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9th Spacecraft Charging Technology Conference

4-8 April, 2005

EPOCHAL TSUKUBA, TSUKUBA, JAPAN

Book of Abstracts

20090106244

Hosted by

JAXA

Japan Aerospace Exploration Agency

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NASA SEE esa

Sponsor's Speech

Opening Ceremony, the 9th Spacecraft Charging Technology Conference

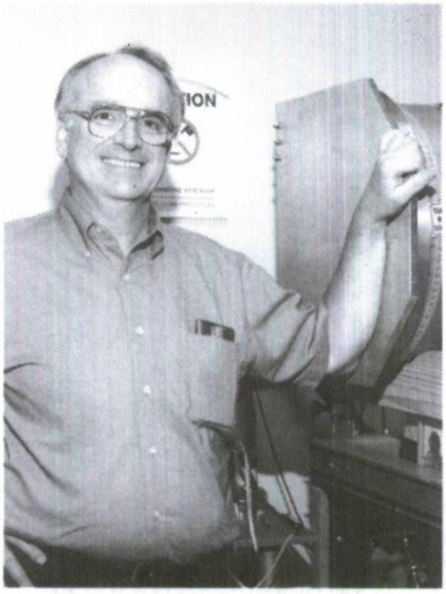
Shu T. Lai
Air Force Research Laboratory
Space Vehicles Directorate
Hanscom AFB., MA, USA

Good morning. Welcomc to this Conference. I would like to thank the leadership of the 9th Spacecraft Charging Technology Conference for inviting me to give a sponsor's speech at the opening ceremony. The Air Force Research Laboratory's (AFRL) Space Vehicles Directorate is plcased to offer its support to the leadership in hosting the 9th Spacecraft Charging Technology Conference in Tsukuba, Japan, in April 2005. AFRL has a long history of co-sponsoring this important topical conference and we look forward to this tradition. This conference is the 9th in thc series. AFRL has been a sponsor or co-sponsor of each of the Conferences in the series.

The first one was sponsored by AFRL and NASA and was held at the US Air Force Academy, Colorado Springs, 1978. Mr. Charlie Pike was the AFRL person organizing and chairing the first Conference. In fact, the first four conferences were all held at the US Air Force Academy in Colorado. The fifth one was held in Naval Postgraduate School, Monterey, CA. It was organized by Dr. Chris Olsen. The sixth one was at Hanscom AFB., Massachusetts, and was organized mainly by Dr. David Cooke and also myself. We always appreciated the strong attendance by our international colleagues. The European Space Agency Technology Centre's (ESTEC) sponsorship of the 7th Conference, held in Holland, set a significant milestone in the history of the conference series making it a truly international event. Dr. Alain Hilgers was the organizer of the Conference in Holland. The 8th Conference was held at NASA Marshall Space Flight Center, Huntsville, Alabama. It was organized by Mr. Jody Minor. Each one of the Conferences in the series has been a great success. The topical discussions have greatly helped in making progress in the fields of spacecraft charging, spacecraft interactions, and related areas.

Now comes the 9th Spacecraft Charging Technology Conference. This one is hosted by Japan Aerospace Exploration Agency (JAXA) of Japan, making it again a truly international event. Dr. Tateo Goka is the chair of the organizing committee, as you know. AFRL is pleased to offer its support to this Conference. We have been budgeting funds specifically for the travel of AFRL personnel and contractors. AFRL would like to publish the conferencc proccedings as an AFRL Technical Report. AFRL Technical Reports are not copyrighted, so there should be no problem with duplicate publication by JAXA. In addition, we have identified a program at AFRL/AOARD's Tokyo office for providing financial support to the Confercnce. We realize that the support is not much but we hope that our pledge of support has aided the Conference in securing the highest level of JAXA endorsement for this activity. In conclusion, AFRL wishcs this Conference a great success.

In Memoriam
Arthur Robb Frederickson



A. Robb Frederickson, known as "Robb," died of pancreatic cancer at Huntington Memorial Hospital, Pasadena, California, on April 5, 2004. A Senior Member of IEEE, Robb served as Chairman of the Boston Section IEEE Nuclear and Plasma Sciences Chapter for 10 years and as Secretary for 5 years. In addition, Robb was Treasurer of the 1974 IEEE Radiation Effects Conference.

Robb grew up in Berkeley Heights and Westfield, New Jersey, the son of the former Arthur R. and Bertine B. Frederickson. At Rensselaer Polytechnic Institute he received his B.S. in Physics in 1965. Robb received his MS and PhD degrees in physics from the University of Massachusetts at Lowell. His dissertation, "Recombination-enhanced Defect Diffusion in Semiconductor Devices, Carbon in Silicon," foreshadowed his wide-ranging studies in microelectronic devices and their interactions with the space environment.

After 31 years at Air Force Cambridge Research Laboratory and its reincarnations (AF Rome Air Development Center, AF Geophysics Lab, AF Philips Lab, and AF Research Lab) at Hanscom AFB, Massachusetts, Robb moved to Pasadena, CA, in 1997 when he joined Caltech's Jet Propulsion Laboratory.

He had over 71 publications, presented innumerable talks at conferences worldwide, was granted 5 patents, and authored a well-received book on radiation effects on dielectrics. He capped his Air Force career as Principal Investigator of the CRRES satellite Internal Discharge Monitor. This experiment proved to be a landmark in the study of spacecraft charging and gained him international fame for the seminal findings on the characteristics of internal spacecraft discharges that came from it.

Representative of the way his many friends and colleagues felt about him, one of them recently summarized his contributions: "He demonstrated great physical originality and brought a critical perspective to our spacecraft charging work," and that was merely his first career.

In his second career in Pasadena, he became a recognized expert at JPL and Caltech in spacecraft charging effects. Creating his own plasma physics laboratory, he studied the effects of plasma and space radiation on spacecraft materials and characterized the space radiation belts. Robb consulted for many JPL and commercial spacecraft programs about the effects of plasma and radiation, and helped develop optimum spacecraft designs to avoid the deleterious effects of the space environment. In addition, he took a lead role in training young engineers, helping to develop the next generation of JPL engineers and scientists. His research results, through review papers, scientific meetings, and publications, have permanently secured his place as an international expert in spacecraft design.

Robb received the Air Force Systems Command Technical Achievement Award in 1969. He is listed in *Who's Who in Science and Engineering* and in *Who's Who in America* (Marquis). Robb served the Hanscom Chapter of Sigma Xi as Treasurer and Secretary. In Massachusetts, Robb was active in the Carlisle Boy Scout Troup and in Colonial Wireless. Other memberships include the American Physical Society, ARRL and both the Caltech and JPL Ham Radio Clubs.

Robb leaves his wife of 33 years, Christine, his sons Timothy and Nathan, a daughter Julie, a daughter-in-law Tracey, and a granddaughter Olivia, a great joy of his last year. In addition to his immediate family, he is survived by a brother Alan of New Jersey, four nieces and a nephew, two aunts and many cousins—all on the East Coast.

HENRY GARRETT
Jet Propulsion Laboratory

9th Spacecraft Charging Technology Conference

Schedule of Events

Sunday, April 3rd

15:30–17:00 Pre–Registration at Tsukuba International Congress Center 2F

Convention Hall 200

17:00–18:30 Welcome Reception at Restraint Espoir

Tsukuba International Congress Center 1F

Monday, April 4th

7:30–8:30 Registration

8:30–8:50 Opening Remarks

Special Invited Talks

8:50 Hitoshi Kuninaka – ISAS/JAXA
Spacecraft Design and Flight Status of HAYABUSA – Asteroid Explorer Propelled by Microwave Discharge Ion Engines (051kun)

9:35 Tatsuo Takada – Musashi Institute of Technology
Pulse Acoustic Technology for Measurement of Charge Distribution in Dielectric Materials for Spacecraft (114tak)

10:20 **Break (20 minutes)**

10:40 Tateo Goka – Japan Aerospace Exploration Agency (JAXA)
JAXA Space Environment Measurement –Overview & Plan– (115gok)

10:55 Mitsushige Oda – Japan Aerospace Exploration Agency (JAXA)
Solar Power Satellite and Electrical Discharge (116oda)

11:40 Masahito Tagawa – Kobe University
Atomic Oxygen-Induced Erosion of Spacecraft Materials -Potential Hazards for Spacecraft Systems- (093tag)

12:25 **Lunch (1 Hour 25 minutes)**

Session 1: Spacecraft Plasma Interaction and Ground Test

13:50 Dale Ferguson – NASA/MSFC
Arcing in LEO – Does the Whole Array Discharge? (016fer)

14:15 Hirokazu Tahara – Osaka University
Ground-Based Experiment of Electric Breakdown of Spacecraft Insulator Surface in Ambient Plasma Environment (001tah)

14:40 Leon Levy – ONERA
Arc Propagation on Space Power Transfer Systems: A First Approach Study (048lev)

15:05 Henry Brandhorst – Auburn University
Hypervelocity Impact to Solar Cells in a Plasma Environment (056bra)

15:30 David Rodgers – QinetiQ
Development of the Spacecraft Plasma Interaction Guidelines and Handbook (106rod)

15:55 **Poster Session 1**

Takuya Harada – Shibaura Institute of Technology
Evaluation of Irradiation Effects on Silica Glass by Electron-Beam (031har)

Takashi Kawasaki – Kyushu Institute of Technology
ESD Test of a Large Solar Array Coupon in GEO Plasma Environment (035kaw)

Kenta Mitsuhashi – Musashi Institute of Technology
Development of Optical Measurement System for Internal Charge Distribution in Insulating Materials (066mit)

Junya Taima – Musashi Institute of Technology
Observation of Internal Charge Behavior of E–Beam Irradiated Polymers Used on Spacecrafts During Elevating Temperature (067tai)

Fuyuko Fukuyoshi – Musashi Institute of Technology
Observation of Internal Charge Behavior in Electron Beam Irradiated Polymers Using Acoustic Method (068fuk)

Kazuhiro Toyoda – Chiba University
Proposal of a Current Regulative Diode for Power Supply in Sustained Arc Test (070toy)

Nelson W. Green – Jet Propulsion Laboratory
Charge Storage Measuremens of Resistivity for Dielectric Samples from the CRRES Internal Discharge Monitor (073gre)

Haruhisa Fujii – Nara National College of Technology
ESD–Triggered Arc Discharges on Solar Array by Electron–Beam Irradiation (083fuj)

Haruhisa Fujii – Nara National College of Technology
Basic Experiment for Vacuum Arc Discharge Triggered by ESD (084fuj)

Brian Wells – Auburn University
Cryogenic and Elevated Temperature Hypervelocity Impact Facility (057wel)

Tadahiro Shimoda – Kyoto University
Asymmetry of Photoelectron Distribution and its Dependence on Spacecraft Potential Obtained from GEOTAIL Data (089shi)

Boris Vayner – Ohio Aerospace Institute
Thin Film Solar Array Samples in Simulated LEO Environment (006vay)

DELU QU – Kyushu Institute of Technology
Laboratory Test of Arcing on Satellite Power Cables (027-qu)

Jeongho Kim – Kyushu Institute of Technology
Threshold Conditions to Induce the Sustained Arc on the Solar Array Panel of LEO Satellite (041kim)

Keigo Ishisaka – Toyama Prefectural University
Relationship Between the Geotail Spacecraft Potential and the Electron Density in the Near Tail Regions (030ish)

Myron Mandell – Science Applications International Corporation
NASCAP-2K Simulations of VLF Plasma Antenna (076man)

Teppei Okumura – Kyusyu Institute of Technology
Degradation of Solar Cell Electric Performance Due to Arcing in LEO Plasma Environment (028oku)

Yasumasa Kasaba – ISAS/JAXA
Evaluation of DC Electric Field Measurement by GEOTAIL Observations & Simulations (040kas)

Yuri Ruzhin – Russian Academy of Science
The Spacecraft Charging by the Electron Beam Injection at Ionosphere (022ruz)

Minoru Iwasa – Tokyo Institute of Technology
Study of the Plasma Interference with High Voltage Electrode Array for Space Power Application (095iwa)

Denis Payan – CNES
Charging of Coaxial Lines with Floating Core at Geosynchronous Altitudes. (107pay)

Hironori Maejima – Japan Aerospace Exploration Agency
On-Orbit Power Fluctuations of ADEOS-II (108mae)

Benoit Thiebault – ESA (098thi)
Potential Barrier in the Electrostatic Sheath Around a Spacecraft

Benoit Thiebault – ESA
Simulation of the Cluster floating probe signal (099thi)

Noriyoshi Onodera – Mitsubishi Electric Corporation (109ono)
Robust Design of Satellite Systems Against Spacecraft Charging

Satoshi Hosoda – Kyushu Institute of Technology (110hos)
Development of 400V Solar Array Technology for LEO Spacecraft

Yoshio Shikata Kyushu Institute of Technology
Optical Spectral Analysis of ARC Plasma on Solar Array in GEO Plasma Environment (111shi)

Evgeny Nikolskiy – Russian Federal Space Agency Lavochkin Association
The Analysis of Spacecraft – "Arcon-1" Charging Under Data of Flight Measurings and Mathematical Simulation (081nik)

Gary Galica – Physical Sciences Inc..
Energetic Charged Particle Spectrometer for the Space Environment Reliability Verification Integrated System (SERVIS-1) Satellite (087gal)

Vitaly Shpakovsky – Russian Academy of Science
The Dynamics of the Spacecraft Potential During Electron Beam Injection (023shp)

Benoit Thiebault – ESA
Measurements of SMART-1 Plasma Environment (100thi)

John Alred – Boeing
Time to Effect of Plasma-Induced Arcing on ISS Anodized Aluminum Surfaces (121alr)

Tuesday, April 5th

Session 2:

Ground Testing Techniques and International Standardization

8:30	Emmanuel Amorim – CNES *Electrostatic Discharges on a $1M^2$ Solar Array Coupon – Influence of the Energy Stored on Coverglass on Flashover Current* (011amo)
8:55	Ludovic Gaillot – EADS-ASTRIUM *Secondary Arcs on Solar Generators – Emags2 Test Campaign* (062gai)
9:20	Boris Vayner – Ohio Aerospace Institute *Large Solar Array in Dense Plume Plasma: Ground Test* (007vay)
9:45	Dale Ferguson – NASA Marshall *NASA GRC and MSFC Space-Plasma Arc Testing Procedures* (112fer)
10:10	**Break (20 minutes)**
10:30	Claude Berthou – Alcatel Space *Plasma ESD qualification test procedure of Alcatel Space Solar Array* (045ber)
10:55	Mengu Cho – Kyushu Institute of Technology *Japanese Practices of Solar array ESD Ground Tests* (053cho)
11:20	Henry Garrett – The Jet Propulsion Laboratory *Towards a New Surface and Internal Charging Design Guideline for the 21st Century* (013gar)
11:45	Denis Payan – CNES *Solar Array Test Set-Up Proposed in the Frame of European Standardization.* (019pay)
12:10	**Excursion / Lunch on Bus**

Wednesday, April 6th

8:30 **Round Table Discussion on ESD Test Standard**

10:10 **Break (20 minutes)**

Session 3:

Spacecraft Plasma Interaction and Environment Specification

10:30 Shu Lai – Air Force Research Laboratory
Why do Spacecrafts Charge in Sunlight? Differential Charging and Surface Condition (009lai)

10:55 Joseph Minow – NASA/Marshall Space Flight Center
Radiation and Internal Charging Environment's for Thin Dielectrics in Interplanetary Space (021min)

Session 4: Modeling, Database and Numerical Simulation

11 :20 Jean–Francois Roussel – ONERA
Spacecraft Plasma Interaction Software (SPIS): Numerical Solvers Methods and Architecture (042rou)

11:45 Julien Forest – Artenum Company
SPIS–UI, a New Integrated Modeling Environment (IME) for Space Environment Simulation Codes. (097for)

12:10 Benoit Thiebault – ESA
Tests and Validation of a New Spacecraft Plasma Interaction Software, SPIS (101thi)

12:35 **Lunch (1 Hour 25 minutes)**

14:00 Hideyuki Usui – Kyoto University
Development of Geospace Environment Simulator (061usu)

14:25 Shinji Hatta – Kyushu Institute of Technology

Multi-utility Spacecraft Charging Analysis Tool (MUSCAT): Development Overview (054hat)

14:50 Sebastien Clerc – Alcatel Space

3D-Simulation of the Satellite Charge in Magnetosphere Environment (005cle)

15:15 Myron Mandell – Science Applications International Corporation

Nascap-2k Spacecraft Charging Code Overview (075man)

15:40 Joseph Wang – Virginia Polytechnic Institute & State University

A First-Principle Based Virtual Testbed for Spacecraft (082wan)

16:05 **Poster Session 2**

Denis Payan – CNES

Electrostatic Behaviour of Dielectrics Under Geo-Like Charging Space Environment Simulated in Laboratory. (020pay)

John Dennison – Utah State University

Payload to Investigate Their Effects on Electron Emission and Resistivity of Spacecraft Materials (079den)

Yokota Kumiko – Kobe University

Atomic Oxygen-Induced Erosion of Polymeric Materials Under Surface Charging Condition (094yok)

Sebastien Clerc – ALCATEL SPACE

The Importance of Accurate Computation of Secondary Electron Emission for Modeling Spacecraft Charging (025cle)

Adrian Wheelock – AFRL/VSBX

Simulations of Current Coupling in Ion Beam-Neutralizer Interactions (090whe)

Crispel Pierre – CNES

Secondary Arc Modelisation on Satellite Solar Generators (085pie)

Koji Horikawa – Osaka University
Numerical Calculation of Ablation and Plasma Expansion Induced by Electric Breakdown of Spacecraft Insulator Surface in Ambient Plasma Environment (002hor)

Linda Parker – Jacobs Sverdrup/MSFC Group
Analysis of Surface Charging for a Candidate Solar Sail Mission Using NASCAP-2K (071par)

Masaki Okada – ROIS/NIPR
Development of Unstructured-Grid EM Particle Code for the Spacecraft Environment Analysis (059oka)

Rikio Watanabe – Musashi Institute of Technology
1D Monte-Carlo Simulation of Charge Accumulation Process Inside Teflon film (038wat)

Sachio Akebono – Kyushu Institute of Technology
Circuit Analysis of Effects of Solar Array Arcing on Spacecraft Power System (034ake)

Sebastien Clerc – ALCATEL SPACE (026cle)
Validation of Daylight Charging Capabilities of the Sparcs Code

Valery Mileev – Moscow State University (043mil)
Modeling of Spacecraft Dielectric Materials Internal Charging

Hirokazu Tahara – Osaka University
Ground-Based Experiment of Electron Collection by an Electrodynamic Bare Tether (003tah)

Hirokazu Tahara – Osaka University
Plasma Plume Characteristics of Electric Thrusters (004tah)

Juan Sanmartin – Universidad Politecnica de Madrid
Performance of Coupled ED-Tether / Ion Thruster System (064san)

Kengo Yanagi – Ibaraki University
Effect of Magnetic Fields on Sustainment and Dynamics of Low Current DC Vacuum Arcs (029yan)

Myron Mandell – Science Applications International Corporation
Ion Engine Plume Interaction Calculations for Prometheus I (077man)

Satomi Kawamoto – Japan Aerospace Exploration Agency
Electrodynamic Tether Systems for Debris Removal (036kaw)

Yasushi Okawa – Japan Aerospace Exploration Agency
Preliminary Testing of Carbon-Nanotube Field Emission Cathodes for Electro Dynamic Tethers (039oka)

Boxue Du – Tianjin University
Discharge Characteristic of Gamma-Ray Irradiated Polybutylene Naphthalate (033-du)

Shana Figueroa – Air Force Research Laboratory (092fig)
Ion Scattering in a Self-Consistent Cylindrical Plasma Sheath

Sebastien Jourdain – Artenum Company
LibreSource: A New Virtual Lab for the SPINE Community (105jou)

Takayuki Harano – Kyushu Institute of Technology
Preliminary Study on Sustained Arc due to Plasma Excited by Debris Impact on the Solar Array Coupon (017har)

Ikkoh Funaki – Japan Aerospace Exploration Agency
Scaled-Down Experiment of Spacecraft Charging with Artificial Plasma Emission (117fun)

Hidenori Kojima – University of Tsukuba
Experimental Study of Magnetic Sails (118koj)

Brandon Reddell – Boeing
Probabilistic Analysis of ISS Plasma Interaction (119red)

Brandon Reddell – Boeing
Floating Potential Measurement Unit Langmuir Probe Testing and Data Reduction Techniques for ISS Applications (120red)

Aroh Barjatya – Utah Space University
Vehicle Charging on a Sounding Rocket Payload (104bar)

Charles Swenson – Utah Space University

The ISS Floating Potential Measurement Unit [(122swe)]

Charles Swenson – Utah Space University

Plasma Impedance Probe Diagnostics: Model and Data [(123swe)]

Richard Briët – The Aerospace Corporation

Optimization of Spacecraft Charging Mitigation Requirements [(124bri)]

Richard Briët – The Aerospace Corporation

Scaling Laws for Pulse Waveforms from Surface Discharges [(125bri)]

Thursday, April 7th

Session 5: Plasma Propulsion and Tethers

8:30 Erik Engwall – Uppsala University
Cold magnetospheric Plasma Flows and Spacecraft Wakes:PicUp3D Simulations and Cluster Data (049eng)

8:55 Juan Sanmartin – Universidad Politecnica de Madrid
Spherical Collector Versus Bare Tether for Drag, Thrust, and Power Generation (063san)

9:20 George Khazanov – NASA/MSFC (014kha)
Analysis of Bare-Tether Systems as a Thruster for MXER Studies

9:45 Hironori FUJII – Tokyo Metropolitan University (055fuj)
A Proposed Bare-Tether Experiment on Board a Sounding Rocket

10:10 **Break (20 minutes)**

Session 6: Satellite On-orbit Investigations

10:30 Gary Galica – Physical Sciences Inc.
Scintillator-Based Low Energy Particle Imaging (086gal)

10:55 Evgeny Nikolskiy – Russian Federal Spase Agency Lavochkin Association (080nik)
S/c "Cosmos - 2393" Charging Under Data of Flight Measurings

11:20 Joseph Fennell – Aerospace Corp
HEO Satellite Surface Charging in1995-2002 (058fen)

11:45 Klaus Torkar – IWF/OAW
Active Spacecraft Potential Control: Results from the Double Star Project (018tor)

12:10 **Lunch (1 Hour 20 minutes)**

13:30 Lev Novikov – Moscow State University
Analysis of On-Board Magnetosphere Plasma Data and Geosynchronous Spacecraft Charging (044nov)

13:55 S.V.K. Shastry – ISRO Satellite Centre
On-orbit Data from Surface Charge Monitor Payload of GSAT-2 Spacecraft (010svk)

14:20 James Roeder – The Aerospace Corp.
Differential Charging of Satellite Surface Materials (065roe)

14:45 Anders Eriksson – Swedish Institute of Space Physics
Charging of Conductive Spacecraft in the Auroral Zone (103eri)

15:10 **Break (20 minutes)**

Session 7: Charging of Polar Orbiting Satellites

15:30 Phillip Anderson – University of Texas at Dallas
Spacecraft Charging Hazards in Low Earth Orbit (091and)

15:55 Masao Nakamura – National Institute of Information and Communications Technology
Space Plasma Environment at the ADEOS-II Anomaly (078nak)

16:20 Shirou Kawakita – Japan Aerospace Exploration Agency
Investigation of Operational Anomaly of ADEOS-II Satellite (060kaw)

16:45 Takanori Iwata – Japan Aerospace Exploration Agency
Solar Array Paddle for the Advanced Land Observing Satellite (ALOS): Charging Mitigation and Verification (113iwa)

17:10 Mengu Cho – Kyushu Institute of Technology
ESD Tests of Solar Array Paddle on a Polar Orbiting Satellite (052cho)

Friday, April 8th

Session 8: Numerical Simulation of Spacecraft Interaction with Thruster Plume and Solar Wind

8:30 Iain Boyd – University of Michigan
Numerical Simulation of Hall Thruster Plasma Plumes (088boy)

8:55 Alain Hilgers – ESA
Modelling of SMART-1 Interaction with the Electric Thruster Plume (102hil)

9:20 Victoria Davis – Science Applications International Corporation
Ionospheric Currents to the Special Sensor Ultraviolet Limb Imager on DMSP (069dav)

9:45 Gennady Markelov – AOES BV
Modifications of SPIS Software and Modeling of Plasma Flow Around SMART-1 (096mar)

10:10 **Break (20 minutes)**

Session 9: Material Characterization

10:30 John Dennison – Utah State University
Evolution of the Electron Yield Curves of Insulators (072den)

10:55 John Dennison – Utah State University
Proposed Modifications to Engineering Design Guidelines Related to Resistivity Measurements and Spacecraft Charging (074den)

11 :20 Virginie Griseri – Laboratoire de Génie Electrique de Toulouse
Space Charge Detection and Behaviour Analysis in Electron Irradiated Polymers (037gri)

11:45	**Presentation of 10th SCTC**
11:55	**Closing Remarks**
12:00	**Lunch (1 Hour 35 minutes)**
14:00	**Tour of Tsukuba Space Center**

Invited Talk

Invited Talk

Notes :

SPACECRAFT DESIGN and FLIGHT STATUS of HAYABUSA・ASTEROID EXPLORER PROPELLED by MICROWAVE DISCHARGE ION ENGINES

Hitoshi Kuninaka

ISAS/JAXA

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E-mail : kuninaka@isas.jaxa.jp

At May 9th, 2003 the MUSES-C asteroid explorers was launched directly into the deep space by the M-V rocket and named HAYABUSA, which means falcon in Japanese. It was just a moment that the microwave discharge ion engines were space-borne after the 15-year research and development. The Ion Engine System on HAYABUSA explorer was designed and assembled using four ion engines, of which main feature is the plasma production by microwave without solid electrodes as life critical components. The spacecraft onboard the ion engines was also designed and tested on the ground so as to adapt the high velocity plasma beams. The HAYABUSA explorer aims to bring back surface material of an asteroid to Earth. The vacuum exposure during two weeks after the launch and various kinds of test runs in a month enabled the ion engines to accelerate HAYABUSA asteroid explorer continuously at a several meters per second in a day from July 2003. In the first year HAYABUSA has stayed near Earth on the 1year Earth synchronous orbit, on which it has stored delta-V generated by the ion engines. On May 19, 2004 HAYABUSA passed by 3,000km above the Pacific Ocean and changed its orbit toward the asteroid. At the end of November 2004 the total accumulated operational time exceeds 19,000 hour and unit with propellant consumption 19kg and resultant delta V 1,200m/s. This paper will reports the flight status of HAYABUSA spacecraft and spacecraft design and assemble methods for the microwave discharge ion engines.

Notes :

PULSE ACOUSTIC TECHNOLOGY for MEASUREMENT of CHARGE DISTRIBUTION in DIELECTRIC MATERIALS for SPACECRAFT

Tatsuo Takada

Musashi Institute of Technology

Pulsed Electro-Acoustic (PEA) method for measuring the space charge distribution in dielectrics has applied to the basic investigation of spacecraft charging studies. The PEA method was developed in the electric power engineering field. People in a field of high voltage engineering at 1970's had wanted to observe an electric charge distribution in insulation layer of high voltage dc submarine cable because the electrical breakdown of the dc cable was occurred by an electric field distortion caused by charge accumulation in insulation layer. In 1980's we started to developing some new technologies using laser-pulse irradiation, nano-second acoustic pulse generation and elliptical polarized light modulation etc., to observe the electric charge distribution in dielectric materials. Now theses technologies can be applied to the engineering field such as a spacecraft charging, a high energy charge particle irradiation, a photo-conduction materials of copy machine and an integrated electronic circuit board etc..

We have developed two methods for measuring the space charge distribution; one is Pulsed Electro-Acoustic (PEA) method, and another is Pressure Wave Propagation (PWP) method. The PEA method is widely used to measure the space charge distribution because of its simple operation and low electrical noise etc.. We have used the PEA method to measure the space charge distribution in test sample, such as bulk plastic materials of 1.0 to 10 mm in thickness, sheet polymeric materials of 50 to 500 mm in thickness and plate glass materials of 0.5 to 2.0 mm in thickness etc.. The dynamic electric charge behavior under application of dc voltage could be observed for short interval of 10 ms to over 1 hour. Then we can observe the dynamic eclectic charge behavior in the insulating materials just before electrical breakdown events.

In order to observe the space charge accumulation in dielectric materials under condition of space environment, such as an electron beam irradiation (0.5 to 2.0 MeV), a proton beam irradiation (1.0 to 6.0 MeV) and a gamma ray irradiation (0.5 to 2.0 MeV), the PEA method was applied to the basic investigation of spacecraft charging studies. In this report, I will present you about the typical measurement results including new obtained data.

Notes :

JAXA SPACE ENVIRONMENT MEASUREMENT
-OVERVIEW & PLAN-

Tateo Goka

Japan Aerospace Exploration Agency (JAXA)

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In the inner magnetosphere, studies of the flux variability (short- and long-term) of radiation belt particles are particularly important not only for improving our understanding of the relevant phenomena associated with them, but also for engineering considerations viz. a viz. spacecraft anomalies due to space environment effects (Electro-Static Discharge: ESD, and Single Event Upset: SEU). Therefore, ongoing radiation measurements and monitoring by satellites is a requirement. For this purpose, we have carried out TEDA (Technical Data Acquisition Equipment) and SEDA (Space Environment Data Acquisition Equipment) mission in JAXA. The current status of radiation measurements using JAXA satellites is reviewed. Also, electrostatic charge potential measured by Potential Monitors (POM)onboard 3 satellites (ETS-V for 10 years in GEO orbit, ETS-VI in GTO orbit, ADEOS in Polar LEO orbit) is reviewed. I will report space environment measurement plan and spacecraft charging technology research plan (in-situ measurement including POM onboard ETS-VIII in GEO orbit and simulation).

Notes :

SOLAR POWER SATELLITE and ELECTRICAL DISCHARGE

Mitsushige Oda
Japan Aerospace Exploration Agency (JAXA)
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The space solar power satellite (SPS, SSPS) are being considered for many years as an alternative energy source of future when the fossil fuel will lost its position as an economical and easy-to-get energy source. There are several types of SSPS. The most popular type of SSPS is a system that collects solar energy in space using the photovoltaic cells, and sends the collected energy to the ground by microwave.
In this system, electrical energy generated by the solar cells must be sent to the microwave transmitters using some kind of electrical cables. In designing SPS, mass of this electrical cable can not be neglected since total length of the power transmission cables between the photovoltaic cells and the power transmission devices is long. If we want to reduce mass of the cables, we must use thin cables. However this means loss by the resistance will be large and the loss will become heat. If we will use high voltage to reduce the resistance loss, we have to face the electrical discharge. Therefore the electrical discharge problem will have big impact on design of SSPS.

Notes :

ATOMIC OXYGEN-INDUCED EROSION of SPACECRAFT MATERIALS -POTENTIAL HAZARDS for SPACECRAFT SYSTEMS-

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Space environment is complicated. Many environmental factors influence each other and, as a result, it sometimes provides unexpected effect to spacecraft systems or materials. Charging/discharging phenomena may cause serious problem to a high-voltage spacecraft system. Polymers are used for electrical and thermal insulation in many satellites, and are one of the targeting materials for satellite charging problems. On the other hand, polymers used for such purposes are often used in the unpressurized area of the satellite and suffered atomic oxygen bombardment, which is a dominant atmospheric component of the upper atmosphere of the Earth. It has been widely recognized that many polymeric materials are eroded by atomic oxygen.

In-flight atomic oxygen exposure experiments have been faced not only contamination problems, but also influenced by many other environmental factors such as ultraviolet, temperature, surface charging and so on. These uncertain environmental parameters make the fundamental understanding of the reaction of hyperthermal atomic oxygen with materials difficult. On the other hand, difficulty in acceleration of the electrically neutral atomic oxygen up to the orbital velocity of spacecraft (7800 m/s = 5 eV) in the ground-based experimental facility has been limited an opportunity for laboratory experiment. Only a laser-detonation atomic oxygen source, which is sometimes called "PSI-type source" is capable for forming high-intensity, hyperthermal (5 eV) atomic oxygen beam. Total 7 space-related organization, companies and universities in the world are now operating this type of atomic oxygen source for space applications. However, we still do not have sufficient data to describe how materials react when hyperthermal atomic oxygen hits the spacecraft surfaces at various conditions.

In this presentation, recent knowledge on atomic oxygen-induced erosion of spacecraft materials is surveyed. Experimental results obtained by laser-detonation atomic oxygen beam sources in Kobe University will be presented. The configuration of the experimental facility [1], basic properties of atomic oxygen-induced erosion of polymers [2, 3], and a quantitative study in synergistic effect of atomic oxygen with ultraviolet [4] are the major topics in this presentation. Research results reported by the US or European groups will also be introduced. Potential effects of atomic oxygen-induced erosion of materials to spacecraft charging problems will be mentioned.

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Notes :

Session 1

Spacecraft Plasma Interaction and Ground Test

Session 1

Notes :

ARCING in LEO - DOES the WHOLE ARRAY DISCHARGE?

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The conventional wisdom about solar array arcing in LEO is that only the parts of the solar array that are swept over by the arc-generated plasma front are discharged in the initial arc. This limits the amount of energy that can be discharged. Recent work done at the NASA Glenn Research Center has shown that this idea is mistaken. In fact, the capacitance of the entire solar array may be discharged, which for large arrays leads to very large and possibly debilitating arcs, even if no sustained arc occurs. We present the laboratory work that conclusively demonstrates this fact by using a grounded plate that prevents the arc-plasma front from reaching certain array strings. Finally, we discuss the dependence of arc strength and arc pulse width on the capacitance that is discharged, and provide a physical mechanism for discharge of the entire array, even when parts of the array are not accessible to the arc-plasma front. Mitigation techniques are also presented.

Notes :

GROUND-BASED EXPERIMENT of ELECTRIC BREAKDOWN of SPACECRAFT INSULATOR SURFACE in AMBIENT PLASMA ENVIRONMENT

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In the future, LEO spacecraft will be larger and higher powered. Because of the balance of leakage currents through ambient space plasma, their main conductive body will have a higher negative potential without plasma contactor operation. When spacecraft operate with a higher voltage, more intensive electric breakdown, i.e. arcing, is suspected to occur on the surface. In this study, ground-based experiment was carried out to understand the arcing phenomenon and to examine influences of ambient space plasma on the arcing process. Simulating plasmas were generated by electron cyclotron resonance discharge. When arcing occurred on negatively-biased anodized aluminum sample plates or Kapton films in the plasma environment, the time variations in arc current and bias voltage were measured. Arc spot diameter was also measured. The experimental results showed that both the peak arc current and the total charge emitted by arcing increased with initial charging voltage and neutral particle number density. The diameter of arc spots increased with initial charging voltage although it was almost constant regardless of neutral particle density. Accordingly, high voltage operation of LEO spacecraft might bring drastic degradation of the insulators by arcing, depending on ambient plasma conditions, particularly with high neutral particle density; that is, a high arc current is suspected to flow.

Notes :

ARC PROPAGATION on SPACE POWER TRANSFER SYSTEMS: A FIRST APPROACH STUDY

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Electric arcs observed between solar cells have been identified as the origin of power losses on orbit. Such arcs have been identified as vacuum arcs and have been shown to produce severe damage with functional destruction if the solar array provides sufficient current (1A) under sufficient voltage (20V) to sustain it. In the work presented here, we deal with power transfer systems encountered in equipment like SADM (Solar Array Drive Mechanism) involving many packaged circuits which are designed to carry high current (>5A) under voltages higher than 50 volts. This study is an investigation of vacuum arc propagation from one circuit to another. The experiments were performed on samples made with several metallic tracks and up to three electrical circuits (6 tracks) were used at the same time. The experimental set-up consisted in triggering a first arc between two adjacent tracks while biasing two, three or four more tracks, and to monitor any kind of connection or propagation. The results obtained show that a vacuum arc can propagate from one circuit to another: arcs triggered by propagation between tracks separated by more than 10mm were seen. This propagation is due to the expansion of the arc plasma produced by the first arc towards other tracks. From our experiments we have learned that any kind of connections between the electrical circuits are possible but a vacuum arc would more easily propagate by connecting to one of the tracks where the first arc was triggered. Arc movement on one track has been also observed along several centimetres. The damages produced by such arcs were heavy and destructive: complete vaporization of anode tracks was observed. On cathode tracks, only partial melting and vaporization due to cathode spots of vacuum arcs were seen.

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HYPERVELOCITY IMPACT to SOLAR CELLS in a PLASMA ENVIRONMENT

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As satellite power levels of advanced spacecraft climb above 20 - 50 kW, higher solar array operating voltages become attractive. Even in today's satellites, operating spacecraft buses at 100 V and above has led to arcing, so the issue of spacecraft charging remains a design problem. In addition, micrometeoroid impacts on all of these arrays may also lead to arcing if the spacecraft is at an elevated potential. For example, tests on space station hardware disclosed arcing at 50V on anodized Al structures that were struck with hypervelocity particles in Low Earth Orbit (LEO) plasmas. Thus an understanding of these effects is necessary to design reliable high voltage solar arrays of the future.

Most solar arrays arc in a space plasma environment (sometimes catastrophically) at voltages below 200 to 300V. Thus there is an absence of reliable information on which to base high power satellite designs. The effect of high velocity micrometeoroids on future high voltage arrays is one of the unknowns for spacecraft design. Existing NASCAP-GEO models can provide guidance about solar cell string design, encapsulation approaches and field control for the expected environment, but the effects of micrometeoroid penetration in that environment are not included.

To assess these effects, contemporary GaAs modules and samples of the Stretched Lens Array (SLA) modules were exposed to hypervelocity impact with 100 μm diameter soda lime glass spheres at velocities up to 12 km/sec. The two strings of GaAs cells were held at differential voltages above 60V at absolute voltages near 200 V. A plasma environment typical of Geosynchronous Earth Orbit was also present. The SLA was held at voltages up to 1000V. Several types of arcs were observed in the GaAs strings. Peak bias voltages were limited by cover glass overhang, but in the case of the SLA no arcing occurred.

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DEVELOPMENT of the SPACECRAFT PLASMA INTERACTION GUIDELINES and HANDBOOK

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As part of ESA's programme to develop European capabilities in the field of spacecraft plasma interactions, a project is under way to develop a repository of useful information for spacecraft plasma interactions. This is intended to provide an easily accessible source of the important design requirements and guidelines relating to spacecraft charging and other spacecraft plasma interactions. In addition, it collects together useful equations and tabular information from the literature. These are also presented as figures as much as possible, to make their meaning clear. One goal is to show the ways in which requirements can be satisfied for common circumstances. The document also directs readers to relevant computational tools and laboratory facilities, so that more detailed investigations can be carried out. Through this it aims to promote the use and sharing of resources in Europe.

The contents of the handbook include: general plasma parameters, spacecraft sheaths and wakes, tethers and electric propulsion, surface charging, internal charging and solar array effects.

The Spacecraft Plasma Interaction Guidelines and Handbook will exist in two forms; a hardcopy text and graphics version and an interactive electronic version. The content of both versions will be the same except that the equations and figures of the text version will be able to be solved and dynamically generated in the interactive version. The emphasis is the rapid calculation of simple formulae.

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Session 2

Ground Testing Techniques and International Standardization

Session 2

Notes :

ELECTROSTATIC DISCHARGES on a 1M² SOLAR ARRAY COUPON – INFLUENCE of the ENERGY STORED on COVERGLASS on FLASHOVER CURRENT

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Satellites on a geostationary orbit interact with the space environment: while the satellite body is negatively charged, dielectrics build up a differential charge. The differential voltage created is called inverted voltage gradient situation and can trigger electrostatic discharges (ESD) on solar arrays that can degenerate in some cases into destructive secondary arcs between adjacent cells. Since power losses have been produced by such phenomena, realistic ground tests of ESD on solar arrays are a main concern. In this paper we present experiments made on a large solar array coupon as part of a CNES study performed at ONERA in order to set a European standard in ground tests (ECSS E20-06). This work is associated with a Japanese study and aims to set the basis of an ISO standard for ESD and secondary arcs. The solar array coupon is constituted by 19 strings of 16 silicon solar cells and its size is 592x1333 mm. During experiments, the coupon was biased at 5000V and the inverted gradient potential situation was created by secondary electron emission produced by an electron beam striking the solar array surface. Blow-off and flashover currents were measured, and surface potential measurements were performed before and after ESD. Influence of the external capacitance value used during experiments was also studied. The results obtained show clear correlation between the discharge duration, the energy stored on cover glass and the coupon size. Potential distribution measurements performed after ESD show partial neutralization on a limited surface where the differential voltage between the cover glass surface and the coupon decreased from +1000V to +150V. Neutralization all over the coupon was also observed for some discharges.

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SECONDARY ARCS on SOLAR GENERATORS - EMAGS 2 TEST CAMPAIGN

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EMAGS 2 is the continued effort from ESA to better understand the Solar Array Triggered Arc Phenomena and to define how the tests should be performed.

It has involved one industrial and one laboratory worldwide known for his expertise in solar array ESD testing.

An extensive test campaign on solar array samples has been performed from June 2003 to June 2004.

The first objective of EMAGS 2 was to compare the characteristics of the secondary arcs obtained in an electronic charging environment with those triggered in a plasma environment (inverted voltage gradient conditions in both cases) in order to assess their equivalence or differences.

Indeed, a test in plasma (contrary to a test under electronic irradiation) allows to trigger more easily primary and secondary discharges but is not automatically representative of an electron charging environment.

Comparative tests have been performed on dummy samples (Cu) with similar test set-ups except for the bias voltage of the coupon (-500/1000V in plasma instead of -5kV in electron), and so, for the capacitance simulating the satellite one (3 different capacitances tested in plasma).

These tests allowed to define a set-up in plasma providing results representative of those obtained in electron.

Considering these conclusions, the three next series of tests have been performed on solar array samples in plasma.

- Comparison GaAs / Si: An important issue of this study was to assess the voltage / current thresholds for secondary arc triggering according to the cell type (GaAs triple junction or Si).
- Variable gap study: The impact of the gap value on the ESD forming has been intensively studied with several silicon samples with gap between adjacent cells varying from 0.5 mm, 0.9 mm, 2 mm.
- Cumulative effects: cumulative effects of repetitive primary and secondary (aborted) discharges have been investigated on GaAs samples.

Notes :

LARGE SOLAR ARRAY in DENSE PLUME PLASMA: GROUND TEST

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Large solar array (2m x 4 m) was tested in arc jet plume plasma environment. Test was conducted in voluminous vacuum chamber (12 m diameter and 19 m height) that prevented the back reaction of chamber walls on plasma distribution. Vacuum equipment (ten diffusion pumps) provided background pressure about 1 μTorr. Plume plasma diagnostics was performed by using spherical and cylindrical Lamgmuir probes, emissive probes, and Faraday cup. Optical spectra of hot plasma near the nozzle were measured and analyzed. Contamination of solar array surfaces was determined by employing two QCM probes. Chemical composition of plasma was monitored by quadrupole mass spectrometer. Current collections and arc thresholds were measured by biasing array with an external power supply. Acquired data were used to make important conclusions concerning the influence of plume plasma on solar array operation.

Notes :

NASA GRC and MSFC SPACE-PLASMA ARC TESTING PROCEDURES

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Tests of arcing and current collection in simulated space plasma conditions have been performed at the NASA Glenn Research Center (GRC) in Cleveland, Ohio, for over 30 years and at the Marshall Space flight Center (MSFC) for almost as long. During this period, proper test conditions for accurate and meaningful space simulation have been worked out, comparisons with actual space performance in space flight tests and with real operational satellites have been made, and NASA has achieved our own internal standards for test protocols. It is the purpose of this paper to communicate the test conditions, test procedures, and types of analysis used at NASA GRC and MSFC to the space environmental testing community at large, to help with international space-plasma arcing testing standardization. To be discussed are:

1. Neutral pressures, neutral gases, and vacuum chamber sizes.
2. Electron and ion densities, plasma uniformity, sample sizes, and Debye lengths.
3. Biasing samples versus self-generated voltages. Floating samples versus grounded.
4. Power supplies and current limits. Isolation of samples from power supplies during arcs. Arc circuits. Capacitance during biased arc-threshold tests. Capacitance during sustained arcing and damage tests. Arc detection. Preventing sustained discharges during testing.
5. Real array or structure samples versus idealized samples.
6. Validity of LEO tests for GEO samples.
7. Extracting arc threshold information from arc rate versus voltage tests.
8. Snapover and current collection at positive sample bias. Glows at positive bias. Kapton pyrolization.
9. Trigger arc thresholds. Sustained arc thresholds. Paschen discharge during sustained arcing.
10. Testing for Paschen discharge thresholds. Testing for dielectric breakdown thresholds. Testing for tether arcing.
11. Testing in very dense plasmas (ie thruster plumes).
12. Arc mitigation strategies. Charging mitigation strategies. Models.
13. Analysis of test results.

Finally, the necessity of testing will be emphasized, not to the exclusion of modeling, but as part of a

complete strategy for determining when and if arcs will occur, and preventing them from occurring in space.

Notes :

PLASMA ESD QUALIFICATION TEST PROCEDURE of ALCATEL SPACE SOLAR ARRAY

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Plasma ESD qualification test procedure of Alcatel Space Solar Array. The detrimental effects of spacecraft charging are well known, particularly when the charging leads to arcing on solar array and consequently to permanent power loss. Motivated by these concerns, Alcatel Space has undertaken several test campaigns to qualify solar array design and to investigate margins.

This paper describes the state-of-the-art of Alcatel Space solar array ESD laboratory test set-up and procedure. Primary discharges are created using a plasma source and a negative polarisation of the sample rear side. The required power to feed the arc can be delivered by a Solar Array Simulator with low capacitance power sources. The test sample is fully representative of a flight model.

Notes :

JAPANESE PRACTICES of SOLAR ARRAY ESD GROUND TESTS

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As the power level of Geostationary satellites increases, there is more demand of careful ground test on solar array insulation strength. International atmosphere surrounding commercial telecommunication satellites calls for common international standard on test conditions.

In this paper, we review ESD tests of solar array carried out in recent years at Kyushu Institute of Technology jointly with the Japan Aerospace Exploration Agency. The purpose of this paper is to describe the test conditions with the reasons of selecting those conditions and how we certified the insulation strength of given solar array designs against electrostatic discharges in orbit. The paper is aimed to stimulate discussion among experts on the issue of international standardization of solar array ESD test.

Notes :

TOWARDS a NEW SURFACE and INTERNAL CHARGING DESIGN GUIDELINE for the 21st CENTURY

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In 2003, Dr. A. R. Frederickson initiated a program to recast and combine the primary guides for mitigating the effects of spacecraft charging. That effort had the ambitious goal of taking the existing NASA guidelines for preventing surface electrostatic charging, NASA-TP-2361 (Purvis et al., 1984), and internal electrostatic charging, NASA-HDBK 4002 (Whittlesey, 1998), and bringing them up to date with recent laboratory and on-orbit findings. With the death of Dr. Frederickson in 2004, that study devolved to the principle authors to complete. This paper will describe the status of those on-going efforts to combine and update the two guidelines with emphasis on the proposed contents and on the differences and similarities between surface and internal charging mitigation techniques. It is planned to have a draft revision ready for review by the spacecraft charging community by the fall of 2005, with 2006 dedicated to implementing reviewers' comments and additions leading to a new, officially approved NASA guideline by the fall of 2006.

Notes :

SOLAR ARRAY TEST SET-UP PROPOSED in the FRAME of EUROPEAN STANDARDIZATION.

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The validation of new solar cell for geostationary satellite application sets the problem of arcing between solar cells triggered by electrostatic discharges (ESD). The main difficulty of this test is to properly simulate the transient functioning of one string during the first microsecond of an electrostatic discharge and to take into account all the energy stored on a panel. The CNES Solar Simulator with all the physical data we need to take into account to properly simulate de functioning of the string is presented in this paper.

Since the power losses on the Tempo and Panamsat satellites attributed to electrostatic discharges in 1997 (15% of the power in three months), the international scientific community has looked into the problem of sustain arc on solar array.

It was therefore necessary to draw up a discharge theory, with a sequencing of events, by taking into account the energy available at the moment of discharge on the satellite (presented at the 7th SCTC)– as on this energy depends the amplitude of the primary discharge, and therefore the associated thermal effect, which totally conditions the arc.

On the other hand, manufacturers, now encountering problems of electrostatic nature on satellites (section losses on high voltage solar arrays), present the problem in terms of efficient solutions. Manufacturers and laboratories still test solar array samples with their own assembly configurations which, for a single sample, can either be disastrous or have not the slightest effect. Indeed, depending on whether the energy developed in the primary discharge comes from a capacitance of 100pF or one of 1µF, the result as regards the direct effects of the primary discharge and the sustained arc risk will be completely different.

To test solar array samples a priori in the confinement of an enclosed vacuum, it is necessary to define a laboratory test set-up, which represents what the solar array comes across in the geostationary orbit. To finalise this test set-up, we had to know the error made when using a 4 cells sample instead of a real large solar array in terms primary discharge representativity, available energy stored, and electrical response of the simulating circuit.

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Session 3

Spacecraft Plasma Interaction and Environment Specification

Session 3

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WHY DO SPACECRAFTS CHARGE in SUNLIGHT? DIFFERENTIAL CHARGING and SURFACE CONDITION

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Why do spacecrafts charge in sunlight? The first reason concerns differential charging between the sunlit and dark sides. A monopole-dipole model describing the differential charging potential distribution yields interesting theoretical results. We compare the results with observations. The second reason concerns reflectance. Much attention has been paid in recent years to the effect of surface conditions on secondary emission, which plays an essential role in spacecraft charging. In comparison, little or no attention has been paid to the effect of surface condition on photoemission, which plays a dominating role in spacecraft charging. We present theoretical reasoning why highly reflective mirrors generate substantially reduced photoemission. We have calculated, by using the Langmuir orbit-limited current balance equation in 1-D, 2-D, and 3-D, the different surface potentials of various surface materials under typical space plasma conditions, satellite surface reflectivity values, and sunlight incidence angles. We present numerical results confirming that with substantially reduced photoemission, highly reflective surfaces would often charge to high negative potentials in sunlight.

Notes :

RADIATION and INTERNAL CHARGING ENVIRONMENTS for THIN DIELECTRICS in INTERPLANETARY SPACE

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Spacecraft designs using solar sails for propulsion or thin membranes to shade instruments from the sun to achieve cryogenic operating temperatures are being considered for a number of missions in the next decades. A common feature of these designs are thin dielectric materials that will be exposed to the solar wind, solar energetic particle events, and the distant magnetotail plasma environments encountered by spacecraft in orbit about the Earth-Sun L2 point. This paper will discuss the relevant radiation and internal charging environments developed to support spacecraft design for both total dose radiation effects as well as dose rate dependent phenomenon, such as internal charging in the solar wind and distant magnetotail environments. We will describe the development of radiation and internal charging environment models based on nearly a complete solar cycle of Ulysses solar wind plasma measurements over a complete range of heliocentric latitudes and the early years of the Geotail mission where distant magnetotail plasma environments were sampled beyond XGSE = -100 Re to nearly L2 (XGSE ~ -236 Re). Example applications of the environment models are shown to demonstrate the radiation and internal charging environments of thin materials exposed to the interplanetary space plasma environments.

Notes :

Session 4

Modeling, Database and Numerical Simulation

Session 4

Notes :

SPACECRAFT PLASMA INTERACTION SOFTWARE (SPIS): NUMERICAL SOLVERS METHODS and ARCHITECTURE

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The development of a new software for spacecraft plasma interactions modeling was started in Europe at the end of 2002. This Spacecraft Plasma Interaction Software (SPIS) is developed for and by SPINE community (Spacecraft Plasma Interaction Network in Europe) on an open source basis. The ESA contractors, ONERA and Artenum, are in charge of the development of SPIS framework and main numerical modules. In the framework of this collaborative development, SPINE community has several roles. It first expressed its requirements. It is now testing and validating the code on challenging test cases, and is starting to develop extra modules.

The software framework is based on the integration or interfacing with available open source tools for CAD, 2D meshing, 3D meshing, GUI, post-processing and graphical display. The numerical routines allow the modeling of the plasma dynamics (kinetic or fluid, electrostatic with possible extension to electromagnetic) and it's coupling with the spacecraft (equivalent circuit approach). The modeling of all types of environments and devices is (LEO/GEO/PEO..., EP/solar arrays...). The emphasis was put on the modularity of the code to allow the interoperability of the modules, through an object-oriented (OO) approach throughout the code (Java language, Jython script language).

Requirements and design of SPIS code were defined during the first semester of 2003. The first version of the code was released in Spring 2004, and extended versions later in 2004 and 2005. This paper will mostly focus on the numerical core of the solvers and their OO architecture, and will address the following major topics.

Non-linear Poisson equation is solved following an implicit Newton-type method on an unstructured mesh (finite elements). The spacecraft circuit solver automatically generates equivalent components for local coating modeling, and allows the definition of extra device between electric super-nodes (a group of elements, e.g. at solar array level). All material interactions are taken into account following state of the art models, and the code flexibility will allow their improvement following parallel experimental activities. Extra capabilities will be described such as particle transport models, particle sources, etc. The OO architecture of the solvers will eventually be described. It is a key factor for the modularity of such an

open code.

Notes :

SPIS-UI, a NEW INTEGRATED MODELLING ENVIRONMENT (IME) for SPACE ENVIRONMENT SIMULATION CODES.

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In the frame of the SPIS project (see http://www.spis.org), an advanced, customizable and flexible User Interface, SPIS-UI, has been developed. The main requirements of SPIS-UI were, first, to offer to the users an interface the most generic and adaptable possible, able to link all components of the complete modeling chain from the pre to the post-processing phase. In second, SPIS-UI should present many levels of access and control of the simulation kernel, SPIS-NUM, and data, from an advanced GUI to a direct manipulation of objects via a script language (Jython) or the possibility of dynamically modify source codes. One difficulty was to integrate in a structured manner a large set of very heterogeneous pre-existing tools (CAD tools, meshers, data importers...) and simplify the most possible the definition of initial and boundary conditions for the simulation kernel. To handle this objective, SPIS-UI was designed around a generic Task Manager that coordinates the various actions to be done. Each component and tool is wrapped into generic tasks, dynamically pluggable into the framework. Open Source and fully written in Jython and Java in an Object Oriented Approach, SPIS-UI accepts external components written in a large set of interpreted and native languages (Python, Java, C/C++, FORTRAN • . The consistency of shared data (CAD, mesh, properties • is maintained along the modelling chain through a generic Common Data Bus. The framework is completed by various extensible post-processing modules and customizable 2D/3D visualization pipelines based on the VTK technology.

This approach has already allowed the possibility to adapt the framework to a very large number of configurations depending on the specific needs of each system to model, users' needs or other models, as the kernel of the PicUp3D code. Also, this approach allows cross-validation of various simulation codes and open the perspective of future extensions with multi-physic and multi-models elements.

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TESTS and VALIDATION of a NEW SPACECRAFT PLASMA INTERACTION SOFTWARE, SPIS

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A new spacecraft plasma interaction software, SPIS, has been developed in the frame of the Spacecraft Plasma Interaction Network, SPINE (see http://www.spis.org/spine/). This software can a priori simulate the kinetics processes of ions and electrons taking into their space charge and their interaction with spacecraft surfaces. It is open source and freely available world-wide. While the development and the qualification of the software functionalities is under the responsibility of a consortium led by ONERA under ESA contract, the test and validation of the applicability of the code to physics problem is under SPINE responsibility. The validation programme includes step-by-step applications of the software to sheath modelling in simple geometry, artificial plasma injection, and spacecraft charging. We report here on the progress along this programme including Langmuir probe tests with spherical and cylindrical geometry, and comparison with other software for more complex geometries.

Notes :

DEVELOPMENT of GEOSPACE ENVIRONMENT SIMULATOR

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In the space development and utilization, it is very important to understand the interactions between spacecraft/structures and space plasma environment as well as the natural phenomena occurring in space plasma. In order to evaluate the spacecraft-plasma interactions quantitatively to contribute to the progress of space utilization and space technology, we aim to develop a proto model of Aerospace environment simulator (GES) • by making the most use of the conventional full-particle, hybrid and MHD plasma simulations. For the development of GES, we have been using the Earth simulator which is one of the fastest supercomputer system in the world. GES can be regarded as a numerical chamber in which we can virtually perform space experiments and analyze the temporal and spatial evolution of spacecraft-plasma interactions. GES will be able to provide fundamental data regarding various engineering aspects such as the electrostatic charging and electromagnetic interference of spacecraft immersed in space plasmas, which will be useful and important information in determining designs and detailed specifications of spacecraft and space systems.

In the present paper, we will briefly introduce GES and some preliminary results obtained with the Earth simulator. We will also mention the contribution to MUSCAT which we have newly started developing as a numerical tool for analyzing the spacecraft charging.

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MULTI-UTILITY SPACECRAFT CHARGING ANALYSIS TOOL (MUSCAT): DEVELOPMENT OVERVIEW

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Since the failure of ADEOS-II, charging of polar orbiting satellites has become a serious issue. The Japan Aerospace Exploration Agency has decided to develop a computational tool that can calculate charging status of a polar orbiting satellite jointly with Kyushu Institute of Technology. The simulation code is a combination of a particle-in-cell method and a particle Tracking method and can be used not only for a polar satellite but also for a GEO satellite or a low inclination LEO satellite. The aim of the simulation code is to give satellite designers chances to identify the charging hazard in the satellite design phase with user-friendly interface. The development of software named, Multi-utility Spacecraft Charging Analysis Tool (MUSCAT), started in November, 2004. Overview of development plan and current status of the simulation code will be presented.

Notes :

3D-SIMULATION of the SATELLITE CHARGE in MAGNETOSPHERE ENVIRONMENT

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In this talk, we are interested in the resolution of the stationary Vlasov-Poisson system modelling the interactions between plasma and satellite in a three-dimensional frame, without reemission.

The ion and electron currents received by the surface of the satellite are computed by using the back-trajectories algorithm. It consists in following back the trajectories of the particles which impact the surface of the satellite.

For the computation of the space charge in the whole space for any geometry of the satellite, we have built a stationary algorithm based on the back-trajectories algorithm. The retrograd particle trajectory from infinity to the satellite can be considered as a direct trajectory from the satellite to infinity, so that the distribution function along this direct trajectory is zero. We determine all the trajectories for which the distribution functions are zero. For the other points in the phase space where the distribution functions are not equal to zero, the simple argument of the conservation of the total energy is enough to determine these distribution functions without computing their trajectories. This is true only if the distribution functions are isotropic at infinity. So, the charge density is computed in the whole space. In order to validate this algorithm, we look at the case of a sphere. In this case, we are able to compute the analytical expression of the charge density. The asymptotical behaviour of the space charge and the potential are studied. A one-dimensional algorithm with spherical symmetry has been built for the resolution of the Vlasov-Poisson system and is a good validation for the three-dimensional algorithm. The obtained results are satisfactory.

The extension to the case with secondary emission will also be presented.

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NASCAP-2K SPACECRAFT CHARGING CODE OVERVIEW

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Nascap-2k is a modern spacecraft charging code, replacing the older codes NASCAP/GEO, NASCAP/LEO, POLAR, and DynaPAC. The code builds on the physical principles, mathematical algorithms, and user experience developed over three decades of spacecraft charging research.
Capabilities include surface charging in geosynchronous and interplanetary orbits, sheath and wake structure and current collection in low-Earth orbits, and auroral charging. External potential structure and particle trajectories are computed using a finite element method on a nested grid structure and may be visualized within the Nascap-2k interface. Space charge can be treated either analytically, self-consistently with particle trajectories, or by importing plume densities from an external code such as EPIC (Electric Propulsion Interactions Code). Particle-in-cell (PIC) capabilities are available to study dynamic plasma effects.
Auxiliary programs to Nascap-2k include Object Toolkit (for developing spacecraft surface models) and GridTool (for constructing nested grid structures around spacecraft models).
This talk will show examples of the various calculations that can be done with Nascap-2k, as well as validation by comparison with analytical results.

Notes :

A FIRST-PRINCIPLE BASED VIRTUAL TESTBED for SPACECRAFT

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Future space missions will require spacecraft integrated with advanced propulsion and power systems, such as very high power electric thrusters, high specific power solar arrays, large solar sails, and others. These advanced systems are drastically different from the state-of-the-art systems, and so will be their impacts on the host spacecraft and science instruments. Due to the complex nature of the physics, a lack of effective experimental means to diagnose the interaction effects on ground for in-flight conditions, and a lack of flight validation opportunity, no broadly applicable framework currently exists for one to accurately evaluate the risks from the integration of these new technologies and to derive a clear, physics based design guideline. This paper presents an overview of our research to develop a first-principle based virtual testbed on spacecraft-environmental interactions for spacecraft using advanced propulsion technology. Specifically, we discuss a) the development of a new, multi-purpose particle simulation package that is capable of handling the complex geometry associated with a real spacecraft while maintaining the computational speed of a standard particle code; b) the development of a user interface integrated with a computer aided design tool, a virtual reality environment, and a shared, network visualization environment; and c) the application of this virtual testbed to study interaction effects for spacecraft using single and multiple high-power ion thrusters and large solar sails.

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Session 5

Plasma Propulsion and Tethers

Session 5

Notes :

COLD MAGNETOSPHERIC PLASMA FLOWS and SPACECRAFT WAKES: PICUP3D SIMULATIONS and CLUSTER DATA

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Behind a spacecraft in a supersonic ion flow, a negatively charged wake will form. In a sufficiently tenuous plasma, where the potential of a sunlit spacecraft becomes positive, the size of the ion wake can become very much larger than what may be expected from the spacecraft geometrical size. This occurs if the energy of the flowing ions is less than what is needed to overcome the positive spacecraft potential. As the ion flow must be supersonic, this means that the plasma flow has to be quite cold. It turns out that such conditions are quite common in the terrestrial magnetosphere, particularly in the polar wind region. The polar wind cold plasma flow at geocentric distances outside 10 Earth radii can have ion flow kinetic energy around 10 eV, ion temperature of a few eV and be so tenuous that the potential of a spacecraft reaches above 20 V. This plasma is rarely measured on scientific spacecraft, as the ions cannot reach an ion detector mounted on the highly positive spacecraft. Consequently, there are only a few studies of its properties and distribution. However, the cold ion density can be estimated by comparing the number of ions actually detected on the spacecraft with independent density estimates, for example spacecraft potential measurements or wave signatures. Another effect of the negatively charged wake is its impact on electric field measurements on the spacecraft, which shows a clear signature of the wake electrostatic potential. While this is a contamination to the measurement of the natural electric field, it is also potentially useful to derive flow properties like velocity and temperature. In order to understand the problems and exploit the possibilities of this effect, we have used the PicUp3D code to perform PIC simulations. We present the simulation results and compare them to data from the Cluster spacecraft, whose comprehensive nstrument payload are quite ideal for studies of this character.

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SPHERICAL COLLECTOR VERSUS BARE TETHER for DRAG, THRUST, and POWER GENERATION

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Spherical collectors versus bare tethers for drag, thrust, and power generation Juan R. Sanmartin, Universidad Politecnica de Madrid, Madrid, Spain Enrico C. Lorenzini, Harvard-Smithsonian Center for Astrophysics, Cambridge, USA Performances of ED-tethers using either spherical collectors or bare tethers for drag, thrust, or power generation, are compared. The standard Parker-Murphy model of current to a full sphere, with neither space-charge nor plasma-motion effects considered, but modified to best fit TSS1R results, is used (the Lam, Al'pert/Gurevich space-charge limited model will be used elsewhere). In the analysis, the spherical collector is assumed to collect current well beyond its random-current value (thick-sheath). Both average current in the bare-tether and current to the sphere are normalized with the short-circuit current in the absence of applied power, allowing a comparison of performances for all three applications in terms of characteristic dimensionless numbers. The sphere is always substantially outperformed by the bare-tether if ohmic effects are weak, though its performance relatively improves as such effects increase.

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ANALYSIS of BARE-TETHER SYSTEMS as a THRUSTER for MXER STUDIES

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The concept of the electrodynamic propulsion has a number of attractive features and has been widely discussed for different applications. A number of system designs have been proposed and compared during the last ten years. In spite of this, the choice of the proper design, for a specific mission, is far from evident. Such characteristics of tether performance as system acceleration, efficiency, etc. should be calculated and compared. The code that calculates the current for bare and partly insulated tethers with circular (wire) and rectangle (tape) cross-sections is presented. It takes into account the corrections to the OML current due to the tether cross-section geometry and the magnetic field produced by the tether current. There are two options in this code: for current calculation with the prescribed energy supply and with the prescribed end-point potential. This permits us to calculate the parameters characterizing tether performance. Results for the current calculated for tethers with different designs for the currently proposed Momentum eXchange Electrodynamic Reboost (MXER) Tether System are presented.

Notes :

A PROPOSED BARE-TETHER EXPERIMENT on BOARD a SOUNDING ROCKET

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A sounding rocket experiment is proposed to carry out two experiments by the conductive bare-tether; I) the test of the OML (Orbital- Motion- Limited) theory to collect electron, and II) the test of technique to determine (neutral) density profile in critical E-layer. The main driver of the missions is the experiment is conducted in LEO and will show no danger to other satellites as the tether missions YES1, SEDSAT, and ProSEDS, which is cancelled just for afraid of collision with the ISS orbit. Also, the sounding rocket mission is possible to demonstrate the bare tether technology in low cost, simple mission concept, fast realization. The present sounding rocket experiment is expected to be the first conductive bare tether experiment.

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Session 6

Satellite On-orbit Investigations

Session 6

Notes :

SCINTILLATOR-BASED LOW ENERGY PARTICLE IMAGING

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Physical Sciences Inc. (PSI), in cooperation with the Boston University Center for Space Physics, and under the sponsorship of the Air Force Research Laboratory Space Vehicle Directorate, has developed and tested a lightweight, multi-configuration sensor to monitor the space weather environment. The scintillator-based Low Energy Imaging Particle Spectrometer (LIPS) is ideally suited to monitoring the lower energy (20 to 1000 keV) charged particle environment. The LIPS design is also compatible with the weight, volume, and power requirements of nanosatellites (<0.5 kg, <0.5 W). The LIPS design does not rely upon a magnetic sector to discriminate between particle types; rather it takes advantage of particle cross-section characteristics and scintillator properties to discriminate. We have already proven the feasibility of our approach; i.e., using thin films of materials to create particle-specific detectors, fiber-optically coupled to a position-sensitive photo multiplier tube. The result is a tremendous savings in sensor weight and volume. We present modeling predictions and sensor performance during ground calibrations.

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S/C "COSMOS – 2393" CHARGING UNDER DATA of FLIGHT MEASURINGS

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Flight experiment on measuring of s/c "Cosmos – 2393" charging on HEO has been prepared and initiated in Lavochkin Association in 2002. The purpose of experiment is data retrieval about parameters of s/c charging and study of its regularities. For experiment equipment "Zond – Zaryad" analogous to that which has been installed on "Mir"station is used [1]. The equipment provide the opportunity of the permanent control of the parameters on s/c orbit. The following parameters are measured:

ﾷ constant electrical field;

ﾷ variable electrical field;

ﾷ current on a sensor.

The first observed data were submitted on the 8-th SCTC.

In the article connection of measured parameters (for the period since March 2003 till November 2004) with a position of s/c in orbit, its orientation and geomagnetic activity are shown. In particular, at geomagnetic storms it is marked anticipatory (at some o'clock earlier) change of parameters of s/c charging in the relation to parameter Kp obtained under data of ground measurings.

1. In Flight Measurement of the Outside Surface Potential of "Mir" Orbital Station Klimov, S.I. et al. SP-476 7th Spacecraft Charging Technology Conference p.303, 2001

Notes :

HEO SATELLITE SURFACE CHARGING in 1995-2002

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In the past, large amounts of satellite charging data have been take at geosynchronous orbit and by DMSP and other satellites in low altitude orbits. There have been limited satellite charging measurements at intermediate to high latitudes at high altitudes. In this presentation we will show the results of spacecraft frame potentials that have been measured over several years on a highly elliptical orbit satellite, HEO 95-034, with an apogee of ~7.2 Re, ~63 deg. inclination and 12 hour period. The charging signatures were obtained using an electron-proton plasma spectrometer. The measurements are all of charging levels < -30 volts and summarized according to their local time and L range of occurrence. L was calculated using an IGRF field model. For example, the resultant plots for satellite frame potentials ≤ -100 volts shows that the occurrence of satellite frame charging at these high latitudes mimics the charging local time patterns observed by geosynchronous satellites. In addition, these data show that such charging extends from our minimum L of ≤ 4 Re to L > 10 Re. In all, more than 1000 extended charging intervals were observed during the late 1995 through mid 2002 period. For each charging interval that occurred, while transiting the inner magnetosphere, the maximum frame voltage and the minimum and maximum L values during charging were obtained. At times, signatures of differential charging were also observed. The differential charging was evidenced in the electron distributions in the same manner as the signature of satellite frame charging is exhibited in the ion distributions. The charging statistics for both frame and differential charging will be shown as a function of L, local time and charging levels and will be discussed in terms of the environmental conditons observed.

Notes :

ACTIVE SPACECRAFT POTENTIAL CONTROL: RESULTS from the DOUBLE STAR PROJECT

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Ion emitter instruments ASPOC (Active Spacecraft Potential Control) have been used successfully in several magnetospheric missions including Cluster. An improved version has been developed for the equatorial spacecraft of the Chinese-European Double Star mission (TC-1) launched in December 2003. The modifications include an new design of the ion emitter modules, as well as revised manufacturing and test procedures. As a result, higher currents than in previous missions can be achieved.
The main objective of the investigation lies in a reduction of positive spacecraft potential in order to minimize the perturbations to the plasma measurements on board, in particular to the plasma electron instrument PEACE. These data show an almost complete suppression of photo-electrons when ASPOC is emitting at 30 to 50 μA beam current. The angular distribution of the electrons in the presence of the ion beam is investigated in detail. The suppression of photo-electrons is slightly less effective in the direction opposite to the ion beam. The measurement of ambient electron distributions is highly improved, and no signatures of potential barriers were found.

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ANALYSIS of ON-BOARD MAGNETOSPHERE PLASMA DATA and GEOSYNCHRONOUS SPACECRAFT CHARGING

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Analysis of measurement data obtained for hot magnetosphere plasma fluxes in geosynchronous orbit has enabled to establish correlation between geosynchronous spacecraft charging and the plasma parameters. The measurements were done onboard the Russian ELECTRO [1] geosynchronous spacecrafts during years 1995-1997. Differential energy electron and proton spectra were measured with electrostatic analyzers in 0.1-12 keV energy range.

Time dependences of the plasma particle two-Maxwell distribution function parameters (plasma density and temperature) were obtained for full set of the experimental data (approx. 700000 energy spectra).

Well-defined correlation between the spacecraft charging (e.g. potential value) and the electron flux parameters (ration of the hot component current to the cold component current) was revealed. Distortions the measured electron and proton spectra caused by negative spacecraft potential were registered by spectrometers. Estimations of the potential values were done in terms of the distortions for various flight conditions using COULOMB computer tool [2]. Spacecraft charging was observed not only in the Earthцт shadow (during vernal and autumnal equinoxes), but in the sunlit orbit segments. Negative potential values in the 0.2-7.0 kV interval were observed.

1. Novikov L.S., Sosnovets E.N., Mar'in B.V., Makletsov A.A., Mileev V.N., Feigin V.M. In-flight investigations of geosynchronous spacecraft charging, Proceedings of the 9th International Symposium on Materials in Space Environment (16-20 June 2003, Noordwijk, The Netherlands) ESA SP-540, pp. 677-680.
2. Krupnikov K.K., Mileev V.N. , and Novikov L.S. A Mathematical Model of Spacecraft Charging ('COULOMB' Tool), Radiation Measurements, 1996, Vol.26, No.3, 513-516.

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ON-ORBIT DATA from SURFACE CHARGE MONITOR PAYLOAD of GSAT-2 SPACECRAFT

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The surface charge monitor (SCM) payload of GSAT-2 spacecraft comprises four detector heads and an SCM electronics (SCME) module. The SCM has been operating continuously since May 14, 2003, except when switched OFF during the period from August 14, 2003 to September 12, 2003.

The spacecraft charging currents from the GEO plasma, in general, have densities in the range from less than 0.01nA/cm2 to about 1nA/cm2. The SCM payload provides a means of monitoring these currents.

The detector heads, mounted on the external surface of the spacecraft panels, are directly exposed to the space plasma environment. The SCME houses a DC/DC converter module and provides telecomm and / telemetry interfaces to the spacecraft.

The response of SCM circuit to variations in the detector current has been investigated by employing the MIROCAP-6 circuit simulation software. The simulation results agreed with the results obtained by exposing SCM detector head to low-energy electrons in an electron beam test facility

Observations from on-orbit data:

· The magnitude of the on-orbit charging current density indicates the charging of detector heads to negative potentials with respect to spacecraft structure.

· The maximum current density of 0.32nA/cm2 recorded on June 17, 2003 has been a response to geomagnetic storm caused by the Solar flare event observed on June 16, 2003.

· The SCM responded to the geomagnetic disturbances following the solar flare of October 28, 2003. A peak current density of 0.45nA/cm2 has been recorded on October 29, 2003.

· In response to coronal mass ejection event (CME) of November 18, 2003, the detector recorded a peak current density of 0.38nA/cm2 on November 20, 2003.

· The maximum current density of 0.27nA/cm2 recorded on November 9, 2004 has been a response to geomagnetic storm caused by the Solar flare and CME observed on November 7, 2004

· The above enhanced current density events recorded by the SCM correlate well with the geomagnetic storm events recorded on GOES-10/12 spacecraft.

Notes :

DIFFERENTIAL CHARGING of SATELLITE SURFACE MATERIALS

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High electrostatic potentials of satellite surfaces can result in electrostatic discharges (ESD) that can damage or interfere with the satellite electronics. Measurements of the satellite conducting frame potential relative to the plasma environment are typically used to provide information on the statistical probability of charging. The frame potential is usually measured from the spectrum of the ions incident to the satellite or by voltage probes outside the satellite plasma sheath. But ESD events are more directly related to the differential charging of the various surface materials on the satellite. We will present satellite differential charging observations of several sample materials from the Satellite Surface Potential Monitor (SSPM) on the SCATHA mission. The SSPM data will be compared with measurements of the satellite frame potential and with the ESD events detected by the SCATHA discharge monitor. An example during an eclipse of the SCATHA satellite shows that the SSPM sample potentials are observed to anticorrelate with the frame potential. The ESD events for this intervals occur near the edges of the eclipse interval during high levels of SSPM potentials but only moderate frame potentials.

Notes :

CHARGING of CONDUCTIVE SPACECRAFT in the AURORAL ZONE

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Even spacecraft designed to avoid charging effects by having a conductive surface may sometimes charge to substantial potentials, occasionally into the kilovolt range. This is clearly shown in a study involving data from the Freja spacecraft, traversing the auroral zone at 1700 km altitude.

We present event studies and results of a statistical investigation as well as numerical simulations to show the effect of auroral electrons on the spacecraft. In the observations, simulations up to kilovolts are indeed sometimes observed, though lower values are much more normal. The simulations agree well with observed charging in many cases, but were not able to reproduce the events with kilovolt charging.

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Session 7

Charging of Polar Orbiting Satellite

Session 7

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SPACECRAFT CHARGING HAZARDS in LOW EARTH ORBIT

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The space environment in low-Earth orbit (LEO) has until recently been considered quite benign to high levels of spacecraft charging. However, it has been found that the DMSP spacecraft at 840 km can charge to very large negative voltages (up to -2000 V) when encountering intense precipitating electron events (auroral arcs) while traversing the auroral zone. The occurrence frequency of charging events, defined as when the spacecraft charged to levels exceeding 100 V negative, was highly correlated with the 11-year solar cycle with the largest number of events occurring during solar minimum. This was due to the requirement that the background thermal plasma density be low, at most 104 cm-2. During solar maximum, the plasma density is typically well above that level due to the solar EUV ionizing radiation, and although the occurrence frequency of auroral arcs is considerably greater than at solar minimum, the occurrence of high-level charging is minimal. Indeed, of!

the over 1200 events found during the most recent solar cycle, none occurred during the last solar maximum. This has implications to a number of LEO satellite programs, including the International Space Station (ISS). The plasma density in the ISS orbit, at a much lower altitude than DMSP, is well above that at 840 km and rarely below 104 cm-2. However, in the wake of the ISS, the plasma density can be 2 orders of magnitude or more lower than the background density and thus conditions are ripe for significant charging effects. With an inclination of 51.6 degrees, the ISS does enter the auroral zone, particularly during geomagnetic storms and substorms when the auroral boundary can penetrate to very low latitudes. This has significant implications for EVA operations in the ISS wake.

Notes :

SPACE PLASMA ENVIRONMENT at the ADEOS-II ANOMALY

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The fatal anomaly of the Advanced Earth Observing Satellite -II (ADEOS-II), that was due to the electric power loss from 6kW to 1 kW, occurred off the coast of Peru from 16:12 to 16:16 UT on 24 October, 2002. It was considered to be caused by the short/open of the electronic cables in the harness connected the solar paddle and the satellite body. That had been probably triggered by the charging (and discharging) of the electronically floating Multi-Layer Insulation (MLI) blanket of the harness by the energetic auroral electrons when passing through the northern aurora oval just before the anomaly. Unfortunately the data of the onboard energetic charged particle detector was not available at that time due to the data transmission schedule. However the NOAA-17 satellite, that was almost 30 km above the ADEOS-II, observed the extraordinary enhancement of energetic auroral electrons at the night side aurora oval. This event was an overture of the following historical big storms (so-called, Halloween and Thanksgiving storms). We will discuss the on-orbit space plasma data and the solar-terrestrial environments related to the ADEOS-II anomaly.

Notes :

INVESTIGATION of OPERATIONAL ANOMALY of ADEOS-II SATELLITE

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ADEOS-II satellite experienced anomalous operation on October 25th, 2003. The generated power on the satellite suddenly decreased from 6 kW to 1 kW. We conducted the experiments in order to investigate the degradation of the power of the satellite. When the satellite went though the aurora region around the pole of the earth, the multi layer insulator film wrapping the power cables would be charged by low energy electrons. ESD tests for the power cables were carried out. Trigger arc discharges occurred between the film and the cables with cracks produced by thermal cycles. Subsequently, a secondary arc electric discharge occurred between the cables themselves. After several discharges, this secondary arc caused sustained arc which burned out the cables. The heat caused by arc tracking between the hot and return cables made them burn out. This is the mechanism of operational anomaly on ADEOS-II satellite.

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SOLAR ARRAY PADDLE for the ADVANCED LAND OBSERVING SATELLITE (ALOS): CHARGING MITIGATION and VERIFICATION

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The Advanced Land Observing Satellite (ALOS) is the JAXA's flagship for high-resolution Earth observation. A large solar array paddle was developed for ALOS, which requires the electrical power generation of 7kW for three advanced mission instruments and a high-speed mission data handling system. With the deployed dimension of 22x3m in a polar orbit, this 9-panel rigid paddle has 8000 insulator Silver-Teflon thermal sheets and 72 exposed bypath diode boards on its back (i.e., shadow) face, and 23000 cover-glass integrated silicon solar cells on its front (i.e., sunny) face. An assessment of polar auroral plasma environment and a charging analysis for the ALOS paddle suggested that large negative potentials on the dielectric back-surfaces and at satellite ground may be induced through the ALOS's auroral passage in off-nominal conditions. Charging and arcing of the ALOS's baseline panel design was verified in laboratory experiments for the electron beam radiation and the plasma interaction, which simulated charging situations near the poles. Both the back face and the front face were tested, and arc thresholds were identified. Possibility of sustained arc and surge voltage, as well as survivability against estimated accumulation of arcs, was investigated. The results indicate that the back face has small negative arc thresholds for both the insulator surface potential and the spacecraft ground voltage. Surface flashover was observed over the Silver-Teflon coating.

Although both the back and front faces demonstrated immunity against sustained arcs, a varitety of approaches to mitigate the back face's susceptivity for charging and arcing were experimentally examined. The conductive adhesive that surrounded the baseline Silver-Teflon sheets and covered the CFRP face-sheets eliminated arcs at edges of the thermal coating and the face-sheets. The Kapton film that shielded the diode board protected the exposed power line from arcs. An ideal reference, ITO coated Silver-Teflon thermal sheet attached to the substrate with conductive adhesive, completely eliminated surface charging and arcs. Modifications to the baseline design adopted the conductive adhesive surface coating and the diode board film. This paper presents an overview of the ALOS's paddle design, with a particular emphasis on the characteristics related to charging . A series of the electrostatic discharge tests for the baseline and mitigated designs are summarized with their results. Operational considerations

against geo-magnetic storms are also described.

Notes :

ESD TESTS of SOLAR ARRAY PADDLE on a POLAR ORBITING SATELLITE

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The Advanced Land Observing Satellite (ALOS) that will be launched by the Japan Aerospace Exploration Agency in 2005 will carry a large solar array paddle of 22mx3m in a polar orbit One solar array circuit generates 56V to 58V and 2.7A. The wake side of paddle can be charged to a highly negative value. Laboratory experiments and numerical simulation are carried out to investigate charging and arcing phenomena on the paddle. The backside has exposed bypath diode boards and Silver-Teflon® thermal coating. The front side has 69mmx36mm Silicon cells.

We irradiated the backside with an electron beam to simulate charging situation near the North Pole. The return line of the powercircuit was grounded so that the coupon surface had so-called normal potential gradient with highly negative potential on the insulator surface. Surface flashover was observed once the insulator potential exceeded -7kV, that could occur according to the numerical simulation under the worst condition in orbit.

The backside of the paddle was also biased to a negative potential under LEO plasma environment to simulate charging situation near the South Pole, where aurora electrons charge the satellite body potential negatively. Frequent arcs were observed on CFRP surface with a negative potential of -70 ~ -80V. Possibility of sustained arc and surge voltage between hot and return ends of diode power circuit was investigated. Although no sustained arc was observed under the nominal operation condition, mitigation methods to decrease the risk of secondary arc at the diode board were also tested.

We also irradiated the front side with an electron beam to simulate the charging situation near the South Pole. No sustained arc was observed between strings under the nominal operation condition. Even when cracks were deliberately formed near interconnectors, no sustained arc was observed between strings and the substrates.

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Session 8

Numerical Simulation of Spacecraft Interaction with Thruster Plume and Solar Wind

Session 8

Notes :

NUMERICAL SIMULATION of HALL THRUSTER PLASMA PLUMES

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Plasma propulsion systems are finding increased application to many space missions due to their improved performance over chemical rockets. An important issue in the successful deployment of this technology is the safe integration of the plasma thrusters onto a spacecraft. Of specific concern is the possible interaction with the spacecraft of the plasma plume created by each thruster. Energetic ions emitted by the thruster can directly sputter materials when they impact on spacecraft surfaces leading to decreased functionality of those surfaces and possible re-deposition of the eroded materials on adjacent surfaces. Hall thrusters are one of the most important types of plasma propulsion and are used for station-keeping on communications satellites, and are under development for space exploration missions. In this study, we describe recent work to develop a detailed description of the plasma plume from Hall thrusters. A hybrid fluid-particle approach is employed in which electrons are modeled by solving their conservation equations, and the heavy particles are simulated using Monte Carlo particle methods. In this study, the effects of variations in thruster operating conditions (voltage and power), and propellant (xenon, krypton, bismuth) are studied numerically using the hybrid simulation method. The results are examined specifically in the context of effects on the plasma environment created around a spacecraft by the plasma plumes generated under these varying conditions.

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MODELLING of SMART-1 INTERACTION with the ELECTRIC THRUSTER PLUME

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Results from SMART-1 plasma measurements indicate that the spacecraft is in general floating 10 to 20 Volts negative with respect to the ambient plasma. An analytical model based on the interaction of the charge exchange plasma with the spacecraft and the solar panels is shown to be in good quantitative agreement with most of the observations. Progress for simulating the overall interaction between the thruster plume and the spacecraft by the use of PIC simulation codes is also presented.

Notes :

IONOSPHERIC CURRENTS to the SPECIAL SENSOR ULTRAVIOLET LIMB IMAGER on DMSP

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We compute ionospheric currents to the Special Sensor Ultraviolet Limb Imager (SSULI) on the DMSP F16 spacecraft to determine if these currents enter the imager and interfere with its operation. While the SSULI is performing nominally, a high level of noise is observed in the spectral data. A preliminary investigation concluded that the noise correlates with the measured spacecraft potential and that ions passing through the imager along the light path to the detector might explain the noise. Our task is to quantify the ion current entering the imager as a function of spacecraft potential and to determine if proposed design changes will eliminate the problem on future missions.

We use Nascap-2k to compute space potentials and resulting sheath currents around the DMSP spacecraft. We conclude that ion currents of about 0.3 microamperes enter the SSULI over a range of angles from the ram direction. To address what happens to this current after it enters the imager and its modulation by spacecraft potential, we perform calculations that include a portion of the interior of the imager. In these calculations, most of the current is attracted to the mirror and the interior walls and does not follow the light path through the imager to the detector.

The amount of current that would interfere with the imager's operation is orders of magnitude less than the ram current. To compute the current following the light path, we use the reverse trajectory approach. This approach is appropriate for the computation of currents to surfaces for which most of the external phase space is blocked. In this approach, ions are tracked outward from the imager to relate the distribution function at the imager's interior surface to the ambient ram ion distribution function. The distribution function is then integrated to give the current.

Notes :

MODIFICATIONS of SPIS SOFTWARE and MODELING of PLASMA FLOW AROUND SMART-1

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Since 2002 ESA has began a development of SPIS software to simulate an interaction of space plasma with spacecraft surface materials and compute a plasma induced charging. SPIS is open source software and is written with Java which is object oriented programming language. This simplifies future development and modification of software. There is a number of models: an electrostatic 3D article-In-Cell (PIC) plasma, photo and secondary emissions of electrons, electric circuits etc. SPIS also has an artificial plasma source to simulate a plume exhausted from electric propulsion system (EPS). However, current version does not have models for a neutral flow exhausted from EPS and ion-neutral collisions. The main product of ion-neutral collisions is charge-exchange (CEX) ions which have velocities of neutrals at the beginning and can turn back and contaminate spacecraft surface.

The paper applies a simple and fast approach to simulate CEX ion plasma around spacecraft. The approach is based on successive application of DSMC-based software for a neutral flow, axisymmetric PIC-MCC for a plume flow, and 3D PIC software for plasma flow around spacecraft. DSMC software was used as it. Whereas axisymmetric PIC-MCC software was modified to use DSMC results and sample CEX ion production map. New SPIS classes to handle DSMC and PIC-MCC results and generate CEX ions were developed and tested for axisymmetric plume case. The obtained results are in good agreement with PIC-MCC prediction.

3D computations of CEX environment around SMART-1 were performed with modified SPIS software for zero and -18 V spacecraft potentials with quasineutral assumption and Poisson solver. It is shown that a quasineutral assumption is not valid in the vicinity of solar array mounts and behind solar arrays. A full-length paper will also present an implementation of Monte Carlo collision model into SPIS and 3D PIC-MCC simulation of plasma flow around SMART-1.

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Session 9

Material Characterization

Session 9

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EVOLUTION of the ELECTRON YIELD CURVES of INSULATORS

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Electron emission and concomitant charge accumulation near the surface of insulators is central to understanding spacecraft charging. We present a study of changes in electron emission yields as a result of internal charge build up due to electron dose. Evolution of total, backscattered and secondary yield results over a broad range of incident energies are presented for two representative insulators, KaptonTM and Al2O3. Reliable yield curves for un-charged insulators are measured and quantifiable changes in yield are observed due to <100 fC/mm2 fluences. We find excellent agreement with a phenomenological argument based on insulator charging predicted by the yield curve; this includes a decrease in the rate of change of the yield as incident energies approach the crossover energies and as accumulated internal charge reduces the landing energy to asymptotically approach a steady state surface charge and unity yield. We also find that the exponential decay of yield curves with fluence exhibit an energy dependant decay constant, α(E). Finally, we discuss physics based models for this energy dependence. To understand fluence and energy dependence of these charging processes requires knowledge of how charge is deposited within the insulator, the mechanisms for charge trapping and transport within the insulator, and how the profile of trapped charge affects the transport and emission of charges from insulators.

Notes :

PROPOSED MODIFICATIONS to ENGINEERING DESIGN GUIDELINES RELATED to RESISTIVITY MEASUREMENTS and SPACECRAFT CHARGING

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A key parameter in modeling differential spacecraft charging is the resistivity of insulating materials. This parameter determines how charge will accumulate and redistribute across the spacecraft, as well as the time scale for charge transport and dissipation. Existing spacecraft charging guidelines recommend use of tests and imported resistivity data from handbooks that are based principally upon ASTM methods that are more applicable to classical ground conditions and designed for problems associated with power loss through the dielectric, than for how long charge can be stored on an insulator. These data have been found to underestimate charging effects by one to four orders of magnitude for spacecraft charging applications.

A review is presented of methods to measure the resistivity of highly insulating materials—including the electrometer-resistance method, the electrometer-constant voltage method, the voltage rate-of-change method and the charge storage method. This review is based on joint experimental studies conducted for NASA at the Jet Propulsion Laboratory and at Utah State University to investigate the charge storage method and its relation to spacecraft charging. The different methods are found to be appropriate for different resistivity ranges and for different charging circumstances. A simple physics-based model of these methods allows separation of the polarization current and dark current components from long duration measurements of resistivity over day- to month-long time scales. Model parameters are directly related to the magnitude of charge transfer and storage and the rate of charge transport. The model largely explains the observed differences in resistivity found using the different methods and provides a framework for recommendations for the appropriate test method for spacecraft materials with different resistivities and applications. The proposed changes to the existing engineering guidelines are intended to provide design engineers more appropriate methods for consideration and measurements of resistivity for many typical spacecraft charging scenarios.

Notes :

SPACE CHARGE DETECTION and BEHAVIOUR ANALYSIS in ELECTRON IRRADIATED POLYMERS

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During magnetic storms spacecrafts get charged due to the interaction of the plasma environment with metallic and dielectric external surfaces such as thermal blankets. Electrostatic discharges, caused by bulk charging of satellites insulating materials are identified to be a major cause of anomalies. Actually hazardous discharges might cause electronic interference and result in electronic devices disruption. A qualitative and quantitative knowledge of the charge built up is important to reduce these problems from the earliest stage of the design. Charging conditions depends not only on the satellite position in space but also on the period of time. In our case, we are dealing with the electronic environment of vehicle located on the geostationary orbit that corresponds to the path used by most of the commercial satellites. Thanks to a specially equipped irradiation chamber called SIRENE, the electronic spatial environment on the geostationary orbit can be reproduced experimentally in the laboratory. Electrons of controlled energy are delivered by a Van de Graaff accelerator and/or an electron gun. In the present case, dielectric samples are always irradiated by a quasi-monoenergetic electron beam. To study the irradiated samples properties, the chamber has been equipped with space charge detection system based on pulsed electro acoustic method, a surface potential probe and surface current detection system based on the Split Faraday Cup technique. The combination of results obtained by these set-ups makes possible the charge behaviour analysis during irradiation and relaxation. Different irradiation and relaxation conditions have been tested on various materials such as Teflon® and kapton® specimen. The aim of this work is to show the complementarities of the results and to understand the charges behaviour once they have been injected into the material under specific conditions. Most of the measurements have been performed in-situ during electronic irradiation and during relaxation in vacuum.

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Poster Session 1

Poster Session 1

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EVALUATION of IRRADIATION EFFECTS on SILICA GLASS by ELECTRON-BEAM

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Late in 1990, a short circuit accident occurred on a geostationary orbit environment (GEO) satellite due to sustained arc on high voltage solar array. The arc plasma generated by a trigger arc absorbs the electrical charge on cover glass and the conductivity increases. The high density arc plasma short-circuits the adjacent array strings and the solar array supplies the current to the trigger arc that can grow into damaging sustained arc. Therefore, it is important to understand the mechanism of charging phenomena in silica glass, which is a basic component of the cover glass.

We studied irradiation effects of keV-order electron beam irradiation on silica glass by means of cathodoluminescence (CL). The CL measurements were performed under the electron-beam irradiation using a scanning electron microscope as a electron beam source. The CL spectrum was measured in a monochromatic mode, while the time response of the CL was measured in a panchromatic mode, where the integrated intensity of the whole CL spectrum was studied.. Samples were silica glasses containing various concentrations of impurity. Also, thermal oxide, borosilicate glass, and Ge-doped silica were studied for comparison.

The CL spectra exhibit 460 nm bands due to an intrinsic process by recombination of self-trapped excitons of SiO2. Impurity-related CL bands were also observed at 300-400 nm and 650 nm. The time response of the intrinsic CL band at 460 nm comprises growth and decay components. The growth rate of CL increases with the OH content and other impurities. The decay of the CL intensity can be understood in terms of the accumulation of damages such as defect generation in silica glass by 15-keV electrons. Correlation of the CL behavior with charging effect of silica glass is discussed in terms of the trapping and defect generation in silica glass.

Notes :

ESD TEST of a LARGE SOLAR ARRAY COUPON in GEO PLASMA ENVIRONMENT

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In substorm environment, as high energy electrons flow into a satellite, arcing may occur on solar arrays. At our laboratory, the space environment proof test to evaluate the performance of the solar array against arcing is being performed. In a ground examination, it is required to simulate GEO conditions appropriately. However, making examination conditions unnecessarily severe is not welcomed. It leads to the increase of cost to take mitigation based on such an examination result. Since most test facilities are not large enough to accommodate a full solar panel, laboratories usually use coupons. Usually, the capacitance of coverglass is simulated by connecting a capacitor in the external circuit. But, there are big differences among research organizations about how much capacitance is appropriate. The international standard about the ground test conditions does not exist. The purpose of this research is to investigate the influence of coverglass on arcing phenomena on GEO satellite solar array. The experiment is performed by using a large solar array coupon(400x400mm). The coupon has 50 Si cells. The electron beam was used to simulate the substorm environment. At first, we used aluminum foil to scatter the electron beam on the whole surface of solar array coupon. We measure the surface potential on solar array immediately after the arc inception by using a non-contacting surface potential probe. Extent of neutralization differ for each arc. We will report the experimental results in the conference.

Notes :

DEVELOPMENT of OPTICAL MEASUREMENT SYSTEM FOR INTERNAL CHARGE DISTRIBUTION in INSULATING MATERIALS

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Spacecrafts are exposed to the harsh environment where the high-energy charged particles, such as electrons and protons, are scattering. When a large amount of the charged particles is irradiated to the insulating materials of the spacecraft, an electrostatic discharge may occur. The electrostatic discharge sometimes gives a serious damage to the systems of spacecraft. Therefore, it is necessary to study a relationship between electrostatic discharge and charge accumulation in insulating materials to increase the lifetime and the reliability of spacecraft by keeping the spacecraft from having such accident. To investigate the process how the charged particles accumulate into the bulk of the insulating materials, we have developed the internal charge measurement system using optical method. We have already confirmed using pulsed electro-acoustic method (PEA method) that the irradiated electrons accumulate in the bulk of an acrylic resin when the electron beam is irradiated to it.

However the PEA method is available under only a restricted measurement condition. Therefore, we have attempted to develop a widely usable measurement system using optical method. In this method, the intensity of polarized laser light passed through the transparent insulating materials including internal electric field distribution is measured. Since the intensity is proportional to the square of electric field, we can calculate the electric field distribution by measuring the intensity. In this report, to estimate the reliability of the optical measurement system, we compare the distribution of the accumulated charge observed using the optical method to that obtained using PEA method.

Notes :

OBSERVATION of INTERNAL CHARGE BEHAVIOR of E-BEAM IRRADIATED POLYMERS USED on SPACECRAFTS DURING ELEVATING TEMPERATURE

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We have investigated the observation of internal charge behavior of e-beam irradiated polymers with a wide range of temperature. Since the polymeric materials have many superior properties such as light-weight, good mechanical strength, high flexibility and low cost, they are inevitable materials for the spacecrafts. In space environment, however, the environment where satellites exist is so severe, because the temperature changes from -120 to 150 degree centigrade and high energy charged particles drift about. The polymers sometimes have serious damage by irradiation of high energy charged particles. When the polymers of the spacecrafts are irradiated by high energy charged particles, some of injected charges accumulate and remain for long time in the bulk of the polymers. Since the bulk charges sometimes cause of the degradation or breakdown of the materials, the investigation of the charging and decay processes in the polymeric materials with variations in temperature are important to decide an adequate material for the spacecrafts. In this report, we observed the decay processes of accumulated charge in e-beam irradiated polymers, such as Kapton or Teflon, during elevating temperature using pulse electro-acoustic measurement system. From the results, it is found that the various accumulation and decay patterns are observed in each material. It seems to be useful and be helpful to increase the reliability of the polymers for the spacecrafts.

Notes :

OBSERVATION of INTERNAL CHARGE BEHAVIOR in ELECTRON BEAM IRRADIATED POLYMERS USING ACOUSTIC METHOD

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The spacecraft flying in GEO is always exposed to plasma and/or radioactive-rays such as -, - and -rays. In such conditions, dielectric materials like cover glasses of solar battery or thermal blankets are charged up. Especially in GEO, they are exposed to be irradiated by high energy electron beam. In the case of high-energy electron beam irradiation, the electrons are injected into the bulk of dielectric materials. The unexpected discharge accident may result in an error for the mission of spacecraft. Because there are no basic data based on practical experiments, it seems difficult to simulate the accumulation and relaxation processes of injected charge in dielectric materials. In other words, it is difficult to expect when and how an accident of ESD caused by an accumulated charge will happen on spacecraft. Therefore, we need to measure the charge distribution in the bulk of dielectric materials. We have been developing a system for measuring such a charge distribution in dielectric materials using, so-called, PIPWP (Piezo-induced pressure wave propagation) method. Recently, we have improved our system to measure the charge distribution in dielectric materials under electron beam irradiation in vacuum. Using this system, we succeeded in measuring a charge distribution in e-beam irradiated Kapton® and PTFE samples in vacuum condition. The peak position of the negative charge was related to the energy of electron beam. After irradiation, we also observed a decay process of accumulated charge. It was found that the amount of accumulated charge in PTFE is higher than that in Kapton®.

Notes :

PROPOSAL of a CURRENT REGULATIVE DIODE for POWER SUPPLY in SUSTAINED ARC TEST

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Many spacecrafts suffered serious troubles on orbit as their bus voltages became higher. The most dangerous trouble is sustained arc. The sustained arc means that the short-circuit between adjacent strings of solar arrays is formed and the current flows continuously by photovoltaic power of itself. To prevent this destructive troubles in advance, the ground tests are needed to justify whether the sustained arc occurs or not.

The initial energy of sustained arcs is supplied from the trigger arcs occurring at the gap between strings. The energy of trigger arcs is fed by the charge between spacecraft and space, and the charge of coverglass. On the ground test, this energy is simulated by a external capacitance. The potential difference between adjacent strings is simulated by a DC power supply. When the trigger arc occurs in the gap, this power supply can feed the power. If this power supply has a large internal capacitance, this can feed bigger energy than on orbit, and the test condition becomes much different from real one. The power supply demands a small internal capacitance to simulate the real condition.

We proposed a Current Regulative Diode (CRD) for sustained arc test. The CRD is a semiconductor and keeps the value of current even if the potential difference of the CRD changes. We can use any type of power supply for driving the CRD. We measured the characteristic of short-circuit of the power supplies with the CRD and compared CRD power supply with other power supplies. We also performed a sustained arc test using the CRD power supply.

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CHARGE STORAGE MEASUREMENTS of RESISTIVITY for DIELECTRIC SAMPLES from the CRRES INTERNAL DISCHARGE MONITOR

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Resistivity values were experimentally determined using charge storage methods for six samples remaining from the construction of the Internal Discharge Monitor (IDM) flown on the Combined Release and Radiation Effects Satellite (CRRES). Tests were performed over a period of seven weeks in a vacuum of ~5×10-6 torr an average temperature of ~25 °C to simulate a space environment. Samples tested included FR4, PTFE, and alumina with copper electrodes attached to one or more of the sample surfaces. Micaply FR4 circuit board material was found to have a resistivity of ~1×1019 ohm-cm. Fiber filled PTFE exhibited little polarization current and a resistivity of ~4×1020 ohm-cm. Alumina, bled away charge at a rapid rate making resistivity calculation difficult; the best estimate of resistivity is ~1018 ohm-cm. Experimentally determined resistivity values were two to three orders of magnitude less than found using standard ASTM test methods. The 1 min wait time suggested for the standard ASTM tests is much shorter than the measured decay times. The primary currents used to determine ASTM resistivity are caused by the polarization of molecules in the applied electric field rather than charge transport through the bulk of the dielectric. Testing over much longer periods of time in vacuum is required to allow this polarization current to decay away and to allow the observation of charged particles transport through a dielectric material. Application of a simple physics-based model allows separation of the polarization current and dark current components from long duration measurements of resistivity over day- to month-long time scales. Model parameters are directly related to the magnitude of charge transfer and storage and the rate of charge transport.

Notes :

ESD-TRIGGERED ARC DISCHARGES on SOLAR ARRAY by ELECTRON-BEAM IRRADIATION

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Arc discharges occurring on solar array are threat to the spacecraft systems operating with high voltage. It is necessary to make the discharge mechanism clear and to reflect in the solar array design in order to maintain the high reliability of the future large spacecraft systems. From this viewpoint, we have investigated the arc discharge characteristics between a pair of real GaAs solar cells irradiated by electron beam. The characteristics were obtained as parameters of the gap length and the applied voltages between the cells, and the charge quantity released from the external capacitors at the time of the occurrence of ESD. From the analysis of the waveforms of transient discharge current, the soundness of the array insulation design is evaluated.

Notes :

BASIC EXPERIMENT for VACUUM ARC DISCHARGE TRIGGERED by ESD

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TEMPO-2 satellite was estimated to be damaged by sustained arc on the solar array in 1997. ADEOS-2 satellite of Japan was also supposed to be damaged due to arcs in wire-harnesses of the solar paddle in 2004. In order to make the conditions of the occurrence of these arc discharges clear, it is necessary to evaluate the arc discharge phenomenon in vacuum qualitatively. Therefore we tried to investigate the arc discharge characteristics between the electrodes consisting of a pair of thin metal plates or small wires triggered by ESD released from a capacitor. These characteristics were obtained as parameters of the gap length and the applied voltage between the electrodes, the current flowing to the electrodes, the amplitude of the ESD and so on.

Notes :

CRYOGENIC and ELEVATED TEMPERATURE HYPERVELOCITY IMPACT FACILITY

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The Hypervelocity Impact Facility at Space Research Institute, Auburn University has recently completed a facility upgrade that permits the impact testing of space materials within the cryogenic and elevated temperature range. Sample temperatures within the range of 40 – 420K have been achieved for polymer films. These wide temperature range capabilities add to the facilities current testing experience with impact initiated plasma discharge testing for solar cell arrays.

The facility utilizes a plasma drag gun to accelerate a variety simulated micrometeorite materials in the 50 to 150 μm range to velocities between 5 and 12 km/s. For each test 5 to 50 particles impact the surface of the target sample within an impact area of approximately 15 cm in diameter. The test chamber can accommodate samples up to a meter wide for ambient and heated tests, and 48 cm for cryogenic samples. The gun and test chambers are evacuated by He cryopumps and dry roughing pumps to produce a clean, oil free environment. Utilizing a streak camera and PMT detection system, the correspondence between individual particle size, speed and impact site can be determined. Standard post analysis yields: micrographs of each impact site, dimensions of the pertinent impact characteristics and individual particle velocity and size estimates.

Notes :

ASYMMETRY of PHOTOELECTRON DISTRIBUTION and ITS DEPENDENCE on SPACECRAFT POTENTIAL OBTAINED from GEOTAIL DATA

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In the Earth's magnetosphere where the spacecraft potential is positive in most cases, photoelectrons emitted from the spacecraft surface are pulled back and some of them are detected by electron analyzer onboard spacecraft as an unnecessary contamination.By analyzing them contained in the data of LEP/EA-e instrument onboard GEOTAIL spacecraft, we investigated velocity/energy distribution functions of photoelectron, and its relationship to the spacecraft potential.

We found that the ratio of the dawnward flux to the duskward flux of photoelectrons increases when the spacecraft potential is large, and decreases when it is small. Further analysis by plotting the ratio as a function of the photoelectron energy normalized by the spacecraft potential, E/Vsc, revealed that the ratio is maximized when the value of E/Vsc is about 0.3. This result implies the existence of azimuthal electric field in addition to the radial electric field which is regarded to be dominant in an ordinary case around the spacecraft surface, although the generation mechanism of that component is not clear.

Notes :

THIN FILM SOLAR ARRAY SAMPLES in SIMULATED LEO ENVIRONMENT

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Ten different samples of thin film solar arrays were tested in low density plasma. Tests were performed in vertical vacuum chamber (2 m diameter and 3 m height) equipped with four oil diffusion pumps and in horizontal chamber (1.8 m diameter and 2.5 m length) both providing background pressures below 1 μTorr. Cooled traps operation in vertical chamber and cryogenic pump in horizontal chamber secured quite clean environments with some traces of water vapor. Current collections and breakdown thresholds were measured for all samples. Degradation of some sample surfaces due to interaction with plasma electrons was established. The results obtained in this test were used to make recommendations to array manufacturers for further development of space graded solar arrays.

Notes :

LABORATORY TEST of ARCING on SATELLITE POWER CABLES

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In the Low Earth Orbit, cracks are often made at the power cable of a satellite by debris, heat cycle, etc. Triple junctions are formed from these cracks. Then arcs may occur. On a satellite, the cables of HOT and RTN are placed close to each other in many cases. Therefore, sustained arc may occur if arc tracking is formed. If sustained arc takes place, the cable is carbonized. At the worst case, it will be burned out. If it becomes so, the satellite may lose power from the solar panel. The purpose of this study is to investigate the conditions by which self-sustained electrical discharge is generated on cables. An experiment was carried out in a vacuum chamber which simulated Low Earth Orbit environment. A substrate made of CFRP (Carbon Fiber-Reinforced Plastics) was used. The experiment was carried out by placing the cables with cracks on the CFRP surface or on Kapton facesheet over the CFRP. Sustained arcs were observed with certain conditions.

Notes :

THRESHOLD CONDITIONS to INDUCE the SUSTAINED ARC on the SOLAR ARRAY PANEL of LEO SATELLITE

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Near future, satellites will need power generations over 100 volts, therefore, they must endure that voltage, not causing the sustained arc. In general, the bigger solar array power generation is, the larger possibility to induce secondary or sustained arc is. Sustained arc causes the permanent loss of power. Unfortunately, the critical value of power generation to induce the sustained arc was not yet established. In this study, we carried out the laboratory tests on charging and arcing of the solar arrays for satellites in a simulated Low Earth Orbit (LEO) environment. The experiment was focused on the secondary and sustained arc phenomena, which could cause permanent power loss. At first, we measured the relations between secondary/sustained arc occurrence and generated power of satellite. Finally, we investigated the critical voltage and current to induce secondary or sustained arc. From the results, we found out that the secondary arc could well occur as the power increases under the environment. And, we also found out the fact that it can cause very dangerous situation by even very small trigger arc to increase power to some critical extents under LEO environment. In our experiment, we used two types of solar array coupon, one is not coated with Row-Temperature Vulcanized silicone rubber (RTV), and the other is coated with RTV between strings. The detail of experimental results will be presented in the conference.

Notes :

RELATIONSHIP BETWEEN the GEOTAIL SPACECRAFT POTENTIAL and the ELECTRON DENSITY in the NEAR TAIL REGIONS

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The spacecraft potential has been used to derive the electron density surrounding the spacecraft in the magnetosphere and solar wind. The previous studies have examined the relationship between the spacecraft potential and the electron density in the distant tail regions and obtained an empirical formula to show their relation. However the electron density obtained by the empirical formula is often overestimated in the near tail regions with high electron temperature.
In this study, we investigate the relationship between the Geotail spacecraft potential and the electron density/temperature in the near tail regions during the period from November 1994 to March 1997, and improve the empirical formula considering the electron temperature. Then we discuss on the investigation of distribution of low energy plasma in the near tail region by comparing the electron density obtained by the improved empirical formula with that measured by the low-energy particle instrument onboard the Geotail spacecraft.

Notes :

NASCAP-2K SIMULATIONS of VLF PLASMA ANTENNA

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Nascap-2k is a modern spacecraft charging code, replacing the older codes NASCAP/GEO, NASCAP/LEO, POLAR, and DynaPAC. Nascap-2k implements all capabilities and algorithms from DynaPAC (Dynamic Plasma Analysis Code), allowing the user to perform dynamic plasma response calculations such as were done in the late 1980's in connection with the SPEAR II program.

Recently there has been interest in the plasma response to large, very low frequency (3 kHz to 20 kHz) antennas in MEO. As this antenna frequency is less than either the plasma frequency or the electron gyrofrequency (both nearly 300 kHz for plasma density of 10^9 m^-3 and magnetic field of 0.1 gauss), a quasi-static simulation is appropriate.

In this paper we will present antenna simulations using Nascap-2k in both hybrid PIC and full PIC modes. In the former, plasma oscillations will be suppressed, while in the latter they will be excited. Accuracy of the simulations will be assessed by comparison with lower-dimensional simulations of similar cases.

Notes :

DEGRADATION of SOLAR CELL ELECTRIC PERFORMANCE due to ARCING in LEO PLASMA ENVIRONMENT

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To supply large electric power to a spacecraft, we must operate solar arrays at a high voltage to save the cable mass and reduce the joule loss. When a solar array is operated at a high-voltage (>100V) in LEO environment, arcs occur on the array surface. The arcs occur on the so-called triple junction, that is a junction of conductor and dielectric materials exposed to the surrounding plasma. The arcing prevents solar arrays from operating at the high-voltage. Mitigation techniques against arcing are necessary to realize future large space platforms such as solar power satellite, space factory and others. The arc mitigation techniques are sought in the following two directions: (1) Suppress the inception of arcing completely (2) Minimize the damage caused by each arc. In this paper we focus on the second item.

In order to investigate the degradation of solar cell due to arcing, we carried out laboratory experiments. The experiment was performed inside a vacuum chamber. Voltage-current characteristic of solar cell was measured inside the chamber after a pre-determined number of arcs to investigate degradation of solar cell. The external circuit was connected to solar cells through the vacuum chamber wall. The damaged position of solar cell due to arcing was identified by using thermal emission analysis. We investigated threshold arc energy which causes degradation. The arc energy is calculated from arc current and array potential.

From the result, it was found that arcs formed current leak paths along the side surface of degraded solar cell. Each current leak path was formed by only one arc. As the arc energy became large, the current leak path was formed at higher probability which led to a higher probability of solar cell degradation. We have identified the threshold of arc energy to cause the cell degradation.

Notes :

EVALUATION of DC ELECTRIC FIELD MEASUREMENT by GEOTAIL OBSERVATIONS & SIMULATIONS

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We summarize the characteristics of the DC electric field measurement by the double probe system, EFD-P, aboard GEOTAIL. The accuracy and correction factors for the gain (effective length) and off-set, and their relationships to ambient plasma conditions are provided. Those will be compared with the numerical charging simulations of the spacecraft.

Notes :

THE SPACECRAFT CHARGING by the ELECTRON BEAM INJECTION at IONOSPHERE

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The classical estimations give the maximum potential of spherical satellite (spacecraft) during the electron beam injection in the ionosphere that much exceed the value obtained in real rocket experiments even those conducted at night and without artificial neutralization.

The possibilities of compensation are considered on the basis of theoretical models taking into account the collective beam-plasma interactions. The examples of numerical modeling of the charge-discharge dynamics are provided for the APEX spacecraft conditions. It is shown that changing the beam pulse form and rise rate (dI/dt) we can change the structure of space charge zone around the spacecraft and have the chance to create the strong disturbance or resonance conditions in the time interval of (p/wp) Ј t Ј tion. From these positions we consider the APEX facilities used to control the inflow current, the spectrum of energetic particles, the high- frequency oscillations and the spatial distribution of optical emissions in the injector vicinity with high time resolution.

Notes :

STUDY of THE PLASMA INTERFERENCE with HIGH VOLTAGE ELECTRODE ARRAY for SPACE POWER APPLICATION

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We are studying the problems associated with high voltage power systems in space. Especially we are interested in the potential distribution of the solar array that is resistant to the electrical discharge. We have carried out an experiment on the interaction between the high voltage solar array and the ionospheric plasma. In our experiment, an array of electrodes is used to simulate the inter-connectors of the high voltage solar array. We have found that the current flowing into each electrode strongly depends on the potential distribution generated by the array of the electrodes.

Notes :

CHARGING of COAXIAL LINES WITH FLOATING CORE at GEOSYNCHRONOUS ALTITUDES.

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For telecommunication satellites, RF equipment switching can lead to coaxial cables electrically floating. In a geostationary orbit, space environment can charge floating cables to potentials as high as some tens of volts. With commutations, the charge stored on the core flow out and the generated current can trigger interference or damages on MOS transistors. This paper deals with experimental results at electrostatic behaviour level for such cables under electron irradiation. At the experimental side, two coaxial cables used in satellites (an overshield brand of 7 coaxial cables and a cable connected to a switch) are irradiated by an electron beam source simulating a geostationary orbit equivalent energy range and flux intensity. During these experiments, we monitored the current collection and the voltage gained on core and/or cables shielding after their irradiation in order to evaluate the probable occurrence of an electrostatic discharge

This chapter is aimed at presenting experimental results for the electrostatic behaviour of coaxial lines. Several cables samples, usually used onboard geostationnary spacecraft, were submitted to high energetic geostationary representative flux levels. Floating core and/or floating shield voltage, current collected on core and shield and Esd transients were measured to evaluate risky events on electronics.

All the tests hereafter presented were performed in GEODUR vacuum chamber implemented at ONERA DESP (space environment department) in Toulouse.

This experimental test facility is capable to reproduce the penetrating electrons of the charging space environment thanks to a high energetic electron gun. Measurements of typical features as voltages on surface and transient currents are possible.

Study results synthesis

From the experimental results on cnes and radial cables

A few peculiar points have to be underlined

Potential levels, at equilibrium or under discharge lead after irradiation for the conductive parts whether they are floating or on a high impedance link (core or proof) are directly related to the preliminary outgassing period.

Core registered currents during irradiation depend strongly on the connecting mode of the shielding (floating or connected to the ground).

In the best case where the shielding is at the ground the potential reached at the core is yet critical. Hence for a 40pA/cm²flux a -320 V potential is measured on CNES cable whereas it reached −1200 V for Radiall cable.

With just a few hours of orbital position we are exposed to the stringent conditions.

The interpretation of these results must consequently be rather considered as: to make representative testing of what happen under orbital conditions it is mandatory to let the cable under testing being largely enough outgassed to promote worst conditions of measurements that could have been otherwise let on seen.

The shielding connected mode deeply influences the current level on the core during irradiation.

So far on RF circuits cables used between switches imply an absolute need for charge collecting resistors on each cable that could temporarily be floating along with the commutation schemes.

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ON-ORBIT POWER FLUCTUATIONS of ADEOS-II

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Japan's Earth Observation Satellite, ADEOS-II, was stopped the operations on October 24th, 2003. Before the anomaly, 100W fluctuations of generation power were known by evaluation of the telemetry. 100W corresponds to the power of an array circuit, therefore short or open circuits of array circuits were considered. JAXA had investigated the cause of the short or open circuits and performed some ground experiments to verify the scenario. This paper describes the result of the cause investigations.

Notes :

POTENTIAL BARRIER in the ELECTROSTATIC SHEATH around a SPACECRAFT

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In this paper a fully self-consistent model of the plasma around an electron emitting central body in a spherically symmetric geometry is used to analyse the electrostatic sheath around an idealized spacecraft in low density plasma. It is shown that non-monotonic potential with negative potential barrier can exist all around a positively charged spacecraft even in the case of realistic illumination pattern. The magnitude and the location of the potential barrier in regions of the magnetosphere with large Debye length plasma are discussed.

Notes :

SIMULATION of the CLUSTER FLOATING PROBE SIGNAL

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In this study, a numerical model of the relationship between the double probe potential with respect to spacecraft potential and the plasma parameters (density, temperature • has been developed. The mathematical model has been validated by comparison with numerical simulation results relying on a narrower set of hypotheses. A fit of the data between 1 and 80 cm-3 has been performed which allows to calibrate the model with suitable parameters for the photoelectron emission. The model can then be alternatively used for either density or temperature estimate. Uncertainties and range of validity are discussed.

Notes :

ROBUST DESIGN of SATELLITE SYSTEMS against SPACECRAFT CHARGING

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The spacecraft charging is one of major concerns of satellite systems. Space plasma can build up high differential voltages resulting in electrostatic discharges (ESD), sometimes causing anomalous behavior of spacecraft systems. We have studied the influence due to charging phenomena on satellite systems and mitigation techniques to control charging, electromagnetic interference propagation and electronics immunity. In this paper, some examples of our systems design and against the spacecraft charging the concerned issues are described. Standardized guidelines for designing satellite systems against ESD are needed. On the other hand, it is desirable that the requirements shall be appropriate and practical for satellite systems design, manufacturing and testing.

Notes :

DEVELOPMENT of 400V SOLAR ARRAY TECHNOLOGY for LEO SPACECRAFT

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In order to realize 400 volts operation in LEO, arcing caused by interaction between the spacecraft and the surrounding LEO plasma must be overcome. We have been investigating a design of high voltage solar array. This paper is the summary report for all the tests. All coupons were designed along following strategies; (1) prevent ambient ions from breaking into triple junction (2) decrease the electric field at triple junction. All of them achieved good results for arc mitigation. Especially, the coupon with ion barrier film which made from ETFE (called " film coupon") has never suffered any arc at ・400V. We carried out the tests under realistic orbit environment for the film coupon. The performed tests are as follows; endurance test for 30 years operation, debris impact, heat cycle, UV&AO exposure, film contamination, electro-static charging by contact, ambient pressure variation and arcing on substrate. It was confirmed that the covered film performed successfully in all the situations. This coupon has never arced in more than 25 hours which is equivalent to 1% power degradation in 30 yeas operation. Also, the electric output changed little by the covered film contamination and degradation. No primary arc occurred at all vacuum pressure and no charging by contact. With respect to debris impact, sufficient results were obtained with respect to resistance characteristics. The covered film had little damage even if the film support was hit directly. It was confirmed that the film attachments was well support the heat cycle +-90 degrees C by reducing tension of film.

Conductive substrate suffered many arcs at ・400V biased. Also, a sustained arc phenomenon between cells and substrate was induced by debris impact. The utilization of flexible substrate is adequate to application for 400V solar array in LEO environment.

Notes :

OPTICAL SPECTRAL ANALYSIS of ARC PLASMA on SOLAR ARRAY in GEO PLASMA ENVIRONMENT

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As the power level of Geostationary satellites increases, discharge phenomena on solar array are becoming serious threat to safe operation. There is more demand of international standard on ground test conditions (test environment, test circuit, test duration and external capacitance). Especially, the value of external capacitance that feed energy to the primary arc is very important. The external capacitance is corresponding to the absorbed electric charges on the insulator surface. A low capacitance value doesn't provide a sufficient current to trigger a secondary arc. While, we must avoid using excessive value capacitance that lead to degradation of cell electrical power output. The amount of external capacitor currently employed differs among research institutions.

We focused the relation between the electrical conductivity of the primary arc plasma and the amount of external capacitor in order to evaluate the range of proper amount of external capacitor. Electrical conductivity is strongly related to plasma temperature. Thus, we measure the plasma temperature by the emission spectroscopy. A sample coupon has 12 cells with typical size coverglass (7cm x 3.5cm x 100um). All the tests was carried out under the condition of inverted gradient, where the coupon was biased negatively and irradiated by the electron beam. Identified spectra were hydrogen, Carbon molecular, Silver from measured spectra of primary arc. Especially, C2 swan band was measured clearly from arc initiation to extinction with good repeatability. Thus, temperature was deduced by relative intensity of each blanch of C2 swan band. Primary arc temperature was about 1eV. Also, temperature and peak current of Primary arc had correlation each other. The relation between temperature and capacitance value will be presented at the conference.

Notes :

THE ANALYSIS of SPACECRAFT "ARCON-1" CHARGING UNDER DATA of FLIGHT MEASURINGS and MATHEMATICAL SIMULATION

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Flight experiment on measurement of s/c "Arcon-1" charging in the orbit in height about 1800km has been prepared and led by Lavochkin Association within one year to the period 2002-2003. The purpose of the experiment was data acquisition about parameters of s/c charging and studying its laws. The equipment "Zond-Zaryad" (similar to that has been established on "Mir" station[1]) for this experiment was used. The equipment provided the opportunity of the permanent control of the parameters on s/c orbit. The following parameters were measured:

ﾷ constant electrical field;

ﾷ variable electrical field;

ﾷ current on a sensor.

The analysis of a part of observed data was submitted on 8-th SCTC.

Mathematical simulation was led on the basis of observed data for definition of parameters different surface segments of s/c charging. The analysis of s/c charging was conducted with the help of the program for s/c charging calculation on low-altitude orbits [2], a designed by Skobeltsyn Institute of Nuclear Physics Moscow State University. This analysis was done taking into account data about constant electrical fields and currents. In this program effect on s/c of electrons of ionospheric plasma, directional streams ram ions, auroral electrons and light is taken into account. In the report outcomes of matching of full-scale data and mathematical simulation for lines of s/c typical cases loading are reduced.

1. In Flight Measurement of the Outside Surface Potential of "Mir" Orbital Station Klimov, S.I. et al. SP-476 7th Spacecraft Charging Technology Conference p.303, 2001
2. Makletsov A.A., Mileev V.N., Novikov L.S., Sinolits V.V. Effective Ion Current Computation Algorithm for Modelling of LEO Spacecraft Charging. Proceedings of 7th Spacecraft Charging Technology Conference (23-27 April 2001, ESTEC, Noordwijk, The Netherlands) SP-476, pp. 555-556.

Notes :

ENERGETIC CHARGED PARTICLE SPECTROMETER for the SPACE ENVIRONMENT RELIABILITY VERIFICATION INTEGRATED SYSTEM (SERVIS-1) SATELLITE

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Unmanned Space Experiment Free Flyer

We present the design, calibration and early mission flight data from a new radiation sensor, the Light Particle Detector, designed specifically to quantify the orbital environment responsible for microelectronics damage including deep dielectric charging. It supports Japan's Space Environment Reliability Verification Integrated System. We have developed a new, high-energy, charged particle spectrometer as part of the Space Environment Reliability Verification Integrated System-1 (SERVIS-1) satellite developed by the Institute for Unmanned Space Experiment Free Flyer (USEF) under contract from Japan's New Energy and Industrial Technology Development Organization (NEDO) The LPD, or Light Particle Detector, characterizes the proton, electron and ion fluxes and energy distributions incident on the SERVIS-1 spacecraft. The LPD is one component of a comprehensive Environment Monitor System for Space (EMSS) being developed by the Mitsubishi Precision Co., Ltd., for USEF. The EMSS comprises the LPD energetic particle spectrometer, a Single Event Upset Monitor and several distributed FET dosimeters.

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THE DYNAMICS of the SPACECRAFT POTENTIAL during ELECTRON BEAM INJECTION

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In interpreting the results of active space experiments with charged particles injected in the near Earth space it is important to determine the change of potential of the injecting spacecraft and the disturbed plasma characteristics in its vicinity. The structure of the plasma- beam system in the vicinity of a beam-injecting spacecraft is rather intricate.

APEX was a diversified project with an emphasis on the neutralization problem and the dynamics of the spatial charge zone of spacecraft. Since the injection heights range was from 400 to 3500 km where additional neutralization mechanisms do not work there was a possibility to study the total scope of the above mention problems.

It is shown that by changing the injection current in accordance with a certain law we can obtain a medium with the required properties in the spacecraft environment. A comprehensive analysis will allow an appropriate injection regime to be selected. The time and space variations of electron density in space charge zone of spacecraft are determined by the magnitude and the sign of injected current variations as well as by the experimental parameters.

Notes :

MEASUREMENTS of SMART-1 PLASMA ENVIRONMENT

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SMART-1 is the first spacecraft of the European Space Agency using a plasma engine as a main propulsion system. SMART-1 successfully reached a moon orbit in December 2004. It is also equipped with various plasma diagnosis instruments to study the electrostatic environment of the thruster and its potential impact on the spacecraft. Measurements from the EPDP plasma instruments are analysed in the both cases when the electric thruster is operated or not and onboard calibration is performed by comparing with other instruments. Characteristics from the charge exchange plasma are presented and it is shown that the spacecraft potential is floating within 10-20 volts negative with respect to the plasma.

Notes :

TIME to EFFECT of PLASMA-INDUCED ARCING on ISS ANODIZED ALUMINUM SURFACES

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The presentation identifies anodized aluminum surfaces, chromic-acid and sulfuric-acid anodized, on the ISS which could arc due to plasma charging.

An assessment of the effects of plasma-induced arcing on these ISS nodized surfaces, with and without concurrent micrometeoroid and orbital debris (MM/OD)impacts has been completed. The purpose of this study was to assess the loss of thermal control capability of the anodized surfaces on ISS if the Plasma Contactor Units were not continually operated as ISS is constructed and operated. The study considers six cases for the arcing Chromic Acid Anodized Aluminum (CAA) surfaces or Sulfuric Acid Anodized Aluminum (SAA) with no Micrometeoroid/Orbital Debris impacts (MM/OD), with MM/OD impacts that completely penetrated the anodize layer, and with MM/OD impacts that partially penetrated the anodize layer. The Marshall Space Flight Center (MSFC) performed analyses to provide the data giving the size and depth of penetration of MM/OD impacts. All six cases documented the thermal control properties of the anodized layers throughout the ISS life without PCU operations.

Notes :

Notes :

Poster Session 2

Poster Session 2

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ELECTROSTATIC BEHAVIOR of DIELECTRICS under GEO-LIKE CHARGING SPACE ENVIRONMENT SIMULATED in LABORATORY.

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Due to their dielectric nature and under the effect of the different forms of radiation encountered in space, dielectrics accumulate electrical charges up to the point where electrostatic discharges may occur. To prevent and avoid harmful interference due to discharges, their behaviour under irradiation must therefore be investigated in the laboratory before they are used in space applications. A current and widely used practice is to submit the tested materials to the bombardment of monoenergetic electron beams. Such a practice ignores the presence in space of a spectrum of electrons with energies reaching several MeV, and leads solely to surface charging and surface potentials generally higher than those really induced in space. The new approach consist in using in laboratory an electron source as similar as possible to the one existing in orbit. This paper describes the result obtained in the SIRENE facility, which was developed for simulating the spatial geostationary or MEO environment during great geomagnetic activity comparatively with monoenergetic irradiation. The range of available electrons in SIRENE goes from 10 to 400 keV. It points out the effect of the radiated induced conductivity. This paper provides results obtained on classical dielectrics as Teflon® Kapton® and Duroïd® of different thickness under mono and multi-energetic irradiation at ambient temperature and at -150°C. This paper showed also that even in the case of very low flux of high energy electrons, the surface voltage measured remains deeply affected

Based on those results the usual habit to use monoenergetic irradiation for qualification of space dielectrics should be reconsidered. For example surface voltage obtained of Teflon® is lower than the one obtained on kapton® under the same irradiation condition.

Notes :

PAYLOAD to INVESTIGATE THEIR EFFECTS on ELECTRON EMISSION and RESISTIVITY of SPACECRAFT MATERIALS

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A study of the effects of prolonged exposure to the space environment and of charge-enhanced contamination on the electron emission and resistivity of spacecraft materials, the ***S****tate of* ***U****tah* ***Sp****ace* ***E****nvironment &* ***C****ontamination* ***S****tudy (****SUSpECS****)*, is planned for flight aboard the MISSE-6 payload. The ***M****aterials* ***I****nternational* ***S****pace* ***S****tation* ***E****xperiment (****MISSE-6****)* program is designed to characterize the performance of candidate new space materials over the course of approximately four to eight month exposure periods on-orbit on the International Space Station, with a target flight date of mid-2006. The study is conducted by the Utah State University Materials Physics Group, in cooperation with the USU Get-Away Special Program and ATK Thiokol. Electron emission and transport properties of materials are key in determining the likelihood of deleterious spacecraft charging effects and are essential parameters in modeling these effects with engineering tools like NASCAP-2K code. While preliminary ground-based studies have shown that contamination can lead to catastrophic charging effects under certain circumstances, little direct information is presently available on the effects of sample deterioration and contamination on emission properties for materials flown in space.

Approximately 40 samples will be mounted on panels on both the ram and wake sides of the ISS. They have been carefully chosen to provide needed information for different ongoing studies and a broad cross-section of prototypical materials used on the exteriors of spacecrafts. Much of the pre-flight testing has already been done in conjunction with previous studies through the NASA Space Environments and Effects Program and other projects. The materials will be tested for resistivity and dielectric strength, and for electron-, ion-, and photon-induced electron emission yield curves and emission spectra. Characterization measurements include optical and electron microscopy, reflection spectroscopy, resistivity and Auger electron spectroscopy. In addition, studies of the service life of composite and ceramic materials of the ATK Thermal Protection Systems and Lightweight Structure Systems will evaluate chemical and mechanical properties as a function of depth from the AO and UV exposure surface.

Notes :

ATOMIC OXYGEN-INDUCED EROSION of POLYMERIC MATERIALS under SURFACE CHARGING CONDITION

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Space environmental effect (SEE) is emphasized to achieve long-term low-risk mission for spacecraft. One of the difficulties in studying SEE is its synergistic effect in real space environment. For example, the synergistic effect of 172 nm vacuum ultraviolet on atomic oxygen-induced erosion of polyimide enhances its erosion rate more than 300% depending on the ultraviolet intensity even though ultraviolet itself does not give any mass-loss on polyimide [1].

Spacecraft charging has been recognized as a serious problem on electronic systems aboard spacecraft. It sometimes seriously damages spacecraft system due to discharge. Moreover, the surface charging may affect the atomic oxygen-induced erosion of materials. The only literature reported was the effect of electron beam irradiation in the atomic oxygen erosion of polysulfone reported by King and Wilson [2]. They found that electron beam irradiation or bias voltages applied to the back plate of polysulfone increased the signal of reactive products from the target (CO and CO2), which suggested increase in mass-loss. They examined only for polysulfone. If surface charging also influenced the erosion rate of polyimide, surface-charging phenomenon during flights or ground-based tests needs to be considered in future material erosion studies.

In this study, we examined the effect of surface charging on the atomic oxygen-induced erosion of polyimide, which is a standard material for witness sample in most of the material exposure tests in low Earth orbit. Laser detonation atomic oxygen beam source at Kobe University, which delivers 5 eV atomic oxygen beam was used for simulating high-energy collision of atomic oxygen in low Earth orbit space environment. A quartz crystal microbalance (QCM) technique, which was established to study synergistic effect of atomic oxygen and ultraviolet on polymer erosion [3], was applied to measure the erosion rate of polyimide under surface-charging conditions. Effect of bias voltage on the atomic oxygen-induced erosion phenomenon of polyimide was analyzed and discussed.

References

[1] Yokota K., Ohmae N., Tagawa M., High Performance Polymers, Vol.16, No.2 (2004) pp.221-234.

[2] King, T., Wilson, W, AIAA Defense and Space Programs Conference and Exhibit, Huntsville, AL, 1997, AIAA 97-3901.

[3] Tagawa M., Yokota K., Kinoshita H., Ohmae N., Proceedings of the 9th International Symposium on Materials in a Space, Noordwijk, The Netherlands, ESA SP-540, June 16-20, 2003, pp.247-252.

Notes : ....

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THE IMPORTANCE of ACCURATE COMPUTATION of SECONDARY ELECTRON EMISSION for MODELING SPACECRAFT CHARGING

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The secondary electron yield is a critical process in establishing the charge balance in spacecraft charging and the subsequent determination of the equilibrium potential. Spacecraft charging codes use a parameterized expression for the secondary electron yield δ(Eo) as a function of incident electron energy, Eo. A critical step in accurately characterizing a particular spacecraft material is establishing the most efficient and accurate way to determine the fitting parameters in terms of the measured electron yield data and physics-based theoretical models. Simple two- or three- step physics models of the electron penetration, transport and emission from a solid are typically expressed in terms of the incident electron penetration depth at normal incidence or range R(Eo), and the mean free path of the secondary electron, λ(E). We review the models for δ(Eo) derived from various forms of the range expression R(Eo), including the Sternglass model based on the Bethe expression for the stopping power, several power law expressions of the form R(Eo)=b1Eon1 with different n1, and a more general empirical bi-exponential expression R(Eo)=b1Eon1+b2Eon2. Expressions are developed that relate the theoretical fitting parameters (λ, b1, b2, n1 and n2) to experimental terms (the energy Emax at the maximum secondary electron yield δmax, the first and second crossover energies E1 and E2, and the asymptotic limits for δ(Eo→∞)). In most models, the yield is the result of an integral along the path length of incident electrons. Special care must be taken in computing this integral. An improved fourth-order numerical method is presented, and its effectiveness is shown to be a significant improvement as compared to standard second-order methods. The fitting procedures and range models are applied to several measured data sets to compare their effectiveness in modeling the function δ(Eo) over the full range of incident energies, and in particular for determining crossover energies and critical temperatures.

Notes :

SIMULATIONS of CURRENT COUPLING IN ION BEAM-NEUTRALIZER INTERACTIONS

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Neutralization of ion beams in electric propulsion applications is a well-known phenomenon. The physics behind the robust matching of ion and electron currents and densities, are not. As electric propulsion devices move into micro and macro regimes with colloids, FEEPs, and thruster arrays, thruster-neutralizer interactions are under increasing scrutiny. A series of 2D simulations using PIC codes are presented, detailing starting and steady state interactions. It is shown that starting conditions require careful matching of currents to propagate without space charge effects while steady state conditions are robust regardless of ion or electron currents. Additionally, numerical effects are seen to mimic current coupling under certain conditions, but no robust physical mechanism for current coupling has been seen in PIC.

Notes :

SECONDARY ARC MODELISATION on SATELLITE SOLAR GENERATORS

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We are interested in the modeling of secondary arcs formation on satellite solar generators. Power on satellites is generally provided by solar generators which use semiconductor solar cells to convert solar radiation energy into electrical power. Solar generators consist of individual solar cells of about a few cm² in surface, which are connected in series into 'strings' to provide the required potential difference (usually ranging from 50 to several hundreds of Volts). Strings are then connected in parallel to deliver the requested power to the satellite equipments. Solar cells are made of high quality semiconductor materials (usually Germanium) and are very expensive to fabricate. It happens too often that after a certain operation time, an entire portion of the solar generator undergoes a permanent failure. The reason for it is the occurrence of an electrical arc which shortcuts one or several strings.

At the beginning of the scenario which leads to the secondary arc, the satellite charging in the earth environment plasma triggers a primary discharge between a cell interconnect and the dielectric which is used in the protective layer. The so-created plasma plume expands and eventually connects and shortcuts two neighbouring solar cells. The potential difference between the two cells (which is generated and maintained by the operation of the cells themselves) induces the transition of the primary discharge into a secondary arc. Once this arc is established, it pyrolyzes the insulating kapton substrate and transforms it into a conductor, which provides a permanent solid-state shortcut between the two cells, thereby irremediably deteriorating this part of the solar generator. The second phase of the discharge scenario, namely the expansion of the plasma plume and the transition from the primary discharge into the secondary arc can be modeled.

To describe the expansion of the plasma plume, a fluid description for the ions and electrons with the Poisson equation for the electric field is generally used (Euler-Poisson model). But, in our case where space charge effects occur on very short space scales, numerical simulations of this model are very expensive. Then, a current-carrying quasineutral model is derived from the two-fluid Euler-Poisson system by letting the ratio of the Debye length to the typical size of the device tend to zero. This manipulation leads to a numerical method that approach the solution given by the Euler-Poisson model in a precise and efficient way.

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NUMERICAL CALCULATION of ABLATION and PLASMA EXPANSION INDUCED by ELECTRIC BREAKDOWN of SPACECRAFT INSULATOR SURFACE in AMBIENT PLASMA ENVIRONMENT

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In the future, LEO spacecraft will be larger and higher powered. Because of the balance of leakage currents through ambient space plasma, their main conductive body will have a higher negative potential without plasma contactor operation. When spacecraft operate with a higher voltage, more intensive electric breakdown, i.e. arcing, is suspected to occur on the surface. In this study, unsteady physical processes inside the arc spot, such as ablation and heating of insulator, and plasma generation and acceleration etc, were studied using Computational Fluid Dynamics (CFD). Direct-Simulation-Monte-Carlo Particle-In-Cell (DSMC-PIC) plasma simulation was also carried out to examine influences of ambient space plasma on plasma expansion processes outside the arc spot. The calculated results showed that the spot diameter increased with time by intensive ablation; a large amount of plasma was produced inside the arc spot. Furthermore, neutral particles in addition to charged particles around spacecraft, outside the arc spot, played an important role in expansion of arc plasma by intensive ionization near the arc spot. Accordingly, high voltage operation of LEO spacecraft might bring drastic degradation of insulator surface by arcing, depending on insulator material properties and ambient plasma conditions.

Notes :

ANALYSIS of SURFACE CHARGING for a CANDIDATE SOLAR SAIL MISSION USING NASCAP-2K

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The characterization of the electromagnetic interaction for a solar sail in the solar wind environment, and identification of viable charging mitigation strategies, is a critical solar sail mission design task, as spacecraft charging has important implications both for science applications and for sail lifetime. To that end, we have performed surface charging calculations of a candidate 150-meter-class solar sail spacecraft for the 0.5 AU solar polar orbit and a 1.0 AU L1 orbit. We construct a model of the spacecraft with candidate materials having appropriate electrical properties using Object Toolkit and perform the spacecraft charging analysis using NASCAP-2k, the NASA/AFRL sponsored spacecraft charging analysis tool. We use nominal and atypical solar wind environments appropriate for the 0.5 AU and 1.0 AU missions to establish current collection of solar wind ions and electrons. In addition, we include a geostationary orbit case to demonstrate a bounding example of extreme (negative) charging of a solar sail spacecraft in the geostationary orbit environment. Results from the charging analysis demonstrate that minimal differential potentials (and resulting threat of electrostatic discharge) occur when the spacecraft is constructed entirely of conducting materials, as expected. Examples with dielectric materials exposed to the space environment exhibit differential potentials ranging from a few volts to extreme potentials in the kilovolt range.

Notes :

DEVELOPMENT of UNSTRUCTURED-GRID EM PARTICLE CODE for the SPACECRAFT ENVIRONMENT ANALYSIS

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We have developed a 3-dimensional electromagnetic particle simulation code with the unstructured-grid system. This code solves Maxwell's equations which is discretized with tetrahedral elements in 3D simulation space. Plasma particles are also traced by solving the equations of motion with the Buneman-Boris method. The main advantage of this code is the adaptability of modeling more realistic shape of a spacecraft than the orthogonal grid code. Thus, this simulation code is suitable for analyzing the plasma environment in the vicinity of a spacecraft especially in the region within a Debye length from the surface of the spacecraft as well as the spacecraft charging phenomena.

Notes :

1D MONTE-CARLO SIMULATION of CHARGE ACCUMULATION PROCESS INSIDE TEFLON FILM

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Dielectric breakdowns and electrostatic discharges occurring on/inside spacecraft surfaces have been considered as one of major causes of spacecraft malfunction and breakdown. Especially, satellites in the GEO(GEosynchronous Orbit) suffer from severe bombardment of high-energy electrons and other radiative particles. In order to avoid failures, detailed analyses of charging and discharging process inside surface materials such as Teflon are essential. Most previous studies focused on surface charging in a low-energy plasma environment and was theorized that differential surface charging may result in catastrophic discharges. Recently, however, it has been pointed out that there is possibility that internal charging is also related to discharging of spacecraft besides surface charging. Although there are some practical estimations of discharge criteria based on empirical equations, numerical simulations based on the first principle are important to understand the phenomena. In the present research, the charge accumulation processes inside a Teflon film are investigated with one-dimensional Monte-Carlo simulation. Elastic and inelastic scattering processes are considered in the collisions between electrons and atoms consisting of Teflon (CF4). Electron-phonon interaction and trapping effect are also included in the estimation of total cross section. Computed charge density distributions are compared with other simulation results and experimental data. The innovative measurement technique is used to obtain the charge density distribution inside a Teflon film. A PC based parallel computer is introduced to accelerate the computation so that the realistic time scale is attained.

Notes :

CIRCUIT ANALYSIS of EFFECTS of SOLAR ARRAY ARCING on SPACECRAFT POWER SYSTEM

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When solar array is operated at a high-voltage, arcs occur on the array surface. Our task is to examine effects of conductive current generated by solar array arcing on spacecraft circuit. First, an equivalent circuit of a small solar array module is formulated. The equivalent circuit is made for various types of solar cell such as silicon, gallium arsenic (hetero junction), and triple junction cells, etc. The equivalent circuit is composed of current source, diode, resistance, inductance and capacitance. First, we measured the frequency characteristic and I-V characteristic of a given solar array, and modeled the equivalent circuit which had the same characteristics by electronic circuit simulation software (SPICE). Spacecraft has hundreds of solar cells which are connected in series and/or parallel. We modeled an entire solar panel by connecting each unit solar array module. We also added the equivalent circuit of the arc inception and expansion. We simulated observed transient current in ground experiments using the equivalent circuit.

Notes :

VALIDATION of DAYLIGHT CHARGING CAPABILITIES of the SPARCS CODE

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Validation test cases for the SPARCS code (S. Clerc et al., 8th SCTC, 2002) are presented. More specifically daylight charging of spacecraft is addressed. The main mechanism for charging in daylight is the formation of a potential barrier which leads to partial recollection of secondary electrons (SE) and photo-electrons.

We first validate the recollection of SE using the particle tracking capabilities of SPARCS. The test case involves a positively biased metallic surface. The spherical symmetry and the plane diode configurations are studied. The SE are assumed to follow either a Maxwellian distribution or the semi-empirical model of Chung and Everhart, and an isotropic cosine angular distribution. Numerical results for the net total yield agree with the analytic expression. The case of a negatively biased sample is also studied. Here the total yield is unchanged but a focusing effect occurs (Nickles et al., 6th SCTC, 200). Results are presented for the plane diode geometry. Comparison is made to data for energy- and angular-resolved electron emission distributions from sputtered, polycrystalline gold surfaces taken as a function of increasingly negative sample bias voltage. In this case, the finite size of the sample leads to qualitative differences with the 1D model.

To validate the potential barrier formation due to surface charging, we compute the charging of a dielectric sphere in sunlight. Due to the complexity of this problem, an exact analytical solution cannot be found. Instead, a simplified monopole-dipole approximation is shown to give a reasonable agreement.

Finally, the computation of daylight charging benchmark of a model telecom spacecraft is presented. Agreement with published results (Davis et al. 8th SCTC, 2002) is good.

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MODELLING of SPACECRAFT DIELECTRIC MATERIALS INTERNAL CHARGING

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The RDOSE model [1] developed in Skobeltsyn Institute of Nuclear Physics of Moscow State University, and GEANT program were used for mathematical modelling of spatial distribution of internal electric charge arising in dielectric elements of the spacecraft design under influence of electrons of the Earth's radiation belts.

The RDOSE model intended for engineering calculations uses the ray tracing method for calculation of equivalent thickness of the protective shield for the chosen spacecraft design element, and for computation of the internal charge of thermalized electrons in this element. For description of geometrical model of complex spacecraft, the universal graphic interface using the modern programming languages (C, C++) and data descriptions languages (XML, X3D), Gtk+ ToolKit and visualization tools (OpenGL, VTK) was created. Practical application of the software developed is shown by the example of computation of the internal charge distribution in three-dimensional spacecraft model.

The model constructed on the basis of the GEANT program complex describes dynamics of the spacecraft dielectric materials internal charging process under influence of high energy electrons. In the report, results of computation of the dielectric materials internal charging for monoenergetic electron beams and electrons of the Earth's radiation belts are presented. The account of the self-consistent internal electric field of the thermalized electrons results in significant change of the stopped electron distribution, produces changes of back-scattered and transient particles outputs, and occurrence of secondary accelerated high energy runaway electrons.

1. Krupnikov K.K., Makletsov A.A., Mileev V.N., Novikov L.S., Sinolits V.V. Computer simulation of spacecraft/environment interaction · Radiat.Measur., 1999, Vol. 30, pp. 653-659.

Notes :

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GROUND-BASED EXPERIMENT of ELECTRON COLLECTION by an ELECTRODYNAMIC BARE TETHER

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Bare-tether systems are one of the greatest-efficiency electrodynamic tethered systems. The system with an uninsulated portion of the metallic tether itself to collect electrons from the space plasma is operated as a thruster or a power generator on a satellite. Ground-based experiments were carried out to understand phenomena of electron collection by a bare tether in space. Metallic tether samples were exposed to a simulating Low-Earth-Orbit plasma flow as varying tether sample diameter and length, and plasma velocity. A magnetic field was also applied. The current flowing from the plasma flow to a tether sample was measured with varying biased voltage to a tether sample. The potential of a tether sample on the plasma potential was normalized with voltage corresponding to electron temperature, and the collection current with thermal diffusion current. The following results were obtained: (1) the normalized collection current increased with normalized tether sample potential; (2) the tether sample diameter did not influence the normalized collection current characteristics although an increase in tether sample length decreased the normalized collection current in this experiment; (3) the normalized collection current increased with plasma velocity; and (4) the existence of magnetic field raised the normalized collection current because of the edge effect of a tether sample or the three-dimensional effect. A high collection current above the orbital-motion-limied current could be achieved with a magnetic field. Accordingly, the collection current characteristics of a bare tether in space are considered to strongly depend on plasma velocity and surrounding magnetic field.

Notes :

PLASMA PLUME CHARACTERISTICS of ELECTRIC THRUSTERS

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Electric thrusters are promising thrusters for near-future space missions. The exhaust plasma plume characteristics strongly depend on thruster species. In this study, three kinds of plasma thruster were investigated in order to understand their plasma plume characteristics. Several plasma diagnostic measurements were carried out. The low-power radiation-cooled arcjet thruster RAT-VII was operated with a mixture of hydrogen and nitrogen simulating hydrazine. In the downstream plume region, the electron temperature characteristics were almost flat around 7000 K. The electron number density on the central axis intensively decreased from the order of 1018 m-3 near the nozzle exit to the order of 1014 m-3 at an axial position of 50 cm. In radial distributions, the electron density decreased radially-outward although the Mach number increased. In the quasi-steady MPD thruster MY-III, the plasma was slightly expanded radially-outward downstream within the extrapolation line of the divergent nozzle regardless of discharge current and gas species although it was intensively expanded outside the line. Both the electron temperature and the electron density decreased radially-outward. The angle of radial expansion for hydrogen was relatively small compared with cases for argon because of an intensive thermal pinch effect for hydrogen. Using the low-power Hall thruster THT-IV, the ion current distribution was examined. The peak of ion current density on the central axis increased with coil current, i.e., with magnetic field strength, and in ranges from -60 to -20 deg and from +20 to +60 deg the ion current density decreased. As a result, an increase in magnetic field strength produced a more convergent ion beam. Two peaks near ion energies of 160 and 250 eV in the ion distribution function existed with a discharge voltage of 200 V. They correspond to single and double charged ions, respectively. A number ratio of the single to double charged ions was roughly estimated to be about 10 %.

Notes :

PERFORMANCE of COUPLED ED-TETHER / ION THRUSTER SYSTEM

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Use of propulsion systems that couple electyrodynamic tethers to ion thrusters, as suggested in the literature, is discussed. The system establishes electrical contact with the ionospheric plasma, at the anodic end of the tether, by ejecting ions instead of collecting electrons; also, the ion thruster adds its thrust to the Lorentz force on the tether. In this paper, we analyze the performance of this coupled system, as measured by the ratio of mission impulse (thrust times mission duration) to the overall system mass, which includes the power subsystem mass, the tether subsystem mass, and the propellant mass consumed in the ion thruster. It is shown that a tether acting by itself, collecting electrons at its anodic end, substantially outperforms the coupled system for times longer than a characteristic time of the ion thruster, for which propellant mass equals the power subsystem mass; for shorter times performances are shown to be similar.

Notes :

EFFECT of MAGNETIC FIELDS on SUSTAINMENT and DYNAMICS of LOW CURRENT DC VACUUM ARCS

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Vacuum arc is applied in various areas. They are vacuum circuit breakers, metal film growth, metal ion plantation to materials and so on. Lately, application to the attitude control of space satellite (thruster) is reported. In general, low current vacuum arc is hard to sustain for a long time. In this paper, we try to stabilize low current DC vacuum arcs by using various types of magnetic field distribution.

In our experiment, low current DC vacuum arcs, less than 20A, were ignited by the opening of a butt contact made of zinc. The velocity of the opening was 5mm/s and the final gap length was 5 to 20mm. One or two magnets were arranged behind the two contacts and four types of magnetic field distribution were formed. They were, (1) without magnetic field, (2) axial magnetic field distribution by two magnets, (3) repulsive magnetic field distribution by two magnets, and (4) diffusible magnetic filed distribution by one magnet.

Experimental results were summarized as follows.

(1) Without magnetic field, the cathode spot moves at random on the plane cathode surface and the arc plasma is diffusing. The arc voltage was constant for increasing gap length.

(2) In the axial magnetic field, the movement of the cathode spot is limited and the arc plasma is constricted. The arc voltage rose with increasing gap length but the arc continued a bit longer than without magnetic field.

(3) In the repulsive magnetic field, the arc vanished quickly after the opening of the contact.

(4) In the diffusible magnetic filed, the characteristics differed by the position of the magnet. When a magnet was arranged behind the cathode, the cathode spot rotated in the direction of Lorentz force. On the other hand, when a magnet was arranged behind the anode, the arc plasma was constricted in font of the anode and the lifetime became longer even the arc voltage rose with increasing gap length.

Notes :

ION ENGINE PLUME in TERACTION CALCULATIONS for PROMETHEUS I

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The Prometheus I (Jupiter Icy Moons Orbiter) spacecraft design calls for multiple ion thrusters, each with considerable higher beam energy (~5 keV) and beam current (~5A) than has previously flown in space. The challenges posed by such powerful thrusters relate not only to the thrusters themselves, but also to designing the spacecraft to avoid potentially deleterious effects of the thruster plumes. Accommodation of these thrusters requires good prediction of the highest angle portions of the main beam, as well as knowledge of elastically scattered and charge exchange ions, predictions for grid erosion and contamination of surfaces by eroded grid material, and effects of the plasma plume on radio transmissions. Nonlinear interactions of multiple thrusters are also of concern.

In this paper we describe two- and three-dimensional calculations for plume structure and effects of the Prometheus I ion engines. Many of the techniques used have been validated by application to ground test data for the NSTAR and NEXT ion engines. Predictions for plume structure and possible sputtering and contamination effects, as well as preliminary results for multiple plume interactions, will be presented.

Notes :

ELECTRODYNAMIC TETHER SYSTEMS for DEBRIS REMOVAL

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As a countermeasure for suppressing space debris growth, the Institute of Space Technology and Aeronautics, Japan Aerospace Exploration Agency (JAXA), is investigating a space debris removal system and assessing the viability of electrodynamic tether (EDT) technology as a high efficient orbital transfer system. A small EDT package collects electrons from ambient plasma using a bare tether and emits electrons by field emitter array cathodes that utilize carbon nano tube. This package presents a possible technique for lowering the orbit of a debris removal system without the need for propellant. To establish the EDT technology, a test flight experiment using an upper rocket is studied. First, numerical simulations are performed for a mission analysis such as available currents, Lorenz force, orbital changes, and the stabilities of the tether. The status of the trial fabrications and some evaluation test of the tether such as electric discharge are also described.

Notes :

PRELIMINARY TESTING of CARBON-NANOTUBE FIELD EMISSION CATHODES for ELECTRODYNAMIC TETHERS

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A preliminary experiment of field emission cathodes (FECs) was conducted aiming at applying for electrodynamic tethers (EDTs). The FEC is a promising electron-emitting device for space use because it is operated without working gases and has robust mechanism. The EDT can be a totally propellant-free propulsion system in orbit if the FEC is used as an electron-emitting device and a bare wire as an electron-collector. In various types of FECs, carbon-nanotube (CNT) type emitters are suitable for EDT application because CNT-FECs have high tolerance to high-voltage breakdown and are operable under relatively low vacuum condition. In the experiment, two types of multi-wall CNTs were tested; CNTs made by thermal chemical vapor deposition (CVD) and ones by arc discharge process. The electron extraction tests were performed using no gate electrodes but face-to-face flat anode electrodes. Experimental results showed that 1) an electron emission current over 120 mA was obtained from 40 mm diameter CVD-CNTs but current stability was not satisfactory and 2) a stable electron current of 2 mA was extracted from 4 mm diameter arc-CNTs. The fabrication process limited the dimensions of arc-CNTs in this case. In both the experiments, electron emission current density over 10 mA/cm2 was achieved and this current density was large enough for future EDT application. Next steps are developing gate electrodes for efficient electron extraction and testing the influences of ambient environment on FEC operation.

Notes :

Discharge Characteristic of Gamma-ray Irradiated Polybutylene Naphthalate

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Polymer materials with superior properties have been widely utilized as electrical insulation. Because the polymer containing carbon atoms in their molecular structures, most of it used for electrical insulation may suffer from dielectric breakdown. With the increasing uses of electric and electronic devices in space and nuclear power stations, those polymer insulation materials are inevitably exposed to various kinds of environments. Accordingly, it becomes necessary to investigate the influence of the pressure and radiation on insulation materials. In this research, the total dose of gamma-ray irradiation effects on the erosion depth and discharge quantity have been studied. The experiment was carried out by a dc impulse voltage under reduced pressure. Polybutylene naphthalate, which was irradiated in air up to 100 kGy and 1 MGy with dose rate of 10kGy/h using a 60Co gamma-source, has been used as the test sample. The changes of erosion depth and discharge quantity are discussed with decreasing the atmospheric pressure in the range from 100kPa to 1kPa and the frequency of applied impulse voltage in the range from 100Hz to 200Hz. It is found that both of the erosion depth and discharge quantity decreased with increasing the total dose of gamma-ray irradiation and decreasing the atmospheric pressure. The erosion depth and discharge quantity increased with increasing the frequency of applied otaimpulse voltage. A fractal analysis of discharge currents on dielectric surface was investigated. The results show that the changes in the fractal dimension depend upon the total dose of gamma-ray irradiation and the reduced pressure. Furthermore, the relationship between the fractal dimension and discharge quantity was demonstrated.

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(092fig)

Ion scattering in a Self-consistent Cylindrical Plasma Sheath

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Results are presented from a study of charged particle scattering about a charged wire in an ionospheric plasma. The one dimensional case assumes an infinite wire in an unmagnetized plasma with finite and equal ion and electron temperatures. Because particle energy and angular momentum are conserved in such a formulation, the results have the potential to provide a standard against which to compare more complicated electrodynamic tether simulations. Results indicate that higher plasma shielding limits the range of impact parameters that experience significant scattering, and that attracted particles entering tangent to the sheath experience increased scattering. The results also suggest the less intuitive result that the total scattering cross-section is also reduced for significant shielding.

Notes :

Ion Scattering in a Self-Consistent Cylindrical Plasma Sheath

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(TBD)

Notes :

LIBRESOURCE: A NEW VIRTUAL LAB for the SPINE COMMUNITY

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The ESA SPIS project (www.spis.org) has fundamentally changed the way to develop large simulation software by the resort to the Open Source strategy and a collaborative development. In this approach, the initial software development team (ESA, ONERA and Artenum) should be progressively relayed by a community of users and developers in the frame of the SPINE network and coming from both scientific and industrial communities. Because SPINE members are scattered about the world, the need of tools to work together and share data and models (source codes, inputs files, results, documents...) became quickly critical.

This was addressed by the development of a Virtual Common Laboratory available through the Web, the SPINE server (www.spis.org). This server offers a set of functionalities from a control versioning system for underdevelopment software (SPIS, PicUp3D) to data exchange areas and awareness tools (forum, mailing lists, meetings organisation). The possibility to define very refined users • rights and groups of users authorises to build various public and private areas and sub-projects related to each thematic working group. The SPINE's Virtual Lab is fully based on LibreSource (www.libresource.org), a new collaborative platform developed by the INRIA/Artenum consortium in the frame of the French RNTL program. Open Source, fully based on the JAVA J2EE technology and using standard protocols (http), LibreSource is a modular platform gathering the functionalities of community oriented platforms like Zope/Plone and development oriented platforms like SourceForge/CVS. Its design and the technological choice done allow future extensions and make LibreSource highly customisable. LibreSource includes a new and very powerful control version system (So6) issued from research models of the INRIA/ECCO team. It allows the possibility of complex software production chains (development/validation/publication). LibreSource include also possibilities of data sharing and online structured archives offering a full text indexation and a quick search. The SPINE's Virtual Lab has already outlined new methods of collaboration and opens a new way to work for heterogeneous or geographically distributed scientific and industrial communities.

Notes :

PRELIMINARY STUDY on SUSTAINED ARC due to PLASMA EXCITED by DEBRIS IMPACT on the SOLAR ARRAY COUPON

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Recently solar array has become higher in potential and larger in capacitance. Therefore, possibility of collision between space debris and enlarged solar array has been suggested. Actually, a lot of debris and dust impacts were confirmed on fuselage of recovered satellite SFU and solar array paddle of satellite Eureca. If space debris collides with solar array of an orbit satellite, it causes accrual of high-density plasma by debris impact induced and dielectric breakdown of satellite component. The phenomenon called discharge may occur. This discharge short circuit and current does not flow into a load of the satellite. And the very worst event by this discharge is operational end of the satellite. However, any events of discharge phenomenon by debris impact cannot be still confirmed. But we cannot ignore such possibility of discharge by debris impact.

The purpose of the present paper is to investigate discharge condition due to debris impact which yields us reduction of electric power of solar array, and to reduce influence of the impact on satellite missions. In this study, a solar array coupon was tested under hypervelocity impact in which a projectile was launched by a two-stage light gas gun installed in KIT. As a result, we verified discharge event in the hypervelocity impact ground test.

Notes :

Scaled-down experiment of spacecraft charging with artificial plasma emission

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In order to characterize transient charging phenomena of a spacecraft with artificial ion beam emission, charging of an electrically floated body with a small ion source was experimentally studied in a vacuum chamber. Charging rate of the beam emitting electrically floated body (dVbody/dt) was controlled by a parameter, Ib/C, where Ib corresponds to the ion beam current, and C is the capacitance of the floated body. For a wide range of Ib/C parameter, it was found that dVbody/dt=Ib/C at the beginning of the charging. Then the charging slows down because the current emitted from the body drastically decreased as a result of a space charge limited ion flow developed in front of the ion source. This space charge limited flow shows oscillatory behaviors of ions at frequencies around 10 Hz, 1 kHz , and 300 kHz, corresponding to ion movement in the space charge limited region.

Notes :

EXPERIMENTAL STUDY of MAGNETIC SAILS

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For the case of the Earth, a large-scale interaction between the Geomagnetic field and the solar wind occurs. It is intensively discussed that, as a result of such interactions, what amount of the momentum of the solar wind will transfer to the Earth. A magnetic sail realizes analogous interactions between the solar wind and an artificial magnetic field produced around a spacecraft, to obtains a drag force in the direction of the solar wind. In order to demonstrate the momentum transfer process of the magnetic sail, we design an experimental simulator of the magnetic sails that operates in a space chamber. Preliminary results showed some strong interactions between the high-density and high-velocity plasma flow and an artificial magnetic field, hence the possibility of the magnetic sail simulator is provided; however, further improvement is required to realize a collision-less solar wind plasma flow in the laboratory.

Notes :

PROBABILISTIC ANALYSIS of ISS PLASMA INTERACTION

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To date, the International Space Station (ISS) has been one of the largest objects flown in lower earth orbit (LEO). The ISS utilizes high voltage solar arrays (160V) that are negatively grounded leading to pressurized elements that can float negatively with respect to the plasma. Because laboratory measurements indicate a dielectric breakdown potential difference of 80V, arcing could occur on the ISS structure. To overcome the possibility of arcing and clamp the potential of the structure, two Plasma Contactor Units (PCUs) were designed, built, and flown. Also a limited amount of measurements of the floating potential for the present ISS configuration were made by a Floating Potential Probe (FPP), indicating a minimum potential of –24 Volts at the measurement location.

A predictive tool, the ISS Plasma Interaction Model (PIM) has been developed accounting for the solar array electron collection, solar array mast wire and effective conductive area on the structure. The model has been used for predictions of the present ISS configuration. The conductive area has been inferred based on available floating potential measurements. Analysis of FPP and PCU data indicated distribution of the conductive area along the Russian segment of the ISS structure.

A significant input to PIM is the plasma environment. The International Reference Ionosphere (IRI 2001) was initially used to obtain plasma temperature and density values. However, IRI provides mean parameters, leading to difficulties in interpretation of on-orbit data, especially at eclipse exit where maximum charging can occur. This limits our predicative capability. Satellite and Incoherent Scatter Radar (ISR) data of plasma parameters have also been collected. Approximately 130,000 electron temperature (Te) and density (Ne) pairs for typical ISS eclipse exit conditions have been extracted from the reduced Langmuir probe data flown aboard the NASA DE-2 satellite. Additionally, another 18,000 Te and Ne pairs of ISR data from several radar locations around the globe were used to assure consistency of the satellite data. PIM predictions for ISS charging made with this data correlated very well with FPP data, indicating that the general physics of spacecraft charging with high voltage solar arrays have been captured. The predictions also provided the probabilities of occurrences for ISS charging. These probabilities give a numerical measure of the number of times when the ISS will approach or exceed the vehicle plasma hazard conditions for each configuration.

In this paper we shall present the interaction mechanisms between the ISS and the surrounding plasma and give an overview of the PIM components. PIM predictions are compared with available data followed by a discussion of the variability of plasma parameters and the conductive area on the ISS. The ISS PIM will be further tested and verified as data from the Floating Potential Measurement Unit become available, and construction of the ISS continues.

Notes :

FLOATING POTENTIAL MEASUREMENT UNIT LANGMUIR PROBE TESTING and DATA REDUCTION TECHNIQUES for ISS APPLICATIONS

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The Floating Potential Measurement Unit (FPMU) is a package of several plasma instruments designed and built by Utah State University (USU) under contract to the NASA Johnson Space Center (JSC). The FPMU will be attached to the International Space Station (ISS) at camera port 2 on the end of the S1 truss to measure the local potential of the ISS relative to the plasma and to measure the local plasma properties of density and electron temperature. These measurements will be used to validate existing ISS charging models that are used in the Plasma Hazard Evaluation Process (PHEP). This process governs the hazard control of both the vehicle and the crew during Extra Vehicular Activity (EVA).

The FPMU consists of a Floating Potential Probe (FPP), a Plasma Impedance Probe (PIP), a Wide-sweep Langmuir Probe (WLP), and a Narrow-sweep Langmuir Probe (NLP) with associated electronics. The data from the FPMU will be collected via a ground station at the Mission Control Center (MCC) at JSC and transferred to the Environments Team of the Boeing ISS Program Office for data reduction and interpretation. Due to the crucial nature of this data for the PHEP, the ISS Plasma Team has developed a FPMU Contamination Test and Analysis plan. The plan will include experimental investigations as well as computational analysis. The overarching goal is to determine the impact of contamination on measurements of plasma electron temperature (Te), potential (Vsp), and density (Ne) made by Langmuir probes. Both the FPMU qualification unit and one FPMU Flight Unit, S/N 4, have been sent to the NASA Marshall Space Flight Center for testing in the MSFC plasma chambers. The test will also assess the capability to clean FPMU plasma probes. In parallel to this testing, Boeing is leading the development of the theoretical methods to be utilized to correct data from contaminated Langmuir probes.

This paper describes the analysis methodology used to correct contaminated FPMU Langmuir probe data.

The results are directly compared with the results obtained from a clean reference Langmuir probe, where the corresponding data was taken during the same test. Strategies for the on-orbit checkout and verification of the FPMU after ISS installation, such as using Incoherent Scatter Radar data for comparison, will also be discussed.

Notes :

VEHICLE CHARGING on a SOUNDING ROCKET PAYLOAD

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The rocket investigation "Scattering Layer in the Bottomside Equatorial F-region Ionosphere" was part of the NASA EQUIS II campaign. Two salvos of sounding rockets were launched from Roi Namur in Kwajalein on August 7th and 15th of 2004. Each of the salvos consisted of one instrumented and two chemical release payloads. The instrumented rockets were launched westward into equatorial spread F precursor that was first observed from ground using the Altair radar. The instrumented rockets reached an apogee of ~421 km. The instruments consisted of an internally heated Sweeping Langmuir Probe (SLP), a fixed bias DC Probe (DCP), a Plasma Impedance Probe consisting of a Plasma Frequency Probe and a Plasma Sweeping Probe built at Utah State University.

The instrument suite also included an Electric Field Probe built by Penn State University. The ratio of the SLP area to that of the payload skin was on the order of ~120. This led to a systematic fluctuation of the payload floating potential as SLP swept from -1 V to +5 V. This paper presents an analysis of the rocket surface charging and extraction of temperature from 'warped' Langmuir Probe sweeps.

Notes :

THE ISS FLOATING POTENTIAL MEASUREMENT UNIT

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The Floating Potential Measurement Unit (FPMU) is an instrument developed to study the spacecraft surface charging of the International Space Station (ISS). It has been developed by The Utah State University Space Dynamics Lab under contract to NASA. Vehicle charging of the ISS is an interesting problem due to its extended size and the use of high voltage solar arrays with exposed interconnects. The FPMU consists of four instruments, a floating potential probe, two Langmuir probes and a plasma impedance probe. These probes will measure the floating potential of the ISS, electron density, and electron temperature with redundancy. The FPMU is being integrated into the International Space Station at one of the existing external camera locations that places it in clear ram flow of the space plasma. Operational constraints of the ISS will result in the FPMU being used to obtain snapshots of data and not as a continuous monitor of the ISS charging and environment. Currently the FPMU is awaiting launch when the Space Shuttle returns to service. This paper presents an overview of the FPMU instrument and calibration.

Notes :

PLASMA IMPEDANCE PROBE DIAGNOSTICS: MODEL and DATA

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The electrical impedance of an antenna exposed to the space environment is dependent upon the parameters of the space plasma in which it is immersed. This effect can be used as a diagnostics for electron density, collision frequencies, etc. Utah State University has recently flown a number of Plasma Impedance Probes on sounding rockets with NASA including dipoles, monopoles, and patch antennas. There are several analytic theories for the impedance of monopole or dipole antenna in a space plasma. There are no analytic theories for a patch antenna. Utah State has developed a Plasma Fluid Finite Difference Time Domain (PF-FDTD) simulation that can be used to model various antenna geometries. Antenna impedance data from various geometries are presented and compared with analytic and the PF-FDTD simulation. Preliminary results of the extraction of electron density, electron neutral collision frequency, and electron temperature along the rocket trajectory are presented.

Notes :

OPTIMIZATION of SPACECRAFT CHARGING MITIGATION REQUIREMENTS

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Spacecraft Charging Mitigation is an art form that dates back to the 1960's. The most common practice involves controlling the resistance per unit surface area, and/or the resistance per unit volume of dielectric materials. This is implemented with proper bonding to reference ground to provide a charge bleed-off path.

We show how grounding configuration affects the effectiveness of the mitigation method, and we propose a general rule-of-thumb that guarantees optimal results.

Notes :

SCALING LAWS for PULSE WAVEFORMS from SURFACE DISCHARGES

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Since the 1970's, Balmain, Bogorad, and others proposed seemingly different scaling laws for using small-scale laboratory test results to estimate the peak of current transients from electrostatic discharges on thermal blankets, solar panels, and other large dielectric surfaces on satellites in earth orbit.
We derive a general scaling law and show that the scaling laws proposed by Balmain, Bogorad, and others are special cases. Moreover, we show that for arbitrary shape of the surface area of a dielectric, one can predict both the pulse shape, IESD(t), and the rate of change, dIESD(t)/dt, of an ESD-induced transient.

Notes :

Notes :

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