Review of the Apollo Electromagnetic Compatibility Program

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Prepared by:

A. G. Weygand

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ABSTRACT

Review of the current electromagnetic interference control activities of the individual NASA Centers for Apollo and analysis of the potential problems of lightning protection, electrostatic charging phenomena and effects, and voltage breakdown of electrical equipment in the critical pressure region do not indicate the need for redirection of the current efforts in these areas. However, the need for more timely action in the coordination of the inter-center effort for achieving electromagnetic compatibility of the integrated elements in Apollo is indicated.

Review of the electromagnetic compatibility validation programs of the individual NASA Centers for Apollo reveals the existence of two different test philosophies to assure electromagnetic compatibility of the integrated systems of the individual stages of the launch vehicle and of the separate spacecraft modules, respectively. Clearly, testing in accordance with the requirements of a military specification for integrated system electromagnetic compatibility as will be performed on the individual stages of the Saturn launch vehicles will yield greater assurance of the electromagnetic compatibility of the integrated systems than will functional testing while monitoring subsystems performance for malfunction as will be performed on the individual spacecraft modules and on the integrated Apollo space vehicle. It is recognized that specification testing of the individual spacecraft modules is not feasible in view of the tight schedule and that functional testing must be used to demonstrate electromagnetic compatibility. However, since very limited functional testing is currently planned to demonstrate the compatibility of the LEM and the CSM, it is recommended that the test program be expanded to include functional compatibility testing of the LEM and the CSM in their various mission configurations.

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REVIEW OF THE APOLLO ELECTROMAGNETIC COMPATIBILITY PROGRAM

1.0 INTRODUCTION

A review has been made of the electromagnetic interference control and electromagnetic compatibility validation activities in the Apollo Program in order to assess its adequacy and to recommend modifications to these activities if appropriate. Control of electromagnetic interference and the electromagnetic compatibility of the various stages of the launcn vehicle, the two spacecraft, the associated ground support equipment, and the launch facilities are obvious elements of an acceptable Apollo system design.

Electromagnetic compatibility of the overall system implies that any piece of electrical and electronic equipment will not malfunction or be significantly degraded in performance by the electrical and electromagnetic environment it shares with other elements of the integrated system and will not add radiated or conducted interference to that environment which could result in the malfunction or performance degradation of any other piece of equipment.

In the pursuit of this work the following subtasks were performed:

(a) review of the current electromagnetic interference control and electromagnetic compatibility validation activities of the individual NASA Centers from an overall Apollo Program viewpoint,

(b) review of the electromagnetic compatibility experience in the Gemini Program for possible relevance or application to the Apollo Program, and

(c) analysis of problems of special pertinence or novelty to Apollo including lightning protection, electrostatic charging phenomena and effects, voltage breakdown in electrical equipment occurring at critical pressures, and formal courses on EMC proposed for NASA use.

The results of the investigations conducted under the subtasks described above are summarized in Appendices A through G of this report.

A memorandum* summarizing the electromagnetic interference control activities of the NASA Centers for the Apollo Program has been written and includes discussions of the electromagnetic interference control specifications in effect, center management control procedures, and inter-center panel activities.

Conclusions and recommendations for the existing electromagnetic compatibility program for Apollo are included in Section 2.0 of this report. Potential problem areas uncovered during this review are discussed in Section 3.0.

2.0 CONCLUSIONS AND RECOMMENDATIONS

It became apparent that the personnel working in the Apollo Program are very aware of the need for electromagnetic compatibility programs of the integrated systems. Although the electromagnetic compatibility programs of the various NASA Centers for Apollo were, in some cases, delayed in starting, the separate programs are, for the most part, progressing well at this time and no major redirection of the current intra-center efforts is indicated. The coordination of the inter-center effort for achieving electromagnetic compatibility of the integrated elements in Apollo also began late and the need for more timely action is indicated.

The following recommendations are made for the improvement of the existing electromagnetic compatibility program for Apollo:

(a) A single organization at the Kennedy Space Center should be vested with the responsibility for coordinating all of the efforts of the Kennedy Space Center in the electromagnetic compatibility area.

(b) The test program should be extended to verify the electromagnetic compatibility of the Lunar Excursion Module and the Command and Service Module. To accomplish this objective, the test program should include electromagnetic compatibility tests of or functional tests to demonstrate the electromagnetic compatibility of the Lunar Excursion Module and the Command and Service Module (1) when in the docked con-

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*"Summary of the Electromagnetic Interference Control Activities of the MSFC, the KSC, and the MSC for the Apollo Program," A. G. Weygand, Bellcomm TM-65-2021-7, August 9, 1965.

figuration used for the translunar phase of the mission and (2) when separated and in various orientations likely to be encountered during the separation and the rendezvous and docking phases of the mission.

(c) A timely exchange should be implemented between the NASA Centers of waivers to the requirements contained in the inter-center Interface Control Documents on electromagnetic compatibility.

(d) The electromagnetic interference controls on the electrical interface between the space vehicle and its electrical ground support equipment are the responsibility of the Electrical Systems Panel. However, the Instrumentation and Communication Panel should review their EMI control documents to assure the regulation of sources of EMI that could affect communications.

(e) The Instrumentation and Communication Panel should evaluate the electromagnetic compatibility tests and validation programs for the end items of the various centers and the overall test program for the integrated space vehicle, ground support equipment, and the launch facilities to determine whether the test program will give a high degree of confidence in the overall electromagnetic compatibility of all systems in Apollo.

(f) The "Electromagnetic Compatibility Awareness Course," sponsored by the Apollo Program Office (MAR), should not be linked to the Apollo Program exclusively but should be considered for use as a NASA-wide course.

(g) The measures recommended in the study of lightning protection for MILA Launch Complexes 34, 37, and 39 should provide adequate protection within their intended scope after they are implemented.

(h) Although it is recognized that more experimental data would be desirable for verification, space vehicle electrostatic charging and discharging are not expected to cause hazardous situations for the space vehicle or the astronauts during the Apollo missions.

(i) Electrical and electronic equipments of Apollo-Saturn space vehicle should be designed to operate and tested for operation in the critical gas or air pressure region without performance degradation or malfunction. It would be highly desirable to perform comprehensive thermal-vacuum tests on a fully instrumented flight model of the Lunar Excursion Module and Command and Service Module for both nominal and emergency spacecraft conditions with all in-flight systems operating to verify and demonstrate proper voltage breakdown design of spacecraft systems.

3.0 POTENTIAL PROBLEM AREAS

3.1 CENTER MANAGEMENT CONTROL PROCEDURES

The distribution of responsibility and the assignment of specific tasks within the organization of each of the NASA centers to assure electromagnetic compatibility of the end item from the respective center for the Apollo Program are outlined in Appendix A.

Six directorates of the Kennedy Space Center (prior to May 1, 1966) had responsibilities in the various aspects of achieving electromagnetic compatibility of the Apollo-Saturn space vehicle, the associated ground support equipment, the launch facilities, and the launch complex. However, no single directorate has the overall responsibility for coordinating the electromagnetic compatibility activities of the center, although a group within the Information Systems directorate acts as a central surveillance organization for and assists the various directorates as requested in matters concerning electromagnetic interference control and electromagnetic compatibility.

As in many large organizations where several different units are involved in testing equipments, subsystems and systems, active liaison between testing organizations is especially important and the results of these tests should be coordinated for the integrated system. A central EMI control and compatibility organization would be more effective if it were the contact for all electromagnetic compatibility matters involving a center. In this role, it could direct, coordinate, and monitor the overall electromagnetic compatibility technical effort of the center. It is suggested that such a single and authoritative organization be considered for the Kennedy Space Center.

3.2 ELECTROMAGNETIC COMPATIBILITY VERIFICATION IN APOLLO

The agreements between the NASA centers and their major contractors in the area of electromagnetic interference control, integrated equipment and systems tests to be conducted to demonstrate the electromagnetic compatibility of the various systems, and the co-ordination of inter-center effort in achieving electromagnetic compatibility in Apollo are summarized in Appendix B. A similar summary of the program used to achieve and verify electromagnetic compatibility of the integrated systems in Gemini is presented in Appendix C.

For the most part, the electromagnetic compatibility assurance and verification program being pursued for Apollo compares favorably with the Gemini program. A relative reduction in the scope of the program was noted in some areas. To provide verification of the electromagnetic compatibility of the combined operations of the Gemini spacecraft and the Agena D target vehicle in the various orbital configurations (both docked and undocked), a series of radio frequency and functional compatibility tests were performed at Cape Kennedy. The tests included a simulated mission, involving rendezvous and docking of the Gemini spacecraft with the Agena D target vehicle, on the radio frequency test range. Also, proper Gemini/Agena D subsystems operation was demonstrated in a number of vehicle orientations. However, in current Apollo test planning, no LEM/CSM electromagnetic compatibility tests will be performed per se. Functional testing of the hardline electrical interface between the LEM and the CSM will be accomplished with simulation of physical docking by using an umbilical connection. Radio frequency compatibility checks will consist of those tests performed on the entire space vehicle during launch pad operations with the LEM and the CSM oriented in the launch configuration. At no previous time will compatibility tests be performed on the combined LEM and CSM.

It is recommended that a more extensive test program be considered for demonstrating the electromagnetic and functional compatibility of the LEM and the CSM.

3.3 INTER-CENTER PANEL ACTIVITIES

Inter-center panel activities to insure electromagnetic compatibility in Apollo are reviewed in Appendix B.

The Instrumentation and Communication Panel has been assigned the responsibility of coordinating the inter-center effort for the control of electromagnetic interference in Apollo.

This responsibility includes (1) assigning frequencies to the space vehicle transmitters and receivers for minimum interference, (2) assuring electromagnetic compatibility of the interfaces between the spacecraft and the launch vehicle, and (3) assuring electromagnetic compatibility between the space vehicle and the ground support equipment and launch facilities. These tasks have been delegated to the Electromagnetic Compatibility Subpanel of the Instrumentation and Communication Subpanel.

Except for information on frequency allocations, the exchange of information among the centers concerning items pertinent to achieving electromagnetic compatibility of the interface between the launch vehicle and the spacecraft has not been sufficiently complete or timely to be effective. In particular, it is noted that the exchange between the centers of the lists of deviations or waivers granted by each center to the requirements contained in the electromagnetic compatibility interface control documents is not being done in time for a reasonably complete examination of the effect of the waivers.

The interface between the space vehicle and electrical ground support equipment and the launch facilities is currently the responsibility of the Electrical Systems Panel. The Electrical Systems Panel has some interference controls on this interface scattered in various documents, but, for the most part, the controls consist of informal agreements between centers. The Electromagnetic Compatibility Subpanel has decided not to generate an electromagnetic compatibility interface control document covering this interface because it is believed that it is too late in the program for such a document to be useful. Furthermore, the Electromagnetic Compatibility Subpanel has terminated all of its effort in this area and thus allowing the Electrical Systems Panel to control the interface as it sees fit. It is suggested that the Electromagnetic Compatibility Subpanel should collect the interference controls data which have been documented on this interface, and determine the informal agreements between the centers pertinent to achieving electromagnetic compatibility at this interface, and subsequently should review the information for possible indications of gross incompatibility.

There has been some concern expressed by the Reliability and Quality Control Directorate of the Apollo Program Office about the absence of an electromagnetic compatibility specification covering the overall Apollo-Saturn space vehicle. The usefulness of such a document at this late date in the overall Apollo program is debatable. Since for the most part it is too late to make significant changes in design criteria and design requirements without a significant cost and schedule impact, it appears that a more fruitful task for the Electromagnetic Compatibility Subpanel would be to review and evaluate the electromagnetic compatibility test and validation programs for the end-items being processed by the centers and the test plans for the integrated space vehicle, ground support equipment, and launch facilities to determine whether or not the successful completion of the test program will yield a high degree of confidence in the overall electromagnetic compatibility in Apollo.

3.4 ELECTROMAGNETIC COMPATIBILITY AWARENESS COURSE

The Reliability and Quality Directorate of the Apollo Program Office has contracted with the General Electric Company, Apollo Support Department, to prepare a course and a supporting text to promote an awareness of the importance of electromagnetic compatibility planning in Apollo and to disseminate electromagnetic interference control and electromagnetic compatibility information throughout the Apollo Program. Some background information on the Electromagnetic Compatibility Awareness Course is included in Appendix D.

The Electromagnetic Compatibility Awareness Course contains information applicable to any program and not peculiar only to the Apollo Program. Since the course is intended to be an awareness course and not a design course where solutions and fixes to possible problems are outlined, the content of this course will have little impact on the Apollo Program at this late date. Furthermore, outside of occasional references to the spacecraft acceptance checkout equipment for Apollo as an example to illustrate a point in a lecture and of some discussion of the electromagnetic control specifications used in Apollo, the Apollo Program is never specifically discussed.

It is recommended that this Electromagnetic Compatibility Awareness Course not be linked to the Apollo Program exclusively but be a general NASA-sponsored course for use in promoting electromagnetic compatibility awareness in up-coming NASA programs.

If this course were adopted as a NASA-wide course for use in training both NASA and contractor personnel, special care would need to be exercised in interpretating the requirements of contractually imposed military electromagnetic compatibility specifications and in discussing the required contents of electromagnetic compatibility control and test plans. It is recommended that inputs on these subjects be solicited from each NASA center in order to develop and present a unified NASA interpretation of the requirements of electromagnetic compatibility specifications and of the material which should be included in electromagnetic compatibility control and test plans.

3.5 LIGHTNING PROTECTION

The recommendation resulting from an investigation by the High Voltage Laboratory of the General Electric Company of the practices and procedures for protection of personnel, facilities, ground support equipment and space vehicle (in transit or on the launch pad) at Launch Complexes 34, 37 and 39 at MILA are reviewed in Appendix E. The capability to withstand surges generated by lightning of the many different connected equipments within the launch complexes and space vehicle has not been specified in these recommendations. However, the individual NASA centers have been made aware of the potential problem and have been asked to determine whether the voltage or current spikes induced in any connected equipments will be harmful to their equipment and to provide adequate protection if indicated.

The measures for lightning protection of Launch Complexes 34, 37 and 39 recommended by the General Electric Company have been adopted by the Kennedy Space Center. These measures should provide adequate protection of the launch complexes within their intended scope. However, some consideration should be given to the possible problem of galvanic corrosion of buried metallic facilities in the launch complex resulting from the introduction of extensive buried copper grounding systems.

3.6 ELECTROSTATIC CHARGING

The more important of possible electrostatic charging and discharging mechanisms as applied to the space vehicle operations during the launch powered flight, space coast, and lunar surface operations phases of a lunar landing mission and the results of applicable experiments are discussed briefly in Appendix F.

Although the actual processes resulting in electrostatic charging of the space vehicle during engine operation have not been determined, the results of a series of experiments on static firings of rocket engines and the results of some actual flight experiments indicate that the space vehicle will acquire an electrostatic charge during rocket engine operation.

Because the data available are for solid fuel rocket engines and because of the wide variations in the data, it is extremely difficult to estimate the magnitude of the electrostatic charging effect of the liquid fuel engines of the Saturn launch vehicles. Also difficult to estimate is the magnitude of the charging effect on the space vehicle of the triboelectric charging processes when (a) the boost protective cover is snatched from the spacecraft during the launch sequence, and (b) the space vehicle impacts water, ice or dust particles suspended in the atmosphere during launch.

Results of experiments on two flight vehicles designed to measure potential difference between the system ground points of two stages of the launch vehicle during the separation maneuver, however, indicate that no appreciable potential difference exists between launch vehicle stages during separation. Although these results may not be conclusive, they do suggest that arcing between launch vehicle stages during separation will not be a problem.

Electrical discharges within each stage or spacecraft, except for possible charge realignment via surface streamering from dielectric surfaces to conducting surfaces, will be prevented by adequate bonding of all conducting surfaces with the single point ground of each stage or spacecraft. Surface streamering from the spacecraft boost protective cover to the Launch Escape Tower legs has been recognized as a possible problem area and steps have been taken to protect the electro-explosive devices located in this area from premature activation by electrostatic discharges.

The plasma of the ionosphere will act as an efficient discharge mechanism as the space vehicle rises into the ionosphere during the launch maneuver and will limit the electrostatic charge accumulation on the space vehicle to non-hazardous amounts.

It is known that steady state potentials of space vehicles coasting in earth orbit in the ionosphere will be a few volts negative. For space vehicles coasting in outer space, it is expected that the equilibrium potential of the space vehicle will range from a few volts positive to a few volts negative depending upon which effect predominates, the solar wind or photoemission. If a space vehicle should acquire a large potential by an independent process such as rocket engine operation, the ionosphere will act to discharge the space vehicle rapidly. The solar wind would act in a similar manner, but the discharging rate will be understandably smaller.

The effects of the various postulated electrostatic charging and discharging mechanisms of the solar wind, photoemission, and motion on the lunar surface during lunar surface operations cannot be accurately estimated because of the absence of certain data. Knowledge of the electrical conductivity and the dielectric constant of the materials composing the lunar surface is required to determine the effect of motion on the lunar surface. Knowledge concerning the lunar magnetic field is required to determine the effect of possible shielding of the lunar surface from the solar wind. However, using current best estimates of these physical properties, it is not expected that dangerous charge accumulations will occur during lunar surface operations of the LEM and the astronauts.

In summary, although it is recognized that more experimental data would be desirable to verify some of the assumptions and conclusions, space vehicle electrostatic charging and possible resulting effects are not expected to cause hazardous situations for the space vehicle or the astronauts during any of the Apollo mission phases. The Electromagnetic Compatibility Subpanel of the Instrumentation and Communication Panel has proposed no further theoretical, experimental, or preventive design effort in this area until evidence appears that shows additional activity is warranted.

3.7 VOLTAGE BREAKDOWN

Appendix G contains a discussion of the causes of and problems in space vehicles resulting from voltage breakdown in equipments.

Although voltage breakdown in electrical and electronic equipment can result from the multipactor effect, it is a phenomenon requiring specialized conditions and is not likely to be a problem in Apollo.

The probability of voltage breakdown of electrical and electronic equipment and of antennas is enhanced by operation in a plasma. However, the times during a mission when a plasma will be present can be predicted and the effects can be anticipated and taken into account in the mission and equipment planning. For instance, it is known that during the atmospheric reentry phase of a space mission, communications blackout between the spacecraft and earth based stations will occur as a result of the effects of a plasma sheath surrounding the reentering spacecraft. Consequently, the spacecraft communications equipment operating during this phase will be designed to survive the large reflected powers caused by operation in this severe environment.

The major cause of voltage breakdown of electrical and electronic equipments in space vehicles has been the operation of these equipments in the critical air or gas pressure region. Equipment ambient pressures falling in the critical pressure region can result from a variety of conditions which cannot always be anticipated in the equipment design. Therefore, all high voltage equipment for use in space by man-rated space vehicles should be designed and tested to operate in the critical pressure region.

To date there have been no failures experienced in the Saturn launch vehicles which could be attributed to voltage breakdown phenomena. The Marshall Space Flight Center has been aware of this problem area and for this reason has designed the electrical systems to operate at the lowest practical voltages. In addition, high voltage circuitry and components are potted and circuit boards are cord-wood stacked and have a conformal coating applied.

The Manned Spacecraft Center has also recognized voltage breakdown of electrical and electronic equipment as a possible problem area. Most of the black boxes to be carried in the Apollo spacecraft will undergo vacuum testing separately. During one of these tests, voltage breakdown occurred in the VHF transceiver to be carried by the Apollo Lunar Excursion Module. Subsequent redesign of the packaging of the transceiver made possible satisfactory operation in the critical pressure region.

Although testing of the individual black boxes for voltage breakdown in the anticipated environments is necessary, the integrated spacecraft should also be tested in a simulated mission environment, including the launch phase, under nominal and emergency spacecraft conditions because local environments of equipments in the spacecraft may be significantly different than the anticipated environment. This difference could be caused by decompression of spacecraft, subsequent repressurization, outgassing of materials, and inadequate venting of equipment enclosures. Furthermore, voltage breakdown in wiring and connectors has proven to be a problem in other space vehicle programs and is best tested in their intended configuration and possible environments.

Thermal-vacuum tests on the integrated Lunar Excursion Module and Command and Service Module are planned to be conducted at MSC. Thermal-vacuum tests will also be performed at the Kennedy Space Center on some of the flight spacecraft. However, all systems may not be operating during these tests, notably the communications equipments which use high voltages in some instances.

Voltage breakdown problems including problems in wiring and connectors have frequently been uncovered in spacecraft of other space programs during thermal-vacuum testing of the integrated spacecraft. It would be highly desirable to perform comprehensive thermal-vacuum tests for nominal and emergency flight spacecraft conditions to verify and demonstrate proper voltage breakdown design of the spacecraft systems.

A. G. Weygand

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APPENDIX A

CENTER MANAGEMENT CONTROL PROCEDURES TO ASSURE APOLLO ELECTROMAGNETIC COMPATIBILITY

1.0 General

The electromagnetic interference control and electromagnetic compatibility organizations, their respective responsibilities and their functional relationships within each of the NASA Centers are outlined in the following paragraphs.

2.0 Marshall Space Flight Center

The responsibility for providing electromagnetically compatible operation of the Saturn launch vehicles and associated ground support equipment is shared by two of the laboratories under Research and Development Operations at the Marshall Space Flight Center. The Astrionics Laboratory is responsible for insuring proper design and test of launch vehicle flight equipment and the associated ground support equipment required for prelaunch checkout. The Quality and Reliability Assurance Laboratory is responsible for insuring proper electromagnetic interference control design and test of launch vehicle checkout equipment not required for use in the launch area. The Quality and Reliability Assurance Laboratory also is responsible for the performance or monitoring of the electromagnetic compatibility testing of the integrated stage and associated ground support equipment, but the Astrionics Laboratory is responsible for correction of any incompatibilities revealed during the integrated tests. Both laboratories maintain close contact with the activities of each other in the area of electromagnetic interference control. In addition, both laboratories review all contractor documentation pertaining to electromagnetic interference control in and compatible operation of integrated stage and associated ground support equipment.

The Marshall Space Flight Center has the responsibility for the compatible operation of the overall Saturn launch vehicle in all mission modes although the Kennedy Space Center has the responsibility for the performance of tests to verify this compatibility in the launch area.

3.0 Manned Spacecraft Center

The electromagnetic compatibility effort and responsibility at the Manned Spacecraft Center for the spacecraft and the associated ground support equipment is distributed over the system and subsystem engineering levels. The electromagnetic compatibility systems engineer of the Apollo Spacecraft Program Office defines and specifies the overall system electromagnetic compatibility requirements for the spacecraft and associated ground support equipment and monitors the overall program to insure system electromagnetic compatibility requirements are satisfied. The various subsystem managers within the Engineering and Development directorate are then responsible for translating the overall system requirements into design and test requirements for specific subsystems and for monitoring progress to ensure subsystem compliance with electromagnetic interference control requirements.

The Manned Spacecraft Center has the responsibility for the compatible operation of the overall Apollo spacecraft in all mission modes although the Kennedy Space Center has the responsibility for the performance of tests to verify this compatibility in the launch area.

4.0 Kennedy Space Center

At the Kennedy Space Center six directorates are concerned with different aspects of the electromagnetic compatibility of the space vehicle, the ground support equipment, the launch facilities, and the launch complex. While the responsibility for the level of effort in achieving electromagnetic compatibility in the areas of each respective directorate is retained by that directorate, an organization within one of the directorates acts as a central surveillance organization for and assists the various directorates of the Kennedy Space Center as requested in matters of electromagnetic compatibility. However, the various directorates are not required to keep this organization up to date on electromagnetic compatibility activities taking place under their respective cognizance.

The overall responsibility for verifying electromagnetic compatibility of the various elements within the launch area is divided among the various directorates of the Kennedy Space Center as follows:

(a) Launch Vehicle Operations - demonstration of the electromagnetic compatibility of the launch vehicle with the environment, both conducted and radiated at the launch complex provided by the spacecraft, the ground support equipment, the launch facilities, and the local electromagnetic environment produced by facilities external to the launch area.

(b) Spacecraft Operations - demonstration of the electromagnetic compatibility of the spacecraft with the environment, both conducted and radiated, at the launch complex provided by the launch vehicle, the ground support equipment, the launch facilities, and the local electromagnetic environment produced by facilities external to the launch area.

(c) Quality Assurance - coordination of the electromagnetic interference control specification compliance by electrical and electronic equipments procured by the Kennedy Space Center.

(d) Support Operations - provision of the electromagnetically compatible operation of the communication and data transmission facilities within the launch area.

(e) Engineering and Development - construction of facilities ensuring proper structural bonding, grounding, power grounding and distribution, etc.

(f) Information Systems - provision of assistance to other directorates in the area of electromagnetic interference control as requested including performance of electromagnetic interference testing of individual equipments; preparation of electromagnetic compatibility control plans, test plans, test procedures, test reports, and design guidelines; performance of launch vehicle, spacecraft and ground support equipment compatibility tests including monitoring command-destruct receivers and electro-explosive device firing circuits for possible detrimental interference; performance of ambient electromagnetic interference environmental control in the Merritt Island Launch Area; and management of radio frequency assignment in the Merritt Island Launch Area including the coordination of frequency assignments between NASA and the Air Force Eastern Test Range.

APPENDIX B

PROGRAM TO VERIFY ELECTROMAGNETIC COMPATIBILITY IN APOLLO

1.0 General

Electromagnetic interference control specifications for the most part form the basis of the programs of the various NASA Centers to insure electromagnetic compatibility of the various systems in Apollo. The contractual agreements between the NASA Centers and their respective major contractors and the inter-center panel co-ordination activities in the area of electro-magnetic interference control and the integrated equipment tests to be performed to demonstrate the electromagnetic compatibility of the various systems in Apollo are discussed in the following paragraphs.

2.0 Electromagnetic Interference Control Specifications Contractually Imposed

2.1 Stages of the Saturn Launch Vehicles

The Marshall Space Flight Center has included in the negotiated contracts with the prime contractors for the various stages and the Instrument Unit (IU) of both the Saturn IB and Saturn V launch vehicles the military electromagnetic interference specifications, MIL-I-6181D and MIL-E-6051C, and the military electrical bonding specification, MIL-B-5087A. The design and test of the electrical and electronic equipments of all stages and the IU and associated ground support equipment are required to comply with the requirements of specification MIL-I-6181D. The requirements of specification MIL-E-6051C apply to each completely integrated stage or IU and tests are required to be performed by the contractor to demonstrate a minimum of 6 db of margin between the desired signal and the interference noise at the most critical point of subsystems on either stage of IU. There are currently no electromagnetic compatibility specifications covering the entire Saturn launch vehicle.

2.2 Spacecraft Modules

In the current electromagnetic compatibility program of the Manned Spacecraft Center for Apollo, the prime contractors for the CSM and the LEM are contractually required to design the integrated spacecraft (CSM or LEM) and the associated ground support equipment (GSE) using the military electromagnetic compatibility specification MIL-E-6051C as a guide and attempting to maintain at least a 6 db safety factor against all electromagnetic interference caused malfunctions or performance degradations. Tests to demonstrate the success of this design effort are not contractually required. There are currently no electromagnetic compatibility specifications covering the integration of the CSM and the LEM.

However, the interface control document, "NAA/GAEC Electromagnetic Compatibility Design Criteria ICD", dated June 17, 1965, between North American Aviation, Inc. and the Grumman Aircraft Engineering Company describes the electromagnetic compatibility design requirements that shall be used for the design of all interfaces between the LEM, the Block II CSM, and the Spacecraft - Lunar Excursion Module Adapter.

To minimize the number of electromagnetic incompatibilities in either the integrated CSM or the integrated LEM, the prime contractors are contractually required to design and test all electrical and electronic equipments in accordance with the military electromagnetic inteference specification MIL-I-26600 as amended by the addendum MSC-EMI-10A, or the equivalent. The electromagnetic interference testing of the electrical and electronic equipments will be limited to one of each type of equipment unless modifications are made in the piece of equipment which would require further testing or retesting.

2.2.1 Command and Service Module, and Launch Escape System

The prime contractor for the CSM, North American Aviation, Inc., has required that all deliverable electrical and electronic hardware from subcontractors or from in-house groups except the communication equipments meet the requirements of the North American Aviation, Inc., electromagnetic interference control specification MC999-0002B. The communication equipments are required to meet the electromagnetic interference control requirements included in the communication and data subsystem procurement specification MC999-0023C. The electromagnetic interference control design and test requirements contained in these two documents are comparable and are essentially equivalent to the military specification MIL-I-26600 as amended by the addendum MSC-EMI-10A.

2.2.2 Lunar Excursion Module

The prime contractor for the LEM, Grumman Aircraft Engineering Company, has required that all deliverable electrical or electronic hardware from subcontractors or from in-house groups meet the Grumman Aircraft Engineering Company electromagnetic interference control specification LSP-530-001. The design and test requirements contained in LSP-530-001 are essentially equivalent to the requirements contained in the military specification MIL-I-26600 as amended by the addendum MSC-EMI-10A.

3.0 Inter-Center Panel Activities to Insure Electromagnetic Compatibility in Apollo

The Instrumentation and Communication Panel has taken the initiative in co-ordinating the inter-center effort for the control of electromagnetic interference in the Apollo Program. The Instrumentation and Communication Panel has delegated the responsibility in this area to the Electromagnetic Compatibility Subpanel. The responsibilities of this subpanel as outlined in its charter include allocating frequencies to space vehicle transmitters and receivers on a minimum interference basis and insuring electromagnetic compatibility of the interfaces between the spacecraft and the launch vehicle, and between the space vehicle and the ground support equipment and the launch facilities.

The possible radio frequency interference problems arising from frequency allocations for space vehicle transmitters and receivers have been examined for both the Apollo Saturn IB and Apollo Saturn V space vehicles. Interface Control Document 13M60003, "Saturn-Apollo Frequency Plan," has been issued by the Instrumentation and Communication Panel and contains a statement of the allocated frequencies for each Apollo Saturn IB and Apollo Saturn V space vehicle.

Interface Control Document 13M60004, "Saturn-Apollo Electromagnetic Compatibility Design Criteria," (attached as Appendix H) has also been issued by the Instrumentation and Communication Panel and defines the spacecraft and launch vehicle interface electromagnetic compatibility design requirements. Since this document encompasses some areas which are under the purview of the Electrical Systems Panel, it was co-ordinated with that panel prior to official issuance. Under the provisions of Interface Control Document 13M60004, the Manned Spacecraft Center and the Marshall Space Flight Center each are required to prepare and exchange a list of all deviations and/or waivers to the requirements of this document. These lists are to be examined by the respective centers and any suspected incompatibilities resulting from granting any deviation and/or waiver are to be submitted to the Electromagnetic Compatibility Subpanel for examination. If necessary, the deviation or waiver will be re-submitted by the subpanel to the center which approved it for reconsideration.

4.0 Integrated Systems Tests Performed Prior to Acceptance

4.1 Stages of Saturn Launch Vehicle

Electromagnetic compatibility testing in accordance with the requirements of the military specification MIL-E-6051C will be performed as contractually required on each flight configuration stage of the Saturn vehicle at the checkout facility or manufacturing site of each major contractor. Each stage contractor is required to perform a comprehensive stage system electromagnetic compatibility test which includes testing to demonstrate a minimum safety margin of 6 db between the desired signal and the interference noise at the most critical point of a subsystem, testing to demonstrate no malfunction or performance degradation of a subsystem due to electromagnetic interference, and testing for detrimental transients on power busses while all stage and associated ground support equipments are operated through their normal flight sequence. The test points monitored during these tests are chosen because they indicate performance of flight critical circuits or of inherently noisy circuits. The individual stage contractors have been encouraged to perform as early in the program as possible investigative electromagnetic interference and susceptibility tests with individual stage and ground equipments energized to detect basic incompatibilities and to provide a data base and performance history to aid in the evaluation of future test results.

The Marshall Space Flight Center performs or monitors the performance of radiation interference generation measurements with all intentional radiators operating simultaneously and of radiation susceptibility measurements on the receiving and transmitting subsystems of the individual stages to aid in determining if the stages will be radio frequency compatible when stacked.

4.2 Spacecraft Modules

4.2.1 Command Module, Service Module, and Launch Escape System

Tests will be performed at the North American Aviation, Inc. manufacturing location (Downey) on each flight spacecraft to assure flight readiness of the CM, SM, and the launch escape system and to verify the compatibility of the interfaces between the CM, SM, the launch escape system and the associated ground support and acceptance checkout equipment. These tests will include individual subsystem functional tests, integrated subsystems functional tests and finally simulated mission tests. During the integrated subsystem checkout and the simulated mission tests using the acceptance checkout equipment, the various subsystems will be exercised through various operational modes including abort modes and normal flight modes for the demonstration of operational compatibility of the various subsystems of the spacecraft.

In addition to monitoring performance of the critical spacecraft subsystems, special attention will be given to monitoring transients and other types of interference conducted on both the ac and dc power distribution subsystems during the functional tests and checkout operations. Limited electromagnetic interference and spectrum signature test measurements will be performed on each spacecraft but only as required to assure resolution of electromagnetic interference problems encountered during earlier tests or in earlier spacecraft. This limited electromagnetic interference test data together with the results of the integrated subsystems tests and acceptance checkout data on each spacecraft is expected to provide the necessary information for evaluating the electromagnetic compatibility of spacecraft performance.

Earlier in the program prior to production of flight spacecraft a house spacecraft was used in performing electromagnetic compatibility tests on individual subsystem and integrated systems, in indicating corrective action, and in evaluating design modifications.

4.2.2 Lunar Excursion Module

Tests will be performed at the Grumman Aircraft Engineering Company manufacturing location (Bethpage) on each flight spacecraft and associated ground support and acceptance checkout equipment to demonstrate the compatible operability of the spacecraft subsystems and to verify mission readiness of the integrated spacecraft and associated ground support equipment. These tests will include functional subsystem tests and checkout during subsystem and stage build-up and integrated subsystem functional tests and checkout of the final spacecraft stages. In addition to the integrated spacecraft checkout by the acceptance checkout equipment, the performance of all critical spacecraft subsystems will be monitored during the performance of mission oriented tests when the spacecraft subsystems will be exercised in the various operational modes and combinations.

Electromagnetic compatibility of the subsystems will be confirmed in the house spacecraft which will also be used in spectrum signature tests.

5.0 Test Performed at Cape Kennedy

The functional compatibility and operational readiness of the integrated Apollo Space Vehicle and associated ground support equipment will be demonstrated by the successful completion of a series of tests, checkouts and simulated missions performed at the Merritt Island Launch Area. In general, electromagnetic compatibility of an assembled or partially assembled space vehicle will be checked by performing functional tests on the assembly and monitoring critical subsystems for performance degradation, monitoring ordnance subsystems for interference which could cause premature ignition, and monitoring power distribution systems for transient identification as indications of incompatibility.

5.1 Launch Vehicle Integration

Inspection upon receipt and limited subsystems testing will be performed on the various stages and ground support and checkout equipment to ensure proper subsystem installation and operation prior to stage mating and checkout. After mating operations have been completed, the test program will essentially follow a building block pattern starting with initial checks at the equipment level and progressively expanding into subsystem and system tests and finally integrated system tests. The detailed subsystem and system tests will ensure complete system integrity and will verify proper system operation.

Integrated launch vehicle system tests and checkout will be performed to demonstrate that each system is compatible with other systems, that each stage is compatible with other stages and that all systems are compatible with the associated ground support equipment. Overall tests will then be performed to verify the integrity and compatibility of the launch vehicle and the associated ground support equipment electronic, electrical and mechanical systems while all systems are operating according to a modified and abbreviated countdown with the flight program. Finally a simulated flight test will be performed including the pre-count, countdown, and flight phases of its mission with all launch vehicle and ground support equipment systems operating as close as possible to launch day configuration to checkout all systems and to verify compatibility of operation and readiness for participation in space vehicle checkout of the complete launch vehicle and associated ground support equipments. However, no radio frequency interference tests with transmitters and receivers operating will be performed at this time. Malfunction sequence tests will also be performed to verify the compatibility and proper operation of the launch vehicle systems under non-nominal conditions.

5.2 Spacecraft Integration

Significant reductions in the originally conceived procedures for spacecraft launch operations and checkout at MILA are currently being considered for both the CSM and the LEM which would eliminate much of the redundant tests made at different locations and would aim to perform checkout at as high an assembly level as possible consistent with the cost and schedule slip for repair. Specifically, experience will be gained on reduced testing of CSM 011, 012, and 014 with the objective of obtaining the necessary confidence for shipping CSM 017 directly to the Vertical Assembly Building.

For CSM 011, 012, and 014 after a brief receiving inspection, the CM and SM will be mated and placed in the altitude chamber. After completion of systems verification functional checkout and polarity checks, the assembled CSM and associated ground support equipment will undergo simulated flight tests and simulated flight abort tests to demonstrate the operational performance readiness of the integrated spacecraft for mating with the launch vehicle. After being mechanically mated to the launch vehicle on the pad, integrated systems tests will be performed using a launch vehicle simulator.

For CSM 017 and subsequent, after a brief receiving inspection, the CM and the SM will be mated and be moved to the polarity fixture in the Vertical Assembly Building where polarity checks and systems verification tests will be made. The CSM will then be mated with SLA which had already been mated with the LEM Test Article. This assembly will then be mechanically mated to the launch vehicle and integrated systems tests including simulated launch tests using a launch vehicle simulator will be performed. After successful completion of these tests the spacecraft assembly will be electrically mated with the launch vehicle.

As for the LEM, after a brief receiving inspection of the ascent and descent stages, the LEM ascent and descent stages will be mated and systems verification tests will be performed. Then the LEM and the SLA will be mated and electrical interfaces will be checked.

Combined CSM and LEM tests will be restricted to electrical interface checks when the LEM and the CSM are electrically connected via an umbilical hardline. No RF compatibility tests between the LEM and CSM are planned.

5.3 Space Vehicle Integration

After the launch vehicle and the spacecraft have been mechanically and electrically mated, interface tests will be performed to demonstrate the integrity and functional compatibility of the spacecraft and launch vehicle interfaces, to checkout the space vehicle emergency detection system, and to verify the functional compatibility of space vehicle systems and ground support equipment. (For Saturn IB flights the spacecraft will be mated to the launch vehicle on the launch pad while for the Saturn V flights the spacecraft will be mated to the launch vehicle in the Verticle Assembly Building.) Following these tests will be tests to demonstrate the compatibility and proper operation of all space vehicle and ground support equipment during a nominal automatic firing and flight sequence both with and without umbilical ejections. Malfunction sequence tests will also be performed on the space vehicle.

For the Saturn V flights the space vehicle will then be moved to the launch pad. For the Saturn IB flights the Count Down Demonstration Test (CDDT) will follow which shall be as close as possible to the actual launch countdown. A RF compatibility test of the space vehicle, MILA and ETR RF radiating systems will be performed in conjunction with the CDDT. During the RF compatibility tests all radio frequency systems of the space vehicle, MILA, and ETR will be turned on which will be operating during countdown and launch activities. All launch vehicle and spacecraft RF systems will be interrogated and/or monitored for indications of malfunction and for indications of interference. The RF compatibility tests will be performed with the Mobile Service Structure (MSS) in the launch location and with the MSS removed from the pad area.

After the Saturn V space vehicle has been transported to the launch pad and the ground support equipment has been connected, systems verification tests will be performed to assure that system operation was unaffected by the transportation and that the ground support equipment was properly connected. In addition to malfunction sequence tests on the space vehicle, CDDT and RF Compatibility tests, as described earlier for the Saturn IB space vehicles, will then be performed.

APPENDIX C

PROGRAM TO VERIFY ELECTROMAGNETIC COMPATIBILITY IN GEMINI

1.0 General

The contractual requirements imposed by the Manned Spacecraft Center on the McDonnell Aircraft Company, the prime contractor and the agency responsible for integration of the Gemini spacecraft systems, are not explicit in the area of electromagnetic interference control. Instead, the McDonnell Aircraft Company is made responsible for the control of electromagnetic interference such that equipment items of the spacecraft and associated aerospace ground equipment shall not be adversely affected by interference voltages, currents, or fields present in their operational environment. Moreover these items are required not to produce interference individually or in any combination operationally required which will cause malfunction or unacceptable performance degradation of other equipment items or subsystems of the spacecraft, launch vehicle, target vehicle, and associated aerospace ground equipment.

The basic approach taken by McDonnell for control of electromagnetic interference within Gemini was to place heavy emphasis on equipment design and test. The design and test requirements of the military electromagnetic interference control specification, MIL-I-26600, were included to varying degrees in the Specification Control Drawing for each of the electrical or electronic equipments. The Specification Control Drawing defines the technical requirements for the design and test of each piece of equipment and also forms part of any McDonnell Aircraft Company Purchase Order. Compliance by the subcontractor or vendor is contractually required. Functional compatibility tests were performed by McDonnell to demonstrate electromagnetic compatibility of the integrated spacecraft rather than conducting specific electromagnetic compatibility tests.

The minimization of interference between the transmitters and receivers on-board the Gemini spacecraft, the Titan launch vehicle and the Agena D target vehicle is essential to the success of the Gemini missions. For use in transmitter and receiver frequency allocation, the harmonics of and the cross modulation products of the various on-board transmitter fundamental frequencies were calculated to determine any combinations that might produce radio frequency interference in the on-board receivers.

2.0 Equipments

2.1 Equipment Design and Development Phase

The first formal testing of equipment by subcontractor or vendor as required by the McDonnell Aircraft Company was the "design approval test" which also included electromagnetic interference tests contained in the Specification Control Drawing. This testing was performed on engineering models to demonstrate that the design of the equipment did comply with the requirements contained in the Specification Control Drawing.

Also during this phase, integrated systems tests were conducted by McDonnell. Compatibility of the electrical and electronic subsystems of the Gemini spacecraft was initially tested on the Electronic Systems Test Unit. The Test Unit was constructed early in the program for the purpose of demonstrating the functional compatibility of the electrical and electronic subsystems of the Gemini spacecraft. The mechanical design and the layout of the electrical and electronic subsystems of the final Gemini spacecraft design were approximated as were the cable runs. Other subsystems were not included in the Electronic Systems Test Unit.

Final systems integration tests during this phase were conducted by McDonnell on the Compatibility Test Unit. The Compatibility Test Unit has become the house spacecraft and is configured like a flight spacecraft. Only functional compatibility tests were performed. If a malfunction was detected, corrective action was taken at that time in the appropriate equipment. Any design changes were evaluated on either of these test units.

2.2 Equipment Qualification and Acceptance Test Phase

The next formal testing of the electrical or electronic equipments by the subcontractor or vendor were the qualification tests. Early production units were used in these tests which included rigid checks of the equipments to demonstrate compliance with the design requirements set forth in the approved Specification Control Drawing. Formal failure analysis and corrective action were pursued during these tests. Any waivers or deviations to the requirements in the Specification Control Drawing requested from McDonnell by the subcontractor or vendor were coordinated by McDonnell with the Manned Spacecraft Center.

In the next series of formal tests, the actual flight equipment hardware for each flight spacecraft was tested to measure equipment performance and to ensure proper equipment operation but not to demonstrate compliance with the requirements of the military electromagnetic interference control specification, MIL-I-26600. First the pre-delivery acceptance tests of the equipments were performed on the premises of the subcontractor or vendor. Nearly identical tests were performed by McDonnell prior to installation of the equipments into the spacecraft. It should be noted that these tests were concerned primarily with the performance of the end items supplied by the contractor or vendor before acceptance by the McDonnell Aircraft Company.

3.0 Integrated Systems

3.1 Integrated Spacecraft Systems Test Phase

After installation of the equipments into the spacecraft, performance tests were conducted by the McDonnell Aircraft Company prior to spacecraft acceptance by the Manned Spacecraft Center and shipment to the launch site. These performance tests included module tests, interface verification tests, integrated systems tests, and simulated flight tests. The McDonnell generated test plan does not in itself contain the detailed test procedures for the tests to be performed but does reference other documents called Service Engineering Department Reports. The test equipment to be used, the exact test procedure to be followed, and the test data to be recorded for each test are described in detail in the appropriate Service Engineering Department Reports.

Of prime importance in these tests was demonstration of satisfactory operation of the integrated equipment and of spacecraft operational readiness for its mission. This test philosophy was based on the belief that major electromagnetic incompatibilities would be uncovered during functional tests and that testing to demonstrate compliance with a system electromagnetic compatibility specification, such as MIL-E-6051C, would cause a schedule slip in an already protracted schedule.

It should be noted that one of the tests of the block of tests on the spacecraft included simultaneous operation of all systems to permit intersystem operation and compatibility tests. A "fix" was applied as required in the spacecraft systems to suppress any electromagnetic interference causing malfunction or performance degradation of any equipments or systems.

3.2 Integration of Gemini Spacecraft and Titan Launch Vehicle Phase

An important electrical as well as a mechanical interface exists between the Gemini spacecraft and the Titan launch vehicle. The major electrical interface is the interface between the Gemini guidance computer and the launch vehicle guidance system. To date, there has been no significant problems with the various interfaces between the spacecraft and launch vehicle when mated despite the fact that the first time the two had contact was at the launch site. This success has been attributed to the work of coordinating committees, many meetings, rigidly followed interface specification control documents, and tests performed with a simulated vehicle. The interface control committees were composed of representatives from the Manned Spacecraft Center, the McDonnell Aircraft Company, the Air Force and the Martin Company.

Tests at the launch site of the spacecraft and launch vehicle were minimal to assure there was no damage caused during transportation to the launch site until integration of the launch vehicle and spacecraft was completed on the launch pad. At this time integrated system functional checks were performed and radio frequency testing by the Air Force Eastern Test Range was performed. Finally simulated flight tests were run with assistance from the Eastern Test Range and the Mission Control Center to demonstrate proper performance of the final space vehicle systems and ground support equipments.

3.3 Integration of the Gemini Spacecraft and the Agena D Target Vehicle Phase

Similarly, for later Gemini missions, the target vehicle, the Agena D, will be integrated with the Gemini spacecraft for the first time at Cape Kennedy. At no time during the development of these vehicles was it possible to provide a compatibility test of the two flight vehicles, although several tests were run with a simulated vehicle to provide the necessary design information. However, the collar structure and electronics required on the target vehicle to accomplish rendezvous and physical docking of Gemini spacecraft and the Agena D target vehicle was supplied by the McDonnell Aircraft Company to the contractor for the target vehicle for installation and compatibility tests. As was the case for the interfaces between the spacecraft and launch vehicle, coordinating committees, frequent meetings and interface specification documents played important roles in achieving compatibility of the spacecraft and target vehicle.

To provide a final verification of the compatibility of the combined operations of the Gemini spacecraft and the Agena D target vehicle in the various orbital configurations, a series of radio frequency and functional compatibility tests for Gemini/Agena D were performed at Cape Kennedy. The tests included an orbital flight simulation of the docking maneuver of the Gemini spacecraft and Agena D target vehicle and functional compatibility tests of Gemini/Agena D in both the docked and undocked configurations. The specific objectives of these tests include: (a) demonstration of Gemini/Agena D hardline and radio frequency communication capability to verify satisfactory vehicle interfaces, (b) demonstration of proper performance of Gemini/Agena D subsystems in the combined Gemini/Agena D electromagnetic fields, and (c) demonstration of the proper performance of Gemini/Agena D subsystems during a simulated mission.

APPENDIX D

ELECTROMAGNETIC COMPATIBILITY AWARENESS COURSE

1.0 Background

It has been recognized in past programs that management has been unaware of the importance and necessity of electromagnetic compatibility planning and engineering in the design and development of a system. All too often electromagnetic compatibility of equipments, of subsystems and of the overall integrated system has not been considered until final end-item assembly and test. When interference problems arise during end-item tests, quick fixes are applied, which may compromise the original design, in order to avoid schedule slip and additional cost. However, a major redesign effort is frequently required. Because of this approach to electromagnetic interference control, little information has been gathered or disseminated in a form useful to management and engineering personnel. Consequently, the state-of-the-art of electromagnetic interference control has not kept up with the advances in other fields of electronics.

Recognizing these facts, the Reliability and Quality directorate of the Apollo Program Office has contracted the General Electric Company, Apollo Support Department, to prepare a course and a supporting text to promote an awareness of the importance of electromagnetic compatibility planning in the Apollo Program and to disseminate electromagnetic interference control information throughout the Apollo Program.

The course was to be designed to introduce the personnel, such as project management, design and test engineers and manufacturing engineers, to electromagnetic interference and associated problems and to make them aware of the possible approaches to controlling electromagnetic interference and achieving electromagnetic compatibility of the integrated system. Although it is yet too early to say how widespread the audience will be for this electromagnetic compatibility awareness course, current planning indicates that the course will be given initially to the interested personnel at the Marshall Space Flight Center, and other NASA Centers and then possibly to the prime contractors of the NASA Centers for Apollo-Saturn. The duration of the course will be one full week which will be divided into two parts. The lectures of the first day will be slanted primarily toward management although the lectures will also be of interest to the engineering personnel.

The lectures of the final four days will be aimed at the design test and manufacturing engineers, primarily the design engineers, who have not been intimately concerned with electromagnetic interference control.

A text has been prepared of the electromagnetic interference awareness course and is intended to be a constructive aid to the NASA Apollo team in their electromagnetic compatibility responsibilities. Its purpose is to stimulate interest and further promote understanding in the art and science of electromagnetic interference control throughout the Apollo Program. The first draft of this text entitled, "Electromagnetic Compatibility Principles and Practices," dated May, 1965, was issued with a limited distribution as part of an Apollo Design Reliability series of documents. This first draft of the text is currently in publication by the Government Printing Office and is expected to be widely distributed as a whole or in individual chapters throughout the government and industry. It is hoped that feedback of technical additions and corrections to the material contained in the text will result from the recipients of the documents. The text, it is hoped, will evolve into a set of reference documents which present, in technical depth, the information needed by the engineer to properly design equipments for interference free operation and compatibility within a system, including design guides and techniques, prediction methodology, and testing methods and procedures.

2.0 Presentation of the Electromagnetic Compatibility Awareness Course

A NASA sponsored workshop on electromagnetic compatibility was held in Daytona Beach, Florida, on November 1 through November 5, 1965, which served as a preview of the Electromagnetic Compatibility Awareness Course. Representatives from industry, the various NASA centers, the Department of Defense, and the Electromagnetic Compatibility Analysis Center and some of the members of the G-46 Committee on Electromagnetic Compatibility of the Electronics Industries Association participated in this workshop. The objectives of the workshop were to stimulate discussions of and information exchange on different aspects of electromagnetic compatibility and to obtain comments from acknowledged electromagnetic interference control specialists on the organization of, the effectiveness of the presentation of, and the appropriateness, completeness, and technical accuracy of the material contained in the course and in the draft of the supporting text. The information and comments obtained during the workshop sessions was to be used by the General Electric Company for reworking and refining the Electromagnetic Compatibility Awareness Course.

With Dr. C. Frederick, a recognized leader in the field of electromagnetic compatibility noted for the Frederick Research Corporation four volume publication, "Handbook on Radio Frequency Interference", acting as technical coordinator, the Electromagnetic Compatibility Awareness Course was reworked and the lecture notes were revised. The revised week-long course is divided into 16 one hour lectures with a planned one hour discussion period following each lecture. The revised course was given at the Michoud Operations Plant in New Orleans, Louisiana during the week of April 17, 1966 to representatives from NASA, the Chrysler Corporation Space Division and the Boeing Company. Although the revised lecture notes were issued to the course attendees, the Apollo Program Office publication, "Electromagnetic Compatibility Principles and Practices", was used as the text for this course.

APPENDIX E

LIGHTNING PROTECTION OF APOLLO-SATURN LAUNCH COMPLEXES

There are two important modes by which lightning strikes can cause damage which must be considered in the development of a plan for the protection of launch complexes from lightning strikes. These are (a) direct stroke effects which could cause damage in the form of burning, blasting, or destructive currents or voltage surges in electrical equipment and (b) damage effects of currents or voltages induced in nearby wiring and sensitive electrical equipments by electromagnetic fields produced by the lightning stroke.

An investigation was conducted by the High Voltage Laboratory of the General Electric Company under NASA contract NASw-410 to determine the possible effects of lightning strokes to Apollo-Saturn launch complexes, to review standard practices and possible solutions to unique problems, and to recommend practices and procedures to protect the launch complexes from lightning strikes or to minimize the effects of such strikes. The results of this investigation of protection of personnel, facilities, ground support equipment and space vehicle (in transit or on the pad) on Launch Complexes 34, 37, and 39 have been published in the following two General Electric Company reports which have been distributed to interested personnel at the Kennedy Space Center, the Manned Spacecraft Center, and the Marshall Space Flight Center:

(a) "Lightning Protection for Saturn Launch Complex 39," dated September 10, 1963, and

(b) "Analysis of Lightning Effects on Launch Complex 34 and 37," dated July 1, 1964.

These reports were reviewed and the lightning protection measures recommended were evaluated by the Bell Telephone Laboratories, Inc. The results of this review and evaluation are included in a memorandum attached to this appendix.

An ad-hoc lightning protection committee was formed in 1963 by NASA to direct the effort of the High Voltage Laboratory of the General Electric Company in this investigation and to develop plans for protecting launch complexes 34, 37, and 39 against lightning strikes. This committee, chaired by Mr. H. D. Brewster of the Kennedy Space Center, consisted of representatives of the three NASA centers. A plan was developed for the protection of Launch Complex 39 by the ad-hoc committee using the results of the investigation performed by the General Electric Company. This plan was subsequently established as policy for the Kennedy Space Center by the Director on June 18, 1963. This plan appears in the Kennedy Space Center technical report, TR-4-28-2-D, entitled, "Lightning Protection for Saturn Launch Complex 39," by H. D. Brewster and W. G. Hughes, dated October 18, 1963.

A similar lightning protection plan was developed for Launch Complexes 34 and 37 and was established as policy for the Kennedy Space Center by the Director on November 30, 1964. This plan for Launch Complexes 34 and 37 will be the final objective, but it is recognized that complete implementation of this plan may not be practical or economical if it is imposed on existing facilities for immediate action.

For the most part the lightning protection plans involve activities to be conducted by the Kennedy Space Center. However, the plans include two items which require action from the Manned Spacecraft Center and the Marshall Space Flight Center. These items are (a) that the exterior cable tunnels on the space vehicle should be made of conducting material and (b) that the maximum voltages expected to be induced in circuits provided by the NASA center as a result of a lightning strike should be examined and protective devices put into the circuits if required.

At the present time, the coordination of the inter-center lightning protection activities for Apollo-Saturn has come under the purview of the Electrical Systems Panel.

APPENDIX F

ELECTROSTATIC CHARGING

1.0 General

Some concern has been expressed about possible problems in Apollo resulting from electrostatic charging of the space vehicle during the various phases of the mission. These problem areas include: (a) radio frequency interference during discharging, (b) arcing between two vehicles during separation or during rendezvous causing physical damage, (c) discharges of sufficient rapidity and energy to induce transient pulses of significant size in sensitive electrical circuits, such as electro-explosive devices to cause a malfunction, and (d) adhesion of dust layers to the LEM and extra-vehicular astronauts during lunar surface operations.

Electrostatic charging and discharging mechanisms are discussed qualitatively in the following paragraphs for launch, space coast, and lunar surface operations. Results of some of the experiments performed in this field are also included. Electrostatic charging due to rocket engine operation is discussed in the section on launch operations, but the discussion is also applicable to the other sections if the engines are operated during those phases.

2.0 Launch

Many processes have been postulated by which a space vehicle can become electrostatically charged during the launch phase. The dominant processes are expected to be external triboelectric charging and charging processes resulting from rocket engine operation.

When an electrostatically charged, isolated, electrically conducting body is asymmetrically separated into two bodies, a potential difference will be developed between the two bodies. If the potential difference were great enough, a redistribution of electrostatic charges by a spark, arc-over, or current flow between the two bodies could occur if an ionized gas or other conducting path were present. This phenomenon might be expected to be applicable to stage separation of an electrostatically charged vehicle such as the Saturn launch vehicles.

2.1 Theoretical

2.1.1 Triboelectric Charging

The triboelectric (frictional or contact) charging process has long been recognized as a problem when aircraft are operated in precipitation or in an atmosphere where there are many suspended particles. Aircraft potentials measured in the hundreds of kilovolts have been attained as a result of triboelectric charging in the atmosphere. Triboelectric charging occurs whenever two dissimilar materials with different charge carrier mobilities are brought into contact and then separated. The charging rate and the polarity of the resulting potential depends on the materials involved.

Although it is not likely that Apollo-Saturn will be launched in adverse weather conditions where precipitation is present, a partial cloud cover, ice particles, and possibly other particles may be suspended in the atmosphere through which the space vehicle will rise.

Another possible instance of electrostatic charging through frictional effects will be when the boost protective cover is snatched from the spacecraft when the launch escape system is jettisoned.

2.1.2 Rocket Engine Charging

Many phenomena are expected to occur during rocket engine operation which could serve to charge the space vehicle electrostatically, although some may have negligible effect with respect to other phenomena. It is believed the more significant of these phenomena include:

(a) diffusion of electrons to the rocket engine wall due to the greater mobility of the electrons in the plasma formed in the combustion chamber during rocket engine operation,

(b) triboelectric or frictional charging effects caused by the solid particles in the combustion products striking the rocket engine walls, and

(c) thermionic emission or photoemission resulting from the combustion process within the rocket engine.

To produce large potential differences, the uncollected charges must be mechanically removed to large distances from the space vehicle.

The potential of the space vehicle will be limited by some discharging processes such as corona discharge in the atmosphere or the ability of the charged space vehicle to capture a discharging current from the ionized and charged exhaust from the rocket engine. This recirculation of the charges from the exhaust to this space vehicle will depend upon the relative velocity between the charges in the exhaust and the space vehicle, the mean free path of the charge, electron (or ion depending on polarity of space vehicle) and the charge mobility. Thus, as the space vehicle gains altitude, the recirculation properties associated with the plume should be enhanced.

2.2 Experimental

There have been a number of electrostatic charging experiments performed with small solid-fuel rocket motors during static firings. Although all of these experiments indicate that solid-fuel rocket motor operation is a definite cause of vehicle electrostatic charging, little insight into the actual mechanisms has been gleaned from the experiments. In addition, little use can be made of the results of the experiments in predicting the potentials a space vehicle may attain during a launch because the experiments were performed in an environment different than that which a space vehicle would encounter during an actual launch.

Results of electrostatic charging tests on various small rocket motors performed by the Boeing Company in 1963 were "scaled" to apply to the Minuteman vehicle. The tests showed that a negative charge was deposited on a rocket motor burning a Minuteman-type grain. Extending these results suggested to Boeing that a Minuteman vehicle might attain a potential to cause electrical discharge. In order to control any discharge and decouple it from sensitive circuits in the vehicle, special corona dischargers were developed for easy installations on the vehicle. If electrostatic charging of the Saturn vehicle is critical during launch, this method of discharging the vehicle may have application to the Saturn launch vehicle during passage through the atmosphere.

An instrument was installed on a Nike-Cajun vehicle by the Stanford Research Institute in 1961 to measure the field strength at a point on the Nike-Cajun vehicle skin in order to obtain a time history of the potential of the vehicle during launch powered flight. Polarity discrimination was not incorporated into this instrument nor was the ability to discriminate between a true field produced by an electrostatically charged vehicle and electron or ion collection by the instrument in the ionosphere. During the Nike burning the vehicle reached a peak potential of 26 kilovolts, but during the Cajun burning the peak potential was only about two kilovolts. In addition, fairly high vehicle potentials were measured above 100 km.

A similar experiment was conducted on the Nike-Cajun vehicle by the Stanford Research Institute in 1965. However, the instrument used to measure the electric field strength was equipped with a synchronous detection scheme designed to enable the instrument to distinguish between a true electric field and a drift current caused by ion or electron collection from the environment by the instrument. The results of the experiment paralleled the results of the earlier experiment performed in 1961, with the exception that the potential of the vehicle while in the ionosphere never exceeded 5 kilovolts. It is believed that the indication of high vehicle potentials in the ionosphere during the 1961 experiment was caused by collection of ions or electrons from the ionosphere by the instrument.

Ling Temco Vought Aerospace Corporation instrumented a Scout vehicle to measure the field strength at certain points on the vehicle skin in order to ascertain a time history of the vehicle electrostatic potentials during launch and to measure the potential differences between stages during separation maneuver. Four sensors were used to make two surface field strength measurements and two stage separation potential difference measurements. The NASA Scout Evaluation Vehicle, S-131R, was launched in August, 1965.

Surface measurements of the field strength were made on the second and third stages of the NASA Scout Evaluation Vehicle. The measurements indicated that the vehicle reached a peak potential of less than 1 kilovolt during the rocket motor operation of the first three stages of the Scout vehicle.

Instruments were carried on the NASA Scout Evaluation Vehicle to determine the maximum amplitude of the potential difference between the system ground points of the first stage and the second stage during the first stage separation maneuver, and between the system ground points of the second stage and the third stage during the second stage separation maneuver. A peak instantaneous voltage detector with a rapid rise-time response was used to record the peak potential differences or transients during separation and the results were later telemetered to the earth. During the August, 1965, launch of Scout vehicle, S-131R, there were no indications of recorded potential differences between the two parts of the vehicle during the separation maneuver.

An experiment was carried on SA-10 to measure the potential difference between the S-I booster stage and the S-IV stage of the Saturn I launch vehicle during separation. A continuous voltage measurement was made between the two stages during the separation maneuver using a telemetry channel with a frequency response of 80 cps. An instantaneous peak voltage detector to record any high frequency voltage pulses was not used. The result of the experiment indicated no perceptible difference between the two stages during separation.

3.0 Space Coast

The more important mechanisms for the electrostatic charging of a space vehicle coasting in space are collection of electrons or ions from the environmental plasma, (either the ionosphere or the solar wind), and photoemission. Other mechanisms include (a) thermionic emission, (b) secondary emission, (c) collisions with other bodies such as micrometeors, and (d) field emission.

3.1 Theoretical

3.1.1 Electron or Ion Collection

A space vehicle in space will be continuously bombarded by environmental particles which will be electrically charged, such as electrons, ions, and cosmic rays. The incidence of such particles, in general, will result in a transfer of charge to or from the space vehicle. Since the electrons are much more mobile than the positive ions, the space vehicle will probably accumulate a negative charge. The electric field around the space vehicle produced by the charge on the space vehicle will reduce the number of electrons reaching the space vehicle until equilibrium is reached.

Electron and ion concentrations vary with altitude, time, and solar activity. In addition, the Van Allen belts in the ionosphere contain high energy particles. The discharging or charging capability of the electrons or positive ions are dependent on the energies and the concentrations of the particles and on the relative velocities of the particles with respect to the space vehicle. In the solar wind, the positive ions are characterized by having mostly directed energy, while the electrons are generally randomly directed.

For the case of a space vehicle orbiting in the ionosphere, equilibrium potentials are expected to be a few volts negative, although somewhat higher potentials may be attained when passing through the Van Allen radiation belts. It is also expected that the equilibrium potential of the space vehicle in the solar wind will be limited to a few volts due to this charging mechanism of positive ion or electron collection.

3.1.2 Photoemission

As the altitude of the space vehicle increases, the concentrations of electrons and positive ions in space decreases and the process of photoemission becomes more important as a charging or discharging mechanism. Photoemission is the process by which incident photons of sufficient energy will cause the emission of electrons from a space vehicle bathed in sunlight. The total electron flow from the space vehicle depends on the intensity of the radiation, the area presented by the space vehicle to the sun, the surface material of the space vehicle, and the potential of the space vehicle.

Since most of the electrons emitted from the space vehicle surface due to the photoelectric effect will be low energy, it is expected that the space vehicle equilibrium potential for this process alone will be limited to a few volts positive.

3.1.3 Field Emission

Field emission is the name of the process by which electrons are drawn out of metal surfaces by a large electrostatic field. Under the action of electrostatic fields of

10^6 -10^7 volts/cm and greater, electrons may be drawn out of metal surfaces by the field alone. Fields of this order of magnitude may be reached with fine points and corners for a potential difference as low as 1,000 to 10,000 volts. Since the discharge current rises exponentially with potential difference above this critical value, the largest negative potential of the space vehicle in space is expected to be limited to close to this critical value.

3.2 Experimental

The Explorer VIII Satellite, launched in 1960, with a perigee of 420 kilometers and an apogee of 2300 kilometers contained instruments to measure electron density and temperature, positive ion concentration and mass, and the interaction between the vehicle and the ionized atmosphere. The results obtained from the single-grid electron trap carried by this satellite were analyzed and the satellite equilibrium potential was calculated. It was shown that there is general agreement between the predicted value of equilibrium potential of a few volts and the measured value.

An experiment was carried on the Gemini IV spacecraft to measure the Gemini spacecraft electrostatic potential via the measurement of electric field strength during earth orbital coast. The data from this voltage experiment indicated that very high positive spacecraft potentials (2 x 10^6) existed for extended periods of time. These results were counter to previous experimental results as well as to the current theoretical understanding of electrostatic charging of satellites orbiting in the ionosphere. It was postulated that the apparent spacecraft potential was due to some cause other than true spacecraft potential, such as the effects of positive ion or electron collection by the sensor. Post flight testing of the experiment instrument indicated the instrument to be sensitive to radio frequency energy and fluxes of charged plasma particles incident to the sensing face of the instrument.

The same experiment was carried on the Gemini V spacecraft modified to exclude electric fields from the sensor while allowing the sensor to detect unbalanced spacecraft currents which may have been collected by the sensor carried on the Gemini IV spacecraft. The readings of the sensor telemetered to the earth from the Gemini V spacecraft were very similar to the data obtained from the Gemini IV spacecraft. In view of these results, the data from Gemini IV cannot be interpreted in terms of spacecraft potential but must be interpreted in terms of unbalanced spacecraft currents.

4.0 Lunar Surface Operations

The effects of the various postulated electrostatic charging and discharging mechanisms on lunar operations and crew extra-vehicular activities on the lunar surface cannot be as accurately estimated as are the effects on the launch and space coast operations because of the greater number of unknowns. The more dominant electrostatic charging or discharging mechanisms are expected to be the motion over the dry surface of the moon, the solar wind, and photoemission.

Knowledge of the electrical conductivity and the dielectric constant of the materials composing the local lunar surface and the Extra-Vehicular Mobility Unit is required in order to determine the magnitude and polarity of any electrostatic charge accumulated by an extra vehicular astronaut moving across the lunar surface. It is not expected that this process will result in dangerous charge accumulations.

It is expected that the electrostatic discharging qualities of the solar wind will be sufficient to dissipate any large charge accumulations due to motion over the lunar surface unless the lunar surface is shielded from the solar wind. If the space vehicle and astronauts are on the dark side of the moon, the solar wind could only be effective in dissipating positive charges because the positive ions have mostly directed energy while the electrons are mostly thermalized.

The photoelectric effect will also aid in dissipating a negative charge on the space vehicle or astronauts who are located on the sunlit portion of the lunar surface.

5.0 Conclusions

It would be expected that first stage separation would be the most critical portion of the launch phase with respect to the possible deleterious effects resulting from electrostatic charge accumulations on the skin of the space vehicle. However results of experiments on a Saturn I Launch Vehicle and on a Scout vehicle suggest that arcing between launch vehicle stages during separation will not be a problem.

The plasma of the ionosphere will act as an efficient discharge mechanism as the space vehicle rises into the ionosphere during the launch maneuver and is expected to limit the electrostatic charge accumulation on the space vehicle to non-hazardous amounts. It is known that steady state potentials of space vehicles coasting in earth orbit in the ionosphere will be a few volts negative. If a space vehicle should acquire a large electrostatic charge by an independent process such as rocket engine operation, the ionosphere will act to discharge the space vehicle rapidly.

For space vehicles coasting in outer space, it is expected that the equilibrium potential of the space vehicle will range from a few volts positive to a few volts negative depending upon which effect predominates, photoemission or the solar wind. The solar wind will act in a manner similar to the ionosphere but at a reduced rate in discharging a space vehicle acquiring a large electrostatic charge by an independent process.

The effects of the solar winds, photoemission, and contact friction due to motion on the lunar surface during lunar surface operations are not expected to cause dangerous electrostatic charge accumulations on the LEM or the extra-vehicular astronauts.

In summary, although it is recognized that more experimental data would be desirable to verify some of the above assumptions and conclusions, space vehicle electrostatic charging and possible resulting effects are not expected to cause hazardous situations for the space vehicle or the astronauts during any of the Apollo mission phases.

APPENDIX G

VOLTAGE BREAKDOWN IN ELECTRICAL AND ELECTRONIC EQUIPMENTS

1.0 General

Voltage breakdown in the form of glow, corona, flash-over, streamer discharge, or arc discharge has resulted in or has been suspected of causing the failure of space electronic equipment as well as causing electromagnetic interference. The probability of voltage breakdown is increased by operation of electronic equipment in the critical gas or air pressure region or operation of electronic equipment in a plasma. In addition, breakdown can occur in a vacuum by means of the multipactor effect. These mechanisms will be discussed in the following paragraphs.

2.0 Critical Gas or Air Pressure

The probability of voltage breakdown is enhanced by operating the equipment in reduced gas pressures corresponding to atmospheric pressures found at altitudes from 65,000 to 310,000 feet. For example, within this critical pressure region, the air dielectric will break down with less than 300 volts across exposed flat plates separated by an appropriate distance.

Operation of electronic equipment in the critical pressure region not intended for operation in those environments can occur as a result of (a) partial loss of vacuum during vacuum testing, (b) decompression of pressurized vessels, (c) repressurization after decompression, (d) inadvertent equipment activation during the launch phase as the space vehicle passes through the thinning atmosphere, (e) outgassing of materials producing localized critical pressures under vacuum conditions, and (f) gases trapped in enclosures inadequately vented to the ambient environment.

Following are a few examples of voltage breakdown problems occurring in space vehicle equipments.

(a) It is highly probable that the failure of the television subsystem of Ranger VI was caused by voltage breakdown after inadvertent turn-on of the television subsystem during the launch phase.

(b) During the thermal-vacuum testing of one of the Mariner spacecraft when all electrical subsystems were operating, an inadvertent increase in the pressure in the test chamber subjected the Mariner spacecraft to operation in the critical pressure region with disasterous results to much of the electronic equipment.

(c) Two failures occurred in the camera subsystems of the Nimbus spacecraft when the camera subsystems were accidentally activated when the pressure in the environmental chamber was within the critical zone. Also uncovered during the thermal-vacuum tests on the Nimbus spacecraft were voltage breakdowns between various conductors within partially sealed connectors.

(d) Problems with the photomultiplier of the spectrumheliograph carried by OSO-2 in earth orbit were attributed to high voltage breakdown.

(e) Glow discharges in the transponder used with the rendezvous radar subsystem of the Gemini spacecraft observed during thermal-vacuum testing was traced to partial pressures produced in the poorly ventilated enclosure of the high voltage circuitry by outgassing of a silicon grease.

(f) During thermal-vacuum testing of various Apollo Lunar Excursion Module equipments, voltage breakdowns occurred in the VHF transceiver and in the Unified S-Band transponder.

In addition to voltage breakdown in electrical and electronic equipments, voltage breakdown of space vehicle antenna subsystems is a possible problem area. The field strength required for antenna breakdown is considerably less for operation in a plasma or in the presence of a very thin ambipolar diffusion layer (space charge present) than for operation in non-ionized atmospheres.

3.0 Plasma

It has been shown in a laboratory that arc discharges between two electrodes can occur at voltages as low as 30 volts dc in the presence of a plasma. A scintillation discharge is often sufficient to trigger such an arc discharge.

A scintillation discharge is produced in the following manner. When a thin dielectric film, such as an oxide film, is present on a conductor the voltage difference between the plasma and the conductor is impressed across the thin film resulting in large voltage gradients for modest voltage differences (30 volts). These gradients are sufficient to break down the thin film.

During a series of Agena flights, an occasional 28 volt battery short circuit occurred at the time of booster separation. It was postulated, although never proven, that the exhaust from retrorocket operation entered the aft section of the Agena vehicle and the 28 volt battery was shorted out by an arc discharge between two connector pins in the presence of a plasma triggered by a scintillation discharge. After the hole where the exhaust plasma could have entered the aft section of the Agena vehicle was plugged, the problem of 28 volt battery shorting never reappeared.

There are times during a space mission when the space vehicle antennas are in a conducting medium as would be encountered in the ionosphere or in the plasma surrounding the vehicle during re-entry into the atmosphere and sometimes during launch. The field strength necessary to cause voltage breakdown of an antenna in a conducting medium is considerably less than that required to cause voltage breakdown of an antenna in an air dielectric. Voltage breakdown at an antenna is evidenced by a sharp increase in reflected power and a reduction of power radiated from the antenna.

4.0 Multipactor Breakdown

Multipactor breakdown is a radio frequency resonant discharge dependent upon electrode material and spacing, gas pressure, radio frequency voltage and frequency. This breakdown is caused by electrons being accelerated between two electrodes connected to a radio frequency source and the multiplication of free electrons between the electrons by secondary emission. The exciting voltage must be at a frequency corresponding to the electrode spacing such that a polarity reversal will occur on the electrodes at the moment of electron impact with each electrode. The electrode material must be such that more than one electron will be released from the electrode for each electron impacting the electrode when accelerated by a sufficiently large radio frequency voltage between the electrodes. In addition, the mean free path of the electrons must be sufficiently long to allow the accelerated electrons to impact the electrode alternately in phase with the radio frequency voltage.

5.0 Summary

The major cause of voltage breakdown failure of electrical and electronic equipments experienced in space vehicles to date has been the operation of these equipments in the critical air or gas pressure region. Critical gas pressures in the operating environment of these equipments could be caused by decompression of the space vehicle and subsequent repressurization, outgassing of materials in a vacuum, and inadequate venting of equipment enclosures.

Although failures in electrical and electronic equipments can result from multipactor breakdown which should not be overlooked it is a phenomenon requiring specialized conditions and is not likely to be a problem.

APPENDIX H

SATURN APOLLO ELECTROMAGNETIC COMPATIBILITY DESIGN CRITERIA

ICD 13M60004

A copy of the Interface Control Document (ICD) 13M60004 entitled, "Saturn Electromagnetic Compatibility Design Criteria", approved by the co-chairmen of the Instrumentation and Communication Panel in August, 1965 is attached to this appendix. ICD 13M60004 covers the electromagnetic interface between the Apollo spacecraft and the Saturn launch vehicle. It recognizes the Electromagnetic Compatibility Subpanel of the Instrumentation and Communication Panel as the agency responsible for the examination and subsequent resolution of any suspected incompatibilities of the interface between the spacecraft and the launch vehicle.

The major portion of this ICD is devoted to defining wiring design criteria including requirements on separation, shielding, routing, and bundling based on the classification of circuits to be interconnected. It invokes the conducted and radiated interference requirements of the military electromagnetic interference control specifications. MIL-I-26600 (as amended by MSC-EMI-10A) and MIL-I-6181D on the individual equipments of the spacecraft and the launch vehicle, respectively. In addition to including those design requirements contained in the military electrical bonding specification MIL-B-5087A, this ICD includes other requirements to ensure proper electrical bonding across the interface of the spacecraft and the launch vehicle.

SATURN APOLLO

ELECTROMAGNETIC COMPATIBILITY

DESIGN CRITERIA

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C						
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ORIG	R W Williams	5 Aug 65			[illegible]	[illegible]
ISSUE	NAME	DATE	NAME	DATE	NAME	DATE
	MSC		LOC		MSFC	
APPROVAL						

REPOSITORY: THE GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE ALABAMA
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

PREPARED FOR: INSTRUMENTATION AND COMMUNICATIONS PANEL

APOLLO
INTERFACE DOCUMENT

MSFC - Form 1916 (August 1963)

This document shall not be used for manufacturing, procurement of hardware, inspection of manufactured items or assembly but shall govern pertinent design documentation (Class I & II drawings, etc.). Revisions to this document or the properly identified pertinent design documentation can only be made with approval of the responsible interface authority.

Interface Control Document
13M60004

PROPOSED INTERFACE CONTROL DOCUMENT FOR THE APOLLO SPACECRAFT/LAUNCH VEHICLE ELECTROMAGNETIC COMPATIBILITY DESIGN CRITERIA

13m60004

TABLE OF CONTENTS

LIST OF EFFECTIVE PAGES

1.0 INTRODUCTION

This document establishes the criteria necessary for the generation of electromagnetic interference requirements which, when implemented in the space vehicle, will promote electromagnetic compatibility of the integrated space vehicle configuration.

This criteria provides the basis for a common approach to the task of controlling interfacing electromagnetic interference. This common approach will provide confidence that circuitry originating in the launch vehicle will receive proper and necessary protection from EMI wherever it is routed in the spacecraft and vice-versa. In addition, this common approach will provide compatible electrical bonding procedures at all interfaces; each one of which affects the entire space vehicle.

2.0 SCOPE

This document and the criteria delineated are intended to provide a common philosophy between centers for the control of EMI and the subsequent promotion of electromagnetic compatibility for each space vehicle. Although these criteria provide boundaries within which each center is expected to operate, they do not usurp the prerogative of either center in granting deviations and/or waivers; however, when deviations and/or waivers are obtained, the EMC Subpanel will determine to what extent the ICD is affected and, if necessary, re-submit to the center which approved the deviation for reconsideration.

3.0 DOCUMENTATION

3.1 Associated Interface Documentation

The Interface Document Log indexes documents which, not a part of this ICD, are intended as companion documents to specify applicable information not part of Instrumentation and Communication Panel's responsibility.

3.2 Applicable Documents

The following documents form a part of this ICD to the extent specified herein:

MIL-I-26600 and Addendum MSC-EMI-10A	Interference Control Requirements
MIL-I-6181D	Interference Control Requirements, Aircraft Equipment
MIL-B-5087A	Electrical Bonding, Aircraft

3.3 Precedence of Documentation

This ICD takes precedence over all non-negotiated documents describing the interfacing parameters, relative to electromagnetic compatibility, between the Apollo Spacecraft and Launch Vehicle.

3.4 Amendment to Documentation

Amendments to this document shall be subject to approval by the I&C Panel.

4.0 DEVIATIONS

The generation and implementation of requirements which are contrary to the criteria defined in this document, are the prerogative of each center; however, the EMC Subpanel may review any such deviations for potential EMC incompatibilities.

5.0 EMC CRITERIA

5.1 Equipment

5.1.1 The launch vehicle equipment shall comply with the conducted and radiated requirements of MIL-I-6181D.

5.1.2 The spacecraft equipment shall comply with the conducted and radiated requirements of MIL-I-26600/MSC-EMI-10A.

5.1.3 The location and gain of each transmitter antenna together with the transmitter antenna pattern (established on the space vehicle). shall be made available to the EMC Subpanel of the I&C Panel as they become available, together with the transmitters' full operating parameters, power - modulation - harmonic and spurious contents, type transmission, and mission time operation.

5.2 Circuits

The criteria for promoting electromagnetic compatibility of circuitry crossing the launch vehicle/spacecraft interface is defined in the following paragraphs.

5.2.1 Power

There shall be no electrical interconnection of launch vehicle power with the spacecraft power.

5.2.2 Circuit Returns

5.2.2.1 All circuits shall be routed and twisted with their returns for their entire length. Adjacent pins in connectors shall be used for transferring the circuit across the interface. Each circuit and its return must be fully identified by the Electrical Systems Integration Panel.

5.2.2.2 Except in these cases using coaxial cable shield as a circuit return, all circuit returns shall be maintained everywhere, isolated from structure ground when routed through any portion of the space vehicle not under the control of the NASA center using the circuit information.

5.3 Wiring

The reduction of electromagnetic interference in the interface wiring and the implementation of electromagnetic compatibility requirements for audio frequency and radio frequency classified interfacing circuits shall be accomplished by isolating incompatible circuits through the implementation of the bundling, routing, shielding, and separation requirements presented herein.

5.3.1 Definitions

5.3.1.1 Audio Frequency Circuit

An audio frequency circuit shall be defined as a circuit operating in the frequency range DC to 50 kilocycles or having signal rise and fall times greater than 20 microseconds.

5.3.1.2 Pulse Circuit

A pulse circuit shall be defined as a circuit whose rise or fall time is less than or equal to 20 microseconds, and whose minimum repetition rate is 10 pulses per second.

5.3.1.3 Radio Frequency Circuit

A radio frequency circuit shall be defined as a circuit operating at frequencies in excess of 50 kilocycles.

5.3.1.4 Circuit Classification

Circuit classification shall be defined as the categorizing of circuits into a particular group by applying specific frequency, impedance, and voltage limits (see Table 1) to each circuit.

5.3.1.5 Bundling

Bundling shall be defined as the combining of circuits which have identical circuit classifications for the purpose of forming a single compatible group of circuits.

5.3.2 Parameters Used for Circuit Classification

5.3.2.1 Frequency/Rise or Fall Time

The frequency of the circuit shall be the primary frequency component present in the circuit under consideration. However, a circuit whose waveshape is characterized by a rise or fall time that is less than or equal to 20 microseconds shall be classified by the rise or fall time rather than the primary frequency.

5.3.2.2 Loop Impedance

The impedance considered for classification purposes shall be the loop impedance of the interconnecting circuit, that is, the sum of the load impedance and the source impedance.

5.3.2.3 Voltage or Sensitivity

The voltage considered for circuit classification purposes (Table 1) shall be the maximum voltage at the source output. However, when a circuit is deemed susceptible to external disturbance, the minimum sensitivity of the load shall be used for classification purposes, rather than the voltage. Sensitivity shall be defined as the minimum amount of voltage which when induced into the circuit, will result in the degradation of the circuit operation or quality.

5.3.3 Wiring Procedure

The wiring procedure of Paragraph 5.3.3.3 shall be implemented when the basic requirements of Paragraphs 5.3.3.1 and 5.3.3.2 have been satisfied.

5.3.3.1 Basic Requirement for Audio Frequency and Pulse Circuits

A circuit return or AC neutral shall accompany each of the following in every bundle in which the following are routed.

(a) Each audio frequency or pulse signal

(b) Each DC power lead

(c) Each group of three phase AC "Y" connected power leads

(d) Each single phase AC power lead if routed separately from group of three phase AC power leads

5.3.3.2 Basic Requirement for Radio Frequency Circuits

A return for each radio frequency circuit shall always accompany each radio frequency signal. Under no circumstances shall any RF transmission line be used that does not inherently provide a return and an RF shield (the RF shield and the signal return may be the same in the case of coaxial cable).

5.3.3.3 Wiring Procedure

The following steps shall be followed in the implementation of the wiring procedure.

TABLE I

Freq. or Rise and/or Fall Time	Loop Impedance (ohms)	Voltage or Sensitivity	Bundle Type & Circuit Classification	Wire Type Req'd.	Shield Grounding Reqt's.
Direct Current (DC)	<200	≤1v	ML	TWS	SPG
		>1v to ≤32v	HO	TW	NONE
		>32v	EO	TW	NONE
	200-600	≤5mv	ML	TWS	SPG
	≥600			TWDS	SPG
	>300	>5mv to ≤1v	ML	TWS	SPG
		1v to ≤32v	HO	TW	NONE
		>32v	EO	TW	NONE
>0 to 50 KC or Rise and Fall Times 20 Micro Seconds	< 200	≤1v	ML	TWS	SPG
		>1v to ≤25v	HO	TW	NONE
		>25v	EO	TW	NONE
	200-600	≤5mv	ML	TWS	SPG
	≥600			TWDS	SPG
	>200	>5mv to ≤1v	ML	TWS	SPG
		>1v to ≤25v	HO	TW	NONE
		>25v	EO	TW	NONE
Circuits with rise or fall time 20 microseconds	All	≤5mv	ML	TWDS*	SPG
		>5mv to ≤1v	ML	TWS*	SPG
	<1000	>1v to ≤15v	EO	TWS*	MPG
	≥1000			TWDS*	MPG
	All	>15v	PO	TWINAX	MPG
>50 KC	All	All	RF		MPG

*If the capacit[illegible] per foot is critical, Twinax should be used.

*Twists per foot spe[illegible]ied in Table III.

SYMBOLS USED
KC - Kilocycles
MV - Millivolts
SPG - Single Point Ground
MPG - Multiple Point Ground
TW - Twisted
TWS - Twisted Shielded
TWDS - Twisted Double Shield
RF - Radio Frequency
< - Less than
≤ - Less than or equal to
> - Greater than
≥ - Greater than or equal to

STEP I - CLASSIFY ALL CIRCUITS

All interface circuits shall be classified by assigning a circuit classification (Table I) to each circuit. The circuit classification shall be assigned to each wire associated with each circuit. This circuit classification shall appear on all schematics, wiring diagrams, and ICD's in which the circuit appears.

STEP II - DETERMINE THE WIRE TREATMENT REQUIRED FOR EACH CIRCUIT

Assign to each interfacing circuit the appropriate wire type required as stated in Table I, and determine the shield grounding requirements as specified in Table I for those wire types which require shielding. The wire type and shield grounding shall be shown on all schematics, wiring diagrams, and ICD's in which the circuit appears.

STEP III - BUNDLE ALL CODED CIRCUITS

All interfacing circuits having identical circuit classifications and routed in the same area shall be combined into a single bundle, when possible, and the bundle shall be coded with a bundle code which is the same as the circuit classification of the circuits which it contains. This bundle code shall be designated on all drawings in which the bundle appears.

STEP IV - INSTALLATION OF BUNDLES

All bundles which interface the launch vehicle and spacecraft shall be installed according to the separation requirements of Table II.

TABLE II - EDGE TO EDGE BUNDLE SEPARATION REQUIREMENTS

Bundle	Routed Parallel to Bundle	Separation (In inches for paralled runs of D (FEET) $1' > D$	$1' \leq D < 3'$	$D \geq 5'$
	HO	0	1.0	4.0
ML	EO	0	1.5	6.0
	PO	0	2.0	8.0
	RF	0	2.5	10.0
	EO	0	0.5	2.0
HO	PO	0	1.0	4.0
EO	RF	0	1.5	6.0
EO	PO	0	0.5	2.0
	RF	0	1.0	4.0
PO	RF	0	0.5	2.0

NOTE: If bundle coding has not been accomplished throughout the launch vehicle or spacecraft, the interface wire bundle of one center, running through the other centers vehicle, must be separated from other wire bundles by 6 inches and antenna feeds by 10 inches.

5.3.4 Detail Requirements

Step I - Classify all Circuits. The necessary parameters required to determine the classification of a circuit are, in the order required: (1) frequency, rise or fall time; (2) loop impedance; and (3) voltage or sensitivity. Referring to Table I, a circuit shall be classified by determining circuit parameters in the following order:

5.3.4.1 Frequency/Rise or Fall Time

The first parameter that shall be determined for circuit classification is frequency. If a circuit does not operate at DC or at a continual sinusoidal frequency, then the rise and/or fall times of the waveshape must be considered. A circuit which has a waveshape whose rise or fall times are less than or equal to 20 microseconds, shall be classified by the rise or fall time rather than the frequency. A circuit which has a waveshape whose rise and fall times are greater than 20 microseconds, shall be classified in the frequency range 0 to 50 KC.

5.3.4.2 Loop Impedance

The loop impedance (sum of source and load impedances) is the second parameter that shall be determined for circuit classification. Referring to Table I, loop impedance is not required for circuits with a rise or fall time less than or equal to 20 microseconds or circuits with frequencies greater than 50 KC. The loop impedance, however, shall be determined for each circuit in order to determine the wire type required for each circuit as specified in Table I.

5.3.4.3 Voltage or Sensitivity

The final parameter that shall be determined in order to assign a circuit classification is the maximum voltage that appears on the source output or the minimum sensitivity of the load, which ever value is the lowest.

5.3.5 Step II - Determine the Wire Treatment Required for Each Circuit

5.3.5.1 Wire Type for Audio Frequency and Pulse Circuits

A wire type shall be assigned to each audio frequency and pulse circuit as specified in Table 1. The parameter required for assigning the wire type is loop impedance. The wire types called out in Table 1 are general in nature and do not relieve responsibility of the circuit designer for specifying the allowable capacitance and attenuation characteristics that determine the exact construction of a wire type to meet the general requirements of wire types stated in Table 1 and defined as follows:

5.3.5.1.1 Twisted (TW)

Two or more wires (including signal(s) and return) twisted in accordance with the transposition requirements of Table III. A twist shall be defined as one transposition of a wire with one or more wires.

5.3.5.1.2 Twisted Shielded (TWS)

Two or more wires (including signal(s) and return) twisted in accordance with the transposition requirements of Table III and contained under a single shield.

5.3.5.1.3 Twisted Double Shielded (TWDS)

Two or more wires (including signal(s) and return) twisted in accordance with the transposition requirements of Table III and contained under a double shield.

5.3.5.1.4 Twinax

Two conductor twisted shielded line utilizing radio frequency (>50 KC) shielding and having a dielectric which produces a low constant capacitance between each conductor and the shield, and between conductors.

Table III

Gauge	Two Conductor	Three Conductor	Four Conductor	Six Conductor	Eight Conductor
#6	4 Twists/ft.	3	2	1	1
#8	5 Twists/ft.	4	3	2	1.5
#10	6 Twists/ft.	4	3	2.5	2
#12	7 Twists/ft.	5	4	3	2
#14	8 Twists/ft.	6	4	3	2.5
#16	10 Twists/ft.	7	5	4	3
#18	12 Twists/ft.	8	6	6	4
#20	16 Twists/ft.	12	8	8	6
#22 & Above	18 Twists/ft.	16	10	9	8

5.3.5.2 Wire Type for Radio Frequency Circuits

Interconnection of radio frequency circuits, other than waveguide, shall be shielded coaxial cable, or balanced shielded RF cable with a characteristic impedance equal to or less than 100 ohms.

5.3.5.3 Shield Grounding Requirements

The requirements for shield grounding are given in Table I. The requirements are listed as SPG (single point grounding) and MPG (multiple point grounding) and shall be implemented as follows.

5.3.5.3.1 SPG (Single Point Grounding)

Shields which have the designation of SPG in Table I shall be grounded in only one place. This ground shall be made by direct connection to the signal return at the point that the signal return is referenced to the CSM or launch vehicle ground point (VGP).

5.3.5.3.2 MPG (Multiple Point Grounding)

Shields which are designated as MPG in Table I shall be grounded to structure in as many places as possible. The shields shall be connected to structure at all disconnects.

5.3.5.4 Wire Twisting Requirements

All wiring requiring twisting as called out in Table I shall conform to the twisting requirements listed in Table III.

5.3.6 Step III - Wire Bundling

5.3.6.1 Audio Frequency Bundles

All circuits classified ML in Step I (5.3.4) may be routed together in one bundle, and the bundle shall carry a bundle code of ML. All circuits classified HO may be routed together in one bundle which shall carry a bundle code of HO. Similarly, all circuits classified EO may be routed together in one bundle which shall carry a bundle code of EO, and all circuits coded PO shall be routed together in a bundle which shall carry a bundle code of PO. Under no circumstances shall a circuit be routed in any bundle which does not carry a bundle code that is the same as the circuit classification.

5.3.6.2 Radio Frequency Circuits

Circuits classified as RF in Step I shall be routed independantly of all circuits coded ML, HO, EO, and PO. Care shall be exercised in bundling RF circuits to prevent circuit degradation; ideally, each RF circuit should be routed individually from all other RF circuits.

5.3.7 Step IV - Installation of Bundles

The bundles which have been formed according to Step III (5.3.6) shall be installed by the following rules to provide the required electrical isolation between different signal levels:

a. The fundamental requirement shall be the physical isolation of one bundle type from all other bundles of a different bundle code to achieve electrical isolation between unlike bundles.

b. Each bundle type of one code shall be separated from other bundles with an edge to edge separation (in inches) as specified in Table II.

c. When installation requirements necessitate bundles of different classifications to cross, the crossover shall be made at right angles and a one-half inch separation shall be maintained between the bundles at the crossover.

5.4 <u>Wire Shielding Through the Interface</u>

5.4.1 Shields other than coax shall not carry intentional current.

5.4.2 Shields, if required to be single point grounded, shall pass through the interface on separate connector pins.

5.4.2.1 In the event that shields must be "ganged" to conserve connector pins, only four shields of a given type circuit classification may be ganged. Care must be exercised to insure that a shield ganged at one disconnect to X, Y, and Z circuit shields, is not ganged to A, B, and C circuit shields at another disconnect.

5.5 <u>Bonding</u>

5.5.1 <u>Objective</u>

To insure that each stage of the space vehicle is electrically bonded to the next stage such that the space vehicle is an electrically homogeneous ground reference plane.

5.5.2 <u>Criteria</u>

The mating of structural metallic surfaces at each stage junction and intra stage junctions shall be bare metal to bare metal or an acceptable conductive surface treatment. Acceptable electrical surface treatments include alodine, irridite, or equivalent.

5.5.2.1 The location, area, and method of preparing bonds shall be coordinated through the cognizant intercenter panel and shall appear on the appropriate interface documentation and drawings for each side of the interface. A minimum of one bond joint shall be established over entire faying surface area of each stage to stage junction.

5.5.2.2 Bonding Point Identification

All Apollo SC/LV mechanical (mating) ICD's shall indicate by a bonding identification (B) on each ICD drawing the major joints, or structural mating points at which a bonded joint is established.

5.5.2.3 Dissimilar Metals

Unless otherwise specified on the applicable ICD's, connections between dissimilar metals shall be per Table 1 of MIL-B-5087A.

5.5.2.4 DC Resistance Requirements

The DC resistance of each mated joint at the SC/LV interface shall be less than .005 ohms as measured with a Shallcross Model 670-D milliohmeter or equivalent DC meter.

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