ISS AND SPACE ENVIRONMENT INTERACTIONS IN EVENT OF PLASMA CONTACTOR FAILURE

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ABSTRACT

Solar arrays on the International Space Station (ISS) operate at 160 volts. The negative side of the power system is grounded to ISS structure. The high voltage solar arrays coupled with ISS design and materials properties can lead to detrimental interactions with the ionospheric plasma. A plasma contactor unit (PCU) on ISS provides a "ground wire" to control the voltage between the ISS structure and local plasma and prevents such interactions. An evaluation of possible effects that could result in the event of plasma contactor failure was undertaken and is reported in this paper. In general, any identified impacts on ISS function take several months of contactor malfunction to develop. Possible hazards to crewmembers on EVA are addressed and sufficient controls have been put in place.

INTRODUCTION

When complete, the ISS will be the largest structure placed in orbit. In order to provide energy to run the station and provide power to experiments, the ISS has a large power system consisting of large solar arrays (fig. 1). Each wing is 12 meters wide and 34.5 meters long and there will be eight of them at ISS Assembly Complete. The combined US and Russian power systems will generate 110 kilowatts of power for the ISS. In order to efficiently produce and distribute that much power, the voltage level of the solar arrays and power distribution system has been raised. In the past, spacecraft have operated with a 28 V direct current (DC) bus and the solar arrays were typically a few volts above this. The ISS solar arrays generate 160 V, and the distribution system is at 120 V DC. As is typically the case, the negative side of the power system, including the solar arrays, is grounded to the ISS structure.

Early in the ISS design it was recognized that the operation of such a large complex having a high voltage power system in contact with the ambient, ionospheric plasma would result in spacecraft interactions with the environment (refs. 1-4). As a result of evaluation of these effects, the decision was made to add a plasma contactor system to ISS to control the voltage between the ISS structure and the local plasma, thereby preventing any voltage driven interactions. Two plasma contactors are on ISS, but only one at a time will operate.

VOLTAGES RELATIVE TO LOCAL PLASMA

The 160 V solar arrays on ISS are immersed in the ionospheric plasma. The electron thermal velocity is greater than the ISS orbital velocity so they basically have only their thermal velocity. The ions have a directed energy of near 5 electron volts (eV) due to the almost 8 km/sec orbital velocity of ISS. Therefore, there will be a plasma wake on the anti-ram side of station elements. Because of the much lower mass, greater mobility and much higher thermal velocity of electrons, they are much more easily collected by a surface at a given positive voltage than ions to a surface with the same negative voltage.

The solar arrays generate the 160 V operating voltage by stringing many cells in series. Therefore, a distribution of voltage is present all along the array string. The solar array voltage relative to the local plasma will come to an equilibrium so that no net charge is collected by the solar array. For the 160 V array and defining the local plasma as zero volts, the zero volt point on the solar array will be such that the positive end of the array will be near +20 V and the negative end of the array will be about –140 V. This is illustrated in Figure 2. The ISS structure, grounded to the negative side of the solar array, is mostly covered by dielectric materials and is a poor collector and conductor of ion current so the ISS structure will also be near –140 V.

Figure 1 ISS 5A Configuration

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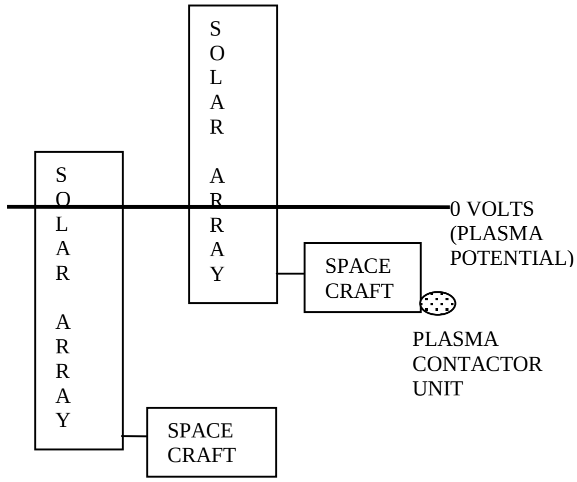

Figure 2 Illustration of Solar Array and Structure Voltage Relative to Local Plasma

The full calculation of floating equilibrium is much more complicated and depends on ISS cell geometry, solar array orientation to the ion flow vector, plasma density and temperature, electric field growth into the plasma, and other factors.

A large majority of the ISS structure is covered with anodized aluminum debris shields. The anodizing process is tailored in order for the surface exposed to space to have the right solar absorptivity and emittance for thermal control. The anodize thickness of the debris shields is on the order of 1.3 microns. The outer surface of the anodized layer is in contact with the plasma and will come to equilibrium with it and, due to the low electron temperature, will be within a volt of local plasma potential. The underlying structure can be at the highly negative voltage due to its grounding to the array. Therefore, the entire voltage difference between the ISS structure and the local plasma drops across this thin anodized surface.

ISS CAPACITANCE AND STORED ENERGY

As described above, the large area thin anodized surfaces on ISS will have up to 140 V across them, therefore becoming large capacitors and storing a lot of energy. The ISS modules will initially be uncharged capacitors, and the electric field will penetrate into the ambient plasma from the dielectric surface and collect ions until the capacitance is fully charged and the voltage fully developed across the dielectric surface. The charging process time is dictated by the available ion current to ISS surfaces. The maximum ion current density is due to ram ions having orbital velocity relative to ISS and is given as $n_e ev$, where n_e is the plasma density, e is electronic charge, and v is velocity.

For surfaces that are perpendicular to the orbital velocity the surface is in contact with the same plasma density but the ion current is dictated by the ion thermal velocity, $(2\ K\ T_e/m_i)^{1/2}$, where K is the Boltzmann constant, T_e is the plasma electron temperature, and m_i is the ion mass. When compared to the ram velocity, the ram ion current density is approximately seven times the thermal current density. Therefore, the surfaces in the ram direction will charge the capacitance seven times faster than surfaces perpendicular to the ion flow.

Because of the large surface area and the extremely thin dielectric, a planar capacitance description can be applied with the capacitance being $C = \varepsilon_0\ \kappa\ A/d$, where ε_0 is the permitivity constant, κ is the dielectric constant, A is the surface area, and d is thickness. Here a value of 5 was used for κ with confidence of being within a factor of two.

Table 1: Capacitance and Energy Stored in ISS Components

ISS COMPONENT	AREA (m^2)	THICKNESS (micron)	CAPACITANCE (farad)	CHARGE SQ METER	CHARGING TIME (SEC)	ENERGY (JOULES)
USA						
Node 1	67.2	1.3	2.30E-03	4.11E-03	3.33	22.08
P6 Truss	19.509	15	5.79E-05	3.56E-04	0.29	0.56
Z1 Truss	8.37	15	2.48E-05	3.56E-04	0.29	0.24
Lab Module	128.9	1.3	4.41E-03	4.11E-03	3.33	42.36
Pressurized Mating Adapter	19.2	15	5.70E-05	3.56E-04	0.29	0.55
TOTAL for 5A	178.729		6.85E-03			65.78
Endcone	12	1.3	4.11E-04	4.11E-03	3.33	3.94
RUSSIAN						
Soyuz	40	125	1.42E-05	4.27E-05	0.03	0.14
Zarya	60	125	2.14E-05	4.27E-05	0.03	0.21
Service Module	100	125	3.56E-05	4.27E-05	0.03	0.34

The thickness of anodized layers is different depending on the anodization process. The large majority of ISS coatings is chromic acid anodize, and they have a thickness of approximately 1.3 microns. By contrast the sulfuric anodized coatings, such as on the truss structure, is approximately 15 microns thick.

Painted surfaces will be even thicker at about 125 microns. Table 1 gives the capacitance for different ISS components up through build 5A when the Lab module is delivered. The area of Russian components was calculated based roughly on size. A plasma density of 10^{12} m^{-3} (typical of low Earth orbit during solar maximum) is used to calculate Table 1 values. Note that the Node 1 and Lab modules have by far the largest capacitance and store the greatest energy. This is due to their large area and thin dielectric.

The area values in Table 1 were provided by the ISS project office and were determined to be the area exposed to the local plasma. The capacitance values are for this entire area. In the laboratory all arcs extinguished when the voltage driving it reached about 20 V. Therefore, (140 V – 20 V) is the voltage used to determine charge available from the capacitance. The Charging Time in Table 1 is the time necessary for the ram ion current to recharge the capacitance. For the highest capacitance it is about 3.3 seconds. Remembering that the ion current density is about a factor of seven lower for perpendicular surfaces, they will charge in about 23 seconds. These times will also represent the time to recharge the capacitance following an arc, which depletes the stored energy. The energy calculated in Table 1 uses the voltage parameter of ($140^2 – 20^2$).

ARCING AND SPUTTERING

The Environmental Workbench (EWB) software package (ref. 5) predicts that without an operating plasma contactor the ISS structure voltage for the early configurations can go to about –140 V. Witness samples of ISS manufactured debris shields and other surfaces were provided for laboratory test. In the lab, the chromic anodized surfaces experienced dielectric breakdown at near –80 V and the sulfuric anodized at near –200 V.

In the case of a plasma contactor not operating, the chromic anodized surfaces will not be able to withstand the voltage that will develop across them and they will arc. How much energy will go into an arc and by what process? It has been shown that the locally dense plasma created at the arc site can carry current and as the arc plasma expands and sweeps across the surface, local current loops discharge into the arc the local energy that is stored capacitively. This is described as a Type 1 arcing process. Another process that doesn't have to be as local is defined as a Type 2 arcing process. When an arc occurs it acts as a low voltage short between the local plasma and ISS structure. The voltage across the ISS "capacitor" can be up to 140 V. Therefore, the outer surface in contact with the plasma that was near zero volts before the arc is now over 100 V positive of the local plasma and easily collects electron current from the local ionospheric plasma. The circuit is completed by electron current flowing to the ISS surface, through the capacitance and along the structure to the arc where it is emitted back to the ionosphere through the arc. This current is limited to the electron thermal current to ISS surfaces but can be tens of amperes.

When an arc does extinguish, the positive array will very quickly collect electron current and drive the structure negative again. The charge required to do this is roughly that associated with the "free space" capacitance of the ISS structure. The dielectric surface in contact with the plasma, representing the other side of the capacitance, will be negative and will collect ions until the capacitance is charged, the time being given in Table 1.

The movement of ions to the negatively charged ISS also can promote sputtering of surfaces. When ISS is negative with respect to the plasma, the exposed surfaces will collect ions from the local plasma. The ions will be accelerated by the voltage difference and sputter the surface on impact.

DOCKING AND EVA CONSIDERATION

ISS can charge negative during portions of its orbit if there is no functioning PCU, but other vehicles, not having a similar power system, will be near plasma potential or zero volts. Various vehicles are docking and undocking with ISS, and components are being manipulated with the ISS and Shuttle Remote Manipulator Systems (RMS). With the large amount of energy stored in the ISS structure at –140 V, a concern was raised regarding contact with vehicles near plasma potential. That is, could an arc occur on contact that would discharge the energy stored in the ISS structure.

A computer model was used to predict currents and voltages following docking. The result indicated that charge transfer and arcing would not occur. This is because the docking vehicle will have some capacitance but is initially uncharged. The charging of this capacitance depends on ion current to the outer surface. This effect resistance is high enough to preclude large energy transfer at the moment of docking. Also a 5 kΩ resistor is in the grapples of the RMS and similar fixtures. This additionally restricts currents during docking. Therefore, large currents and transfer of energy between vehicles during docking is not possible.

Astronauts on extravehicular activity (EVA) raise questions regarding their contact with ISS structure, relative voltages and the possibility of arcing, which could involve the EVA astronaut. If the astronaut is free floating and not in electrical contact with ISS then there is no energy or voltage source which can involve the crewmember in an arcing process. One possibility for coupling was hypothesized. The EVA crew-members are attached to ISS by a stainless steel tether. It is possible for this tether to contact the station at its attachment and the tether can contact exposed metallic structure on the Extravehicular Mobility Unit (EMU). This could move the EMU and astronaut ground negative of the local plasma the same as the ISS structure voltage. In this case, exposed anodized surfaces on the EMU could arc. Very little energy can be stored in the EMU surfaces but, being in close proximity to ISS surfaces, the expanding plasma contacting ISS outer surfaces could provide a low impedance path to complete a circuit through ISS structure and back through the tether. Two plasma contactors functioning during EVA represent two controls. Modeling with EWB has shown that array orientation to the velocity vector can control ISS structure voltage by restricting contact between the cell side of the array and local plasma.

EMI

Arcing will create radiated noise that could interfere with the operation of ISS radio receivers or other ISS avionics. This arcing will create conducted noise currents in the vehicle structure that could couple into and interfere with the safe operation of ISS avionics. Through a series of tests and analyses, the possible impacts of these interference scenarios on ISS avionics were evaluated and found to produce minimal effects.

4A AND 5A CONFIGURATION EVALUATION

Due to delivery of the first US solar arrays on flight 4A, the above described charging and resulting interactions for ISS were not a concern for prior assembly builds. Therefore, the 4A and 5A configurations are the first that require plasma interaction evaluation. The 5A assembly flight delivered the Lab Module that adds area, capacitance and stored energy to the total ISS configuration. The next assembly flight after 5A to add appreciable additional area with thin dielectric is 7A. Therefore, most of the 5A evaluation of ISS interactions with the space plasma will be applicable up to 7A. During the entire assembly process, as more hardware is added, the available energy for arcs will increase and potential arcing sites (those in the velocity vector) will change.

As the ISS configuration matures, the flight attitude will be one described as Local Vertical Local Horizontal, LVLH. In this attitude the velocity vector is along the longitudinal or X-axis of the ISS configuration with the docking Endcone normal vector into the ram direction. The angle between these two vectors can vary by up to 15 degrees. In the 4A to 7A timeframe the primary attitude will be LVLH but part of the time ISS will fly in an attitude where the X-axis is perpendicular to the orbital plane (XPOP), and the ISS rotates about the X-axis once each orbit. Therefore, the velocity vector is normal to the side of the Node in 4A and of both the Node and Lab Module in 5A.

ENERGY AVAILABLE TO ARCS

Table 1 gives the energy stored in the components on orbit during the present configuration. The values given are for the entire surface area. For XPOP configurations half of this area will be in the ram direction. The other half will be in the wake and, taking longer to charge, will not likely provide energy into a continuous arcing condition. Therefore, the values shown in Table 1 are a factor of two greater than anticipated. However, these values were used in interaction calculations to be conservative.

In Table 1 are also values associated with the Endcone of the Lab Module. In the LVLH mode the area projected into the ram direction is the Endcone area. This will have the shortest charging time and, assuming that an arc will occur by the time full voltage is developed across the dielectric in the ram direction, the energy listed under Endcone is that provided from a full charge. However, the capacitance of the sides of Node 1 and the Lab Module will be only partially charged in the 3.3 seconds it takes to charge the ram surface. For a Type 1 discharge process the arc plasma may expand to include part of the side of the Node 1 and Lab Module. Type 2 processes could provide a mechanism to discharge the other Node 1 and Lab Module stored energy into an arc on the Endcone. Therefore, the total energy available for an arc in the LVLH flight orientation for the 5A configuration is given by the value in Table 1 for the Endcone plus that due to partial charging of the remaining capacitance. The value is,

$$E_{5A,\,LVLH} = E_{Endcone} + ½\,[(C_{Node} + C_{Lab})\,\{(V_{Side})^2 - (V_{Ex})^2\} + (C_{P6} + C_{Z1} + C_{Russian})\,\{(V_{Max})^2 - (V_{Ex})^2\}]$$

where V_{Side} is given by 3.33/23 (V_{Max}) + V_{Ex} . The voltage must be added to V_{Ex} since previous arcs will have discharged to a point where they will extinguish, and this represents a voltage start for the recharge process. Carrying these calculations through results in a value of 10 Joules available to an arc for the 5A configuration in the LVLH flight mode.

CONTAMINATION EVALUATION

A central pit occurs in the center of the arc site and Lichtenberg patterns radiate from the pit. The size of the pit and the extent of the radial pattern depend on the energy into the arc. Such arcing produces ejecta that are released both as a vapor and fine particulate. Deposition on other ISS surfaces represents a contamination. Also, sputtering of conductive surfaces and non-conducting surfaces during charging will be contamination sources.

As seen in Figure 1, the line of sight from the Endcone to other ISS structure is extremely limited. The Z1 truss and solar array wings have a small but very limited view at an extreme angle from the Endcone. Contamination is released in roughly a cosine distribution. The overall result for the 5A configuration is that the limited surfaces on which any deposition could occur are such that the deposition amount is negligible. This may not be true for later ISS configurations when additional hardware is added.

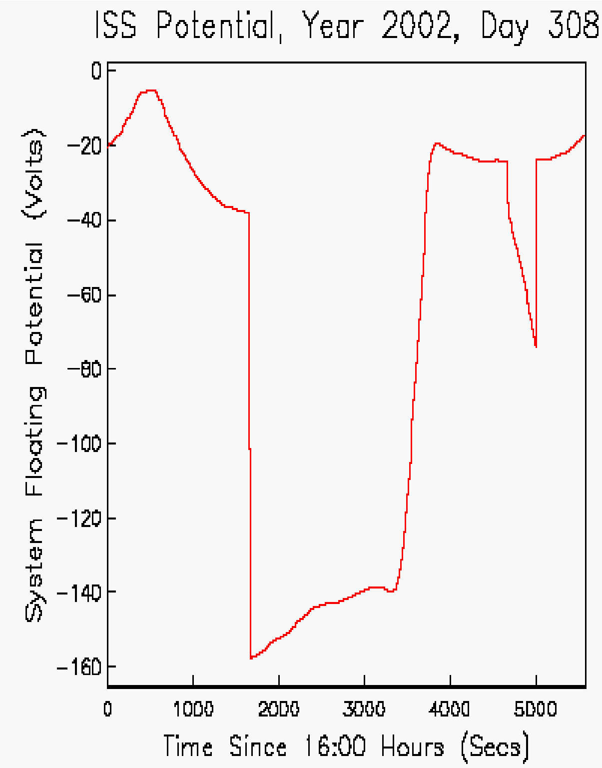

Figure 3 EWB Prediction of ISS Structure Voltage Relative to Local Plasma

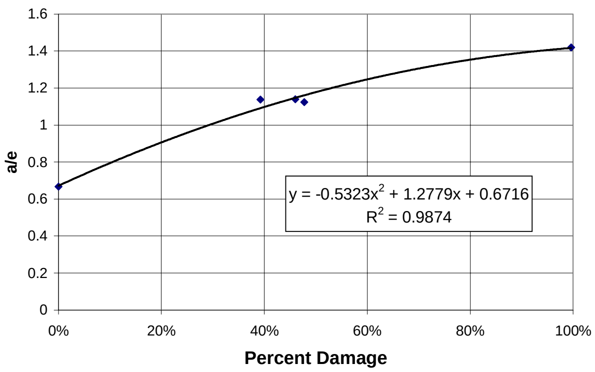

Figure 4 Solar Absorptance/Infrared Emittance Ratio as a Function of Arcing Damage

DEGRADATION OF THERMAL CONTROL

The anodized outer layers of the ISS modules are designed to provide the proper solar absorptivity and emittance for thermal control. Arcing of these surfaces removes the anodization layer at the arc site, exposing the underlying metal and physically alters the surface at the arc site. These changes will alter the thermal control properties, and this will vary as a function of arcs per unit area on the surface. This will also depend on the energy into the arc and the extent of local damage occurring at the arc site. Samples were arced in the laboratory and the solar absorptance and emittance measured as a function of the surface density of arc sites.

Figure 4 is the experimental ratio of solar absorptance to emittance. In order to apply this curve to ISS 4A and 5A configurations, the surfaces that arc and the increase in arcs per unit area must be determined. In LVLH, the Endcones will most prominently arc and in XPOP the cylindrical sides of the Node 1 and Lab Modules will be the arc sites. As seen in Figure 3 about 1/3 of the time in orbit the structure can be driven to a low enough voltage to support arcing. Taking all these values into account, as well as the recharge time for surfaces in the ram direction, the arc density per unit area can be calculated for time in orbit. From this the changes in solar absorptivity and emittance with time in orbit can be determined. Once this is done the altered thermal properties can be used to determine temperature changes on ISS surfaces.

ISS thermal models were utilized and external temperatures determined for the changes in thermal properties due to arcing on surfaces. The result is given in Figure 5. Some touch temperature limits are shown and can be exceeded. However, to reach them requires several months of operation without a PCU and without applying any operational methods to control ISS structure potential.

SUMMARY

The ISS is different than any other system that has been placed in orbit before. Its power system will interact directly with the ionospheric plasma. The resulting interactions with ISS are very much a systems issue as the effects depend on ISS power system design, materials selections and local plasma conditions. None of these interactions can lead to degradation or safety concerns unless both operating PCUs

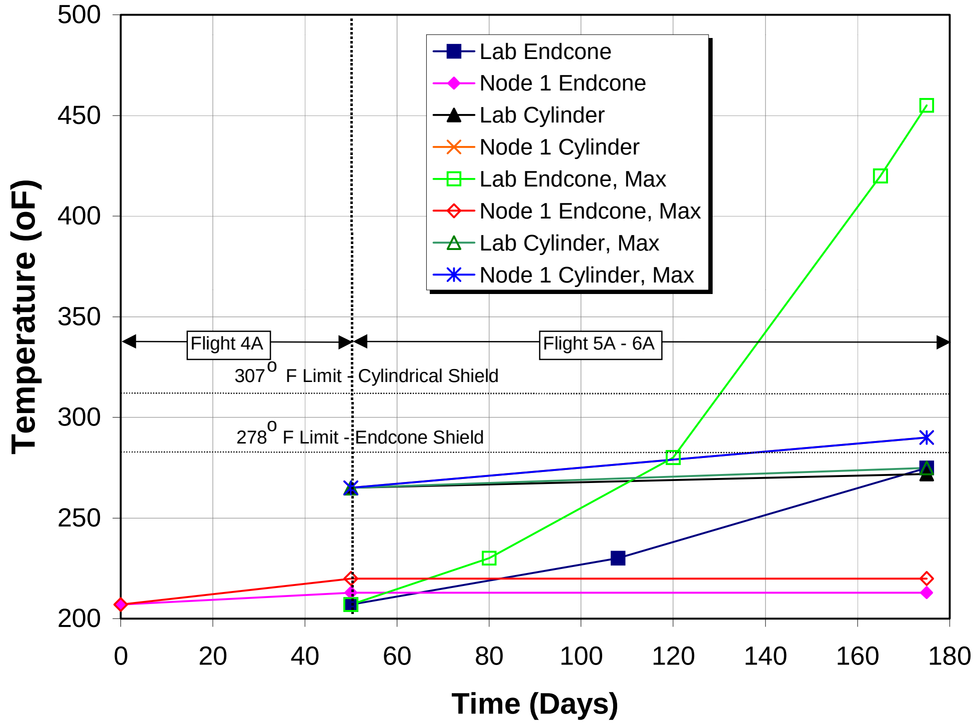

Figure 5. External Touch Temperature Changes Due to Arcing

on ISS have failed. In addition, operational procedures have been developed which can limit charging voltages to benign levels except for a few specific orbital conditions. This was the purpose of the effort to evaluate and quantify effects so that degradation with time was quantified, safety issues identified and the proper level of control developed to provide for successful ISS function.

Some of the interactions presented, such as ISS charging, capacitance, energy storage and dielectric breakdown, are well understood and characterized. As we consider the energy that will go into arcs and the process by which this will occur the certainty is less and more assumptions are made, often necessarily conservative. The resulting impact of arcing on thermal control surfaces is coupled to the same assumptions regarding energy input into the arcs and arc rate. It is believed that the results presented in this paper are not conservative by more than a factor of two. More work is required to understand and characterize these effects on a large physical scale and for orbital conditions. ISS itself may offer the best testbed.

Consideration of EVA safety requires that any scenario that can produce a hazard to the crew be considered, however remote. The likelihood of any such occurrence, even without controls, is very small since several simultaneous conditions are required. Beyond that, both ISS plasma contactors will operate during EVA activities and any one operating will prevent ISS charging related hazards. Additionally, the ISS will be flown in such a configuration during EVA as to prevent charging by keeping the forward side of the arrays into the wake.

More work is required to decrease the uncertainty in some interaction phenomena and assess future builds of the ISS. Interactions of ISS with the ambient plasma are controllable via the PCUs and operational procedures.

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