 0.115×10^{-9} A/cm^2 [*Purvis et al.*, 1984] and the area A of a Boeing 702 model satellite mirror (http://www.hughespace.com/factsheets/702/) to be $A = 41 \times 8$ m^2. The approximate charge Q intercepted by the mirror in a year (τ = 12 months) is given by

$$\begin{aligned} Q &= JA\tau \\ &= 0.115 \times 10^{-9} \times (41 \times 8) \times 10^4 \times (12 \times 30 \times 24) \times 3600 \end{aligned} \tag{7}$$

which is 1.2×10^6 Coulomb. With four mirrors, the annual charge intercepted from the space plasma and stored in the rechargeable battery is approximately 4.8×10^6 Coulomb or 1333 ampere-hours.

[28] The primary purpose of the suggested connection of the mirrors to a rechargeable battery is to prevent mirror-charging in sunlight and hence reduce the hazard of sudden discharges between the mirrors and their adjacent solar panels, which are most precious. The collection and storage of charge by the rechargeable batteries serve an additional purpose. If the solar cell circuit is broken by a hypervelocity meteor/debris impact, the rechargeable battery, which can be located and protected inside the satellite body, may provide emergency functions, albeit not for a long time. For example, transmitting the anomaly data to the ground for diagnostic purpose. Finally, nonconducting mirror surfaces could be coated with conductive reflective materials.

[29] The method discussed above in this section is one of the simpler methods for mitigating spacecraft charging and differential charging. There are other mitigation methods, each having its advantages and disadvantages. For example, *Grard* [1975, 1976] suggested using field emission of electrons from sharp spikes for mitigating spacecraft charging. For a recent comprehensive review on spacecraft charging mitigation methods, see, for example, *Lai* [2003].

11. Summary and Conclusion

[30] Surface reflectivity is important in calculations of photoemission in sunlight. Since photoelectron current often exceeds the ambient currents, it is often the dominating one compared with all other currents. Reflectivity therefore greatly affects surface charging in sunlight. Modern mirrors can achieve very high reflectivity. We conjecture that mirrors on geosynchronous spacecrafts would charge as if they would in eclipse, namely, to several kilovolts negative when the space plasma temperature is high.

[31] We offer this idea as a plausible explanation for the sudden and permanent degradations of the solar panels on the Boeing 702 model satellites. This fleet features a special design of solar panels flanked by mirrors on both sides to intensify the solar radiation on the solar cells. The entire fleet has experienced gradual and permanent solar cell degradations. According to our mirror-charging idea, both the solar panel and the mirrors would charge in eclipses to high negative potentials if the ambient plasma is hot but, upon eclipse exit, the solar panel potential would return to a low charging level while the mirrors remain at high negative potentials. Thus the eclipse exit on the morningside creates a differential charging situation. It is well known that the morningside is more prone to spacecraft charging because the curvature and gradient drifts drive the hot electrons eastward at geosynchronous altitudes. Therefore a sudden development of differential charging up to thousands of volts in the morning sector of the geosynchronous environment is a potentially hazardous situation. Whether a disastrous discharge might follow depends on the existence of a triggering mechanism such as a hypervelocity impact by a meteorite, the conditions for sustaining an avalanche ionization, and the loss mechanisms.

[32] Sputtering by ambient ions on the mirrors is enhanced by the negative charging. If the mirror surfaces are made of various atom species, differential removal rate by sputtering may cause surface roughness and reduction of the efficiency of the mirror. Although the sputtering rate is small, prolonged sputtering every day, not only in eclipse but also in sunlight, would shorten the useful life of the mirrors. Unlike sudden discharges, which may cause damage or stepwise degradation to the solar cells, sputtering causes gradual degradation only to the mirrors. We have suggested a simple method for mitigation of differential charging.

[33] Finally, we suggest that reflectivity should be included as an important parameter in the development of modern spacecraft charging codes such as NASCAP 2000 and future versions. For spacecraft designers, the reflectivity of some highly reflective surfaces such as mirrors, radiators, should be included in their considerations.

Appendix A: Reduced Photoemission

[34] This appendix argues that the the photoemission current can be sufficiently reduced by high reflectivity.

The average ambient electron flux J_e at geosynchronous altitudes is 0.115×10^{-9} A/cm^2 [*Purvis et al.*, 1984], while the photoemission flux J_{ph} from typical surfaces at norrmal conditions is 2×10^{-9} A/cm^2 [*Stannard et al.*, 1981] with variations depending on surface material and conditions [*Grad*, 1973; *Pedersen*, 1995; *Nakagawa et al.*, 2000]. Thus the average ratio is about 20.

$$A = J_{ph}/J_e \approx 20 \tag{A1}$$

For a surface collecting ambient electrons and emitting photoelectrons from the same area, equation (A1) suggests that the ambient electron current I_a can not compete with the photoemission current I_{ph} unless the former increases by a factor of about 20. In storm periods, the ambient current increases [*Purvis et al.*, 1984], the ration A becoming about 4 on the average.

[35] We now argue how even the (nonstormy period) factor 20 can be reduced to a value comparable with the ambient electron current. First, in any geometry, the sunlit area of a spacecraft, or an object in space, is at most half of the total area. While photoemission is from at most half of the total area, ambient electrons impact on all areas. Since the current I equals the flux J multiplied by the area for normal incidence of sunlight ($\theta = 0$), the current ratio A as deduced from equation (A1) is reduced to about 10.

$$A = I_{ph}(0)/I_a \approx 10 \tag{A2}$$

Second, for general incidence angle θ of sunlight, the photoemission current I_{ph} is given by

$$I_{ph}(\theta) = I_{ph}(0)(1 - \psi(\theta)) \tag{A3}$$

where the $\psi(\theta)$ factor considered in equation (A3) is due to the effective area of sunlight received on a surface with incidence angle θ and not related to the smoothness of the surface.

$$\psi(\theta) = \cos\theta \tag{A4}$$

(Actually, the reflectivity $R(\theta)$ in equation (3) increases with θ [*Hughes and Dubridge*, 1932; *Palik*, 1985]. At normal incidence, $R(\theta)$ is minimum. At grazing incidence (90 degree), $R(\theta)$ equals 1. This angle dependence has nothing to do with the effective area but with the physics of light reflectance. We do not even need to invoke this factor in arguing for reduction of the ratio A of photoemission current to ambient electron current. If we do, A in equation (A2) would be reduced even further.)

[36] Lacking any geometrical details of the solar panels, let us, as an exercise, assume that the width of each mirror flanking the solar panel equals that of the solar panel (Figure 2). In order for sunlight to reflect from the mirror onto the panel, the angle of sunlight incidence θ on the mirror has to be 60 degrees (<SAN in Figure A1). At 60 degrees, if the mirror is wider, the mirror area beyond its panel width would not contribute to the sunlght reflection onto the solar panel. If the incidence angle θ is less than 60 degrees, the sunlight reflected from the mirror would not hit the solar panel. (Figure A1). With θ = 60 degrees in equation (A4), we have

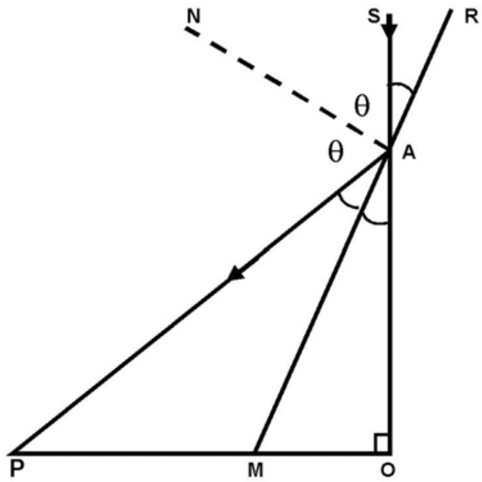

Figure A1. Geometry of reflection angle. PM is a solar panel, MA is a flanking mirror, the length PM equals MA, N A is the normal to the mirror at A, SA is the incoming sunlight from the sun, and AP is the reflected sunlight onto the solar panel. It is simple to deduce that the angle θ is 60 degrees. If the mirror is longer (extended to R), the reflected sunlight from the extended length (AR) would not land on the solar panel.

$$\psi(\theta) = \cos\theta = 1/2 \tag{A4$'$}$$

and therefore equation (A2) becomes

$$A = I_{ph}(\theta)/I_a = 5 \tag{A5}$$

If the angle θ is larger than 60 degrees, the ratio A in equation (A5) would be even smaller.

[37] Finally, with mirror surface smoothness, the reflectivity R can be up to about 0.9 as discussed in the main text of this paper. Combining all three factors, we have

$$A = I_{ph}(\theta = 60^\circ, R = 0.9)/I_a = 0.5 \tag{A6}$$

Thus we have argued that the photoemission current I_{ph} is comparable with the ambient electron current I_e. If one includes the θ dependence of $R(\theta)$, the ratio A in equation (A6) can be lowered further but only slightly at θ = 60 degrees. For our purpose of an estimate, equation (A6) suffices.

Appendix B: Charging Potential With Reduced Photoemission

[38] To illustrate the effect of reduced photoemission on charging potential, we consider the current balance equation in the Langmuir orbit-limited regime, which is often a fairly good approximation for geosynchronous charging calculations. Without loss of generality, we consider the equation in one, two, and three dimensions.

$$I_e(0)[1 - \langle\delta + \eta\rangle]\exp(-e_e\phi/kT_e) - I_i(0)\mu\left(1 - \frac{e_i\phi}{kT_i}\right)^\alpha - I_{ph} = 0 \tag{B1}$$

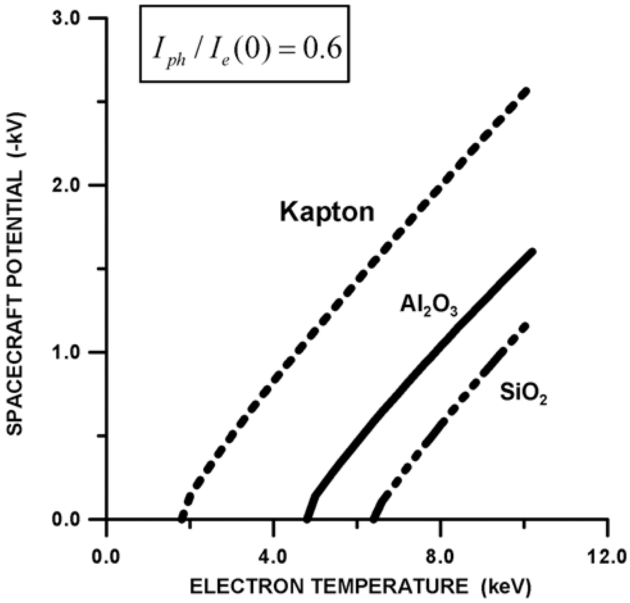

Figure B1. Surface potentials of kapton, aluminum oxide, and silicon. The Langmuir orbit-limited current balance equation has been used. The inputs assume the ambient ion current to be 0.05 that of the ambient electrons $I_e(0)$ at zero potential and the photoelectron current emitted to be 0.6 that of the ambient electrons $I_e(0)$.

where

$$\langle \delta + \eta \rangle = \frac{\int_0^\infty dE E f(E)[\delta(E) + \eta(E)]}{\int_0^\infty dE E f(E)} \tag{B2}$$

In equation (B1), $I_e(0)$ is the ambient electron current at zero spacecraft potential $\phi = 0$, $I_i(0)$ is the ambient ion current at $\phi = 0$, e_e is the electron charge, e_i is the ion charge, T_e is the ambient electron temperature, T_i is the ambient ion

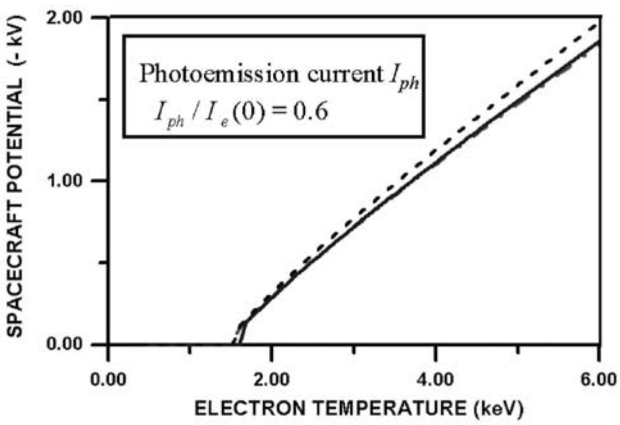

Figure B2. Calculated surface potentials of kapton in 1-D, 2-D, and 3-D with $I_{ph}(0) = 0.6 \times I_a(0)$. The dimensional effects are small because the ion collection term is small compared with the photoelectron current unless at high surface potentials.

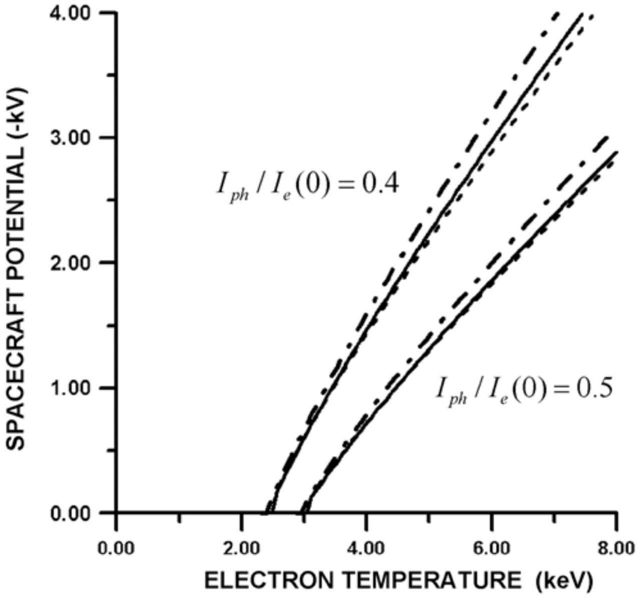

Figure B3. Calculated surface potentials of aluminum oxide in 1-D, 2-D, and 3-D with $I_{ph}(0) = 0.4 \times I_a(0)$ and $0.5 \times I_e(0)$. Dash-dot-dash is for 1-D, solid for 2-D, and dash for 3-D.

temperature, μ is a factor for two dimensions only ($\mu = 1$ for a sphere, $\mu = 1.1$ for an infinite cylinder, and $\mu = 1$ for a plane) [*Mott-Smith and Langmuir*, 1926; *Laframboise and Parker*, 1973; *Lai*, 1994], $\alpha = 1$ for sphere, 1/2 for cylinder, and 0 for a plane, and I_{ph} is the photoelectron current. In equation (B2), δ and η are the secondary and backscattered electron emission coefficients, respectively, and $f(E)$ is the ambient electron distribution function. The notations are as in the work of *Lai and Della-Rose* [2001].

[39] In order to demonstrate that negative charging is possible for mirrors with reduced photoemission, we show equations (B1) and (B2) to calculate the surface potentials. Figure B1 shows that with a photoemission current I_{ph} equal to 0.6 times the ambient electron current I_e, i.e., with the ratio A satisfying equation (A6), the surface potentials of kapton, aluminum oxide, and silcon oxide, charge to negative potentials for a sphere. To show the effects of two-dimensions (2-D) and 1-D, we calculate the surface

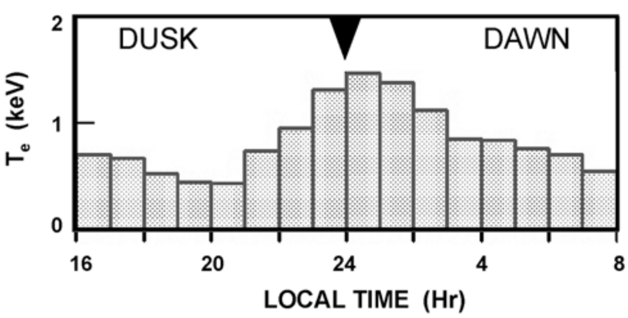

Figure C1. Asymmetry of electron temperature. The average electron temperature is higher after midnight (triangle) as measured by LANL-97A, September 2001.

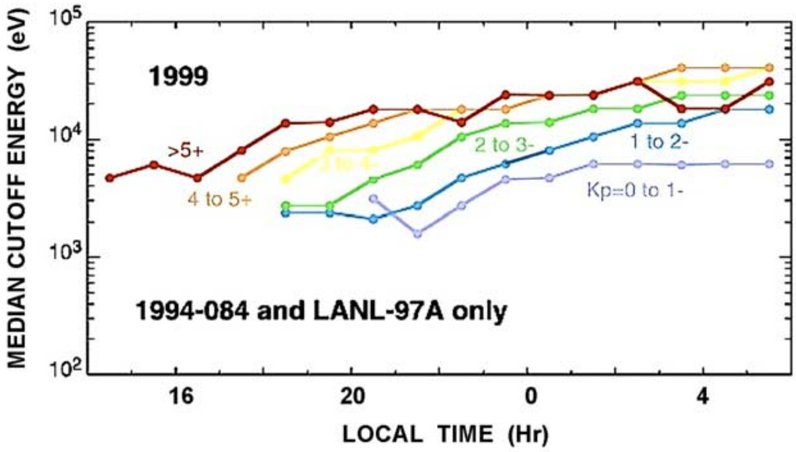

Figure C2. Asymmetry of cutoff energy as a function of local time. The cutoff energy is lower before midnight. (Courtesy of *Thomsen et al.* [2002].)

potentials of kapton in Figure B2. The results show very little difference in the various dimensions because the dimensional effect is governed by the ion collection term, which is small compared with the photoemission term unless at high surface potentials. If the ratio A is less, for example, $A = 0.4$ or 0.5, the behavior of aluminum charging to negative potentials is shown in Figure B3. The computational results demonstrate that it is possible for mirrors to charge to negative potentials in sunlight. We have not even invoked stormy conditions. In stormy conditions, charging of mirrors would be even more likely. In contrast, with sunlight concentrating on the solar panel which is not a mirror, photoemission from the solar panel would prevent it from charging in sunlight.

Appendix C: Space Hazard Upon Eclipse Exit

[40] This appendix explains why it is more hazardous after eclipse exit than before eclipse entrance, especially for a satellite equipped with mirrors of high reflectance. The reasons are based on two space environmental factors, namely, (1) electron temperature and (2) upper energy cutoff of the electron distribution function. This appendix is of general or basic interest and not for a specific case such as, for example, the PAS-7 Satellite anomaly only.

[41] When the satellite is in eclipse, both the mirror and the rest of the satellite body do not emit photoelectrons. They both charge to negative potentials when the ambient electron temperature exceeds the critical temperature of the surface material. The magnitude of the potential generally increases with the electron temperature. Outside the eclipse, the mirrors, emitting little or no photoelectrons despite in sunlight, charge to negative potentials when the ambient electron temperature exceeds the critical temperature. As in eclipse, the level of charging of the mirrors increases with the electron temperature. However, the rest of the satellite body emits photoelectrons outside the eclipse and therefore does not charge to high negative potentials. Thus high-level differential charging between the mirrors and the satellite body may ensue upon an eclipse exit, posing a hazard.

[42] Differential charging can occur similarly before an eclipse entrance. However, the level of differential charging is more pronounced for elipse exit than entrance. A reason is that the hot electrons in the midnight sector of the geosynchronous region tend to drift eastward because of the Earth's dipole magnetic field curvature and gradient, as explained in standard textbooks [e.g., *Kivelson and Russell*, 1995]. Figure C1 shows the typical asymmetry of the electron temperature before and after midnight as observed on a geosynchronous satellite. This typical behavior shows that the electron temperature is higher on the dawnside than on the duskside. Since the onset of charging occurs when the temperature reaches the critical temperature and the magnitude of the charging potential increases with the temperature, charging is more likely on the dawnside than on the duskside of midnight. Indeed, it is well known that the level of spacecraft charging is usually higher on the dawnside than on the duskside of midnight.

[43] The second reason has to do with the electron distribution. Because of drifts, hot electrons are often less abundant on the duskside, forming a distribution with a steep reduction (or cutoff) beyond a cutoff energy. Figure C2 shows the measurements made by *Thomsen et al.* [2002]. The cutoff energy is lower on the duskside than at dawn for all values of kp but more prominently during low kp. The existence of a finite cutoff energy E_{upper} affects the critical temperature. Indeed, a Maxwellian model featuring a finite cutoff energy [*Lai*, 2004] shows that the presence of a cutoff energy raises the critical temperature. That is, the cutoff energy renders charging less likely.

[44] To calculate the critical temperature for a cutoff distribution, one writes down the current balance equation with the integration range from $E = 0$ to $E = E_{\mathrm{upper}}$ instead of infinity. Such a calculation [*Lai*, 2004] yields the result that the critical temperature increasing as the finite cutoff energy decreases. Figure C3 shows the critical temperature (solid curve) for a gold surface with an isotropic incoming electron flux. For example, if $E_{\mathrm{upper}} = 10$ keV, the critical temperature $T^* = \infty$, implying no spacecraft charging. If $E_{\mathrm{upper}} = 20$ keV, the critical temperature $T^* = 5.5$ keV approximately. If $E_{\mathrm{upper}} = \infty$ (no cutoff), one

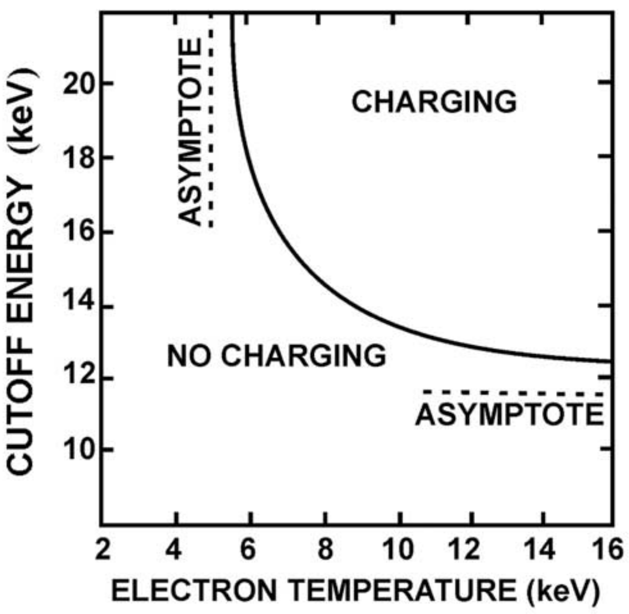

Figure C3. Critical temperature for the onset of spacecraft charging. The dependence on the cutoff energy is shown. Below a critical cutoff energy, spacecraft charging can not occur. (Courtesy of *Lai* [2004].)

recovers the Maxwellian result T^* = 4.9 keV (vertical asymptote).

[45] In conclusion, there is asymmetry in spacecraft charging before and after midnight, the charging level being usually higher on the dawnside. Satellite eclipse occurs around midnight. Hazard of differential charging is more important upon eclipse exit than entrance.

[46] **Acknowledgments.** I would like to thank James A.R. Samson, Yves Petroff, and Frank Zimmermann for confirmation that photoemission is proportional to the complement of reflectance. I am especially thankful to James A.R. Samson for pointing out the presence of cleavages in his sand-blasted sample and to Frank Zimmermann for clarifying the extra source of photoemission from closed chamber walls receivng photons reflected from the highly reflective samples placed inside the chamber. The Los Alamos MPA measurements were obtained from the CDAWeb data service at NASA Goddard Space Flight Center. The author thanks Michelle F. Thomsen for permission to use the data. He is grateful to David Cooke for the PAS-7 news on *Space News*, 1 October 2001, Edmond Murad for invaluable encouragement, David J. Knecht and Alex Smith for correcting a few careless mistakes, Charles Swenson for his appreciation of my AGU presentation of this work, and Maurice Tautz for reading the manuscript with helpful comments.

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