Plasma Vehicle Charging Analysis for Orion Flight Test 1

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Abstract— **In preparation for the upcoming experimental test flight for the Orion crew module, considerable interest was raised over the possibility of exposure to elevated levels of plasma activity and vehicle charging both externally on surfaces and internally on dielectrics during the flight test orbital operations. Initial analysis using NASCAP-2K indicated very high levels of exposure, and this generated additional interest in refining/defining the plasma and spacecraft models used in the analysis. This refinement was pursued, resulting in the use of specific AE8 and AP8 models, rather than SCATHA models, as well as consideration of flight trajectory, time duration, and other parameters possibly affecting the levels of exposure and the magnitude of charge deposition. Analysis using these refined models strongly indicated that, for flight test operations, no special surface coatings were necessary for the Thermal Protection System (TPS), but would definitely be required for future GEO, trans-lunar, and extra-lunar missions.**

I. Introduction

Lockheed Martin (LM), the prime contractor for the Orion vehicle, derived their baseline vehicle electrostatic charging requirements from NASA requirements taken largely from International Space Station (ISS) legacy. These requirements thus primarily reflected the needs of operations in a LEO environment. LM sub-contracted with ElectroMagnetic Applications, Inc. (EMA), to perform certain electromagnetic analyses for the vehicle. Among these was an analysis of vehicle charging expected to occur during the operational test flight of the Orion vehicle. EMA published a charging analysis using NASCAP-2K in Oct of 2011 [1]. This work used known vehicle design and material characteristics, used a classic single Maxwellian distribution per a worst-case GEO environment definition, and employed values for n_e, T_e, n_i , and T_i ,synthesized from multiple sources. Very high magnitudes of surface potential on the vehicle were predicted, as high as 22kV.

NASA Electromagnetic Environmental Effects (E3) at Johnson Space Center reviewed the analysis, and determined the levels were excessively high in comparison to design guidelines contained in standard reference documents [2 – 5].

A Technical Interchange Meeting (TIM) was organized in February of 2013 in Denver, CO, at the EMA facility. Personnel from LM, EMA, JSC E3, and MSFC Natural Environments were invited to participate. In preparation, JSC E3 developed a set of "new and improved" electrostatic charging requirements, and asked MSFC Natural Environments to review them. The TIM was very successful, with a very good exchange of information and philosophy, resulting in LM and EMA agreeing to re-perform the charging analysis using more specifically applicable data, including a better set of electron and ion density and temperature data embodied in AE8/AP8 models, and specific trajectory information for the test flight.

The "new and improved" electrostatic charging requirements were socialized with LM and EMA during the TIM, and were received gracefully and without controversy. These requirements were subsequently incorporated into the Electromagnetic Environmental Effects (E3) document [6] for the Multi-Purpose Crew Vehicle (MPCV) Program.

II. A Repeated Analysis

A. Summary Results

EMA, in accordance with agreements from the TIM in February, re-performed the vehicle charging analysis [7], using AE8 and AP8 yearly average and 100x particle scaling to extrapolate a MEO worst case. In this repeated analysis, they found the average surface potential relative to space plasma fell between -10V and +60V, with the maximum potential difference between all surfaces < 60V. Using a worst case GEO environment (with EFT-1 mission timing/shading), they found the average surface potential relative to space plasma fell between -24kV and +100V, with the maximum potential difference between all surfaces < 10.5kV. They concluded in general that no detrimental ESD effects would occur at charging potentials < 100V, in more than full agreement with [2].

B. Technical Approach

The technical approach EMA used created a detailed Boundary Element Method (BEM) model of the spacecraft in NASCAP 2K using vehicle OML material properties, parameterized for sensitivity analysis. The modeling combined mission magnetic coordinates for EFT-1 with AE8 and AP8 environment model codes found in [8]. With these in place, proceed to determine particle flux for each mission location every 300 seconds, and account for the direction of the sun and eclipses to determine solar effects, if any.

Several standard worst-case plasma environments were considered, including a single Maxwellian Purvis GEO worst case, an double Maxwellian ATS6 worst case, a double Maxwellian single worst day in 1997, and an AE8 AP8 Yearly Average FT-1 Mission. Note the AE8 and AP8 environments were treated as a yearly average, then each carrier flux was increased by factors of 10 and then 100 to observe the impact on

the expected charging. In addition, the AE8 and AP8 environments' temperatures were varied by a factor of 1.5.

Additional considerations included various surface resistance limits, with no material surface resistance greater than a specified value in ohms/square; the BEM model bulk conductivity was adjusted to match an assumed limit: of conductivity = 1 / (surface resistance * thickness).

C. Assumptions

Several assumptions enabled the modeling to proceed apace (see Fig. 1). The spacecraft's orientation with respect to the sun is not static, so the time spent in umbra (shade) may reduce secondary electron flux since the sun's high energy photons are blocked. The effect of the penumbra is ignored – it's either light or dark. The spacecraft is orbiting in a plane (the light does not hit from above or below). The first trajectory is approximated to be very close to the sunlight being incident from (1,0,0). All subsequent trajectories are measured relative to first. Finally, the rotation of the spacecraft around its own axis is ignored.
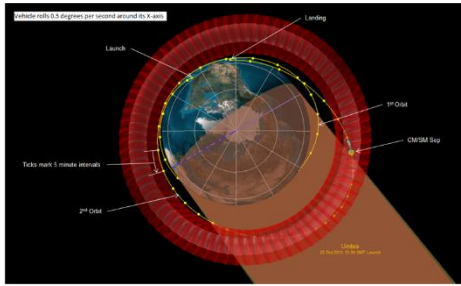

Fig. 1. Diagram of EFT-1 Orbital Paths

III. GRAPHICAL RESULTS

First, the worst-case GEO parameters were considered as a bounding case not necessarily representative of the EFT-1 mission. In each case, the average surface potential for all vehicle surface nodes relative to the plasma was determined. These results indicated the worst-case risk of discharge to the plasma. Next, the maximum potential difference between any two surfaces on the MPCV and Delta upper stage vehicles was assessed. These results indicated the risk of an arc from one surface to another, a critical potential for damage.

For each simulation, the baseline vehicle with material properties was analyzed. Following this, a design rule requiring all exterior surfaces to have a lower surface resistance than a critical value was considered. In these models, all the material bulk and surface conductivities that exceeded each threshold were adjusted down until they met the requirement. The decreased surface resistance had a large effect on the differential results, as the ultimate potential is directly related to the surface resistance requirement.

After the worst-case GEO environment was examined, the ATS6 Mission worst case double Maxwellian parameterization was considered. The results for this assessment were entirely similar to the GEO worst case. The final GEO environment assessed was the standard single-worst case result observed on an instrumented mission. Only the baseline case was considered

for this particular example. Some combined results of the foregoing assessments are shown in Figs. 2 and 3.
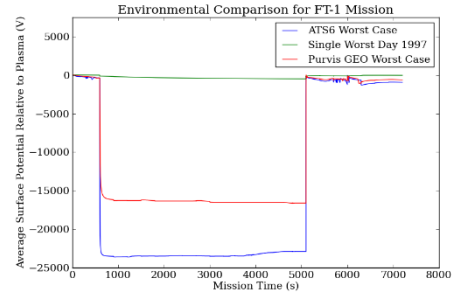

Fig. 2. Worst Case Potentials with Respect to the Plasma
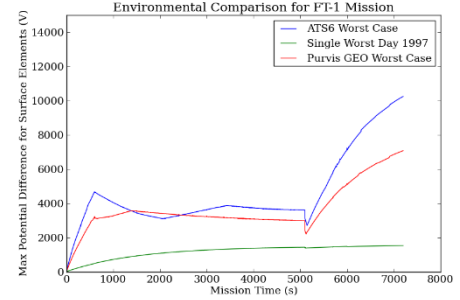

Fig. 3. Worst Case Potentials Between Surface Elements

Having established the worst-case bounding conditions, AE8 and AP8 electron and ion data were fit to a double Maxwellian distribution, and the resultant fluxes were used for subsequent analysis and comparison to the worst-case results.

The curve fitting equations used for the AE8 and AP8 environments are given by:

$$F_{Diff} = \sqrt{\frac{1}{2\pi\theta\, p_D\, m_c}} \left(\frac{E\, p_D}{\theta}\right) n \exp\left(-\frac{E}{\theta}\right)$$

(1)

$$F_{Int} = \sqrt{\frac{\theta\, p_I}{2\pi\, m_c}} \left(\frac{E}{\theta}+1\right) n \exp\left(-\frac{E}{\theta}\right)$$

(2)

where

n = carrier density (m^{-3}); m_c = carrier mass (kg)

E = carrier energy (MeV); θ = plasma temperature (MeV)

p_D = 1.602177E-13 (J/MeV); p_I = 1.602177E-22(J/MeV)

F_{Diff} = differential carrier flux (MeV^{-1} m^{-2} s^{-1})

F_{Int} = integral carrier flux above energy E (m^{-2} s^{-1})

Figs. 4 and 5 show typical double Maxwellian fits for the AE8 and AP8 environments calculated using (1) and (2).
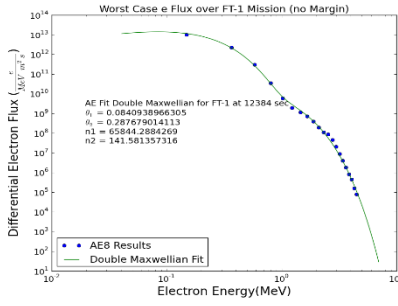
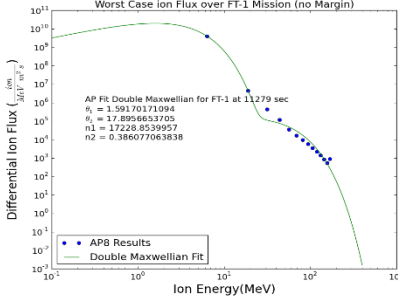Fig. 4. Typical Worst Case Electron Flux Double Maxwellian Fit

Fig. 5. Typical Worst Case Ion Flux Double Maxwellian Fit

Using the same assumptions employed for the GEO environments, EFT-1 mission parameters were adjusted to ensure a worst-case baseline had been found within the AE8 and AP8 environments.
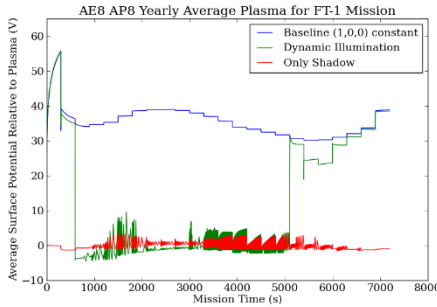

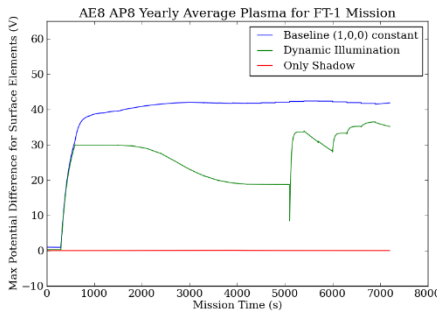Fig. 6. Average Surface Potentials Relative to the Plasma

Fig. 7. Maximum Potential Difference for Surface Elements

The results of this baseline for the calculated AE8 and AP8 yearly average fluence for the EFT-1 mission are shown in Figs. 6 and 7.

Next the number of electrons and ions are independently increased by a factor of 100. The resulting behavior, shown in Figs.8 and 9, shows a non-intuitive, weak dependence on electron and ion densities.
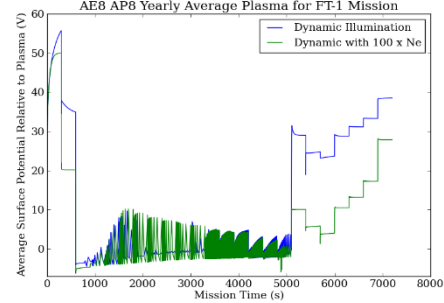
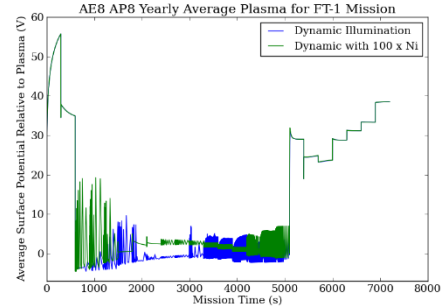

Fig. 8. Effect of Variation of Electron Density

Fig. 9. Effect of Variation of Ion Density

Then the temperature is scaled up by 50% on both electron and ion distributions, and the charging dynamic in the AE8 and AP8 environments is observed to be weakly sensitive to change in temperature as shown in Fig. 10.
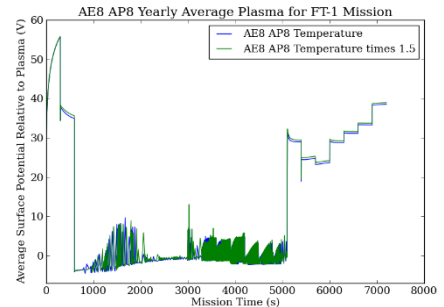

Fig. 10. Effect of Temperature Variation

IV. CONCLUSIONS

In conclusion, the most noticeable effects occur when spacecraft is in umbra. Specific orientation with respect to the sun has no visible effect. Charging is only mildly sensitive to particle densities and temperature. The 'Worst Case GEO' scenarios are predictably much worse. Shading reduces differential charging, and actually increases magnitude of

average potential with respect to the plasma when the potential is negative.

After review of the results of the repeated analysis, the NASA JSC E3 Group determined that the threat to the vehicle during the EFT-1 mission was low enough to assuage any concerns over how the surface of the Thermal Protection System (TPS) was treated for conductivity, and subsequently a waiver was granted for that requirement. It was also clear from that same review that for future missions extending beyond MEO altitudes the vehicle design must demonstrate full compliance with the surface treatment requirement.

REFERENCES

[1]   Appendix p, Delta IV, OFT-1 Spacecraft Charging for Elliptical Orbit, CEV-T-010, Rev 5; EMA-2011-R017 NASA Contract No. NNJ06TA25C, Lockheed Martin Contract No. 4500044566, EMA Contract No. C-1104, October 31, 2011, Electro Magnetic Applications, Inc., 7655 W. Mississippi Ave., Suite 300, Lakewood, CO 80226

[2]   NASA TP 2361, Design Guidelines for Assessing and Controlling Spacecraft Charging Effects

[3]   NASA-HDBK-4002Mitigating In-Space Charging Effects - A Guideline

[4]   NASA-STD-4005, Low Earth Orbit Spacecraft Charging Design Standard

[5]   NASA-HDBK-4006, Low Earth Orbit Spacecraft Charging Design Handbook

[6]   MPCV 70080, Cross Program Electromagnetic Environmental Effects (E3) Requirements Document

[7]   Appendix x, MPCV External Charging during FT-1, System Performance Analysis Report, CEV-T-010, Rev 7; EMA-2013-R042 NASA Contract No. NNJ06TA25C, Lockheed Martin Contract No. 4500044566, EMA Contract No. C-1209, October 1, 2013, Electro Magnetic Applications, Inc., 7655 W. Mississippi Ave., Suite 300, Lakewood, CO 80226

[8]   National Space Science Data Center document (NSSDC/WDC-A-R&S 91-24)