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9. Qualification Model Spacecraft Tests for DEMP, SGEMP, and ESD Effects

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1. INTRODUCTION

This paper describes the development of a satellite design demonstration test program. The test is planned in lieu of a dispersed electromagnetic pulse (DEMP) test at the ARES facility. The test approach is comprehensive in that it includes the effects from electrostatic discharge (ESD), system generated electromagnetic: pulse (SGEMP), as well as DEMP. The comprehensive test concept is based on the similarity of the satellite's, responses to the several environments.

The original FLTSATCOM hardness demonstration apprbach was to test the qualification model spacecraft in the ARES facility to a simulated DEMP environment. The revised test concept focuses on inducing the predicted effects from several environments, rather than exposing the system to a simulation of only one of the environments.

The approach has some risks because one must not only predict the environment but also the effect of the environment. However, the alternate approach, simulating tho environment, also had risks. The compromises which nro made in the construction of simulators lend nt best to an approximation of the desired ohvlranmont. Thus, not only arc simulators themselves approximations, but tho cost associated with a sequence of tests at several different simulators will be large. It was decided to assess system survivability by using n sequence of effect simulations Prom the several predicted electromagnetic environments.

Tho degree to which test objectives can be satisfied by an alternative test approach was fundamental in the test selection process. The purpose of these tests is to demonstrate proper operation while the system is subjected to the predicted effects of the environmbnts. This "simple" objective was motivated by the fact that a complex sequence of subsystem responses and interactions constitutes "system operation." As long as the objective is satisfied, some liberties with the environmental simulation itself may be justified. This is particularly valid if environmental interactions lend themselves to first order coupling analysis involving some operationally passive subset of the total system (Faraday cage structure).

2. SYSTEM DESIGN FEATURES

Figure 1 depicts the major FLTSATCOM communication customers. FLTSATCOM is being built by TRW uhder SAMSO contract, nith the Navy acting as the funding and executive agency. Both the Navy and the Air Force will enjoy the benefits from FLTSATCOM capabilities. The Navy high priority FLTBROADCAST mode consists of a single (redundant) channel providing communication from command facilities to the fleet at large. Similarly, the Air Force is assured a critical communication link with its strategic forces. Both services are furnished an additional number of routine communication channels, the relay channels. Finally, FLTSATCOM provides a wideband channel intended for presidential communications.

FLTSATCOM's construction features, characteristic of a three-axis stabilized spacecraft, are shown in Figure 2. Both receive and transmit antennas, as well as the solar panels, deploy after separation from the ATLAS-CENTAUR launch vehicle. Thermal control of the vehicle is provided by second surface mirrors, paints, and multilayered aluminized mylar and kapton blankets. Significant for the following discussion is the fact that the vehicle itself is constructed as a leaky Faraday cage, The lower, or equipment bay, internally deparated from the upper,

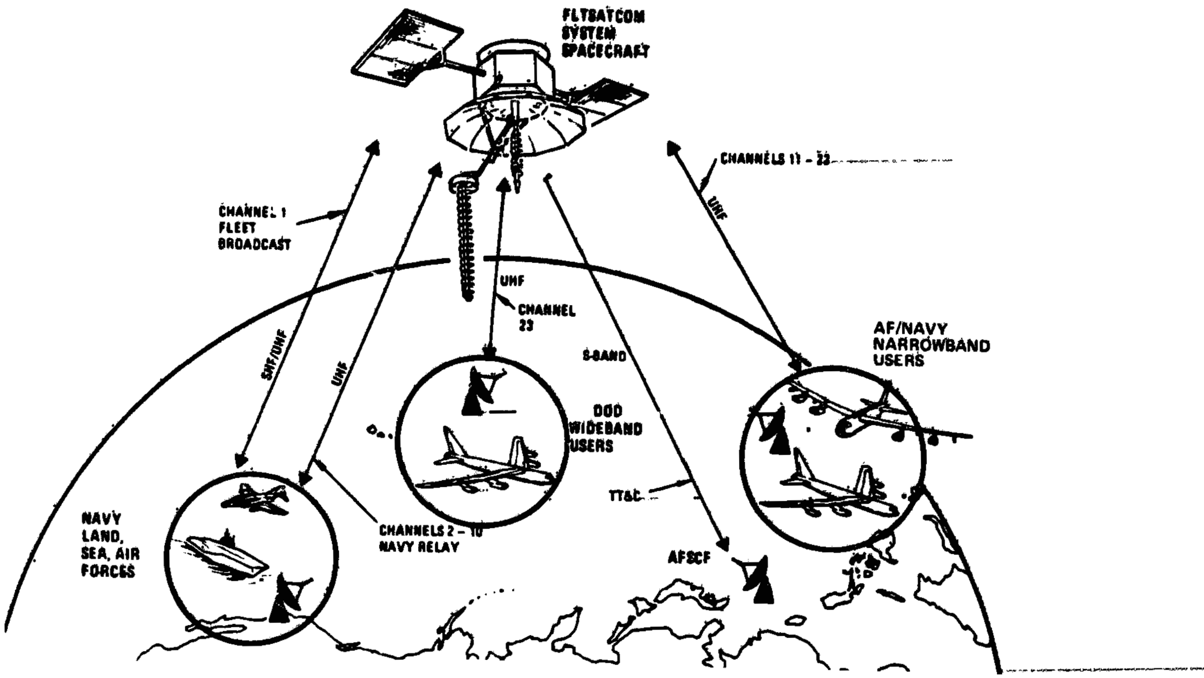


Figure 1. FLTSATCOM Communications Customers

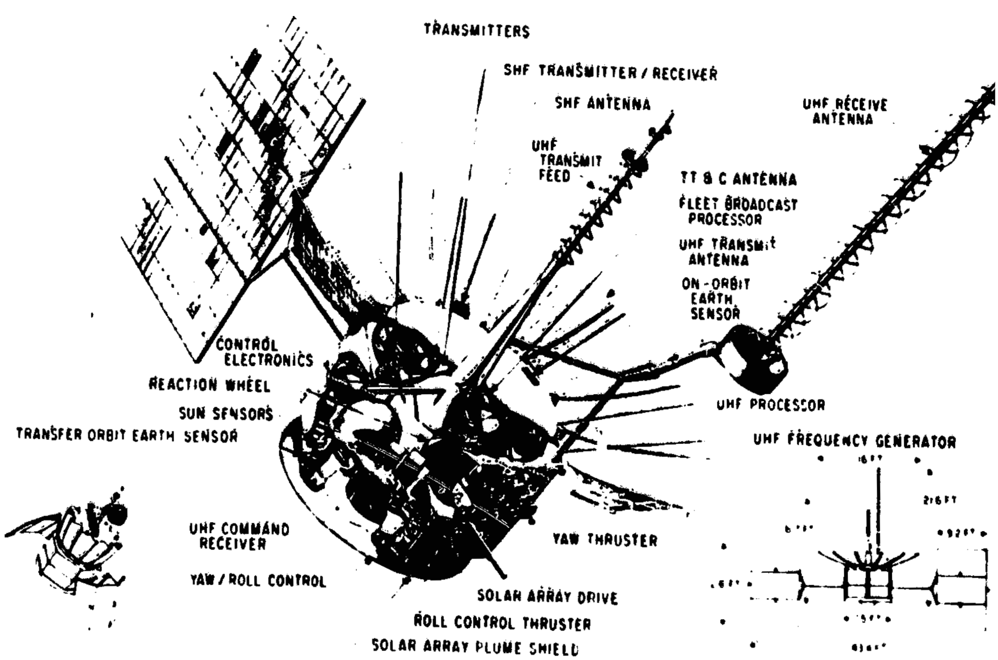


Figure 2. FLTSATCOM Design Features

or payload bay by a shelf, consists of honeycomb panels and a titanium lined rocket motor acting as close-out for electromagnetic signals. The payload bay, housing the transmitters and receivers, is formed by the separating shelf, similar honeycomb sidepanels and the UHF antenna Center dish (connected via fbur braided groundstraps tu the panels). Primary areas of concern from a spatecraft charging point consist of the solar' panel surfaces and the dielectric surfaces of the thermal blankets and mirrors covering the sides. Major pbints of entry for electromagnetic energy are the Solar boom penetratibn points and the thermal blanket covered area between the spacecraft body and the large UHF antenna.

3. TEST DEVELOPMENT CONSIDERATIONS

Figure 3 illustrates the significant electromagnetic environments. Electrostatic discharge results from the build up of electrons on the surface of dielectrics such as the thermal blankets. When the electric field builds up tb the material's breakdown strength, an arc occurs through or at an edge or corner of the thermal blanket. In response to a DEMP, structural currents are edited by the incident electromagnetic wave. In addition, when the spacecraft antennits are tuned to

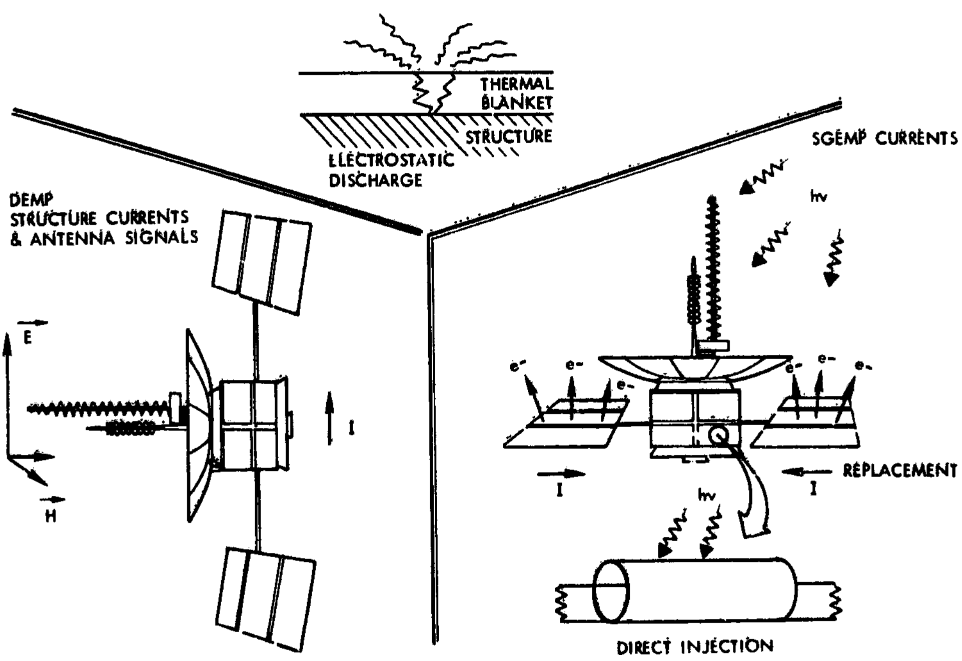


Figure 3. Spacecraft Electromagnetic Environments

frequencies present in the DEMP spectrum, currents are Coupled directly into the communications electronics. There are at least three modes of SGEMP response: external replacement currents, direct injection currents ob cables, and internal cavity fields. Figure 3 shows the external replacement currents caused by the redistribution of electrons on the spacecraft. Incident photons liberate electrons causing a charge imbalance which must be equilibrated. Direct injection currents result from the?interaction of photons with cables. These latter currents will dominate in a well rf shielded system.

For the different environments, key common parameters used to develop a comprehensive test are summarized in Table 1. Note in particular the fact that no requirement to design for the spacecraft charging/discharging environment had been placed on the contractor. Note also that the environment now generally accepted to result in the smallest coupled currents was a design requirement, and the only environment against which the design was to be tested. Peak currents estimated to flow on external surfaces arc?summarized for each environment. The cable core current from direct photon depositibn (cable iiijebtion) is listed separately. Also given are the test facilities which were considered. The confidence statements express not ohly the degree to which coupling estimates were considered Valid, but say something about the difficulties anticipated in conducting each test (measured both by the effort required to develop the?test technique, and by the degree bf design stress evoked by each test).

Table 1. Summary of Phenomenology and Prbgram Considerations

Environment	Requirement	Primary Response/Levels	Test Beds	Confidence
DEMP	Design/ Verification	Field Coupled Skin Currents 10 A/Long	ARES Skin Injection	Good
SGEMP	Design	Skin Currents 200 A/Short	Photon Source (Low)	Good
		Direct Injection 10 A/Short	Cable Injection	
S/C Charging	None	Skin currents 50-1000 A/Med	Arc Discharge Skid Injection	Fair

If the contention that the **FLTSATCOM** design affords **significant** shielding frbm externally **induced** environmental effects is valid, it remains only to demonstrate **design** adequacy to the relatively large photon direct injection currents. Tests exposing cables to a low energy photon source have demonstrated that our estimates of this effect **are good** to a factor of probably two. Since the effects from these environments will result in transients on cables, it **is** efficient to **couple** transients directly to the system via the cable harness of the operating qualification model spaeecraft. Whether it is necessary tb couple currents to **the** entire harness simultaneously, or whether a systematic investigation of all Inter-Connections, or even a sample of those, is adequate, depends a great deal on the functional autonomy of the boxes linked by the cables and the similarity of the circuits used throughout the system.

It **is** necessary to **verify the** assumption that the vehicle does provide good shielding. This is Best done by inducing skin or structure currents arid verifying that the cable response is **as** low as predicted, or **at** least lower thah what will be used during the cable harness direct injection tests.

4. SKIN/STRUCTURAL CURRENT ESTIMATES

Twb aspects of the **electromagnetic** environments have been studied in order to describe the spacecraft's response. The first aspect, particle kinematics, permits the description of a driving function **for** each environment. The second aspects, lumped element modeling, incorporates the drivers into **an** equivalent electrical circuit **for** the spacecraft.

Lumped element electrical circuits of the spacecraft **have** been created for the varibus environments. These consist of resistors, capacitors, inductors and sources, and have varied widely in complexity. **A** model of the spacecraft for **SGEMP** response **is** shown in Figure **4.** The spacecraft has a capacitance to infinity for each of the selected nodes. The nodes are connected by inductances to a number of nodes since phase delay effects Will be important. Space charge limited current drivers from the solar panel are connected in parallel with the **capacitance** to tnfinity to represent the loss of electrons.

Using the niodels described above, estimates have been obtained for the DEMP, SGEMP, **and ESD** surges. **Various** groups Have camputed the response arid because of differences **in** modeling assumptions the results vary somewhat. The composite results are plotted in Figure 5.[1] The high frequency **characteristic** exhibited by the SGEMP reflects the short **rise** time of the x-ray pulse, and the low frequency contetit is **due to** the unipolar time history **of** the pulse. The DEMP has a low frequency cutoff due to ionospheric absorption of the **low** frequencies.

ESD surges are generally thought to be somewhat slower than SGEMP transients and may have a dc component.

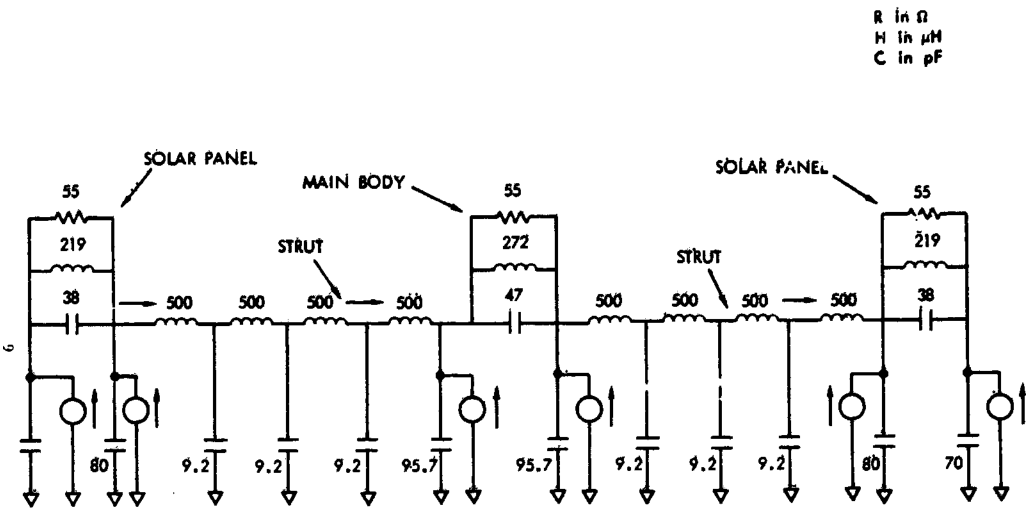


Figure 4. FLTSATCOM Mbdel Used for SGEMP Excitation

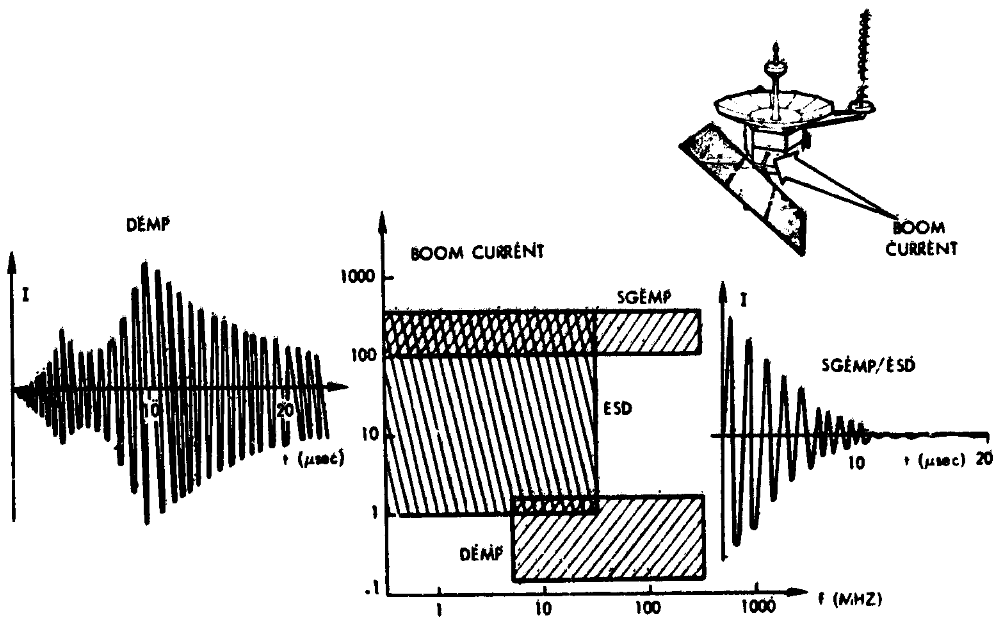


Figure 5. Electromagnetic Environment Characteristics

5. COUPLING APPROACHES

Two test techniques are being developed to simulate the SGEMP and ESD coupled skih currents. A parallel effort is investigating simulation of direct injection into cables. The technique €or driving SGEMP replacement Eurrents was suggested by Mangan, et al.[2] There art? two virtues of the techntque. One. the technique requires no direct electrical connection to the spacecraft. Two, the technique produces surface current responses that agree favorably with SGEMP predictions. In the technique, plates are used to capacitively couple currents to the spacecraft structure. The spacecraft is dielectrically isolated from grohnd by use of a wooden suppdrt structure. Figure 6 shows the skin injection test setup. The functional performance of the spacecraft will be monitored through the telemetry and communications links. This diagnostic approach maintains the dieleetric isolation from ground. Additional response data will be collected on external boom and internal spacecraft cable currents. These measurements will be made with a microwave data link which uses dielectric waveguide to couple the modulated X band carrier.

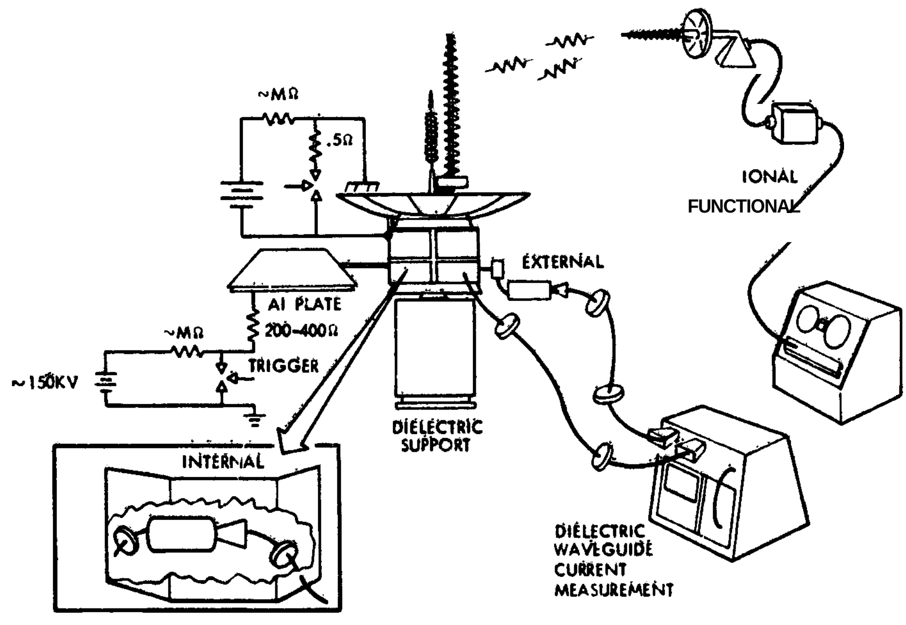
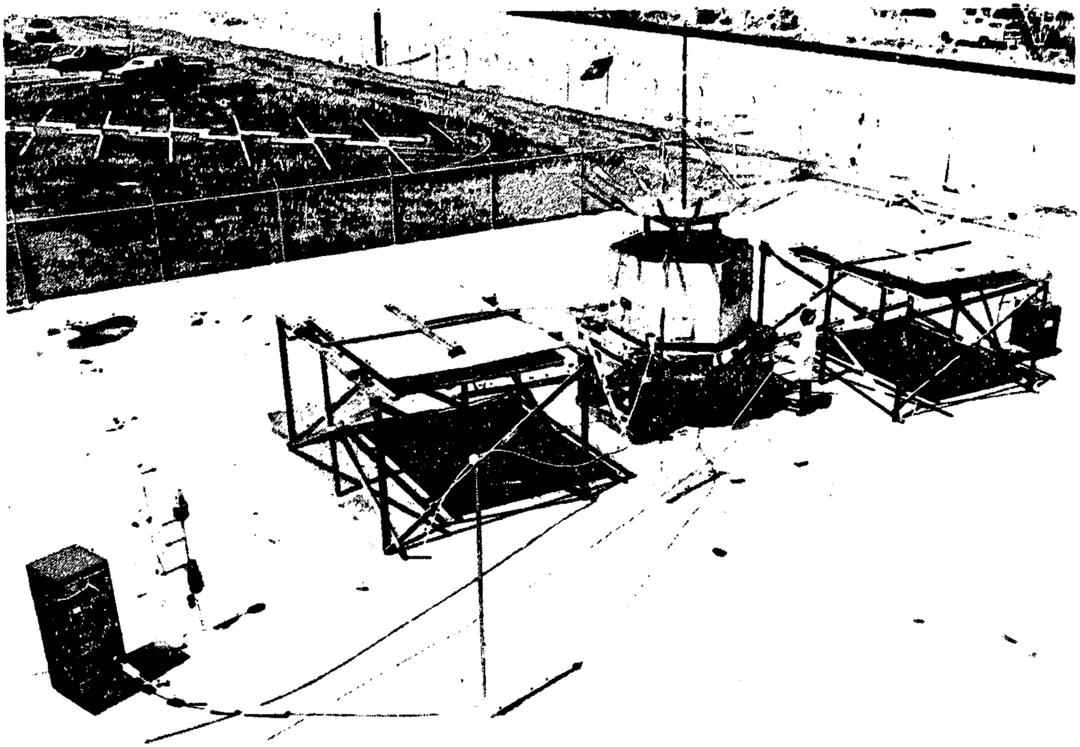


Figure 6. Skin Injection Tests **(SGEMP/ESD)**

A simplified electrical circuit of the replacement current drive technique is given id Figure 7. Either one or two drive plates can be used depending on the mode to be simulated. The capacitance betweed the drive plate(s) is charged tb approximately 66 kV. A triggered spark gap is used to discharge the capacitor through the 200Ω resistor to ground. Upon discharge, a boom Current will fl .v and dependtng oh whether one or two drive plates are used, the spacecraft symmetric or antisymmetric mode will be stimulated.[3]

The test setup pihtured on Figure 6 will also be Used fbr the arc discharge simulation. Spacecraft isolation is required and will be affbrded by a wooden support, the radiating telemetry and communications links, and the dielectric waveguide coupled microwave transmitters to measure currents. The major differende between the two tests is the method of simulating the environment. The arc discharge driver provides more localized coupling. The simulation will be used to induce digcharge currents at a number of different locations around the spatecraft while harness bundle currents and functional performance are monitored, A sheet of copper foil will he placed over the dieleetric material to simulate the layer of electrons which would be collected in the space envirohnient. The copper fbil will be charged to a voltage near the predicted breakdown level with respect to spacecraft structure. The discharge will be created by a triggered spark gap at representative are loeation. An equivalent Circuit for the arc discharge simulation is shown in Figure 8. Figure 9 is a waveform obtained on a pratotype of the pulser to be used duririg the FLTSATCOM tests. The waveform characteristics are controlled by the area of the copper foil (capacitance), lead inductance, anti the discharge resistance.

The! third type of test which will be performed during the FLTSATCOM test program will simulate SGEMP direct injection currents. These tests differ from the replacement current and arc discharge tests in that dielectric isolation is not required. Currents will be injected directly onto the spacecraft signal lines inside the cable shields simulating the SGEMP direct injection mod6 wherein the shielding does nbt provide any protection. This will be the severest test of spacecraft hardness to upset anti permanent damage since, based on predictibns add test data, the injection levels will be on the order of amperes.[4] THC coupling method into spacecraft signal lines is shown in Figure 10.



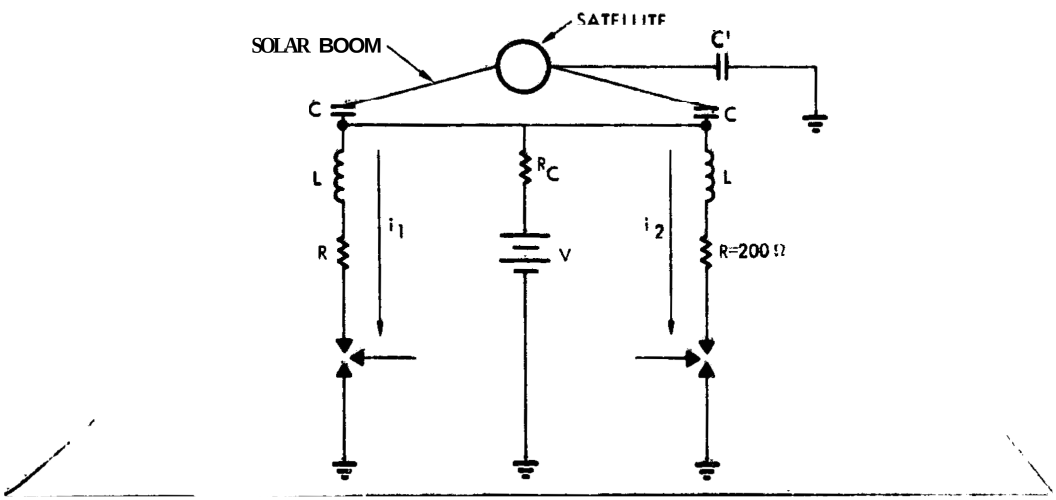
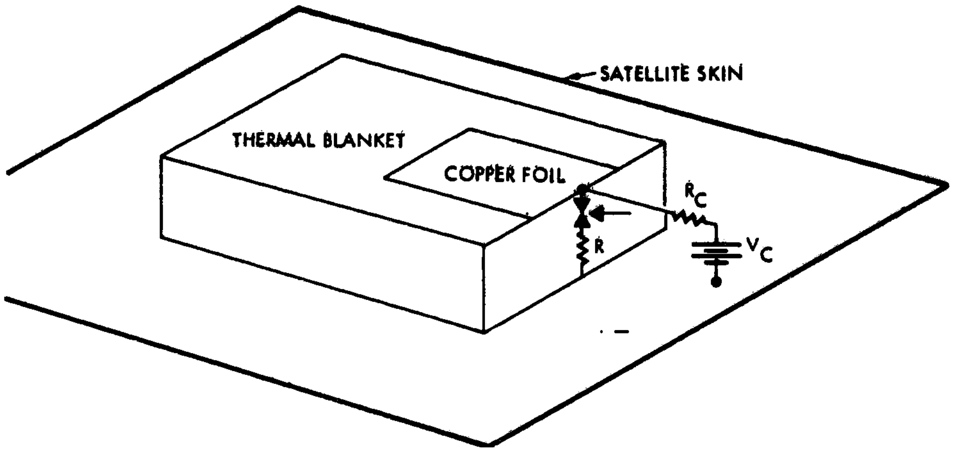
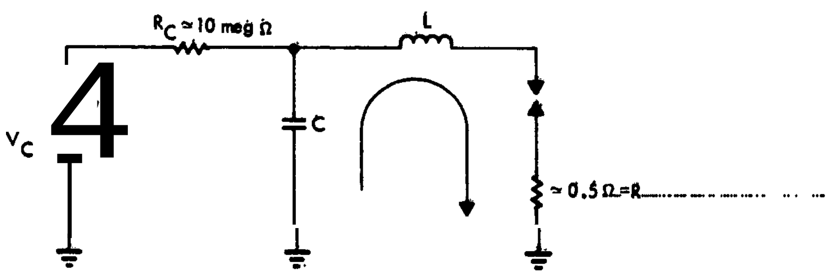


Figure 7. SGEMP Replacement Current Test Setup (with Spacecraft Mockup) and Equivalent Circuit Showing Drive Techntque



ARC DISCHARGE PULSER PLACEMENT



EQUIVALENT CIRCUIT

Figure 8. Equivalent Circuit for ARC Discharge Technique

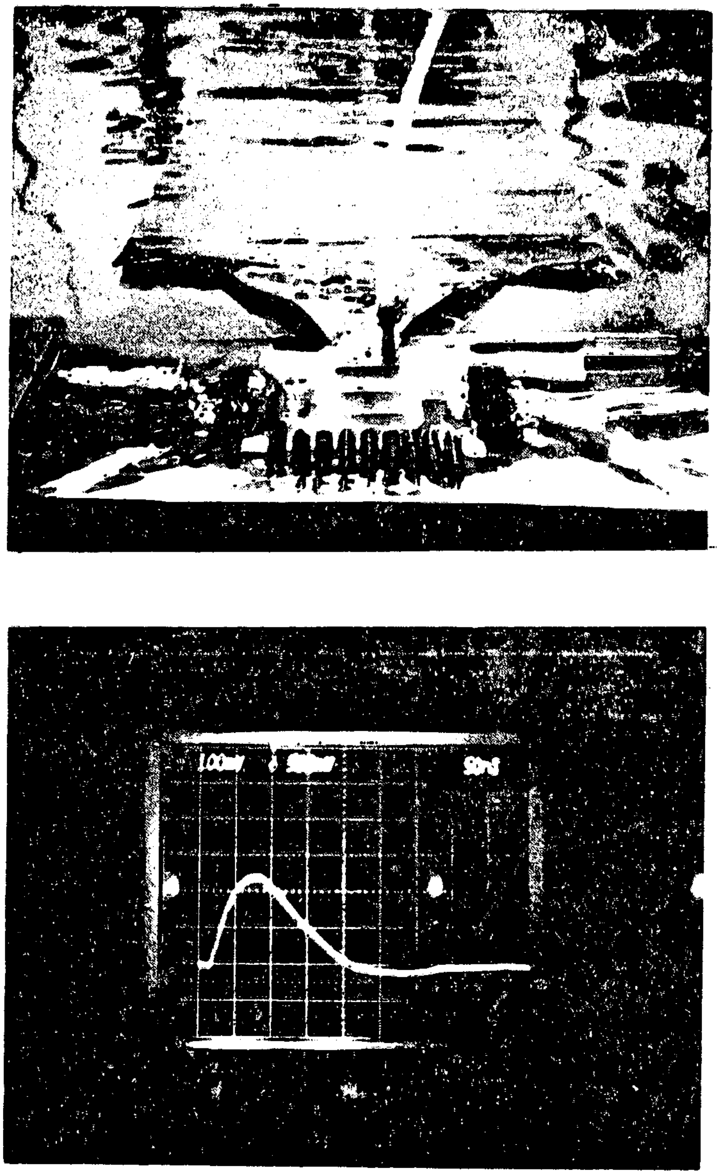


Figure 9. Prototype ARC Discharge Pulser and Waveform Obtained. Charge voltage -5 kV, discharge resistance -0.5Ω, copper foil - 1 m × 0.3 m, thermal blanket -0.008 cm thick

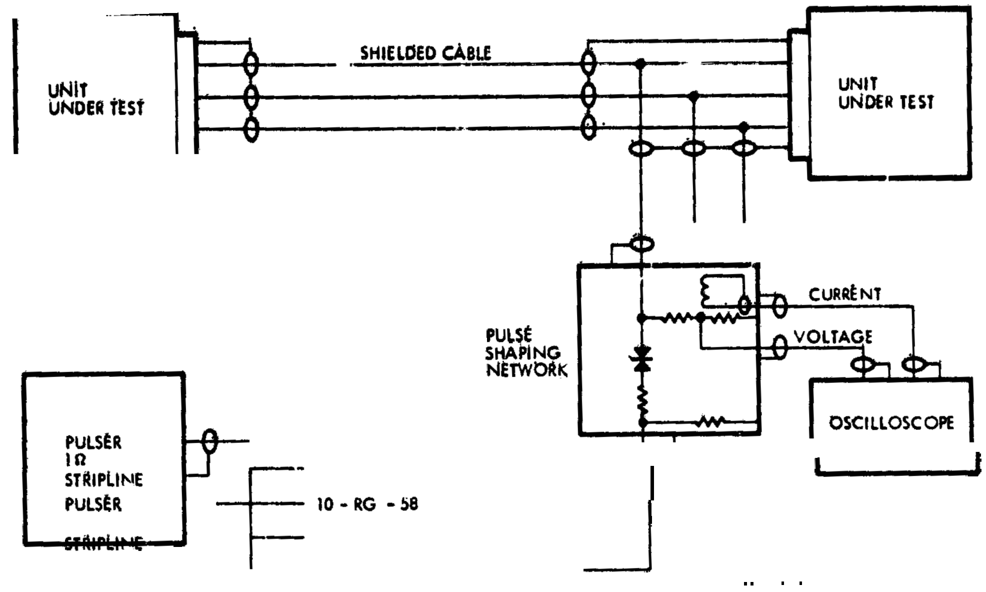


Figure 10. Direct Injection Coupling Technique

6. SUMMARY

We have described a test program which will serve to evaluate a spacecraft design hardened to DEMP, SGEMP, and ESD effects. One of the elements of the program is the injection of skin currents onto the qualification model. spacecraft. The other element is the direct injection of currents into the cable harness. Althbugh direct injection Currents are expected to be the dominaht coupling mechanism, this assumption will be confirmed by measuring harness currents during the skin injection tests simulating external DEMP, SGEMP, and ESD effects. The tests have two significant characteristics. First, the spacecraft will be live and operating so that performance anomalies Will be detected at the systems level. Secbnd, the effects of the photon and electron environments will be induced, no attempt wlll be made to recreate the environment itself.

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