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Contents

7. Common Approach to Solving SGEMP, DEMP, and ESD Survivability

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Abstract

System Generated Electromagnetic Pulse (SGEMP) and Dispersed Electromagnetic Pulse DEMP) are nuclear generated spacecraft environments. Electrostatic discharge (ESD) is a natural spacecraft environment resulting from differential charging in magnetic substorms. All three phenomena, though differing in origin, result in the same problem to the spacecraft and that is Electromagnetic Interference (EMI). A common design approach utilizing a spacecraft structural Faraday Cage is presented which helps solve the EMI problem. Also, other system design techniques are discussed which minimize the magnitude of these environments through control of materials and electrical grounding configuration.

1. INTRODUCTION

The common design approach of the Abstract is applicable to any high altitude spacecraft in an elliptical or synchronous orbit whose altitude subjects it to the spacecraft charging environment and also has a nuclear survivability requirement.

The nature of the ESD, DEMP, and SGEMP environments and the EMI concern is discussed as follows.

2. A COMPARISON OF THE ESD, DEMP, AND SGEMP ENVIRONMENT AND SPACECRAFT PERFORMANCE CONCERNS

2.1 ESD (Electrostatic Discharge)

Electrostatic charging of synchronous spacecraft results from a natural radiation of charged particles collecting on the spacecraft surfaces. Electrostatic discharge occurs when the differential charging of the spacecraft surface materials exceed their dielectric breakdown strength. The resulting system effects of the discharge are electromagnetic interferences such as circuit upset and burnout. ESD occurs in a five-step process. First, a magnetic substorm results in an injection of charged particles, ions and electrons, into the dusk-to-dawn sectors of local time and roughly from 4 to 8 earth radii. Magnetospheric effects cause preferential eastward drift of electrons into the midnight-to-dawn quadrant while ions drift westward. Appreciable electrons have been observed (ATS-5) at 5-30 keV.

The second step is the encounter of the spacecraft surfaces with the hot, negative plasma. If a surface is self shadowed or in eclipse, it will charge to approximately a potential equivalent to the most probable energy of the electron energy distribution, less a potential drop corresponding to secondary electron emission from the surface. If the surface is in sunlight, photoelectric emission will prevent it from charging to a negative potential and in fact the surface may go a few volts positive. It is apparent that different surfaces, due to different exposure to sunlight (and hence photoelectric discharge) and different secondary electron emission and photoemissive properties, will charge to different potentials. Thus step 2 results in differential voltages of several thousand volts appearing at different sites on the spacecraft's exposed surfaces.

The third step is electrostatic discharge whenever the differential potential exceeds the dielectric breakdown of the material.

The fourth step is the electric or magnetic field coupling from the discharge arc into spacecraft harnesses or the irradiation of antenna assemblies associated with the arc (see Figure 1).

The fifth step is the induction of a transient pulse into the circuit with sufficient magnitude to activate the circuit or burnout some of its components, or communication and telemetry interference.

Other possible effects than circuit upset or burnout is direct damage to thermal control surfaces resulting from the arc, discharge, and contamination to surfaces.

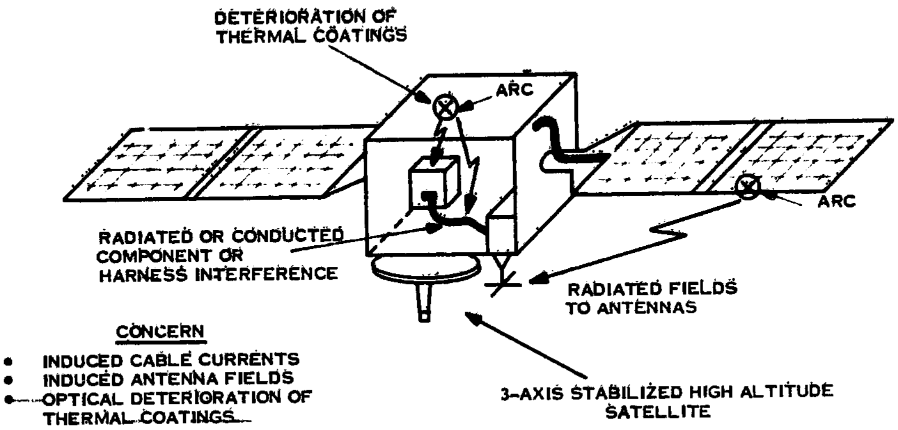


Figure 1. Arcing Induced Satellite Environment Due to S/C Charging

2.2 DEMP (Dispersed Electromagnetic Pulse)

A nuclear weapon detonated in or near the atmosphere generates a copious stream of Compton-electrons. Part of the latter constitutes a time-changing, nonradial current which in turn produces propagating electromagnetic fields. At the spacecraft, the latter have propagated through the ionosphere which acts as a high-pass filter and thus only frequencies above a certain cutoff are observed. Also, frequencies that do propagate to the spacecraft are dispersed and arrive at different times. Impingement of the DEMP on the spacecraft structure and antennas induces structural currents which in turn couple electromagnetic fields into harnesses and communications receiver front ends. This results in EMI and RFI similar to spacecraft arcing (ESD). (See Figure 2).

2.3 SGEMP (System Generated Electromagnetic Pulse)

SGEMP, strictly speaking, is not an external environment but rather a secondary environment produced primarily by the interaction of x-rays with the surfaces and harnesses of the spacecraft. SGEMP can be classified as (1) direct and (2) indirect. The direct refers to coupling of X-rays directly into cables and electronic components. The indirect SGEMP is generated by a two-step process in which first x-rays impinge on surface materials and release secondary electrons through Compton and photoelectric processes. The secondary electrons constitute an accelerated charge and hence generate propagating electric and magnetic fields. The latter couple into cables and circuits according to the particular coupling coefficients that apply to the wavelength of the fields and the geometry of the cables and circuits. Again, transient upset and burnout of the electronics result (See Figure 3).

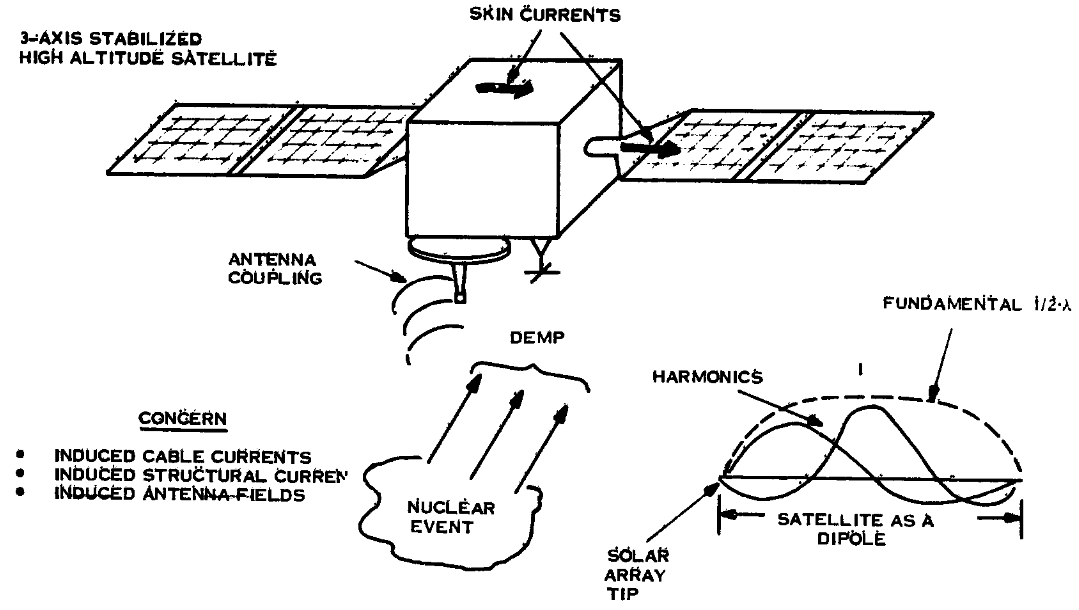


Figure 2. DEMP Induced Satellite Environment (Dispersed Electromagnetic Pulse)

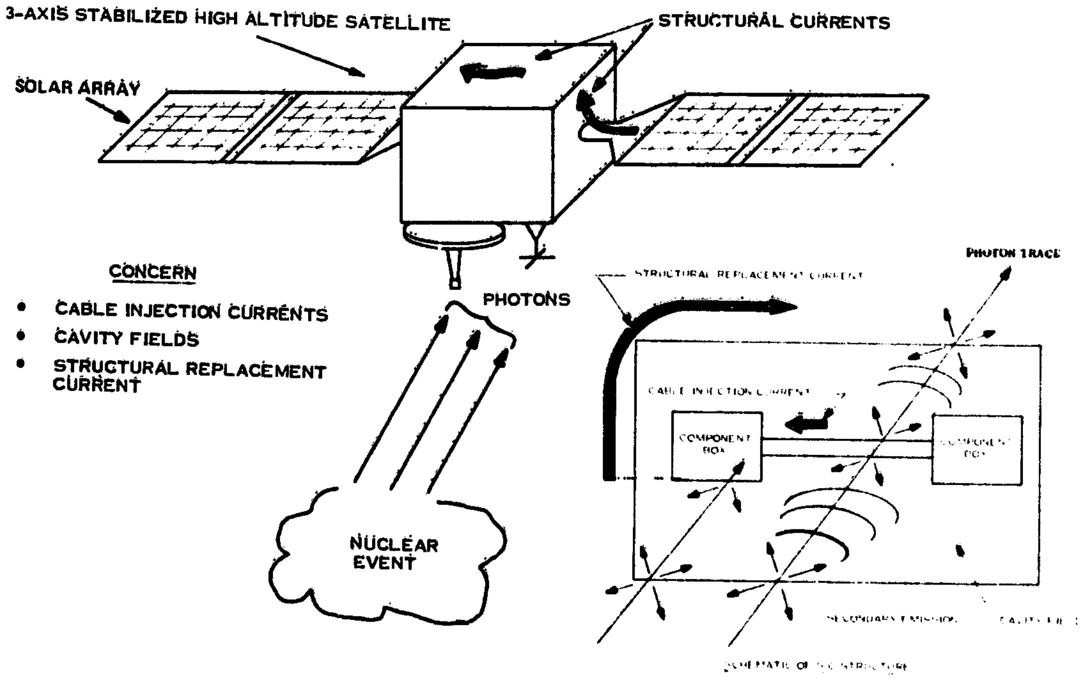


Figure 3. SGEMP Induced Satellite Environment (System Generated EMP)

3. A COMPARISON OF ESD, DEMP, AND SEMP ELECTROMAGNETIC SIGNAL CHARACTERISTICS

Figure 4 illustrates the commonality of the electromagnetic signal characteristics in both the time and frequency domains.* From Figure 4 it can be seen that the fast rise in all three cases produces significant energy in the frequency spectrum out to ~100 MHz and then rolls off at 40 dB/decade. The similarity in the spectrum makes it possible and desirable to find a common design technique which addresses all three phenomena at one time. The common design technique proposed is to design the spacecraft structure such that it encloses the electronics and harnessing in the form of a Faraday cage. Figure 5 illustrates the classical presentation of how the Faraday cage works as an EMI shield. Figure 6 illustrates the comparison between the theoretical shielding effectiveness of an ideal Faraday cage and also what is achievable in practice. Shielding effectiveness in an actual spacecraft is limited by physical construction of the structure which necessitates bonding, riveting, and bolting of structural subassemblies, thus producing metal-

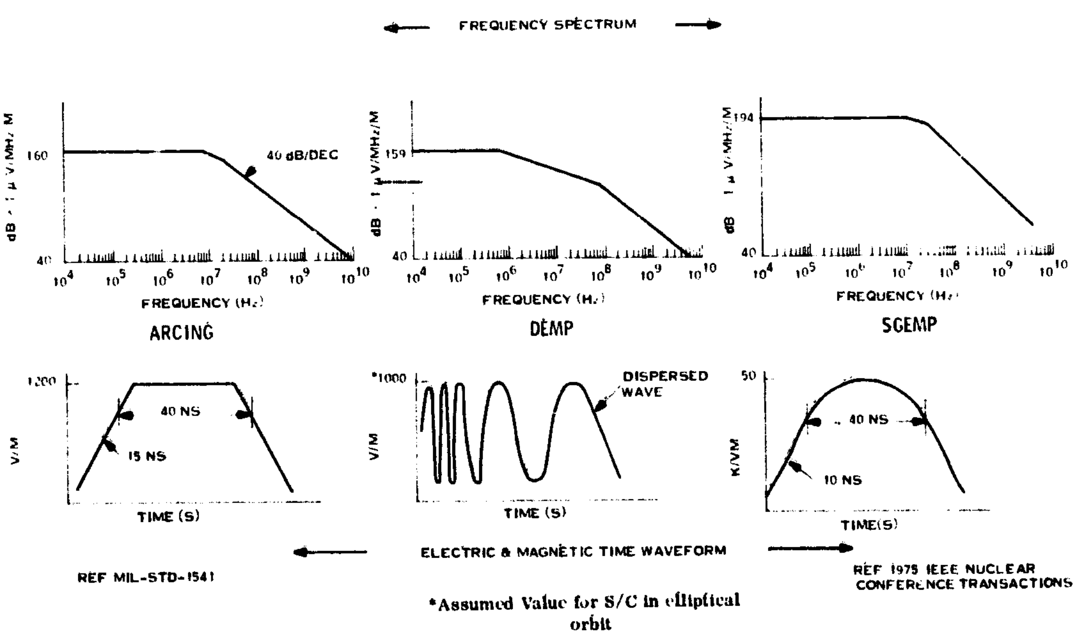


Figure 4. Environment Electromagnetic Signal Characteristics

*The signal characteristics of the photon flux and DEMP and the response levels of the satellite cables and structure have been obtained or derived from the 1974, 1975 and 1976 IEEE Annual Conference Transactions on Nuclear and Space Radiation Effects.

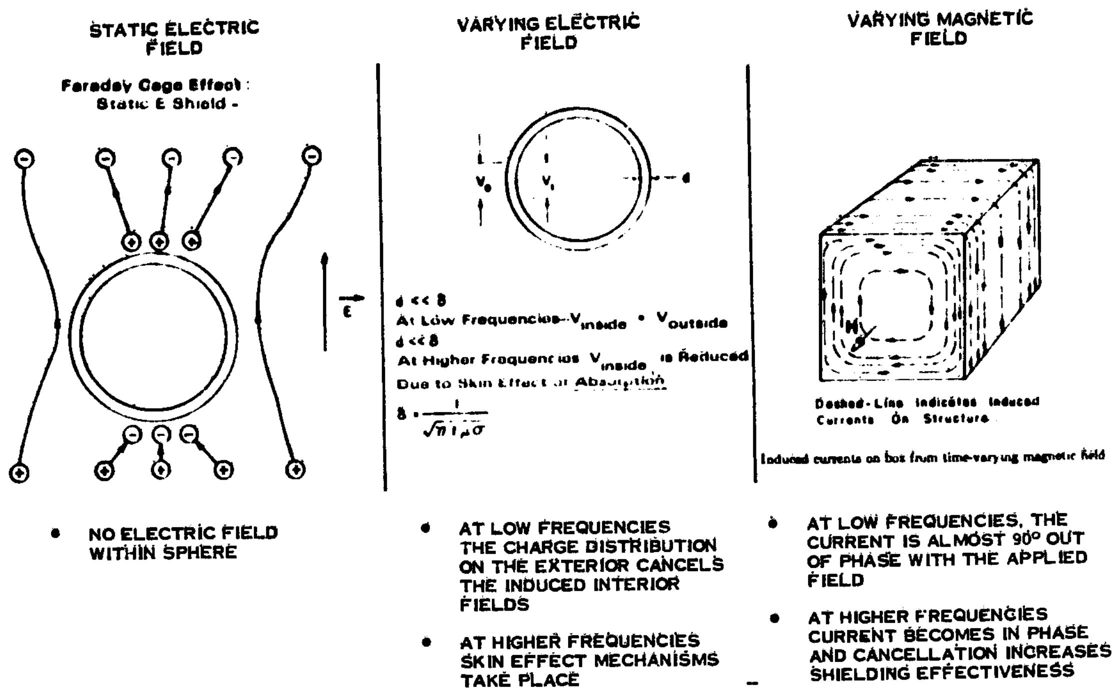
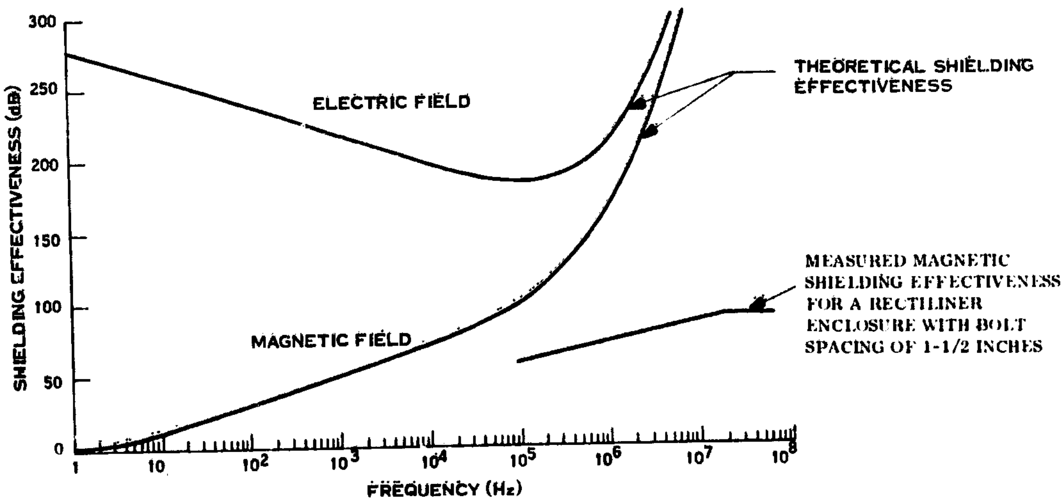


Figure 5. Faraday Cage Shielding Effects



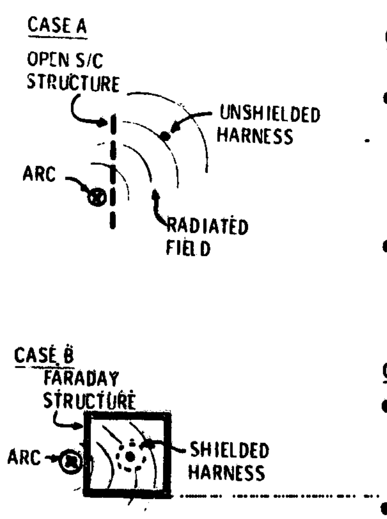
ELECTRIC AND MAGNETIC SHIELDING BEHAVIOR OF A SOLID SHELL ENCLOSURE

Figure 6. Faraday Cage Shielding Effectiveness

to-metal seams allowing electromagnetic leakage. Rivet spacing, number of harness penetrations, access parts, etc., all influence the levels of shielding effectiveness. Fortunately, in terms of spacecraft weight considerations, the use of metal foils conductively bonded between structural elements makes nearly as an effective shielding as does a solid sheet metal enclosure. The actual magnetic shielding effectiveness of a bolted enclosure (1-1/2 in. bolt spacing) is shown in Figure 6. A 30 to 50 dB magnetic shielding effectiveness is achievable.

4. COMPARISON OF EMI LEVELS FOR SHIELDED AND UNSHIELDED CASES

Figures 7 - 9 illustrate in principle how the Faraday shielding attenuates the interferring signal characteristics of Figure 4. It can be seen from the last items in Figures 7 and 8 that the resultant magnitude of the interferring signal both in the ESD and DEMP cases is reduced below circuit component burnout damage levels. The magnitude of the attenuated signal may, however, still be sufficient to cause interference (~0.5 volts). With particularly sensitive digital logic circuits or mission critical functions pulse width discrimination should also be considered as a candidate for inclusion in the EMC design.



CASE A UNSHIELDED CONFIGURATION

- FOR: 1 SQ. FT. KAPTON THERMAL BLANKET; 0.3 JOULE, 10 KV, 40 ns DISCHARGE; 1/3 OF ENERGY CONDUCTED TO STRUCTURE
  THE ARC CURRENT IS:

  $\boxed{I_s = 250 \text{ AMPERES}}$

- FOR: 1 METER UNSHIELDED #AWG 20 WIRE 1-3/4 INCHES ABOVE STRUCTURE
  THE STRUCTURAL CURRENT INDUCED VOLTAGE INTO THE WIRE IS:

  $\boxed{V_w = 6900 \text{ VOLTS}}$

CASE B SHIELDED CONFIGURATION

- IF THE FARADAY CAGE SHIELDING EFFECTIVENESS IS ≥ 40 dB, THE RESULTANT STRUCTURAL CURRENT IS EQUIVALENT TO:

  $\boxed{I_s \; 2.5 \text{ AMPERES}}$ INSIDE THE CAGE

- IF IN ADDITION, THE WIRE HARNESS IS SHIELDED WITH A SHIELDING FACTOR OF 40 dB, THE RESULTANT CABLE INTERFERENCE FROM THE ARC IS:

  $\boxed{V_w = 0.69 \text{ VOLTS}}$

Figure 7. Shielding for Electrostatic Discharge

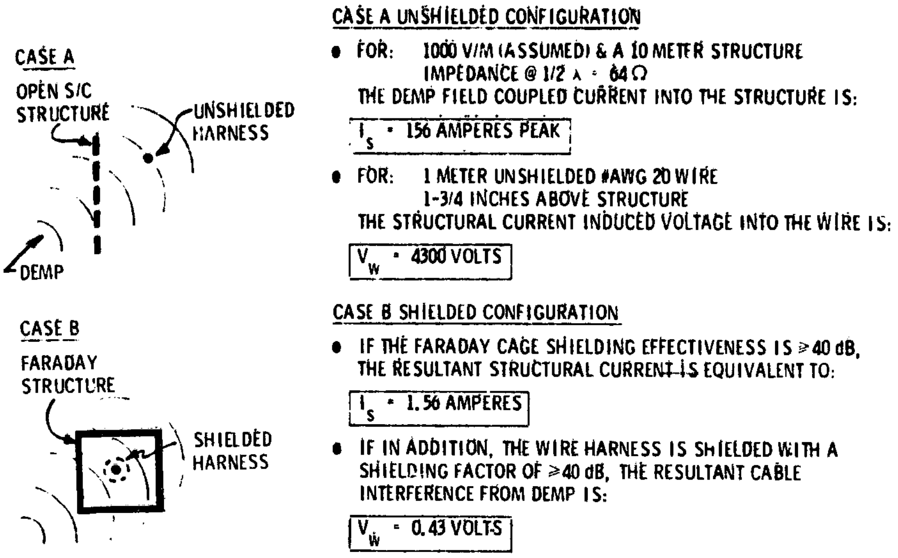


Figure 8. Shielding for DEMP

Figure 9 (the SGEMP shielding case) is a more complex case in that the interferring cable currents are obtained by three different modes:

(1) Coupling to the cable from structural replacement currents.

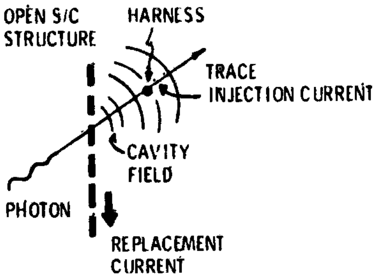
(2) Coupling to the cable by cavity fields.

(3) Direct current injection by x-ray impingement.

Harness and Faraday cage shielding are both effective at suppressing EMI from the structural replacement currents (I_{C2}). Harness shielding, only, is effective at suppressing cavity field induced currents (I_{C1}). Neither harness shielding or Faraday cage shielding are effective suppressing EMI from direct cable x-ray current injection (I_{C3}).

In this latter case, cable current injection must be minimized through the use of low atomic number materials, dielectric shielding, bundling of cable wires, etc. The EMI protection for this particular problem must be addressed by using circuit terminating devices which will limit the inrush current below burnout levels or voltage clamp the input below damage or upset levels.

CASE A



CASE A UNSHIELDED CASE

- IF PEAK FIELD INTENSITY FROM SGEMP IS 1 TO 50 AMPERES/METER PEAK STRUCTURAL CURRENT FOR A 2 METER STRUCTURE IS:

  $\boxed{I_s = 2 \text{ TO } 100 \text{ AMPERES PEAK}}$

- IF CAPACITANCE OF WIRE 1-3/4 INCHES OFF OF STRUCTURE = 2×10^{-11} F/M AND PULSE RISE TIME OF 50 KV/M FIELD = 15 ns, THEN THE INDUCED CURRENT INTO THE WIRE FROM CAVITY FIELDS IS:

  $\boxed{I_{c1} = 60 \text{ AMPS PEAK}}$

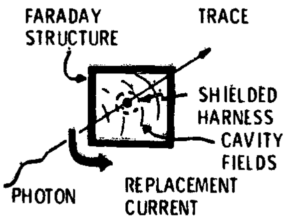
- IF THE STRUCTURAL REPLACEMENT CURRENT IS 100 AMPERES/METER AND THE METER CABLE IS 2 INCHES AWAY

  $\boxed{I_{c2} = 0.66 \text{ AMPERES/METER}}$

- IF THE CABLE TEST DATA PRESENTED AT THE 1976 IEEE NUCLEAR & SPACE RADIATION CONFERENCE BY CLEMENT, WULLER & CHIVINGTON IS SCALED UP TO LEVELS BELOW WHERE THERMAL MECHANICAL SHOCK BECOMES THE OVERRIDING CONSIDERATION, THEN THE INJECTION CABLE CURRENT IS:

  $\boxed{I_{c3} = 1.4 \text{ AMPERES/METER}}$

CASE B



CASE B SHIELDED CONFIGURATION

- CABLE CURRENT DUE TO CAVITY FIELDS
  = I_{c1} - 40 dB CABLE SHIELDING EFFECTIVENESS

  $\boxed{I_{c1} = 0.60 \text{ AMPERES PEAK}}$

- CABLE CURRENT DUE TO STRUCTURAL REPLACEMENT CURRENT AFTER SHIELDING =

  I_{c2} (UNSHIELDED) - 40 dB (FARADAY SHIELD) - 40 dB CABLE SHIELD

  $\boxed{I_{c2} = 0.66 \times 10^{-4} \text{ AMPERES/METER}}$

- CABLE CURRENT DUE TO CABLE INJECTION (AFTER SHIELDING & INCLUDING DIELECTRIC LINING)

  $\boxed{I_{c3} = 1.4 \text{ AMPERES/METER}}$

Figure 9. Shielding for SGEMP

5. OTHER SYSTEM DESIGN TECHNIQUES FOR REDUCTION OF EMI FROM ESD, DEMP AND SGEMP

5.1 ESD EMI Reduction Techniques

The spacecraft discharge phenomenon can be controlled by reducing the differential potential buildup between various outer surface thermal blankets and coatings and the metallic spacecraft structure. This goal is achievable through the use of conductive paints and thermal blanket materials such as astroquartz cloth which has low surface resistance when bombarded by electrons.

In addition, possible discharges between the various inner metallic layers of thermal blanket materials and the resulting degradation of the thermal properties of the blanket can be eliminated by connecting all blanket metallic layers and grounding the composite blanket to the metallic structure. All structural members can also be electrically interconnected to share a common ground potential for prevention of differential structural voltages. All apertures such as the earth coverage antennas can be covered with dielectric/thermal materials (astroquartz cloth) which exhibit high levels of surface and throughput leakage, thereby preventing large charge buildup. In those cases where discharges may not be eliminated, such as on the glass sections of the solar arrays, then the line-of-sight to communications antennas should be eliminated to prevent RFI. Table 1 lists spacecraft design guidelines for preventing arcing.

5.2 DEMP and SGEMP EMI Reduction Techniques

Principal areas of concern from nuclear effects are system transient upsets and permanent degradation of parts and materials. Prevention of damage to piece parts and materials is accomplished through hardened circuit design, nuclear and electromagnetic shielding, and the use of hard materials. All materials, including critical external thermal control materials, must be carefully selected to prevent any significant x-ray induced thermal-mechanical effects and to adequately withstand the natural radiation environments.

The system transient upset and recovery requirements are met by functionally configuring each subsystem to minimize the impact at the component and circuit level. The system is allowed to respond as much as possible without causing undesirable system effects and to functionally recover within the desired time period. This is achieved by ensuring fast circuit recovery through proper piece-part selection and circuit design, preventing the generation and execution of false commands, logic upset, and the use of hardened data storage where required.

Table 2 itemizes design controls for minimizing EMI from SGEMP and DEMP.

Table 1. Spacecraft Design Guidelines for Preventing Arcing

1.	The graphite epoxy used for structural members is textured to be conductive and presents minimum discontinuity in structural ground connections.
2.	All epoxy and other nonmetallic structural bonding adhesives are conductive and present minimum discontinuity in structural ground.
3.	Solar array panels are grounded to each other with grounding jumpers.
4.	Solar array panels are grounded to the spacecraft structure through special slip rings, providing a one milliohm path.
5.	All antennas and support structures are grounded to the main structure.
6.	All electrical components and subsystems are grounded.
7.	Spacecraft thermal blanket materials and coatings have been selected which have high surface leakage and bulk leakage.
8.	All external cable harnesses are shielded and the shield is connected to structural ground at both ends.
9.	All apertures, including RF antenna apertures, are covered with high surface leakage silica cloth composites.
10.	All waveguide elements are electrically connected with spot weld connections and grounded to the main frame.
11.	The support members of all antennas will be connected to the spacecraft structure with conductive epoxy such that each support joint represents approximately a one ohm connection to structure.
12.	All deposited thin film conductors (in thermal blankets or otherwise) shall have a ground strap of sufficient area to carry the transient current loads expected (a 2 joule rating is self-applied).
13.	Electrical resistance from any point on vacuum-deposited conductive films (in thermal blankets or otherwise) to spacecraft structural ground shall not exceed 10 ohms.
14.	Electrical resistance from any point on a thermally isolated substructure to spacecraft structural ground with required grounds in place shall not exceed .01 ohms.
15.	There shall be at least one grounding point on each electrically continuous substructure.

Table 1. Spacecraft Design Guidelines for Preventing Arcing (Cont'd)

16. The electrical resistance of each ground strap bond between the strap bond and the structure shall not exceed .03 ohms.

17. Thermal blanket conducting straps to ground shall be electrically equivalent to a copper conductor of wire of AWG #22.

18. Each structural ground strap shall be electrically equivalent to a copper conductor of wire of AWG #14.

19. Redundant logic is employed in command and other sensitive logic and receiver circuitry.

20. The solar array consists of a honeycomb aluminum base structure with the following layers of materials: coverglass, solar cells, and mica-ply substrate and graphite epoxy. All lateral strips or rows of cells are bonded together by a ground wire at each end of the solar panel, such that the resistance between any two rows of cells does not exceed 5 milliohms.

21. The solar cell string is electrically connected to spacecraft structural ground at the shunt regulator and via natural capacitive paths.

22. All outer solar panels are connected to the inner solar panels by ground wires.

23. Thermal windows on north panel (N.P.) and south panel (S.P.) are covered with second surface mirrors consisting of OSR glass with silver coating on one side. The OSR glass is attached to the panel surfaces with conductive epoxy.

24. Component enclosures and chassis are designed to provide an RF attenuation of 50 dB to radiated fields produced by ESD.

25. All internal and external cables are shielded on a cable bundle basis.

26. Cable shields will be multipoint grounded to the spacecraft structure by low impedance connections.

27. Mounting hardware used to bolt or fasten components to the spacecraft structure shall also serve as ground bonding paths.

28. Nonconductive finishes such as anodized surfaces or painted surfaces will not be used on any of the grounding interfaces.

Table 2. Design Controls for Minimizing EMI from SGEMP & DEMP

DESIGN IMPLEMENTATION	SYSTEM RESPONSE
System Level	
1. All electronics in Faraday Cage >40 dB attenuation	1. Reduces external SGEMP/DEMP field coupling to internal harness
2. Controlled structure and penetration impedances <10 milliohms	2. Controls skin current and the placement current flow minimizes internal coupling
3. Low Z surface materials	3. Minimizes secondary electron emission, reduces external fields and structural replacement currents
Subsystem Level	
1. Internal cavities coated to control secondary emission	1. Minimizes IEMP fields and replacement currents
2. Optimum grounding (Single and Multipoint)	2. Reduces replacement current coupling effects
3. Harness and box RF shielded to >40 dB	3. Reduces cavity field coupling effects
4. Harness design for minimum direct X-ray response	
a. Multiconductor bundle cables instead of flexible coax	4. a. Minimizes weight and SGEMP response
b. Aluminum RF shield plus inside dielectric liner over cable bundles	b. Reduced direct X-ray response
c. Controlled cable routing to avoid replacement current funnel points	c. Minimizes replacement current coupling
d. Multipoint shield grounds	d. Minimizes current coupling transient response
Box Level	
1. All interface and buried circuits protected with terminal protection circuits as required	1. Prevents circuit burnout
2. Circuits designed for maximum practical burnout threshold	2. Minimizes need for terminal pro protection
3. All boxes RF tight to > 40 dB	3. Minimizes cavity field coupling response

6. CONCLUSIONS

EMI from ESD, DEMP, and SGEMP has similar time and frequency domain signal characteristics. Thus, a common design approach to prevent EMI can be implemented through the use of a spacecraft structure configured as a Faraday cage shield. Harness shielding, integral structural grounding, and materials control are also common design features for the minimization of the ESD, DEMP, and SGEMP interference. Figure 10 and Table 3 illustrate the integrated design approach.

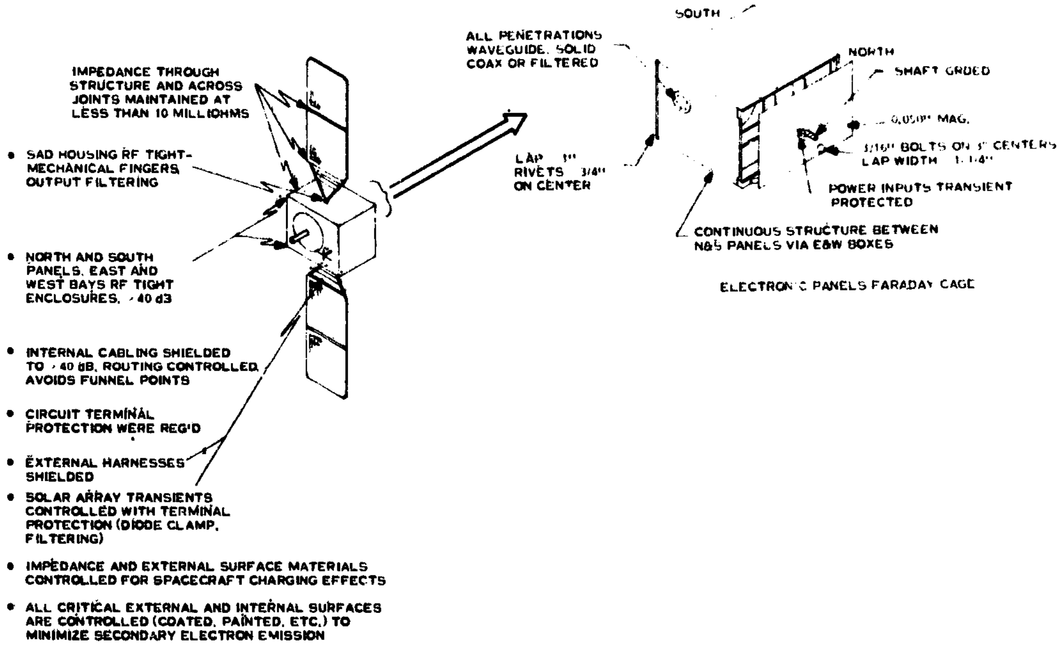


Figure 10. Structural and Materials Implementation of Common Design Approach to SGEMP, DEMP, and ESD Survivability

Table 3. Integrated Survivability/EMC/Spacecraft Charging Design Approach

COMMON DESIGN FEATURE	ENVIRONMENT		
	SGEMP/DEMP PREVENTION & PROTECTION	S/C CHARGING PREVENTION & PROTECTION	EMC BETWEEN SUBSYSTEMS
Structure configured to form a Faraday Cage EMI/RFI shield	Electronics and harness protected from EMP field SGEMP induced structural currents quickly damped	Electronics and harness protected from arcing EMI products	Electronics and harness protected from RFI from transmit antennas
All antennas, solar arrays, structure and electrical components share common low impedance ground	Differential circulating current magnitude minimized	Differential structural voltages eliminated	Shielding effectiveness optimized
All component boxes RF tight	Protects component from cavity fields	Protects components from arcing EMI products	Protected from conducted and radiated S/S interference
All harnesses aluminum shielded	Protects component from cavity fields	Protects components from arcing EMI products	Protected from conducted and radiated S/S interference
All harnesses bundled configured by signal type	Minimizes direct SGEMP current effects	---	Protected from mutual signal interference
Multipoint grounding system	Best grounding configuration for high frequency transients	Best grounding configuration for high frequency transients	Best grounding configuration for RFI protection
All harness entries in and out of structure RF tight	Prevents DEMP fields from being coupled to electronics	Prevents EMP fields from being coupled to electronics	Prevents RF fields from being coupled to electronics
Materials configuration	Low "Z" materials minimizes secondary electron emission effects • Aluminum/magnesium structure • Silica cloth thermal blankets • OSR • Chemglaze	Silica cloth prevents charging and provides thermal control	---