Spacecraft Charging, Plume Interactions, and Space Radiation Design Considerations for All Electric GEO Satellite Missions

Justin J. Likar, Alexander L. Bogorad, Kevin A. August, Robert E. Lombardi, Keith Kannenberg, and Roman Herschitz

Abstract— Use of Electric Propulsion for Geostationary Earth Orbit stationkeeping and Geostationary Transfer Orbit offers satellite operators the opportunity to reduce mission costs and increase revenue by enabling a higher dry mass to orbit compared to traditional chemical or electric propulsion systems. The penalty for such benefits comes, initially, in the form of an increased time-to-orbit whereby the low-thrust Transfer Orbit duration will range from a few months, at best, to possibly 1 yr or more. During the low-thrust transfer, the spacecraft will experience prolonged exposure to portions of the trapped radiation belts to which GEO spacecraft otherwise would not be subjected – most notably the Inner Belt and slot region. Further, the spacecraft will also experience on the order of 10^5 hr of high density, high energy plume plasma – a more challenging operational environment compared to that typically applied to GEO spacecraft. Applicable environments are introduced, an overview of specific operational effects is provided, and design guidelines are highlighted.

Keywords—Spacecraft charging; Hall Current Thruster; Electric Propulsion

I. INTRODUCTION

Many spacecraft manufacturers are offering satellite architectures which include an “All Electric” or “All EP” propulsion system. Consider recent announcements which include the Boeing 702SP (press release 2012), Lockheed Martin A2100 (press release of 9 September 2013), and the ESA Electra partner program. Space Systems / Loral has a growing on-orbit history of using Stationary Plasma Thrusters (SPT) for station-keeping on GEO spacecraft. Benefits for the use of EP for Geostationary Earth Orbit (GEO) stationkeeping and Geostationary Transfer Orbit (GTO) are well documented as EP offers satellite operators the opportunity to reduce mission costs and increase revenue by enabling a higher dry mass to orbit compared to traditional chemical or electro propulsion systems. The acceptance and implementation of EP for space missions has been gradual, beginning with smaller-scale maneuvers such as on-orbit stationkeeping for GEO satellites. Large Delta-V transfers to GEO that primarily rely on an electric stage have been, historically, prohibitive for commercial satellite applications due to the high onboard power requirements and long transit times. Recent trends in commercial GEO satellites, however, show an increase in beginning-of-life (BOL) power-to-mass ratios that may favor the present-term use of EP for orbit raising and insertion into GEO [3]. Generally speaking the GTO duration for All EP missions will be on the order of months but may exceed 1 yr depending on many mission-unique or design-specific variables [1, 2].

In addition to prolonged exposure to the GTO trapped radiation environment a GEO satellite will, as a result of the thruster plume properties, be subjected to a LEO-like plasma environment during thruster operation. The time duration over the transfer mission and on-orbit mission lifetime will easily exceed 10,000 hr. Designers of GEO spacecraft must, therefore, consider design environments and guidelines historically created for LEO spacecraft. Plasma characteristics of the thruster exhaust plume are of primary importance in understanding how the exhaust particles interact with the spacecraft. Several general characteristics of the thruster plume are of significance.

The accommodation of HCTs, and similar advanced EP systems, on spacecraft presents numerous design challenges

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including surface coating / film erosion, surface contamination, current collection by exposed metals / conductors, spacecraft floating potential variations, electromagnetic interference, plume-induced torques, and others beyond the scope of the present study which addresses design challenges associated with spacecraft charging and the space radiation environment.

The purpose of the present manuscript is to summarize the appropriate plasma environments, identify associated performance risks, and also avail satellite designers and operators with credible cost-effective design methods / solutions to ensure mission success is achieved without unnecessary risk and cost burdens. For example, when seeking to achieve the financial benefits of an All EP mission it may be more desirable to trade a, comparatively, complex active charge control system, surface charging management system, and a suitably robust passive system for the mitigation of surface charging caused by the removal of thin-film charging control coatings.

Analysis, simulation, and ground test results are presented for plume induced ESD on various solar cell coupons and plume induced erosion of coatings. Additionally, trapped radiation environment spectra are presented for a generalized GEO + GTO mission using AE9 / AP9 / SPW (IRENE) [3], and AE8 / AP8 for comparison. Dose depth curves, internal charging spectra, and solar cell degradation parameters are also presented and compared with design standards.

II. DESCRIPTION OF ENVIRONMENTS

Recent years have yielded optimized low-thrust transfer mission designs from all launch sites (Baikonur, Equator, Kourou, and Kennedy) and using a variety of launchers (Atlas 5, Falcon 9, Sea Launch, and Delta IV) [1, 2]. Increased Beginning-Of-Life (BOL) power-to-mass ratios, improvements in available power, and competitiveness of available launchers have rendered low-thrust transfers an attractive option for manufacturers and operators alike. Ref. [1] includes several optimized low-thrust transfers for a variety of BOL dry mass values (3700 kg to 8700 kg) and launchers; all transfers are achieved in <130 d although ultimate durations may be 2× to 3× longer in some circumstances.

Fig. 1 [4] summarizes the minimum-time low-thrust trajectory to GEO with an initial orbit of 185 km × 35786 km at 28.5 deg for a 1200 kg dry mass and BOL power of 5 kW. The right hand side of Fig. 1 identifies the approximate altitude regions associated with the radiation and plasma related threats to spacecraft operations that are enhanced or created by the low-thrust GTO. Contributions to radiation dose (ionizing and non-ionizing) as well as plume effects and internal charging effects are discussed further herein. Fig. 2 shows the relative locations of the Van Allen Belts and common mission orbits (GEO Telecommunications and MEO Navigation).

Fig. 1. Altitude profile for minimum-time All EP GTO (left figure) [4]; relative magnitude of radiation and plasma environment risks to spacecraft operation / design (right figure). Dark colors indicate increased risk / threat magnitude whilst light colors indicate lower risk / threat magnitude. For example the threat of Atomic Oxygen (AO) disappears after a few hundred km.
A. Plume Plasma

Most modern All EP systems utilize Hall Current Thrusters. Ref. [5] provides an excellent description of Hall Current Thruster, and Electric Propulsion basics. Fig. 3 includes a basic schematic identifying thruster features. Neutral gas, typically Xenon, is injected into the discharge channel. Electrons emitted by the cathode move into the discharge chamber where they are “trapped” by the radial magnetic field which impedes axial motion and establishes a potential gradient. The electrons collide with, and ionize the neutral Xenon ions which are then accelerated through the potential gradient, producing thrust.

Studied by a number of groups in recent years [6-10], the HCT plume (Fig. 4) consists mainly of ions generated from two main physical processes. The first is due to energetic beam ions produced and accelerated by the thruster fields (primary beam ions). These are the dominant ion species and are the major source of thrust. The second source of ions is due to charge exchange reactions between beam ions and neutral xenon gas.

In typical spacecraft architectures (Fig. 5) nearly all sensitive surfaces are located within the plume wake region (that defined by angles of 90 deg or more as measured from the thrust axis) as shown. Although the distance between thrusters and array surfaces varies by design it is generally in the 1 m to 2 m range. Present best efforts yield predicted plume plasma densities at solar array surfaces in the range of $10^{12} \text{ m}^{-3}$ to $>10^{14} \text{ m}^{-3}$. Predicted ion energies are typically <80 eV although in some specific system architectures, thruster operating modes, and solar array configurations portions of the array surfaces may see ions of energy >120 eV. A number of public and proprietary tools are available to characterize various aspects of the plume environment in the presence of a spacecraft. It is not trivial to extend ground-based plume property characterization to flight-like or on-orbit systems and there are few examples of high quality sensors or sensor suites operating on spacecraft which utilize HCTs. Certainly defining an appropriate plume environment at the spacecraft solar array surfaces will be mission / spacecraft specific and depending upon many variables.

The plasma densities present at solar array surfaces are quite similar to those experienced at LEO and shown to result in damaging arcing on operating arrays [14, 15]. Understanding interactions with such a plasma in all spacecraft operating modes is paramount when designing GEO solar arrays. It is especially important if thrusters are fired during the time a spacecraft is negatively charged by GEO plasma as interactions may result in unexpected synergistic effects that can lead to ESD events and damage of solar arrays [16].

Ion energies present in charge exchange and wake region are sufficient to cause sputtering and contamination of exterior spacecraft surfaces. Although erosion rates for various elements, predominantly metals in the periodic table, have been measured in the past, important (relevant) spacecraft materials exposed to plume (Xenon) ions in the appropriate energy range between 50 eV and 300 eV had not been previously characterized [17].
An operating HCT will affect the spacecraft floating potential, or the potential at which the spacecraft single point ground sits relative to the surrounding plasma environment. The spacecraft potential is a dynamic, equilibrium value, resulting from current balance to exterior spacecraft surfaces; the net sum of currents to spacecraft surfaces must be zero. Positively charged surfaces readily attract electrons whilst negatively charged surfaces attract ions. spacecraft design features which have a profound impact on spacecraft floating potential include grounding methods of the spacecraft Electrical Power System (EPS) and solar arrays, bus voltage, grounding methods of the spacecraft Electric Propulsion system (Hall Current Thruster for example), and selection of exterior materials which include properties such as surface resistivity, bulk resistivity, Secondary Electron Emission (SEE) coefficients, and photoelectric emission coefficients. It is expected that the floating potential of a typical 70 V or 100 V spacecraft will float a few volts negative and within a few volts of the negative spacecraft bus voltage during HCT operation [12, 13]. The ultimate design and operational impacts of this value can be profound as an operating solar array will float to the same value during thruster firings and modern solar cells have been demonstrated to be susceptible to electrostatic discharges at values less than -100 V [14].

B. Natural Plasma

Spacecraft employing an All EP low-thrust transfer to GEO will, consequently, spend an increased time traversing the trapped radiation belts. As a result of the extended time passing through the inner belt and slot region, these spacecraft...
will see correspondingly higher particle fluences ionizing dose, and non-ionizing dose when compared to GEO spacecraft using chemical propulsion for GTO, typically lasting approximately 2 weeks.

The particle environments should be accounted for when specifying design environments for these missions; specific environments are readily obtained using available tools/models. Fig. J through Fig. L include the following:

- GEO trapped electron environment simulated with AE8 Max and AE9 models; does not include GTO (Fig. 6).
- GEO + GTO trapped electron environment simulated with AE8 Max and AE9 models; includes contributions from 220 d transfer (Fig. 7).
- GEO trapped proton environment simulated with AP8 Min and AP9 models; does not include GTO (Fig. 8).
- GEO + GTO trapped proton environment simulated with AP8 Min and AP9 models; includes contributions from 220 d transfer (Fig. 9).

Perturbed Mean was used for all GEO mission simulations (15 yr); Monte-Carlo used for GTO simulations (220 d). All AE9 / AP8 / SPW (IRENE) [3] simulations were performed with the following parameters:

- Number of scenarios in Perturbed Mean >100.
- Number of scenarios in Monte Carlo >100.
- Step size to ensure >100 points per orbit.
- Start time and epoch matter wherefore IGRF model incorporated in 5 yr increments (latest accurate to 2015).
- All results presented herein used Version 1.04.

C. Ionizing Dose, Non-Ionizing Dose, and Deep Charging

Design impacts can be readily ascertained for the trapped particle environments identified in the previous paragraphs.

Fig. 10 and Fig. 11 present the dose-depth curve calculated using Novice Space Radiation transport software with Solid Sphere kernel. Table I includes specific total dose values as read from the curves at a depth reasonable for dose inside of payload or bus electronics boxes, 5.0 mm (200 mil) equivalent aluminum. The difference is obvious as the low-thrust transfer adds ~25 kRad when AX8 model is used and 5 kRad to 15 kRad when AX9 Monte-Carlo model is used. Values are also presented for a typical Design Margin (DM) of 2x.

The effects of the low-thrust transfer on non-ionizing dose is illustrated in Table II which presents percent change in common solar cell performance parameters as calculated using SCREAM [19-20] with trapped particle environments calculated using AX8, AX9 MC Mean, and AX9 MC 95%. The values shown in the table represent difference from value calculated with AX8 trapped particle environments. All calculations include solar protons modelled using ESP 90%. It should be noted that recent results [21] using TacSat-4 CEASE data indicate AP-9 may be under-predicting proton values in the slot region.

Fig. 12 shows the internal (deep) charging flux as calculated using the MC 95% when compared to common design environments (ECSS and NASA). Fig. 13 includes the incident flux transported through a finite aluminum slab of various thicknesses. Practical results are shown in Table III, the incident flux at a depth of 2 mm (78.7 mil) relevant for bus or payload box circuit boards. The results indicate that the worst-case environment as determined via AX9 MC 95% yields incident flux results similar to those calculated with the FLUMIC environment whilst NASA design environment is more severe. These results are not insignificant as it represents (by simple $V = IR$) the difference between a potential of >700 V and <500 V for 1 cm² surface grounded through $10^{15}$ Ω.
Fig. 6. GEO Trapped electrons.

Fig. 7. GEO + GTO trapped electrons.

Fig. 8. GEO trapped protons.

Fig. 9. GEO + GTO trapped protons.
TABLE I. Calculated Ionizing Dose for All EP Missions using Various Trapped Particle Models and Design Margins. All Values in kRad (Si).

<table>
<thead>
<tr>
<th>Mission</th>
<th>AX8</th>
<th>AX8 x2 DM</th>
<th>AX9 MC Mean</th>
<th>AX9 MC Mean x2 DM</th>
<th>AX9 MC 95%</th>
<th>AX9 MC 95% x2 DM</th>
</tr>
</thead>
<tbody>
<tr>
<td>GEO Only</td>
<td>50.6</td>
<td>101.2</td>
<td>74.5</td>
<td>149.0</td>
<td>111.5</td>
<td>223.0</td>
</tr>
<tr>
<td>GEO + GTO</td>
<td>75.1</td>
<td>150.2</td>
<td>80.3</td>
<td>160.6</td>
<td>124.5</td>
<td>249.0</td>
</tr>
</tbody>
</table>


<table>
<thead>
<tr>
<th>Cell Type</th>
<th>Parameter</th>
<th>AX9 MC Mean</th>
<th>AX9 MC 95%</th>
</tr>
</thead>
<tbody>
<tr>
<td>SPL ZTJ</td>
<td>ISC</td>
<td>-19%</td>
<td>+27%</td>
</tr>
<tr>
<td></td>
<td>V OC</td>
<td>-21%</td>
<td>+58%</td>
</tr>
<tr>
<td></td>
<td>P MP</td>
<td>-21%</td>
<td>+36%</td>
</tr>
<tr>
<td>SPL XTJ</td>
<td>ISC</td>
<td>-20%</td>
<td>+32%</td>
</tr>
<tr>
<td></td>
<td>V OC</td>
<td>-21%</td>
<td>+35%</td>
</tr>
<tr>
<td></td>
<td>P MP</td>
<td>-24%</td>
<td>+25%</td>
</tr>
<tr>
<td>Azur 3G28</td>
<td>ISC</td>
<td>-35%</td>
<td>+12%</td>
</tr>
<tr>
<td></td>
<td>V OC</td>
<td>0</td>
<td>+78%</td>
</tr>
<tr>
<td></td>
<td>P MP</td>
<td>-14%</td>
<td>+45%</td>
</tr>
</tbody>
</table>

TABLE III. Incident Electron Flux at 2 mm (78.7 Mil) as Calculated for Various Design Environments.

<table>
<thead>
<tr>
<th>Environment</th>
<th>Incident Flux at 2 mm (pA/cm²)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flumic</td>
<td>0.49</td>
</tr>
<tr>
<td>NASA HDBK 4002A</td>
<td>0.71</td>
</tr>
<tr>
<td>AE9 (95%)</td>
<td>0.47</td>
</tr>
</tbody>
</table>

Fig. 10. GEO Total Dose (15 yr); circle identifies a depth of 200 mil relevant for devices located interior to electronics boxes.

Fig. 11. GEO + GTO Total Dose (15 yr); circle identifies a depth of 200 mil relevant for devices located interior to electronics boxes.
Fig. 12. Integral (deep) charging environment for various design models and several AE9 (95%) results as returned for 1 yr GEO.

Fig. 13. Calculated incident electron flux; circle identifies a depth of 2 mm (78.7 mil) which is appropriate for a circuit interior to bus or payload box.

Fig. 14. Artist rendering of Lockheed Martin A2100 spacecraft identifying areas susceptible to plume effects (www.lockheedmartin.com).
III. DESIGN CONSIDERATIONS

A typical A2100 spacecraft with Hall Current Thrusters is shown Fig. 14. HCT plumes will impact the spacecraft design and operation in multiple ways however the scope of present manuscript is limited to those associated with spacecraft charging and space radiation effects.

A. Material Erosion

Surface erosion of exterior surfaces is, in itself a risk to satellite operation due to variable or degraded thermo-optical for example. However the present manuscript concerns itself with the risks related to spacecraft surface charging. Passive surface charging mitigation, via the use of thin film coatings, paints, and films has long been demonstrated a low-cost, low-complexity, and effective means to protect GEO spacecraft from operational impacts over the vehicle lifetime due to surface charging. Plume induced erosion of these coatings / surfaces renders the electrostatic discharge mitigation properties of exterior surfaces uncertain, at best, or at worst the entire coating may be removed. The resultant surface may possess no surface or bulk conductivity potentially leaving large portions of the spacecraft exterior a threat to accumulate substantial charge and differential voltage.

A variety of publically available and proprietary tools enable prediction of material erosion rates in the presence of a plume plasma. The ultimate utility of such tools for spacecraft design is dependent on the quality of inputs such as material-specific erosion / sputter yields and plume plasma characteristics. Although the quantity and quality of these data are improving interpretation of results is not simple; for example consider the gradual erosion of Multilayer UV Reflective coatings which include a multilayer stack of multiple materials. Similarly, consider that the analytical simulation of thruster plumes used in most codes is based largely upon properties characterized in ground laboratories with a growing, but still limited, amount of flight verification measurements with which to correlate models.

Resulting erosion rate predictions for some typical GEO telecommunications spacecraft are provided as Fig. 15 through Fig. 17. It is interesting to note that, despite obvious differences in thruster types (BPT-4000, SPT-100, and HEMP-3050), unknown operating conditions, spacecraft manufacturer (Lockheed Martin, Space Systems / Loral, and OHB), and exterior surface materials the results seem to trend towards limiting values for two distinct regions on the satellite.

Despite differences in architectures and operations a predicted average erosion depth of 4,000 Å (0.4 μm) and maximum erosion depth between 20,000 Å and 40,000 Å (2 μm to 4 μm) for solar array coverglass and substrate materials including SiO₂, MgF₂, ITO, AZO, ATU, graphite, aluminum, and varieties of Kapton appears consistent.

Owing to its location within the plume backflow region (>90 deg relative to thrust axis) spacecraft body and antenna materials including Optical Solar Reflectors (OSR), Multilayer Insulation (MLI), Germanium, Gold, SiO₂, graphite aluminium, varieties of thermal paints, and varieties of Kapton are predicted to see typical erosion depths of <1,000 Å (0.1 μm) with limited regions of extreme values (>20,000 Å or 2 μm). Particularly noteworthy exceptions to these values include large (>5 m) mesh reflectors which are often gold and may be deployed into far more energetic regions of the plume.

Mitigation measures to preserve the often thin films, coatings, and paints relied upon for passive surface charge mitigation are obvious:

1. Create or obtain a means of rapidly, or repeatedly, estimating the erosion depth or rate of spacecraft exterior surface materials. Rapid or ease-of-calculation is convenient for repeated trade studies. Models / analyses should rely upon credible, physical yields for relevant materials.

2. Optimize the separation / location of the thruster(s) and sensitive spacecraft surfaces such that plume impingement is minimized to the surfaces. Certainly many restrictions / limiting factors exist in system and spacecraft design however including spacecraft charging and erosion depths as part of a robust concurrent engineering or trade study practice can yield positive results with minimal / manageable impacts in attitude control, power, and propulsion system designs or budgets. For example, trade studies should be performed that may consider the following at a minimum:

   a. Thruster plume origin / location (e.g. the spacecraft coordinates of thrust vectors).

   b. Thrust vector angle during GTO and all stationkeeping operations.

   c. Thruster boom; whether thrusters are fixed or deployed.
d. Size (length) of solar array boom(s).

e. Size (length and width) of solar panels.

f. Portions of panels populated by coated solar cell coverglass.

g. Size (diameter) and location(s) of deployed reflectors.

Simple “stay out zones” defined on engineering drawings using view angles to represent the area at risk of plume erosion maybe moderately useful for initial design but are insufficient in final design and should be eschewed in favor of detailed plume erosion modelling per (1).

3. Replace or modify thin-film charge control coatings with an external “sacrificial layer” which protects the conducting coating whilst preserving radiation stability and optical properties both beginning and end of life. Useful limits on coating thicknesses are provided in the previous section; although obviously design and material dependent, values of >4 μm for solar array surfaces and >0.4 μm for spacecraft body surfaces are an appropriate starting point. Fig. 18 depicts recent testing performed by Lockheed Martin during which coatings were subjected to accelerated End-Of-Life (EOL) erosion depths in the presence of a Cylindrical Hall Current Thruster (CHT) plume at the Princeton Plasma Physics Laboratory (PPPL).

Recent testing performed by Lockheed Martin demonstrated that thicker “sacrificial” coverglass coatings over ITO and thicker Multilayer UV Reflector (MUVR) coating over IO maintained their charge dissipative properties after 200 Mrad dose with no measurable degradation in optical properties such as emissivity, reflectance, or transmission. Verification of charge dissipative properties was achieved via electron flood beam exposures in 5 keV to 20 keV electron beam of current densities up to 1 nA/cm².

4. Alternatively, develop coatings, films, or paints which are inherently more robust to plume induced erosion. As SiO₂ processes a sputter range ~0.75× that of MgF₂ (in Xenon at 300 V and 60 deg incidence) variations exist in charge control materials as well – for example AZO, ATO, and ITO can be expected to have differing sputter yields. And, although not reported further herein advanced bulk materials under development, including those which utilize Carbon Nanotubes (CNT) may offer further improvements in plume robustness with additional cost benefits when compared to more traditional coatings or films.

Fig. 15. SGEO sputtering results [11].

Fig. 16. Representative solar array coating erosion analysis [12].
Fig. 17. SiO$_2$ (top) and MgF$_2$ (bottom). Units are Å.

Fig. 18. Coating / film and solar array coupon testing in HCT plume. Coating and film collimators placed to accelerate testing to predicted EOL erosion depth whilst solar array coupons placed to simulate nominal predicted plume property exposure (density, temperature, ion energy, et cetera).

Fig. 19. Primary arc inception voltage in various plasma environments.

Fig. 20. “Triple Point” location at which arcs originate on operating solar arrays.
**Abstract No# 106**

**B. Arcing on Solar Arrays**

It is well known that operating solar arrays are susceptible to electrostatic discharges / arcing when subjected to high density plasma conditions such as those created by HCTs [22-26].

Fig. 19 shows the Primary Arc inception voltage for a number of common solar cell types in a number of plasma environments. Plasma conditions include:

- $10^6$ cm$^{-3}$ with temperature of 0.5 eV as generated with hollow cathode plasma source.
- $10^{11}$ m$^{-3}$ to $10^{12}$ m$^{-3}$, $T_e$ between 0.1 eV and 3 eV, $T_i$ between 30 eV and 50 eV with small percentage of high energy ions (>100 eV) as generated by laboratory Hall Current Thruster [13].
- LEO plasma in presence of International Space Station [15, 27].

Data plotted represent multiple cell types (MJ, ZTJ, Standard Si, ATJM, UTJ, XTJ, and ITJ) from multiple cell vendors obtained during ground testing at Lockheed Martin (Square data points), KIT / JAXA (Filled Circle data points), and NASA (Unfilled Circle data points) as well as on-orbit studies by Lockheed Martin (“X” data points) and NASA (via PASP Plus, “Plus” data points). Data in the figure also represent a variety of solar cell shapes including rectangular (black lines), cropped corner (blue lines), and ~60 cm$^2$ large area (red fill).

The red dotted line at approximately -320 V represents the approximate worst-case floating potential of a conductively coated GEO telecommunications spacecraft although, of course, this value is design dependent (as indicated by the arrows).

Also included in Fig. 19 is the approximate range of spacecraft floating potential during HCT firing although this value is, of course, also highly design dependent and the fuzzy shaded area is merely an approximate reference point [18].

Recent testing (Fig. 18) performed on a small number of UTJ ~60 cm$^2$ large area cells in an HCT plume environment yielded a PA inception voltage of -200 V.

Charge Plate Analyzers (CPA) aboard telecommunications spacecraft at GEO have demonstrated that which had been suspected – negatively charged spacecraft are neutralized by thruster plumes but on the order of a few minutes (5 min to 10 min) rather than near-instantaneously [16]. It suggests that the negatively biased spacecraft (and solar array) will be at risk of arcing / discharge for several minutes whilst exposed to the thruster plume.

Mitigation measures to prevent arcing on operating solar arrays are well described by existing design standards [22-26]; of note, however, is the necessity to ensure that the solar array is immune to arcing, and effects of arcing, in GEO and LEO (recognizing that the LEO environment is quite similar to conditions created by the plume). Additional design measures that improve charging and ESD mitigation in plume environments include:

- Use of mildly conductive solar array substrate materials both on the cell-side and non-cell side as a means of eliminating differential voltages.
- Eliminate or greatly minimize amount of exposed metal such as bus bars, interconnects, and exposed cell edges. An obvious solution is the “grouting” of solar cell edges; although some testing has been performed [28] it is not clear, however, how grout performance is impacted by severe plume exposure and EOL conditions. An attractive approach moving forward may be found in full encapsulation by materials such as ETFE or PMG [14, 30].

Manufacturers have allocated 2-3% degradation in optical performance as a result of plume induced coverglass coating erosion [12]. As shown in Fig. 21 plume induced sputtering and redeposition at cell edges on un-grouted coupons may also induce resistive losses and degradation in cell performance.

The remaining uncertainty associated with the effects of thruster plumes on satellite operation can be further eliminated via integration of small, low-cost, low-power, sensors on host spacecraft. The purpose of such sensors is to provide targeted performance / application measurements of plume effects on the host spacecraft rather than multi-channel, high fidelity, scientific physical measurements; Table IV includes a proposed sensor suite. Optimal utility of this design would be achieved with at least two per spacecraft. Such a sensor suite could be realized quickly at an acceptable cost [29] by leveraging existing instrument designs and providing critical information for designers, operators, and scientists alike.
TABLE IV. PRELIMINARY DESIGN OF NOTIONAL SENSOR SUITE TO FLY ABOARD ALL EP SPACECRAFT.

<table>
<thead>
<tr>
<th>Instrument</th>
<th>Purpose</th>
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</thead>
<tbody>
<tr>
<td>Retarding Potential Analyzer (RPA)</td>
<td>Measure ion energy in plume</td>
</tr>
<tr>
<td>Charge Plate Assembly (CPA)</td>
<td>Measure differential potential</td>
</tr>
<tr>
<td>Planar Langmuir Probe (LP)</td>
<td>Measure plume plasma density, temperature, and potential</td>
</tr>
<tr>
<td>Erosion sensor 1</td>
<td>Measure changes in optical and electrical properties of optical coating</td>
</tr>
<tr>
<td>Erosion sensor 2</td>
<td>Measure changes in electrical properties of metallic trace(s)</td>
</tr>
</tbody>
</table>

Fig. 21. Microscope image of a cell edge after solar cell coupon exposure to >100 hr of HCT plume; thin metallic film is clearly visible on the cell edge resulting in performance degradation.

IV. CONCLUSIONS

Growing use of Electric Propulsion for Geostationary Earth Orbit stationkeeping and GTO offers satellite operators the opportunity to reduce mission costs and increase revenue by enabling a higher dry mass to orbit compared to traditional chemical or electric propulsion systems. The preceding sections review the spacecraft charging and space plasma-related design challenges associated with the long duration low-thrust transfer and exposure to the energetic plume plasma. Relevant design environments are introduced, an overview of specific operational effects is provided, and design guidelines have been highlighted. The design goal remains to eliminate the occurrence of all spacecraft arcing via methods that are (1) technically realistic; (2) as simple as practical but no simple; (3) cost-effective as practical; and (4) obtain flight measurements that be utilized to observe effects real-time and optimize (1), (2), and (3) for future missions.

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Introduction

- Use of low-thrust propulsion offers satellite operators the opportunity to reduce mission costs and increase revenue by enabling a higher dry mass to orbit compared to traditional chemical or electric propulsion systems.

- The penalty for such benefits comes, initially, in the form of an increased time-to-orbit whereby the low-thrust Transfer Orbit duration will range from a few months, at best, to possibly >1 yr.

- During the low-thrust transfer, the spacecraft will experience prolonged exposure to portions of the trapped radiation belts to which GEO spacecraft otherwise would not be subjected – most notably the Inner Belt and slot region.

- Further, the spacecraft will also experience up to $10^4$ hr of high density, high energy plume plasma.

- Complete list of design challenges is large but present focus is on charging / ESD related topics.

Credit: NASA

Low-thrust / All EP GTO results in high trapped particle fluence when compared to chemical propulsion based GTO.

GPS (20,200 km / 55 deg)

Galileo (23,200 km / 56 deg)

GEO (35,600 km / 0 deg)
Overview of Environments

- Manufacturers continue to develop optimized low-thrust transfers for a variety of BOL dry mass values and launchers; best case transfers are achieved in <130 d although ultimate durations may be 2x to 3x longer in some circumstances.
- Figure summarizes minimum-time low-thrust trajectory to GEO with an initial orbit of 185 km x 35786 km at 28.5 deg for a 1200 kg dry mass and BOL power of 5 kW.
- RHS identifies approximate altitude regions associated with radiation and plasma related threats to spacecraft operations enhanced or created by low-thrust GTO.

Credit: Messenger

[Diagram showing trajectory and altitude regions]
Long Duration Transfer Impacts via AX9

User notes and lessons learned

• Number of scenarios in Perturbed Mean >100
• Number of scenarios in Monte Carlo >100
• Step size to ensure >100 points per orbit
• Start time and epoch matter
  – IGRF model incorporated in 5 yr increments (latest accurate to 2015)
• All results presented herein used Version 1.04

<table>
<thead>
<tr>
<th>Analysis Type</th>
<th>Recommended Run</th>
<th>Duration</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total Dose</td>
<td>Perturbed Mean</td>
<td>Several orbits (days)</td>
<td>Plasma + AE9</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Plasma + AP9 + Flare</td>
</tr>
<tr>
<td>Displacement Damage</td>
<td>Perturbed Mean</td>
<td>Several orbits (days)</td>
<td>AP9 + Flare</td>
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<tr>
<td>Proton SEE</td>
<td>Monte Carlo</td>
<td>Full mission</td>
<td>AP9 + Flare</td>
</tr>
<tr>
<td>Internal Charging</td>
<td>Monte Carlo</td>
<td>Full mission</td>
<td>AE9</td>
</tr>
</tbody>
</table>

1Runtime based on 64 Bit 3.33 GHz Intel Xeon CPU (16 GB RAM)
Low-Thrust Transfer Impacts to Dose

- Dose determined at depth reasonable for dose inside of payload or bus electronics boxes (5.0 mm or ~200 mil) eq aluminum
  - The difference is obvious as the low-thrust transfer adds ~25 kRad when AX8 model is used and 5 kRad to 15 kRad when AX9 Monte-Carlo model is used
  - Adds 1.1x to 1.5x

- Mean AX9 values (no DM) offer reduction in eq 1 MeV fluence for common solar cells (ZTJ, XTJ, and 3G28)
  - As calculated via SCREAM
  - 95% values, alternatively, yield >25% increase

- Reality likely to be worse given TACSAT-4 observations on slot region protons

For All EP missions AX8 (x2 DM) > AX9 (95%) and AX9 Mean (x2 DM) > AX9 (95%)

<table>
<thead>
<tr>
<th>Mission</th>
<th>AX8</th>
<th>AX8 (x2 DM)</th>
<th>AX9 MC Mean</th>
<th>AX9 MC Mean (x2 DM)</th>
<th>AX9 MC 95%</th>
<th>AX9 MC 95% x2 DM</th>
</tr>
</thead>
<tbody>
<tr>
<td>GEO Only</td>
<td>50.6</td>
<td>101.2</td>
<td>74.5</td>
<td>149.0</td>
<td>111.5</td>
<td>223.0</td>
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<tr>
<td>GEO + GTO</td>
<td>75.1</td>
<td>150.2</td>
<td>80.3</td>
<td>160.6</td>
<td>124.5</td>
<td>249.0</td>
</tr>
</tbody>
</table>
AE9 Results for Internal Charging

- AE9 (95%) results returned for 1 yr at GEO
- Compared to common design guidelines / design standards
- AE9 extends energy range beyond NASA
- Results are significant
  - Represents (by simple $V = IR$) difference between a potential of $>700$ V and $<500$ V for 1 cm² surface grounded through $10^{15}$ Ω

**Environment**

<table>
<thead>
<tr>
<th>Incident Flux at 2 mm (pA/cm²)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flumic</td>
</tr>
<tr>
<td>NASA HDBK 4002A</td>
</tr>
<tr>
<td>AE9 (95%)</td>
</tr>
</tbody>
</table>

Extrapolate 4002A w/ FLUMIC for transport analysis
Solid sphere kernal (not finite / back slab)
Plume Properties at Spacecraft Surfaces

• In typical spacecraft architectures nearly all sensitive surfaces are located within the plume wake region (>90 deg)
• The distance between thrusters and array surfaces varies by design it is generally 1 m to 2 m
• Predicted plume plasma densities at solar array surfaces in the range of $10^{12}$ m$^{-3}$ to $>10^{14}$ m$^{-3}$ Predicted ion energies are typically <80 eV but may be as high as >120 eV
• See also Paper 124
Predicted Surface Erosion Rates / Depths

- Passive mitigation (thin film coatings, paints, and films) have been demonstrated a low-cost, low-complexity, and effective means of surface charging mitigation.
- Plume induced erosion of these coatings / surfaces renders the ESD mitigation properties of exterior surfaces uncertain.
  - Or at worst the entire coating may be removed.
- The resultant surface may possess no surface or bulk conductivity.
- A variety of publically available and proprietary tools enable prediction of material erosion rates in the presence of a plume.
  - Ultimate utility of such tools for spacecraft design is dependent on the quality of inputs such as material-specific erosion / sputter yields and plume plasma characteristics.
- Despite differences in architectures and operations a predicted average erosion depth of 4,000 Å and maximum erosion depth between 20,000 Å and 40,000 Å (for solar array coverglass and substrate materials including SiO2, MgF2, ITO, AZO, ATO, graphite, aluminum, and varieties of Kapton) appears consistent.
- Spacecraft body and antenna materials including OSR, MLI, Germanium, Gold, SiO2, graphite, aluminum, varieties of thermal paints, and varieties of Kapton are predicted to see typical erosion depths of <1,000 Å with limited regions of extreme values (>20,000 Å).
Basic ESD & Erosion Testing

- Recently a number of ground tests were performed on spacecraft materials & solar cell coupons
  - Photo at right shows test by LM
    - Long duration "nominal" array operation (under bias; not illuminated)
    - Primary arc inception threshold
    - Material erosion within collimators
- Position of samples was determined by analysis
  - Identified portion of CHT plume that best represented predicted plume properties at material surfaces in flight thruster
  - SA coupons positioned in nominal plume
  - Erosion sample collimators positioned to accelerate predicted total EOL erosion depth
    - Scaled BPT-4000 plasma to CHT plasma
Primary Arc Threshold

- Operating solar arrays are susceptible to ESD when subjected to high density plasma conditions (including plume)
- An additional data point is added to the PA inception comparison plot at right
  - Various plasma environments are included
    - $10^{12} \text{ m}^{-3}$ with temperature of 0.5 eV as generated with HC plasma source
    - $10^{11} \text{ m}^{-3}$ to $10^{12} \text{ m}^{-3}$, $T_e$ between 0.1 eV and 3 eV, $T_i$ between 30 eV and 50 eV as generated by HCT
    - LEO plasma at ISS
- Primary arcs likely expected in presence of modest charging environment

Approximate range of floating potential during HCT operation

Full wafer in $\sim 10^8 \text{ cm}^2$ simulated HCT plume
100 hr Plume Exposure

DIV is significantly degraded (4 cell string)

- XTJ 2 In Vacuum Pretest (PPPL)
- XTJ 2 In Vacuum Post test (PPPL)

Current (A)

Voltage (V)

IR Thermography indicates shunt path

Top of cell & CG

Metallic film coating cell edge

Substrate

Interconnect

Metallic material present

Thin film (~1 µm thick)

SEM image (250x @ 20 keV)

VIS image (10^5 x)

Coverglass

Adhesive

MESA etch (Top / Mid cell)

Bottom Ge cell

Observed metallic deposit
Conclusions & Recommendations

• The design goal remains to eliminate the occurrence of all spacecraft arcing via methods that are:
  (1) Technically credible; (2) as simple as practical but no simpler; (3) as cost-effective as practical

• Recommendations include
  – For ESD due to surface modification by plume induced erosion
    • Practice systems-level trade studies
    • Optimize thruster and sensitive surface locations, angles, duty cycles, et cetera
    • Optimize surface dimensions, solar cell layouts, and locations
    • Replace thin conductive coatings with robust coatings to ensure ESD mitigation at EOL
      – Consider thicker coatings, sacrificial layer(s), or new materials
      – Consider “active” charge control
  – For solar array ESD in plume
    • Follow existing design standards (NASA, AIAA, ECSS, JERG, and ISO)
    • Use mildly conductive solar array substrate material
    • Eliminate / reduce amount of exposed metal on array
      – Consider grouting or encapsulation (via ETFE or PMG for example)
  – Certainly for early missions consider use of on-spacecraft sensors

<table>
<thead>
<tr>
<th>Proposed Instruments</th>
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</thead>
<tbody>
<tr>
<td>Retarding Potential Analyzer (RPA)</td>
</tr>
<tr>
<td>Surface Charge Monitor (SCM)</td>
</tr>
<tr>
<td>Planar Langmuir Probe</td>
</tr>
<tr>
<td>Erosion Sensor 1 – optical coating</td>
</tr>
<tr>
<td>Erosion Sensor 2 – ideal metal</td>
</tr>
</tbody>
</table>