

ASSESSMENT AND CONTROL OF SPACECRAFT CHARGING RISKS ON THE INTERNATIONAL SPACE STATION

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Abstract

The International Space Station (ISS) operates in the F2 region of Earth's ionosphere, orbiting at altitudes ranging from 350 to 450 km at an inclination of 51.6 degrees. The relatively dense, cool F2 ionospheric plasma suppresses surface charging processes much of the time, and the flux of relativistic electrons is low enough to preclude deep dielectric charging processes (1,2). The most important spacecraft charging processes in the ISS orbital environment are: 1) ISS electrical power system interactions with the F2 plasma, 2) magnetic induction processes resulting from flight through the geomagnetic field and, 3) charging processes that result from interaction with auroral electrons at high latitude.

Ironically, the cool, dense F2 plasma itself is necessary for the most important ISS spacecraft charging process identified to date. Electrical interactions between the F2 plasma and the 160 V photovoltaic (PV) electrical power system on ISS can produce significant electrical potential differences between ISS conducting structure and the surrounding ionospheric plasma, i.e. ISS floating potentials (FP). The negative-polarity-grounding scheme utilized in the 160 V PV power system leads naturally to negative FP values for ISS conducting structure. A negative FP results in the application of electrostatic fields across the dielectrics that separate ISS conducting structure from the ambient F2 plasma, enabling dielectric breakdown and arcing. Degradation of thermal control coatings and noise in electrical systems can result. ISS is presently equipped two plasma contactor units (PCUs), each of which is designed to control ISS FP to within ± 40 V of the ambient F2 plasma during all phases of ISS construction and operations.

Recently, the continuing review and evaluation of putative ISS charging hazards required by the ISS Program Office revealed that ISS charging could produce an electrical shock hazard to the ISS crew during extravehicular activity (EVA).

ISS charging risks are being evaluated in an ongoing measurement and analysis campaign. The results of ISS charging measurements are combined with a recently developed model of ISS charging (the Plasma Interaction Model) and an exhaustive analysis of historical ionospheric variability data (ISS Ionospheric Specification) to evaluate ISS charging risks using Probabilistic Risk Assessment (PRA) methods. The PRA combines estimates of the frequency of occurrence and severity of the charging hazards with estimates of the reliability of various hazard controls systems, as required by NASA's safety and risk management programs, to enable design and selection of a hazard control approach that minimizes overall programmatic and personnel risk. The PRA provides a quantitative methodology for incorporating the results of the ISS charging measurement and analysis campaigns into the necessary hazard reports, EVA procedures, and ISS flight rules required for operating ISS in a safe and productive manner.

Introduction

Safety and reliability engineering for manned spaceflight is conducted in a regulatory environment, not unlike the regulatory environment that governs civil aviation, characterized by rules, processes, and procedures that may be very different from those applicable to robotic space flight programs or basic space science research projects. The identification, evaluation, and management of spacecraft charging risks for the ISS Program must be accomplished in a manner that is consistent with the regulatory environment that NASA has established for ISS design, development, and flight operations. Basic and applied science studies, flight operations data, and engineering analysis must be integrated in a balanced and objective manner so as to produce safety and reliability products that enable decision-making with a high degree of confidence. Both measurement data and engineering analysis must be subjected to validation and verification processes before acceptance for use in the development of safety engineering products. The safety and reliability products themselves are subject to an exhaustive programmatic review and approval processes. Because the likelihood of occurrence of any risk or hazard can never really be driven to zero, risk management is an essential part of the regulatory process and the question "how much risk is acceptable?" is important, as is the question "how is the risk accepted or declined?" Failure to integrate the science, engineering, operations, and management components of a manned flight risk assessment in an objective and balanced manner can be catastrophically costly. The Columbia Accident Investigation Board Report, Vol. 1, August 2003 provides a detailed example of such a failure.

The regulatory environment applicable to the ISS Program is defined in a series of Space Station Program (SSP) documents. Copies of the subject documents can be obtained from the authors on request, subject to U.S. export control regulations.

- 1) SSP-50200-01, Space Station Program Risk Implementation Plan Vol. 1: Station Program Management Plan Annex E: Safety and Mission Assurance/Program Risk Plan, Revision A Feb. 9, 2000
- 2) SSP-50021, Safety Requirements Document Dec. 12, 1995
- 3) SSP-50175, ISS Risk Management Plan April 10, 2002

- 4) SSP-30309, Safety Analysis and Risk Assessment Document Rev. C July, 1992
- 5) SSP-30599, Safety Review Process, Rev. A Jan. 11, 1995

NASA wide risk management and Probabilistic Risk Assessment (PRA) methods are defined in the NASA Headquarters requirements documents listed below. PRA methods provide a rigorous logical and quantitative approach to answering the three most important questions that arise in the ISS spacecraft charging risk assessment: 1) “What is the likelihood of the hazardous condition?” 2) What is the severity of the consequences of an event caused by the hazardous condition and, 3) “what is the likelihood of failure of the hazard control system?”

- 1) NPG 71250.5A, NASA Program and Project Management Processes and Requirements
- 2) NPG 8000.4, Risk Management Procedures and Guidelines
- 3) NPG8705.x (draft), PRA Applications and Guidelines

ISS is pioneering the use of high-voltage, high power output, photovoltaic power systems on large spacecraft. Spacecraft charging as a result of interaction between photovoltaic electrical power systems with negative polarity grounds plagued early attempts to measure ionospheric temperature and density with satellite mounted Langmuir probes (3). Therefore, the possibility of ISS spacecraft charging hazards or risks resulting from interactions of the 160 V PV array Electrical Power System (EPS) with the ionospheric plasma was recognized early in ISS development (4-13) and ISS is equipped with a ship set of two plasma contactor units (PCUs) (14, 15) to mitigate any possible charging hazards.

It should be noted that negative polarity grounding of the 160 V ISS PV EPS is only one component of the ISS spacecraft charging risk. The choice of thermal control coating on the micrometeoroid and orbital debris (MM/OD) shields employed on the NASA, ESA, and JAXA pressurized elements is an important factor in determining the magnitude of the possible hazards.

As an example, consider a spherical conducting structure that is 10 meters in radius (a reasonable size compared to ISS pressurized elements). If the free space capacitance ($C_{fs} = 4\pi\epsilon\epsilon_0r = 1112 \text{ pF}$; $\epsilon =$ dielectric constant, $\epsilon_0 =$ free space permittivity) of the structure is charged to -134 V , the stored energy is only $E = 0.5CV^2 = 10 \text{ micro Joules}$.

Covering the sphere with a thin dielectric surface coating changes the character of the charging hazard dramatically. On the ram facing side of the sphere, the FP of the external surface of the dielectric film will approach 0 V as a result of positive charge collection from the ionosphere, and -134 V is applied across the dielectric. Now the sphere is best described as a parallel plate capacitor (the conducting structure is one plate and the conducting ionosphere is the other) able to store energy

$$E = 0.5 CV^2 = 0.5 \epsilon\epsilon_0(A_{ram}/d)V^2 = (0.5)(8.85 \times 10^{-12})\epsilon(2\pi r^2/d) V^2,$$

where A_{ram} is the area of the hemisphere able to collect positive charge from the ionosphere. If d is 1 micron (1.3 microns is the thickness of the anodic coating on the US Lab and Node 1 meteoroid and debris shields) and $\epsilon = 5$ for aluminum oxide, the stored energy becomes $E = 250$ Joules. Now, dielectric breakdown of the thin surface coating can discharge the parallel plate capacitor, releasing enough energy to damage the dielectric coating itself and producing enough voltage and current to present a possibly lethal hazard to any EVA crew who may become inserted into the discharge circuit as a result of arcing on exposed dielectric-coated metallic components on the EMU suit. The high-density, low-resistance dielectric-breakdown arc plasma provides the conductive path connecting the negatively charged conducting structure to the positively charged dielectric film surface (5-8,11).

Note that the stored energy is inversely proportional to the dielectric film thickness. Simply increasing the film thickness from 1.0 micron to 100 microns reduces the stored energy from 249.5 Joules to 2.49 Joules while greatly reducing the risk of damage resulting from dielectric breakdown arcing and the current available for electric shock. The thick (>120 microns) dielectric coatings on Sky Lab minimized any charging hazards that might have been generated by the 90 V PV array on that spacecraft. Similarly, the Russian elements of ISS and the ISS Truss Structure contribute little to the charging hazard because surface dielectric coatings are between 10 and 100m microns in thickness.

Stored energy is also directly proportional to V^2 , and reducing the FP at the negative end of the PV array to -40V reduces the stored energy to 0.9 micro Joules for the uncoated conducting sphere, and 22 Joules for the dielectric coated sphere.

The ISS PV array design was subjected to extensive plasma chamber testing to assure that PV array damage due to arcing on the array was not an issue (9,10). Additional plasma chamber testing and analysis of structural and thermal control materials (5-8, 11) combined with the then limited flight data led to design requirements to control ISS structural floating potential to within ± 40 V of the ambient plasma. More recently, the ISS FP requirement for control of the critical arcing hazard only during nominal flight operations has been changed. The new operational requirement, approved by the ISS Safety Review Panel (SRP), is that FP shall be no more negative than -60 V.

Two plasma contactor units (PCUs) designed to actively limit ISS FP to within ± 40 V of the ambient plasma, became part of the basic ISS design during 1990, thereby enabling the use of the negative polarity EPS grounding scheme at low risk by controlling ISS FP in a single-fault-tolerant fashion (14,15). At the time PCUs were adopted by the Program, only the critical hazard associated with dielectric breakdown arcing of specific thermal control coatings had been identified. Electrical interference with critical avionics and inadvertent firing of pyrotechnics as a result of surface dielectric breakdown arcing were both determined to be non-issues by the Russian and American Programs during the course of the safety review process before the first launch of 160 V PV array wings during late November 2000. Critical hazards require a single-fault-tolerant approach to hazard control, though either a zero-fault tolerant hazard control or no-hazard-control (i.e. acceptance or the risk) are also possible after exhaustive review and approval of a suitable Non-Conformance Report (NCR). The NCR quantifies the risk being accepted and justifies the approach, as required by SSP-50021, SSP-30309, and SSP-30599.

More recently, as a result of the continued review and assessment of spacecraft charging risks required by the ISS Program a possible catastrophic hazard, electrical shock to ISS crew during extra vehicular activity (EVA), was identified for the first time. ISS FP is required to be less negative than -40 V on all EVA translation paths and work sites during EVA. Catastrophic hazards require a two-fault-tolerant approach to hazard control, or a single-fault-tolerant hazard control with exhaustive review and approval of an NCR that quantifies the risk being accepted and justifies the approach.

PV array driven charging is only one contributor to possible ISS FP requirements violations. Magnetic induction potentials resulting from the flight of ISS thorough the geomagnetic field have been verified as an important contribution to FP as is shown in the following sections. Additional contributions to FP requirements violations can result of surface and structural charging following flight through auroral electron streams (16,17) though no direct observations of such charging have been made to date. Because PV array driven charging, magnetic induction potential charging, and auroral charging are independent processes to first order, the FP contributions of each simply add at any point on ISS.

Identified ISS Spacecraft Charging Hazards

The expectation or observed occurrence of spacecraft charging does not, in itself, constitute a risk or hazard requiring controls. Proper attention to electrical bonding and grounding practices (isolation of avionics from ground plane disturbances or radiated electromagnetic interference) during the design, development and test phase of a spaceflight program can effectively eliminate spacecraft charging concerns during flight operations. Adherence to the recommendations found in NASA Technical Paper 2361, Design Guidelines for Assessing and Controlling Spacecraft Charging Effects, (18) can eliminate most of the adverse effects of spacecraft charging events. Space Station Program Requirements for electrical grounding requirements (SSP-30240 Rev. C, Dec. 22, 1998) and electromagnetic compatibility (SSP-30423, Rev. E, June 9, 1998) assure that critical ISS avionics and pyrotechnics are not affected by spacecraft charging and dielectric breakdown arcing and the absence of such affects has been verified by testing. Slow degradation of the thin anodic thermal control coatings on the NASA, ESA, and JAXA pressurized elements as a result of dielectric breakdown arcing when FP is more negative than -60 V for long periods of time (> 6 months) is the only critical spacecraft charging hazard identified to date.

The design of the US extravehicular mobility unit (EMU) suit does not comply with the recommendations of NASA TP-2361, SSP-30240 or SSP-30423 as a result of necessary functionality and performance requirements. The only electrical connection between various large areas of partly exposed metallic conductors on the exterior of the suit is the astronaut's body. At the same time, stainless steel safety tethers assure that the astronaut can be grounded to ISS conducting structure in a variety of EVA configurations. It follows that an arc on one of several dielectric coated metallic EMU components can form an expanding, high-density arc plasma that moves outward and may provide a conducting path to discharge all or part of the NASA, ESA and or JAXA pressurized elements through the astronaut (5). The shock hazard to EVA crew is the only catastrophic ISS spacecraft-charging hazard identified to date. The FP control requirement that mitigates the hazard, FP no more negative than -40 V, is a requirement

established by NASA Flight Surgeons and is based on the minimum voltage necessary to drive a potentially lethal current through a perspiring (perspiration is a conductive electrolyte solution that reduces skin resistance dramatically) human being.

ISS Charging: in-Flight Observations and Data

PV array and magnetic induction driven charging of ISS has been the subject of numerous published studies both before and after the activation of the first set of 160 V PV array wings on ISS during early December of 2000 (4-15, 19-25).

PCU emission currents have validated the occurrence of the PV array driven charging mechanism, albeit with currents well below those expected before flight. PCU emission currents have also demonstrated the existence of substantial exposed conducting structure as a result of current collection enabled by magnetic induction voltages in ISS conducting structure.

As discussed below, flight measurement and analysis of U.S. Lab and Node 1 FP, with all FP controls disabled and PV array driven charging enabled, have not been more negative than -28 volts during 2001 (19-25). Plasma chamber testing (5) has shown that the dielectric breakdown voltage for the 1.3-micron thick anodic film on the US Lab and Node 1 meteoroid and debris shields is more negative than -60 volts. Therefore, the PCU system has not been in continuous operation since May 2001.

The ISS FP not-to-exceed-limit for EVA safety is no more negative than -40 V, however, and two PCUs are operated routinely during EVA in a single-fault-tolerant hazard control strategy. A two-fault-tolerant hazard control strategy is not required at this time because a PRA, demonstrating that the likelihood of violating the EVA safety FP requirement is less than 10^{-6} until the next set of 160 V PV array wings is installed on ISS (22-24), enabled approval of an NCR by the ISS Safety Review Panel. Floating Potential Probe measurements (19,20) were used to develop and partly validate a new charging model of ISS (22, 23, 25), which was used in conjunction with the recently developed ISS Plasma Environment Variability Specification (24) to demonstrate the very low likelihood of FP requirements violations.

PCU Emission Current Data

PCU emission current measurements have proven valuable in characterizing PV array driven and magnetic induction charging processes on ISS. PCU emission current from the ISS structural ground to the ambient ionosphere is measured with an LT-1006A operational amplifier in a linear current-to-voltage converter configuration calibrated so that an 8 V amplifier output corresponds to 12 amps of emission current. The amplifier is digitized for telemetry by a 12-bit analog to digital converter (with a sign bit) so that the least significant bit corresponds to 0.004 amps of emission current. The observed baseline noise during flight operations has been on the order of ± 0.005 amps with a zero offset of + 0.015 amps. Circuit analysis shows that the current-to-voltage converter circuit should be linear over the entire range from the noise band to 12 amps.

PCU emission current directly proportional to the net electron collection by the ISS PV-EPS system can be observed as abrupt increases in emission current at ISS orbital sunrise when the PV arrays are operating in sun tracking mode. Another means of measuring net PV-EPS electron collection is available via commanded full shunting of PV-array wings. Net PV-EPS electron collection measurements show the total amount of electron collection from the ionosphere that is not offset by conducting structure ion collection and as such is a good relative measure of the severity of the charging PV-EPS driven charging hazard.

Figure 1 shows measured eclipse-exit plasma contactor emission currents since January 2001. The eclipse exit emission currents show considerable variation both during a given 24-hour day and over the past 3 years. Large variations of Ne and Te with time of day, altitude, ISS latitude and longitude, geomagnetic field, solar activity, and season are expected and can explain much of the observed variability in the eclipse exit emission currents and the implied eclipse exit charging hazard. Detailed models of ISS PV array driven charging show that the magnitude of PV-array-driven charging will vary in a similar way with changes in the natural environment (22-25).

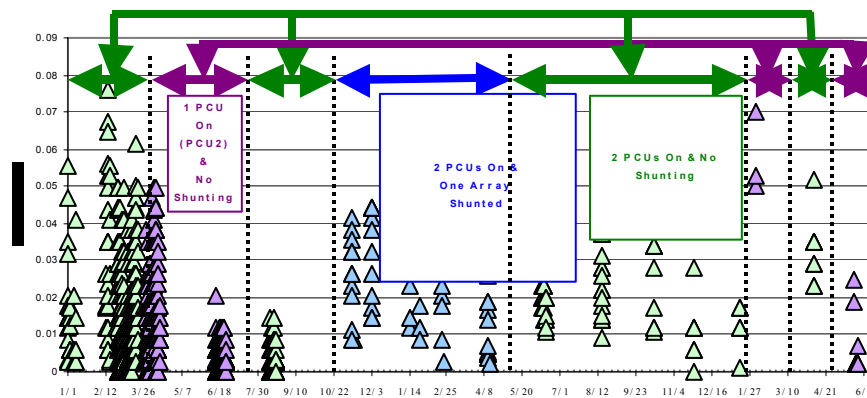


Figure 1. ISS plasma contactor emission current increase at eclipse exit: January 2001 to June 2003; US 160 V PV arrays, sun tracking and not shunted

The ISS PV arrays are automatically shunted whenever the ISS batteries are fully charged and ISS electrical loads do not demand full PV output. PV array wing full shunts can also be commanded from the ground. Commanding a shunt also provides a direct measure of the net PV array electron collection that isn't offset by ion collection by conducting surfaces. Figure 2 shows the orbital flight path corresponding to the commanded-shunting ISS telemetry data shown in Figures 3 and 4.

The top panel in Figures 3 and 4 shows the angular position of the individual PV array wings, P6-2B or P6-4B, in the ISS reference frame. The middle panel displays a number (PVCE-Error 3) that can be used to calculate the number of PV strings in each individual wing that are shunted at the indicated GMT time. The bottom panel shows the net PCU emission current from the ISS ground plane back to the ionosphere and including contributions from both 160 V PV array wings as well as ion and electron collection by ram oriented (i.e. forward facing) conducting structure.

Day 88 2001:88:20:04

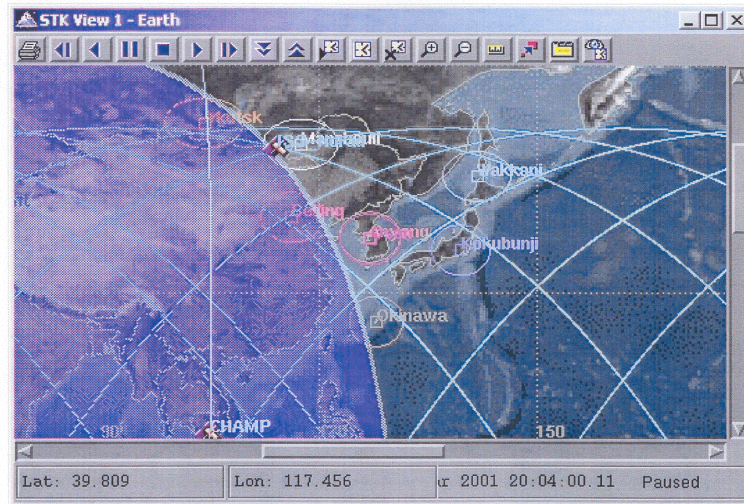


Figure 2. ISS orbital position and flight path corresponding to the commanded PV array shunt data shown in Figures 3 and 4.

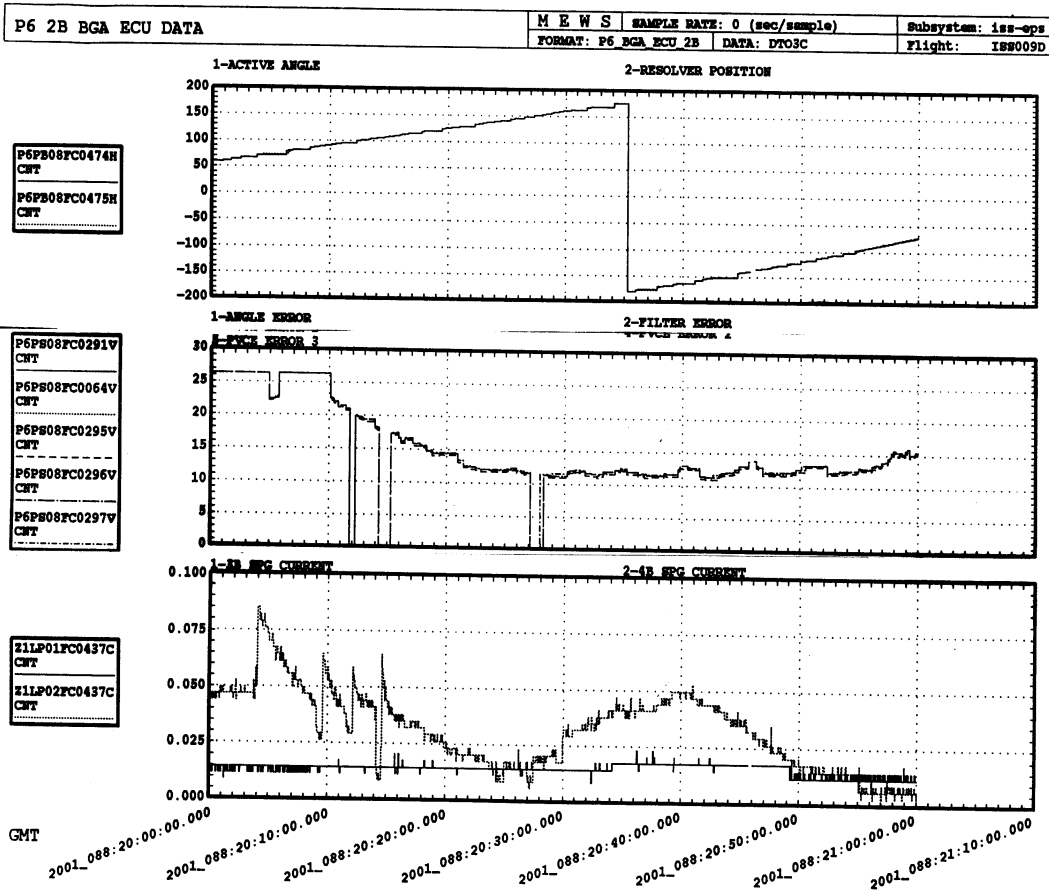


Figure 3: Changes in PCU emission current following commanded full-wing shunting of PV array wing P6-2B. Top panel – PV array wing angle in ISS coordinate system; Middle panel – Shunt voltage (0 = all PV strings shunted); Bottom panel – PCU emission current from ISS ground plane back to the ionosphere. PCU emission current contains contributions from PV array-EPS driven charging processes as well as current collection by ram oriented (forward facing) conducting structure biased by magnetic induction. Approximate number of un-shunted strings = $4.352 \times (\text{PVCE error} - 4.66)$: $\text{PVCE} < 5 \Rightarrow$ commanded shunt; all strings shunted.

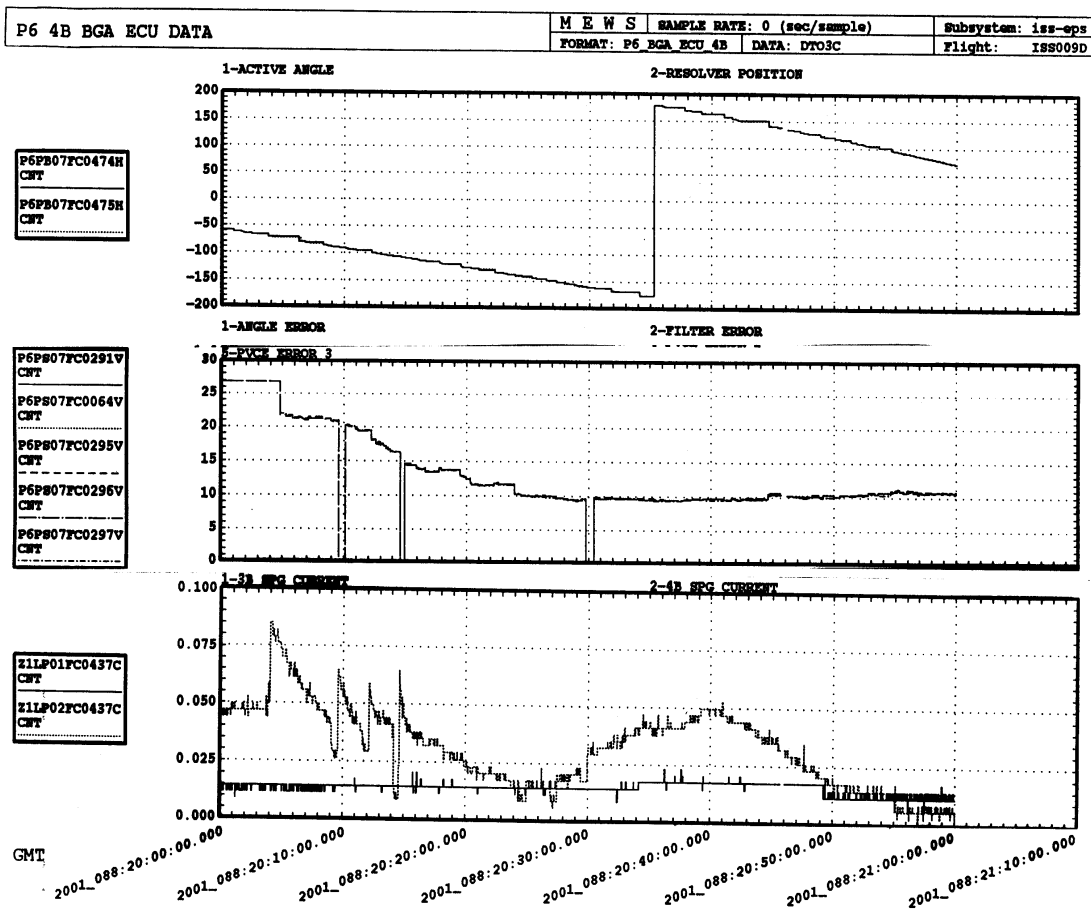


Figure 4: PCU emission current reductions associated with commanded shunting of PV array wing P6-4B. Top panel – PV array wing angle in ISS coordinate system; Middle panel – PV String Shunting PVCE = 0 => all PV strings shunted by command); Bottom panel – PCU emission current from ISS ground plane back to the ionosphere. PCU emission current contains contributions from PV array-EPS driven charging processes as well as current collection by ram oriented (forward facing) conducting structure biased by magnetic induction. Approximate number of un-shunted strings = 4.352 x (PVCE error – 4.66): PVCE < 5 => commanded shunt, all strings shunted.

The 160V ISS PV array cells are able to collect electron current from the ionosphere so as to contribute to charging hazards only when the active surface of the PV array is illuminated, not shunted, and not facing into wake. Figures 3 and 4 both show an abrupt increase in PCU emission current at eclipse exit (2001 88:20:04), (with an immediate small reduction in PVCE indicating that some strings were automatically shunted) with the active surface of the arrays facing forward, nearly perpendicular to the velocity vector. As the PV arrays track the sun past orbital noon (2001 88:20:36) the active surface the PV arrays turn into wake and further electron

collection by the PV array is suppressed. Beginning at 2001 88:30:00 electron collection by forward facing conducting structure becomes prominent again as it was at eclipse exit.

PCU emission current data needs accurate local environmental (Te, Ne) parameters quantitative interpretation. Nonetheless, the existence of the magnetic induction and PV array driven charging processes have been demonstrated and full wing commanded shunting and well as wake orientation have been shown to be viable non-PCU based charging hazard controls. The large changes in eclipse exit emission current, both seasonally and from orbit-to-orbit highlight the important role of environmental variability (24) in characterizing ISS spacecraft charging hazards. The hazards are not continuously present at a fixed level of severity and the hazards may disappear for considerable lengths of time. Therefore, likelihood of occurrence of the hazard as determined by PRA analysis becomes an important part of hazard characterization and control.

Floating Potential Probe Data

The results of in-flight floating potential probe (FPP) (19-25) measurements of ISS FP characterizing both the PV array driven charging process and the contribution of the $\mathbf{V} \times \mathbf{B} \cdot \mathbf{L}$ (\mathbf{V} = spacecraft velocity, \mathbf{B} = geomagnetic field, \mathbf{L} = length of conducting structure) magnetic induction voltages, with the plasma contactor system off, are shown in Figure 5 and Tables 1.

FPP measurements of ISS FP were made during several days in 2001, including intervals when the Space Shuttle was docked to ISS. On January 31, FPP data measurements of ISS FP were made with active side (the side with PV cell strings) of the active surface of the PV arrays in shallow wake flight attitude verifying that wake orientation of the arrays prevents PV array driven charging.

With the plasma contactor system off and PV arrays sun tracking, FPP data was collected on April 10-12, April 15, and on April 21 (before and after Space Shuttle docking). A total of 46 FP measurements characterizing PV array driven charging were made in 2001, encompassing a wide range of ionospheric conditions. Langmuir probe measurements of electron temperature, Te, at eclipse exit ranged from 0.08 to 0.23 eV while electron density, Ne, ranged from 10^9 to $10^{12}/\text{m}^3$. To date, the observed range of PV array-driven charging FP values range from -4 to -24 V. It should be noted that the FPP could not provide Te or Ne if Fp exceeded -10V negative as a result of the limited range of the Langmuir probe sweep voltage.

The April 11 data is fairly typical, despite the geomagnetic storm starting about 13:30 universal time (UT). Figure 5 shows the ISS FP at the FPP measurement point as a function of universal time on April 11, 2001. In Table 1 the total FP for the April 11, 2001 eclipse-exit charging peaks, shown in Figure 2, are broken down into the magnetic induction and PV array driven components for the locations on ISS defined in Figure 6.

Magnetic induction voltage is a significant fraction of the total FP in all cases, and must be considered in any ISS charging assessment. As shown in Figure 2, the agreement between calculated magnetic induction voltage and measurement is excellent in all cases. Figure 6 shows a calculated magnetic induction voltage map of ISS when passing south of Australia on April 11,

2001. Flight south of Australia, in the flight attitude indicated in Figure 6, generates the more magnetic induction voltage on ISS than any other ISS flight path.

The data shown in Figure 5 and Table 1 span 6 orbits or 9 hours. During that time, the rotation of the Earth changed the geographic location of ISS eclipse exit from near the west coast of South America to Australia. The magnetic induction voltage peaks twice on each orbit, at ± 51.6 degrees latitude. Eclipse-exit PV array driven charging peaks are superimposed on the -51.6 latitude magnetic induction peaks. The $+51.6$ magnetic induction voltage peak occurs during eclipse. The measured FP consists only of magnetic induction voltage when ISS is in eclipse or when the PV arrays are shunted or in wake. When sun tracking, the active surface of the PV arrays move into wake at orbital noon. Figure 2 also shows a comparison of magnetic induction voltages calculated using a first principle model (22, 23, 25) with the flight data, demonstrating excellent agreement between the magnetic induction model and the flight data. The ISS magnetic induction voltage map shown in Figure 3 was calculated using the model (22, 23, 25).

Table 1: Post Eclipse Exit ISS Charging Peaks (maximum negative FP in volts) from Figure 2, April 11, 2001 at GMT time indicated (PCU system off)

12:38					14:10					15:41				
	2B	LAB	FPP	4B		2B	LAB	FPP	4B		2B	LAB	FPP	4B
vxB	2.034	-6.51	-3.42	-9.56	vxB	2.571	-7.38	-4.19	-11.5	vxB	2.812	-8.22	-4.96	-13.5
Chg	-19.6	-19.6	-19.6	-19.6	Chg	-17.0	-17.0	-17.0	-17.0	Chg	-15.2	-15.2	-15.2	-15.2
Total	-17.5	-26.1	-23.0	-29.1	Total	-14.4	-24.4	-21.2	-28.5	Total	-12.4	-23.5	-20.2	-28.7

17:15					18:46					20:18				
	2B	LAB	FPP	4B		2B	LAB	FPP	4B		2B	LAB	FPP	4B
vxB	3.429	-9.7	-6.55	-17.2	vxB	3.775	-10.2	-7.35	-19.1	vxB	3.903	-10.1	-7.65	-19.7
Chg	-7.0	-7.0	-7.0	-7.0	Chg	-4.7	-4.7	-4.7	-4.7	Chg	-5.2	-5.2	-5.2	-5.2
Total	-3.5	-16.7	-13.5	-24.2	Total	-0.9	-14.9	-12.0	-23.8	Total	-1.3	-15.3	-12.9	-25.0

Table 1 definitions: vxB=magnetic induction voltage; Chg=PV array driven charging; Total=vxB + Chg; For ISS locations 2B, 4B, Lab and FPP see figure 3

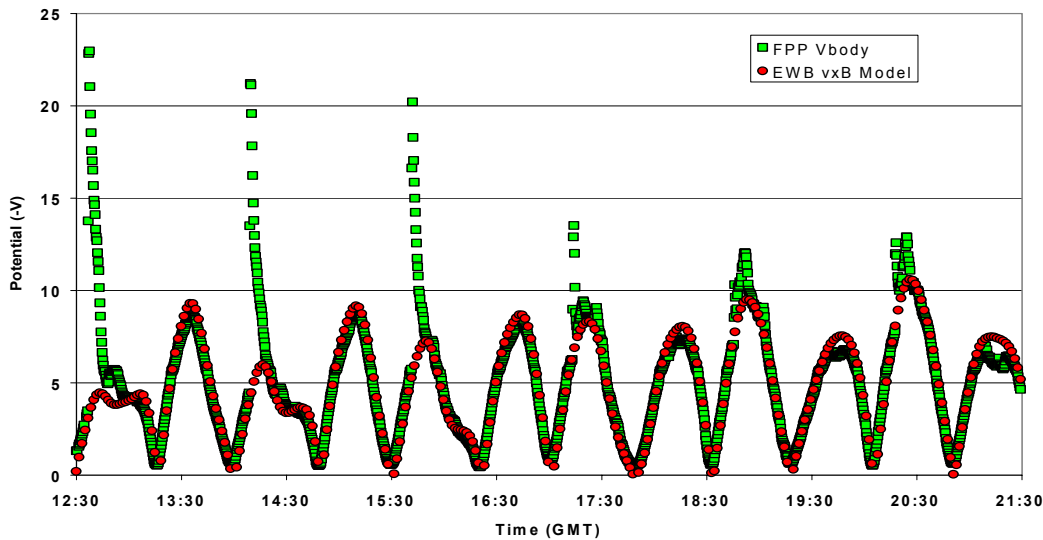


Figure 5: ISS FP at the FPP measurement point (FPP Vbody) with the plasma contactor system off. Calculated magnetic induction FP is compared with measured FP. April 11, 2001

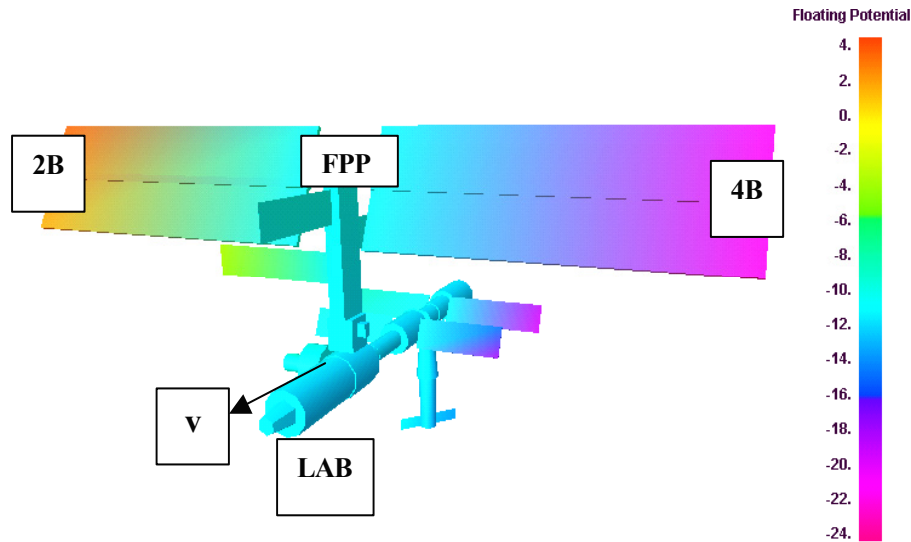


Figure 6: Calculated worst-case magnetic induction voltage map of ISS in +XVV flight attitude (velocity vector v), with 10 degrees down pitch. April 11, 2001

High Latitude Auroral Electron Charging

The possibility of spacecraft charging by auroral electrons at high latitudes, during geomagnetic storms or other geomagnetic disturbances, is a subject of some concern on the part of the spacecraft charging community (16,17, 26-30). Analysis of historical satellite charging and anomaly data for the United States Defense Meteorological Satellite Program (DMSP) satellites and the European Space Agency Freja satellite both suggest that auroral charging may be observed on ISS at high magnetic latitudes (26-30), especially at night during solar minimum. Charging of the Freja and DMSP vehicles has been correlated with ionospheric plasma densities of $10^4/\text{cm}^2$, or less, combined with fluxes of energetic auroral electrons (7-10 keV) greater than 10^8 electrons/($\text{cm}^2\text{sec sr}$) (26-30). The DMSP and Freja satellites both orbit the Earth at or above 800 km, in the topside ionosphere, well above ISS operational altitudes. Nonetheless, the required combinations of ionospheric plasma density and energetic electron flux are expected to occur at ISS altitudes, albeit infrequently, at or near the extreme latitudes of the ISS orbit ($\pm 51.6^\circ$). Inspection of the auroral precipitation maps produced hourly by the US National Oceanics and Atmospheric Administration (NOAA) Polar Orbiting Environmental Satellite (POES) constellation show that ISS passes through the precipitating auroral electrons several times every day, whenever ISS passes south of Australia at night and K_p is greater than about 4 (31).

The question of whether or not flight through the same kind of environment that produces charging and the occasional recoverable anomaly on DMSP constitutes a risk or hazard for ISS or ISS EVA crew remains open. The absence of severe anomalies on the Freja spacecraft in a similar, if not more severe, charging environment highlight the important effect of spacecraft design on spacecraft charging effects. More detailed assessments of the frequency of

occurrence of the auroral charging environment at 350 to 400 km altitude as well as detailed analysis and modeling of the expected ISS and EVA suit charging processes in that environment are in work at this time. ISS and the EVA suits used on ISS are not identical to DMSP or Freja, when treated as electrical systems interacting with the auroral charging environment. Materials properties and materials interactions with the auroral charging environment will likely determine the outcome of the assessments. Secondary electron yields, dielectric coating thickness compared to energetic electron range, and total area of exposed conducting surfaces and vehicle or subsystem capacitance are all important factors. The ISS plasma contactor system also contributes to control of any auroral charging risk by both increasing local plasma density and providing a return path to the ionosphere for any charging of ISS conducting structure produced by auroral electrons.

During the first two years of flight (during the current solar maximum), no ISS equipment anomalies have been reported that correlate with geomagnetic storms or flight through either the diffuse or visible auroras. The ISS crews have reported flying through visible Aurora Australis on at least two occasions. The following excerpt from Commander William Shepherd's deck log of Nov. 10, 2000 is an interesting example.

"11:30: Transited through a very unusual aurora field. Started as a faint green cloud on the horizon, which grew stronger as we approached. Aurora filled our view field from SM (Service Module) nadir ports as we flew through it. A faint reddish plasma layer was above the green field and topped out higher than our orbital altitude."

Figure 3 shows southern hemisphere auroral precipitation maps produced by NOAA POES Satellites 15 (Nov. 10, 2000 10:56 UT) and 16 (Nov. 10, 2000 12:26 UT). Both NOAA SEC satellites show auroral activity levels of 9 and hemispheric powers of 60 to 90 gigawatts, resulting from relatively intense diffuse auroral electron precipitation over Tasmania and southern New Zealand showing that that ISS was well within the precipitating electron environment during the visible auroral arc fly through reported by Cmdr. Shepherd (31). Figure 8 shows an aurora over Greenland photographed by Cmdr. Culbertson during ISS expedition 3.

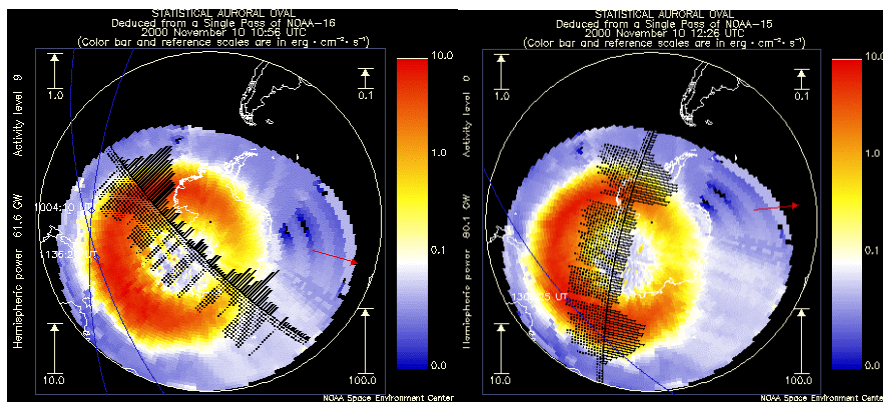


Figure 7. Statistical auroral oval with ISS and NOAA-15 and NOAA-16 ground tracks on November 10, 2000.



Figure 8: Photograph of Auroras over Greenland taken from the ISS Service Module by Cmdr . Frank Culbertson during Expedition 3.

Managing ISS Charging Risks: Probabilistic Risk Assessment (PRA)

The variability of the environment and the resulting variability in the severity of ISS spacecraft charging risks as well as realistic concerns about the reliability of the hazard control hardware leads naturally to the use of Probabilistic Risk Assessment methodology for both assessing and mitigating charging risks.

The ISS Plasma Hazard Evaluation Process (PHEP) consists of five major components all of which provide input into the PRA product.

- 1) The Floating Potential Measurement Unit (32) – A recently developed flight instrument designed to measure FP along with Te and Ne. The data from the FPMU is subject of several stages of validation screening, beginning with parity error checks on telemetry data, continuing with quality-of-fit screening applied to the theoretical relationships to which flight data is fitted for reduction and analysis and completing with comparisons with measurements made on other spacecraft or by ground based incoherent scatter radar facilities. The FPMU is not used for validation of hazard control systems during EVA because the instrument only shows that at least one hazard control is operating when the requirement is for verification of either a single-fault or a two-fault tolerant hazard control system. The FPMU can and will be used for periodic functionality checks on individual hazard controls. The prime function of the FPMU is to provide data to validate the Plasma Interaction Model.
- 2) The Plasma Interaction Model (22, 23, 25) – Simple FPMU measurements on a given day or over a given limited time frame do not provide a reliable indication of the severity of the hazard simply because the environmental factors causing the hazard are so variable. In order to calculate the frequency of occurrence of FP requirements violations a validated model providing reasonably accurate predictions of ISS FP given ISS configuration, attitude, flight path, and flight environment is needed. FPMU data will be used to validate the charging model and only then can the frequency of occurrence of the FP requirements violation can be

calculated using the frequency of occurrence of the environmental conditions leading to the violation. If model validation is not possible then no relaxation of the two-fault-tolerant hazard control based on low likelihood of FP violations is possible.

- 3) The Specification of ISS Plasma Environment Variability (24) provides the likelihood of occurrence of flight environments corresponding to various levels of charging risk. The specification is based on a critical evaluation and statistical analysis of all available satellite and ground based data applicable to the ISS charging problem.
- 4) Critically evaluated and statistically valid test data (33) determining the arcing threshold for various materials on the ISS outer surface and on the outer surface of the EMU suit are essential for setting meaningful FP requirements needed for definition of effective hazard mitigation.
- 5) Probabilistic Risk Assessment provides the methodology needed to logically characterize the necessary and sufficient conditions for a hazard to exist, evaluate the likelihood of occurrence of FP requirements violations, and assess the reliability of the ISS electrical power system and command and data handling components that comprise the hazard control system. If the likelihood of hazard control failure is low enough then a single-fault-tolerant hazard control system can be deemed adequate for mitigation of a catastrophic hazard, even if the likelihood of occurrence of the hazard itself is high enough to require a two-fault-tolerant approach.

These five essential components are assembled in the PHEP to produce the hazard reports, flight rules and procedures that assure the lowest risk and highest benefit for the ISS vehicle and ISS crew.

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