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Spacecraft Charging Issues for Launch Vehicles

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***Abstract*—Spacecraft charging is well known threat to successful long term spacecraft operations and instrument reliability in orbits that spend significant time in hot electron environments. In recent years, spacecraft charging has increasingly been recognized as a potentially significant engineering issue for launch vehicles used to deploy spacecraft using (a) low Earth orbit (LEO), high inclination flight trajectories that pass through the auroral zone, (b) geostationary transfer orbits that require exposures to the hot electron environments in the Earth's outer radiation belts, and (c) LEO escape trajectories using multiple phasing orbits through the Earth's radiation belts while raising apogee towards a final Earth escape geometry. Charging becomes an issue when significant areas of exposed insulating materials or ungrounded conductors are used in the launch vehicle design or the payload is designed for use in a benign charging region beyond the Earth's magnetosphere but must survive passage through the strong charging regimes of the Earth's radiation belts. This presentation will first outline the charging risks encountered on typical launch trajectories used to deploy spacecraft into Earth orbit and Earth escape trajectories. We then describe the process used by NASA's Launch Services Program to evaluate when surface and internal charging is a potential risk to a NASA mission. Finally, we describe the options for mitigating charging risks including modification of the launch vehicle and/or payload design and controlling the risk through operational launch constraints to avoid significant charging environments.**

Keywords—vehicle; broadband; trajectory; discharge; launch constraints

I. Introduction

The space charging environment is an important consideration in the design and operation of both the spacecraft as well as the launch vehicle which must transport the valuable payloads through this hazardous environment. It is a complex environment that poses a variety of risks to the successful operation of space vehicle electronics. The environment is dynamic and has varying plasma temperature and species types, electric and magnetic fields, as well as solar radiation, all of which interact with the space vehicle and give rise to local environments which play a role in the effects on the space vehicle system electronics. This paper focuses on the effects of charging of both launch vehicle and spacecraft systems during transport.

A. Trajectory considerations

There is a wide spectrum of electron energies that comprise the charging environment encountered in space. Typically it is electrons with energies of 1-100keV that drive the surface charging environment, and it is higher energy electrons (>100keV) that cause internal charging due to their ability to penetrate enclosures and deposit charge directly onto sensitive avionics component boards, cable insulation, and ungrounded conductors. The plasma environment varies with altitude and latitude, and charging is primarily a concern for launch vehicle trajectories that are Low-Earth Orbit (LEO), high inclination flight trajectories, geostationary transfer orbits, as well as LEO escape trajectories that pass through Earth's radiation belts. In general, the plasma temperature increases as a function of both latitude and altitude, while the density decreases. Therefore the auroral regions and geostationary Earth orbit (GEO) environments have the higher energy, low density charged particle populations that can cause charging issues. This is partially because the potential to which a spacecraft will charge is directly proportional to the electron temperature, but also because the lower density increases the Debye length of the plasma which describes the plasmas ability to screen potentials that are generated on the surfaces. In a lower density environment, the plasma is not as effective at screening/neutralizing potentials and higher potential differences across the surface of the vehicle can arise.

B. Surface Charging

Surface charging is the most common charging threat encountered during launch as the plasma current for lower energy electrons can be sufficiently high during the limited time between launch and spacecraft separation. During surface charging, the outer surfaces on a space vehicle reach electrical equilibrium with the space plasma environment which includes the low-energy high-density plasmas, high energy electrons, solar radiation and local magnetic field lines. If the entire surface of the vehicle is conductive, it will reach the same potential relative to the surrounding plasma. This is called absolute charging and is typically not a concern as it does not affect the performance of the vehicle. However, if the outer surfaces of the vehicle are not conductive, differential charging can occur, which can be detrimental to both the spacecraft and launch vehicle if fields higher than the breakdown field strength occur. The typical surface resistivity value requirement used to mitigate differential charging effects is <1E9 Ohms/square. However, this requirement is not always met due to design requirement disparities, and these non-conductive surfaces can acquire a high potential difference relative to either the structure of the vehicle or the ambient plasma that can lead to breakdown and subsequent damage to the space vehicle avionics if the energy is coupled into sensitive electronic components. A primary factor that determines the maximum potential a surface can reach is the incident solar radiation, which effectively reduces the amount of negative charge a material can acquire through the photoelectric effect. Solar radiation limits the effective potential a surface can acquire, and must be considered when assessing the risk of surface charging as it is dependent on trajectory as well as the time of launch.

C. Internal Charging

Internal charging is caused from the penetration of high energy electrons through any shielding which can then directly interact with the circuit board of the component. If enough charge is acquired within the circuit board due to these penetrating electrons, a potential difference between components and/or the dielectric of the board can be generated that is high enough to produce an electrostatic discharge directly into sensitive components. Direct discharges of this type can be extremely damaging to components since the energy is coupled directly with minimal losses. However, since the time between launch and payload deploy is limited to typically less than 8 hours for most launch operations, this type of charging is rarely an issue because internal charging currents for >100keV electrons (at GEO near 1pA/cm2), are much lower than surface charging currents for 1-100keV electrons (at GEO near 1nA/cm2), and it is unlikely for the amount of time within the charging environment to be long enough to see any component damage from internal charging effects.

One caveat is when flight operations require the launch vehicle to pass through the radiation belts during highly disturbed periods with elevated energetic electron flux or when multiple phasing orbits are required before the payload is deployed resulting in repeated transits of the radiation belts [c.f., 2,3]. In these cases an analysis for possible internal charging threats is warranted.

II. Charging Evaluation

When the standard surface resistivity requirement of less than or equal to 10^9 ohms/square for charge dissipation is superseded by higher priority thermal constraints, determining the impacts of non-conductive materials in vehicle configurations is unavoidable. Internally, RF devices may be coated with Teflon, Kapton or other similar thermal isolators; while externally, fairings can be coated with fiberglass or non-conductive paints. In such cases when industry standard EMC/EMI and spacecraft charging design requirements have not been met, the associated impacts of discharges to underlying materials, air, or nearby conductors must be examined either through an appropriate analysis or test to demonstrate the vehicle can successfully operate in charging environments.

A. Application

Levying such a charging environment requirement adds analysis tasks to the launch vehicle provider to specify the environment and the spacecraft developer to demonstrate immunity in the presence of discharges. To alleviate performing these additional evaluations when charging is unlikely, the Launch Services Program uses the charging threshold criteria established in NASA-HDBK-4002A [1] as a trigger for applying discharge environment requirements (see Fig 1.)

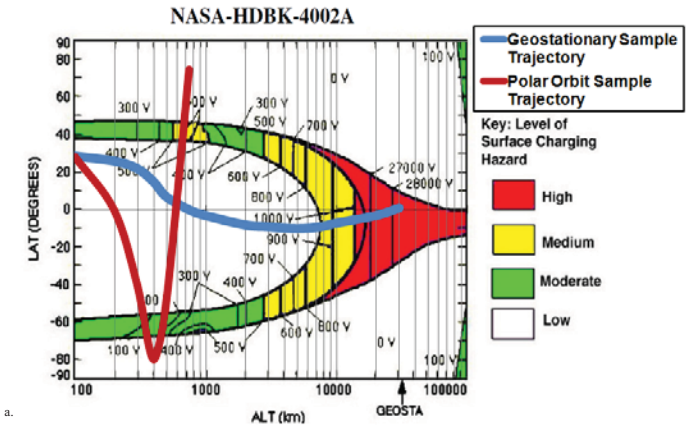

Fig. 1. Charging requirement threshold with example trajectories. Note: trajectories are for illustrative purposes only and do not represent an actual launch trajectory. [1]

If any part of the launch trajectory crosses this "horseshoe" shaped curve in either the moderate, medium or high regions the charging environment is applied and the launch vehicle/spacecraft developer must demonstrate compatibility either through analysis or test. Representative trajectories for polar and geostationary trajectories are shown.

B. Characterization :.

Characterizing the source discharge can be taking from literature discharge models [4],[5] or analysis and testing of the expected culprits. The Air Force, for example mandates a maximum allowable broadband field in their system interface specification for Evolved Expendable Launch Vehicles (EELVs).

Although more precise, determining the environment by test is not without uncertainty. For instance, for polar launches the worst case particle density varies and often the more consistent geosynchronous test condition of 1 nA/cm^2 is used. Humidity conditions prior to testing are also important to prevent charge dissipation during testing that would be

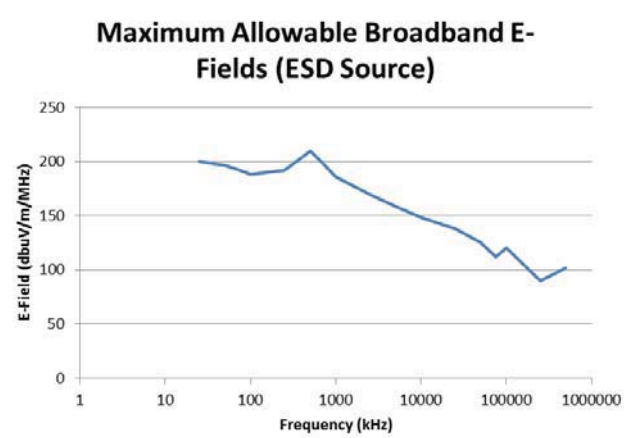

Fig. 2. Sample broadband RF environment from EELV Specification. [3]

inconsistent with the launch configuration where the materials might have encountered environmental "baking." In addition, the measurement set-up should be constrained to the maximum extent possible to ensure the data presented is representative of the discharge. Size of the sample, duplicate "grounding" of the sample and material interfaces are key to develop usable results. Test parameters also play a role in the practicality of the results. For instance, if an antenna with a high antenna factor is used for measurement and then scaled for bandwidth, distance, or multiple events, even the resulting noise floor levels can be problematic. For example, standard EMI antenna measurement bandwidths for discharge detection are in the kHz range and scaling to the required MHz range in the obligatory military standard broadband electric field representation of V/m/MHz may lead to values higher than would have been received by an actual 1 MHz bandwidth device. Bounding measurements of large bandwidths can be useful in setting environment tolerances. Making time domain measurements in addition to standard EMC type radiated emission measurements is one possible control measure.

C. *Immunity Evaluation*

Direct discharge to cables or open pins is typically prohibited. If unavoidable, test evaluation of the victim circuits is necessary. Indirect discharges create broadband radiated emissions such as represented in fig. 2 couple voltages and currents that can interfere with neighboring electronics and/or the spacecraft payload. Latch-up conditions from nearby discharges are rare, but should also be eliminated in the evaluation process. Next, and most common, are effects of discharges to communication devices such as receivers. In this case it is important to consider the actual bandwidth of the receive device and scale the broadband data typically given in 1 MHz bandwidth format. As also can be seen in Fig. 1, the RF environment falls significantly above 1 GHz where many receive devices reside. If the receiver has a large bandwidth, however, this level will be significantly higher than the 1 MHz bandwidth requirement. Finally, the spacecraft should be examined for other sensitive or bandwidth driven devices that are near the discharge for evaluation, noting that most spacecraft are configured with a minimal set of operating hardware in the launch configuration.

III. Mitigation Strategies

Charging mitigation strategies are well documented [1], however it is not uncommon for these mitigation requirements to be usurped by competing requirements. Process improvements can aid in preventing requirement oversight. Still, thermal and other design considerations can drive a design where charging is not mitigated. In such cases, it may be necessary to rely on launch constraints for necessary controls

A. *Integration improvements*

Although processes are typically in place to involve the electromagnetic compatibility (EMC) department in electrical changes made to avionics, changes to materials can occur without consideration to charging implications. Material changes can occur in the face of appropriate material conductivity constraints in the EMC control plan and even when EMC is mandated on drawing sign off, final signatures occur late in the design cycle when changes are difficult to make. Periodic and consistent communication channels between EMC/space charging departments and material/thermal groups is necessary to minimize the charging threats due to late addition of ungrounded conductors or insulation materials.

B. *Launch Constraints*

An option for mitigating spacecraft charging risk is to avoid significant charging environments during launch operations. The relatively short time period between launch and payload separation (or other final critical operations) provide the launch team with an opportunity to consider choosing launch times that minimize the exposure of the launch vehicle to spacecraft charging threats. Use of space weather launch constraints to avoid single event upsets due to solar energetic protons and heavy ions has been used for numerous launch vehicles and widely accepted as a valid method for mitigating upset threats to launch vehicle avionics. Space weather launch constraints for spacecraft charging, in contrast, is a relatively new and largely untested area of launch constraints since most vehicles have either chosen to mitigate charging through good EMC/EMI and spacecraft charging design or have fortunately opted to use flight trajectories with minimal charging risk. We anticipate an increasing use of launch constraints in the future due to novel spacecraft designs that cannot be exposed to the strong charging environments in GEO or launch vehicle design that require use of insulating and/or ungrounded conducting materials.

A. *LEO Auroral Charging*

The primary spacecraft charging threat for launches to low Earth orbit is auroral charging. The midnight boundary index in Fig. 4. provides a record of the latitude at which the equatorial boundary of the auroral oval was encountered by DMSP satelllites over a period from 1 January 1988 through 31 December 2001. Geographic north (blue) and south (red) latitudes at which the DMSP electrostatic analyzers detected significant energetic particle flux is given along with the equivalent equatorial boundary at midnight estimated from each boundary crossing. The plot shows that launch trajectories with inclinations less than about 30^0 are essentially safe from auroral charging. Higher launch inclinations in the approximately 30^0 to 70^0 range may encounter auroral particle fluxes with the encounter probability increasing with latitude.

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Launches on high inclination trajectories above about 70° latitude including the 98° sun-synchronous orbits will pass through the auroral zone on each orbit before payload separation and auroral encounters are guaranteed.

One of the simplest options for high inclination and polar

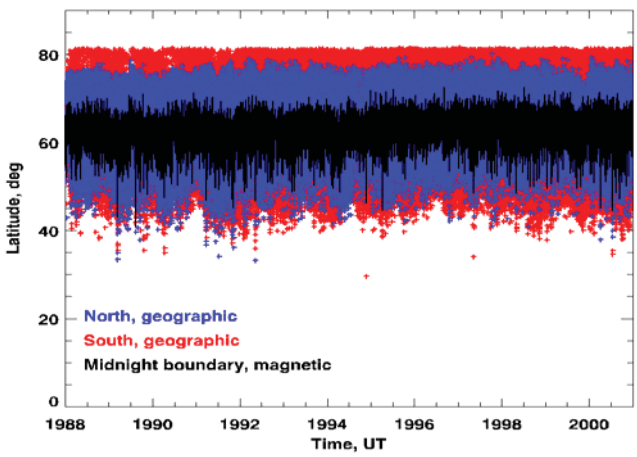

Fig. 4. DMSP Midnight Boundary Index. The index shows the geographic latitude at which DMSP spacecraft encountered the equatorward boundary of the auroral oval in the northern (blue) and southern (red) hemisphere as well as the equivalent boundary geomagnetic latitude at midnight (black) for each of the oval crossings.

launches where auroral charging is a possible threat is to select a launch time that assures the portion of the flight trajectory at latitudes likely to encounter auroral particles will remain in sunlight. Experience with auroral charging in polar environments has demonstrated that auroral charging is suppressed when vehicles are in full sunlight [6,7]. In addition, choice of launch dates that assure the flight trajectory passes through the polar cap in local summer conditions will result in launch operations that take place in conditions where the plasma density in the auroral zone is sufficiently high to minimize auroral surface charging [8,9]. However, this option is often not acceptable for launches to specific sun-synchronous orbits or other high inclination launches that require restricted launch windows at local times in darkness to meet a payload deployment requirement into a specific orbit plane. In these cases, a more active program for monitoring auroral charging conditions will be required.

Launch operations using trajectories that only encounter auroral particle fluxes when geomagnetic activity drives the auroral equatorward towards the flight trajectory are candidates for use of Kp monitoring. The geomagnetic Kp index is a measure of magnetic activity from ground based magnetometers at mid-latitudes and has been shown to be a useful measure of the strength of geomagnetic storm activity. The latitude of the auroral zone has been shown to depend on

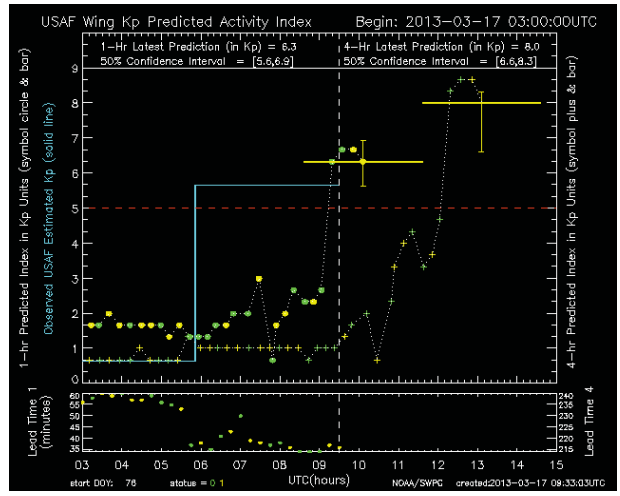

Fig. 5. Air Force Wing Kp Predicted Activity Index. The 1-hour and 4-hour predicted Kp levels (yellow) are compared to measured Kp index (blue). [Source: http://www.swpc.noaa.gov/wingkp/index.html].

geomagnetic activity and good correlations exist between the Kp index and the latitude of the aurora.

Launches to inclinations less than about 70° can avoid the aurora by determining the Kp index describing the level of geomagnetic activity for which the auroral oval will remain at a higher latitude than any point along the flight trajectory. Once this maximum Kp index has been determined is can be used as a launch constraint and a monitoring program is implemented during the pre-launch flight operations to determine if the prescribed level of geomagnetic activity is within the Kp constraint value.

A source of real time Kp measurements that are updated sufficiently often to capture changes in the geomagnetic environment is required to implement the monitoring process. One example that can be used is the Air Force Wing Kp Geomagnetic Activity Index that NOAA's Space Weather Prediction Center (SWPC) provides in near real time. The model estimates values of Kp index at 15 minute time intervals based on correlations between the upstream solar wind conditions at the Sun-Earth L2 point. Kp values are estimated with a lead time of about 30 to 60 minutes in advance because of the finite travel time required for solar wind to travel from L1 to the Earth. Fig 5. is an example of the Wing Kp output from 17 March 2013 during a period of strong geomagnetic activity. Predicted values of Kp=6.3 and Kp=8 for this disturbed period are given for periods 1-hour and 4 hours in advance of the current time (white dashed line).

Another option for monitoring the possibility of a launch trajectory passing through the auroral zone is the use of an auroral oval model constrained by data that gives boundaries of the aurora or auroral particle energy flux. In both cases the equatorward boundary of the predicted auroral zone can be used as a conservative estimate of the lowest latitude at which charging could be a threat. For example, Figure 6 shows output from two implementation of the Ovation auroral model [10]. The top two panels are output from the NOAA SWPC's implementation of the model providing an estimate of auroral viewing probability color coded as a relative intensity level (green) with a boundary showing the lowest latitude at which auroral viewing is thought to be possible (red line). While the model does not provide detailed information on particle flux environments that are responsible for charging it could be used to establish the maximum latitude of the aurora. A

particularly useful feature of the model is the data input is real time solar wind data from the Advanced Composition Explorer (ACE) satellite at L1 and auroral activity is based on correlations with auroral energy deposition models and solar wind conditions at L1. The model therefore provides a predicted level of auroral activity with a lead time of 10's of minutes to about an hour based on the speed of the solar wind.

The bottom two panels are output from the Ovation Prime implementation of the code provided by NASA Goddard Space Flight Centers Space Weather Research Center (SWRC). The SWRC implemention provides color coded plots that indicate the energy flux of precipitating electrons (shown), ions, and combined electrons and ions. This model is updated every 5 minutes and also uses solar wind data from the ACE spacecraft at L2 to provide a prediction of auroral activity.

There is a limit to using these models for predicting charging at the highest latitudes. As discussed earlier in this section the Kp monitoring technique should only be used to avoid conditions that drive auroral towards the equator and across the launch flight trajectory. The technique will fail for launches on high inclination trajectories that pass through the auroral zone each orbit. Auroral charging studies of the DMSP spacecraft in sun-syunchronous 98° orbits to negative potentials exceeding 100 volts have shown that strong auroral charging is not well correlated with the geomagnetic Kp index [8, 11].

B. GEO charging

Launches to GEO, or through GEO to interplanetary destinations, where the launch vehicle remains attached to the payload for periods longer than about 4 to 8 hours will expose the launch vehicle to the strong charging environments in the

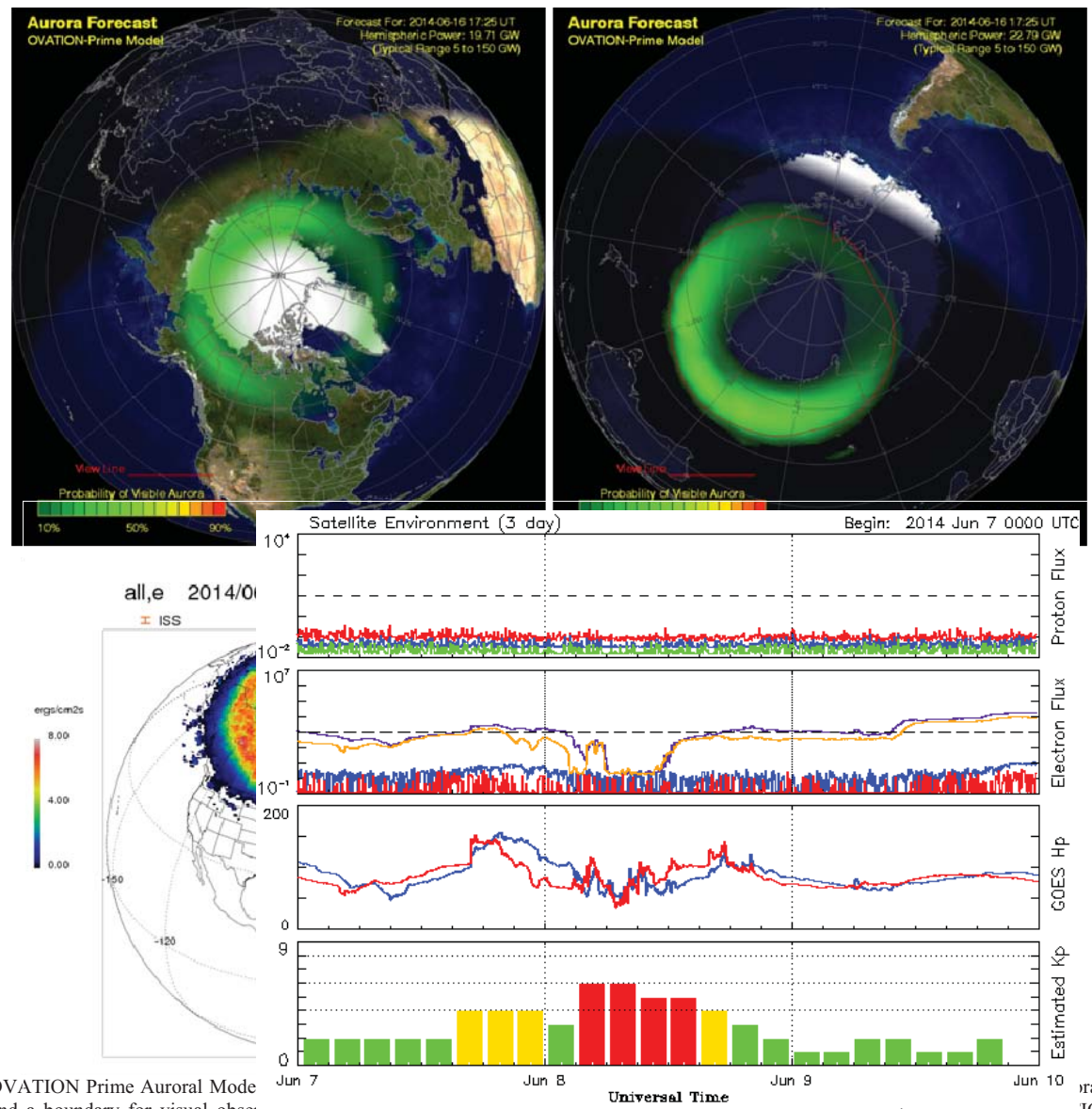

Fig. 6. OVATION Prime Auroral Mode ral viewing (green) and a boundary for visual obse ION model provides maps of energy flux due to auroral electrons. The red track is the orbit of the International Space Station currently included in the SWRC model output. Similar trajectories for launch vehicles could be easily added to the model. (Sources: http://www.swpc.noaa.gov/ovation/, http://www.swpc.noaa.gov/ovation/)

Fig. 7. NOAA SWPC Satellite Environment Data. Real time data from NOAA satellites in geostationary orbit provide useful data to guard against charging on launch operations using geostationary transfer orbits. (Source: http://www.swpc.noaa.gov/rt_plots/satenv.html).

outer radiation belt. Avoiding strong charging in GEO means avoiding geomagnetic storm conditions. Fortunately, GEO charging environments are correlated with the geomagnetic Kp index (or the related Ap index) allowing models of the charging environments to be developed based on the geomagnetic indices [12]. In addition, NOAA SWPC provides real time data that can be used to avoid storm conditions in GEO. Fig. 7. is an example of the 3-day Satellite Environment plot from the period 7-9 June 2014 including a geomagnetic storm where historical data is available for all three days.

The top panel are protons at >10 MeV, >50 MeV, and >100 MeV measured in geostationary orbit typically used for monitoring for solar energetic particles during launch. For the period shown here the proton flux is at background level indicating no enhancement in interplanetary energetic proton levels.

The second panel from the top is the integral electron flux >0.8 MeV and >2 MeV from two satellites in geostationary orbit. These channels are often used by geostationary orbit satellite operators to monitor for internal charging threats since elevated flux of electrons at MeV energies are known to produce anomalies in geostationary orbit. The energetic electrons also provide an opportunity to monitor geomagnetic activity in geostationary orbit since development of strong electric fields in the outer magnetosphere during storm periods will modify the drift trajectories of the energetic particles and drive them towards the magnetopause where they are lost from the magnetosphere [13]. Storm conditions in geostationary orbit that result in strong energetic electron depletions are likely to be accompanied by strong heating of the lower energy 10's keV electrons responsible for charging. Use of this signature to protect a launch vehicle during passage through the outer radiation belt will require development of a quantitative relationship between the level of flux depletion and electron temperature in geostationary orbit. Until this relationship has been established the absence of strong energetic flux depletions may be indicative of benign charging conditions although a study is required to establish that correlation as well.

The component of the Earth's magnetic field perpendicular to the geostationary orbit plane is given in the third panel down in Fig. 7. Magnetic fields in geostationary orbit are disturbed during geomagnetic storms providing another possible parameter to monitor for storm activity. A study is required here as well to determine if there is a quantitative relationship between the magnitude of the magnetic field perturbations and observed charging levels.

The final plot in the bottom panel of Fig. 7. Is the estimated Kp index measured by a ground based magnetometer. The output from the Air Force Wing Kp prediction code provides a better prediction in advance of auroral activity with more timely updates that the value shown in the Satellite Environment plot is preferred over the NOAA estimated Kp index for Kp monitoring.

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