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TASK VIII

PREPRINT

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RTG INTERFACE SPECIFICATIONS

A REPORT COVERING TASK VIII EFFORT
UNDER THE STUDY OF

NASA EVALUATION WITH MODELS OF OPTIMIZED NUCLEAR SPACECRAFT (NEW MOONS)

Wm. S. West, Herbert W. Bilsky and Andrew J. Parker, Jr.

APRIL 1969

GSFC

GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

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NASA Evaluation With Models Of Optimized Nuclear Spacecraft
(NEW MOONS)

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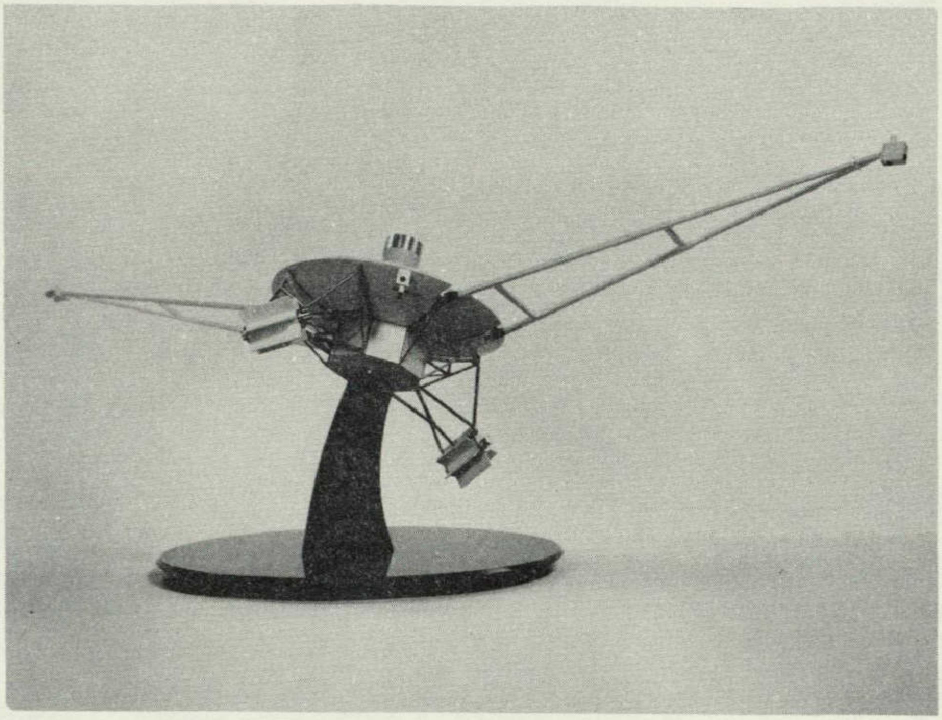

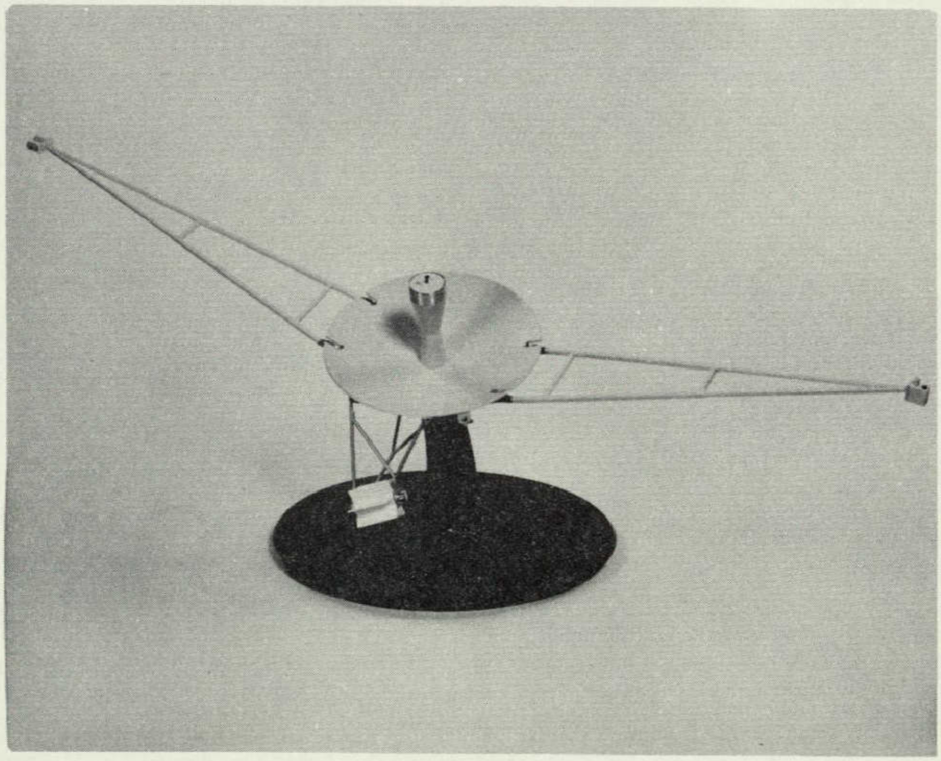

Frontispiece: Two Views of a Model of the Galactic Jupiter Probe Which Served as the "Reference Design" for This Task

RTG INTERFACE SPECIFICATIONS

A Report Covering Task VIII Effort Under The Study

NASA Evaluation With Models Of Optimized Nuclear Spacecraft
(NEW MOONS)

ABSTRACT

This report covers NEW MOONS* study Task VIII, RTG Interface Specifications and presents an illustrative specification containing typical criteria that may be encountered on scientific missions into deep space. These materials could be used as a guide in the preparation of an actual interface specification for RTG-hardware.

*NASA Evaluation With Models Of Optimized Nuclear Spacecraft (NEW MOONS) Contract NAS 5-10441, performed by RCA Astro-Electronics Division, Defense Electronic Products, Princeton, New Jersey for NASA Goddard Space Flight Center, Greenbelt, Maryland.

ACKNOWLEDGMENT

RTG Interface Specifications

NEW MOONS Task VIII

PROGRAM:

In the course of conducting the studies of the NEW MOONS program valuable assistance has been provided by many people representing various organizations. It is considered appropriate to identify those whose contributions were most vital.

Fred Schulman, NASA Office of Advanced Research and Technology and Marcel Aucremanne, NASA Office of Space Science and Applications both realized the necessity for the NEW MOONS studies and provided technical guidance and financial support throughout the program.

Daniel G. Mazur, Assistant Director for Technology and Rudolph A. Stampfl, Deputy Assistant Director both of Goddard Space Flight Center aided in program initiation.

RCA Astro-Electronics Division, the prime contractor, recognized the importance of the NEW MOONS program and has given its support and cooperation toward realizing the objectives of the program. Herbert Bilsky served in the capacity of RCA project manager and technically contributed to the program as well as provided aid in preparation and review of this report.

REPORT PREPARATION:

For this report A. Parker of Hittman Associates, Inc., was the principal investigator and author of the draft materials which were based largely on SNAP-19/Nimbus B criteria and the Galactic Jupiter Probe environmental studies.

Goddard Space Flight Center personnel who have provided important technical information, review and comments include James Trainor, Emil W. Hymowitz, and J. Epstein.

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PREFACE

BACKGROUND AND RELATED INFORMATION

Since the early 1960's, personnel of the Goddard Space Flight Center have been interested in deep-space missions to obtain information concerning the planets, Jupiter, Saturn, Uranus, Neptune and Pluto, as well as information concerning the interplanetary medium. Studies have been performed to establish the feasibility of such missions and various reports were written by Goddard personnel[1] and by others[2].

For almost as long as these missions have been considered, the engineers, scientists and managers at Goddard have realized the necessity for systems, independent of the Sun's energy, to supply the spacecraft electric power requirement. In general, Goddard studies have indicated that there is a weight advantage in using small nuclear power systems such as radioisotope fueled thermoelectric generators instead of presently available solar cells when missions go beyond 2.5 or 3 AU. Further, there are technological and practical uncertainties in projecting use of solar arrays in a range starting beyond 3-5 AU[3] whereas the use of small nuclear power supplies is technically and practically feasible. However, the use of small nuclear systems, while feasible, nevertheless presents technical questions. An in-house Goddard study identified pertinent technological areas requiring study prior to the use of these nuclear generators on spacecraft designed for scientific deep space missions[4]. These areas were divided into the following numbered tasks:

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[1] A selected list of Goddard Space Flight Center deep space reports is presented in X-701-69-170.

[2] A limited list of deep space reports prepared by other centers and contractors is presented in X-701-69-170.

[3] Technical uncertainties involve practical design questions arising from the use of very large solar array areas, their survival through meteroid belts and their system performance when operating at the low temperature and low illumination levels anticipated. This topic is discussed in X-701-67-566 and X-716-67-323.

[4] This study is referred to as NEW MOONS.

Task Number	Task Description — Title	Reference X Document
I	Analysis of Selected Deep-Space Missions	X-701-69-170
IIA	Subsystem Radiation Susceptibility Analysis of Deep-Space Missions	X-701-69-171
IIB	Spacecraft Charge Build-Up Analysis	X-701-69-172
III	Techniques for Achieving Magnetic Cleanliness	X-701-69-173
IV	Weight Minimization Analysis	X-701-69-174
V	Spacecraft Analysis and Design	X-701-69-175
VI	Spacecraft Test Documentation	X-701-69-176
VIIA	Planar RTG-Component Feasibility Study	X-701-69-177
VIIB	Planar RTG-Spacecraft Feasibility Study	X-701-69-178
VIII	RTG Interface Specifications	X-701-69-179
—	Summary Report of NEW MOONS	X-701-69-190

Specific Rationale for Task VIII. For the NEW MOONS Program, two RTG configurations were considered;

- planar, and
- cylindrical.

Either configuration could be designed with Pb Te or Si Ge thermocouples. These and other related matters are discussed throughout the various Task-reports. The objective of Task VIII was to prepare a preliminary interface specification — or a guide for use in the preparation of actual hardware specifications for use on a deep space scientific mission. For the preparation of this document, "GSFC Technical Interface Specification for the Nimbus B/SNAP-19 System" S-450-N1-2 dated July 1966, was used as a model. Environmental criteria appropriate to the

Outer Planet Explorer were taken from X-701-69-189, and from X-701-67-566 which related to the Galactic Jupiter Probe. The illustrative specification presented in this report could be developed to cover a generator of either a planar or a cylindrical configuration. In general the format requirements of GSFC Specification Standard Manual, TID-1, as amended, should be followed.

In general, the spacecraft and subsystems designers will be interested in preparing other criteria and documents associated with the RTG. These documents could include Interagency Agreements, Program Specifications and other materials. These items are not covered in this Task but have been indicated in X-716-67-323, "Advanced Nuclear Systems Study".

The model spacecraft[5] on which the planar RTG was assumed to be used is shown in the frontispiece. While this model spacecraft employs two cylindrical RTG's, a planar configuration could be substituted with appropriate attachment hardware. For a detailed discussion of the design features, operating characteristics and a complete set of manufacturing drawings of a thermal feasibility model reference is made to Task VIIA. To support the sample specification of this Task, reference to other NEW MOONS' Task reports is advisable. The specific areas wherein these reports are useful in this regard are as follows:

- Task IIA establishes the RTG, planetary, and interplanetary radiation environment and develops, parametrically, the RTG-spacecraft separation distance relative to radiation damage on subsystem electronic components.

- TASK III indicates techniques to be used in developing magnetic criteria and establishes the minimum magnetometer-RTG separation distance to limit the magnetic field noise level at the magnetometer to 0.1 gamma.

- Task IV indicates techniques to establish the optimum RTG shielding weight-distance relationship required to limit nuclear radiation to 10 particles $(\gamma + \eta)/cm^2$-sec at the radiation detector and gives the RTG-spacecraft separation distance required to provide for magnetic cleanliness and to provide for satisfactory moment of inertia ratios for a spin-stabilized spacecraft.

- The Task VII reports in addition to providing a description of the basic features of a planar RTG also indicate the inherent protection against meteroid penetration afforded by the design configuration.[6]

[5] "Galactic Jupiter Probe", Phase A Report, X-701-67-566.

[6] An example of the meteoroid flux and spectrum to be encountered during the NEW MOONS mission is given in Appendix B.

- Task VIIB presents preliminary safety analysis calculations which show the capability of a planar RTG surviving reentry heating and also discusses possible locations for mounting planar RTG's on the model spacecraft.

APPLICABILITY TO OTHER PROGRAMS

The NEW MOONS technology and techniques reported may have applicability or some relevancy to additional space missions that may in the future use nuclear systems such as planetary landers and rovers as well as applications spacecraft.

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Specification Number NM-801
Date________________

ILLUSTRATIVE SAMPLE
NOT FOR HARDWARE PROCUREMENT

TECHNICAL INTERFACE SPECIFICATION

FOR THE

NEW MOONS RTG POWER SUPPLY SYSTEM

Prepared By: Power Systems Branch

Approved By:

Technical Officer	Date
Nuclear Systems Manager	Date
Power Systems Manager	Date
Spacecraft Manager	Date
Project Manager	Date

GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

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TECHNICAL INTERFACE SPECIFICATION

FOR THE

NEW MOONS RTG POWER SUPPLY SYSTEM

1. SCOPE

This specification describes the technical interface requirements between the NEW MOONS spacecraft and its RTG power system. The spacecraft is to be developed by NASA Goddard Space Flight Center.

2. APPLICABLE DOCUMENTS

The following documents, are hereby incorporated herein by reference and thereby form a part of this document, except as specifically noted herein. In case of conflict between this specification and the documents referenced herein, this specification shall govern. Current versions of each document, as of the date of this specification, shall apply.

2.1 SPECIFICATIONS. The following specifications are applicable:

(a) NPC 200-4, Quality Requirements for Hand Soldering of Electrical Connections, as amended by GSFC S-323-P-9.

(b) S-450-P-3, GSFC Specification for Semiconductor Power Aging.

(c) S-323-P-2A, Workmanship, Marking, Traceability, Age Control and Packaging Requirements for Semiconductors Procured to Electronic Industries Association (EIA) of Commercial Specifications.

(d) S-323-P-8, Identification and Age Requirements for Semiconductors Procured to NASA or Military Specifications.

(e) GSFC-PPL-7, NASA Preferred Parts List

(f) NPC 200-2, Quality Program Provisions for Space System Contractors.

(g) NPC 200-3, Inspection System Provisions for Suppliers of Space Materials, Parts, Components and Services.

(h) GSFC 300-P1A, GSFC Specification-Printed Wiring Boards.

(i) GSFC S-323-P3, Visual Inspection Requirements for Glass Encased Diodes.

(j) GSFC S-323-P-5, Quality Assurance Requirements for Standard Industrial Equipment.

(k) GSFC S-323-P-6, GSFC Specification-Quality Assurance Provisions for Procured Supplies and Services.

(l) GSFC S-323-P-7, Integrated Circuits.

(m) ______________, GSFC Program specification for the NEW MOONS RTG Power Supply System.

(n) ______________, Environmental Test Specification for the NEW MOONS spacecraft.

(o) NPC 200-1A, Quality Assurance Provisions for Government Agencies.

(p) NPC 250-1, Reliability Program Provisions.

(q) ______________, Specification for Surface Finishes on the NEW MOONS Spacecraft Systems and Subsystem Components.

(r) Magnetic Field Restraints for a Small Galactic Probe, Advanced Plans Staff, NASA-GSFC.

(s) ______________, Preliminary Isotope Fuel Form Specification for the NEW MOONS RTG Power Supply (CRD).

(t) ______________, NEW MOONS RFI Specification.

2.2 STANDARDS. The AEC will establish minimum health and safety criteria which include, but not be limited to the following:

(a) Code of Federal Regulations-Title 10, Part 20-Atomic Energy Commission-Standards for Protection Against Radiation.

(b) Code of Federal Regulations-Title 10, Part 71-Atomic Energy Commission Rules and Regulations for Packaging Radioactive Material for Transport

2.3 DRAWINGS. The following drawings shall apply:

(a) _______________, RTG Power System - Spacecraft Electrical Interface Drawing.*

(b) _______________, RTG Power System - Spacecraft Mechanical Interface Drawing.†

2.4 PUBLICATIONS.‡ The following publications shall apply:

(a) _______________, NEW MOONS Handbook for Experimenters.

(b) _______________, NEW MOONS Project Development Plan.

(c) CG-RIP-2, Classification Guide for the Isotopic Power Program.

(d) Eastern Test Range User's Handbook.

(e) NEW MOONS Task Reports.

3. REQUIREMENTS

3.1 GENERAL DESCRIPTION OF RTG POWER SYSTEM. The RTG power system shall consist of two radioisotope fueled or electrically heated thermoelectric generators, suitably instrumented, a dc-dc conversion system which steps up the RTG's output voltages (if required), telemetry conditioning (if required), required electrical connections, and circuitry, required support mounting structures, ground support equipment, power conditioner (if required), and ground test monitor equipment. All RTG components shall be designed in accordance with the requirements of this specification. The RTG power system must be capable of operating in either a radioisotopic fueled or electrically heated mode and the capability must exist to interchange radioisotope and electrical heating sources without adversely affecting the RTG. Electrically heated models must satisfy both prototype qualification and flight acceptance level requirements.

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*See Appendix E for typical electrical interface drawing.

†See Appendix D for typical mechanical interface drawings for cylindrical RTG's.

‡Copies of NASA originated specifications, standards, drawings and publications required by contractors in connection with this program will be provided by the NASA NEW MOONS Project Office.

Electrically heated and radioisotope fueled RTG's shall exhibit identical characteristics except for the nuclear radiation emissions associated with the latter and additional wiring associated with the former or as agreed upon with the Technical Officer.

3.2 CONFIGURATION AND COMPATIBILITY. The RTG's shall be individually mounted on the spacecraft, see paragraph 2.3, by means of a suitable mounting structure. The module(s) containing the telemetry conditioning equipment and power conditioning equipment (if required), shall be located in one of the spacecraft bays.

3.2.1 Compatibility with NEW MOONS spacecraft and subsystems. The RTG system and all elements thereof shall be compatible, in all respects, with the NEW MOONS spacecraft and its subsystems in all anticipated operational modes.

3.2.2 Compatibility with launch vehicle. The RTG system, as installed on the spacecraft, shall be compatible with the launch vehicle and shroud in all anticipated operational modes. The launch vehicle is the Atlas SLV-3C/Centaur/TE-364-4.

3.2.3 Compatibility with launch and ground support equipment. The RTG ground support and test monitor equipment shall be suited for its intended purpose of complete system checkout as well as handling, transport, installation and removal from the spacecraft. It shall also be compatible with all safety monitoring and support equipment.

3.3 RESPONSIBILITIES. The division of responsibility among NASA, AEC, and other pertinent agencies with respect to interface control, custody, and other items shall be as specified in 2.1.

3.4 ELECTRICAL AND ELECTRONIC REQUIREMENTS AND INTERFACES. The general electrical arrangement of the spacecraft power supply is shown in Figure 1. The following electrical requirements and interfaces are applicable to the NEW MOONS power supply:

(a) The NEW MOONS Power Supply, shown in Figure 1, shall consist of two equally rated RTG's connected in parallel, telemetry and power conditioning, if necessary, required electrical connections, circuitry, and packaging. Detailed electrical interfaces shall be as specified in 2.3.

(b) The RTG power system shall provide a minimum of 120 watts of net dc electrical power delivered at the spacecraft regulated bus for a

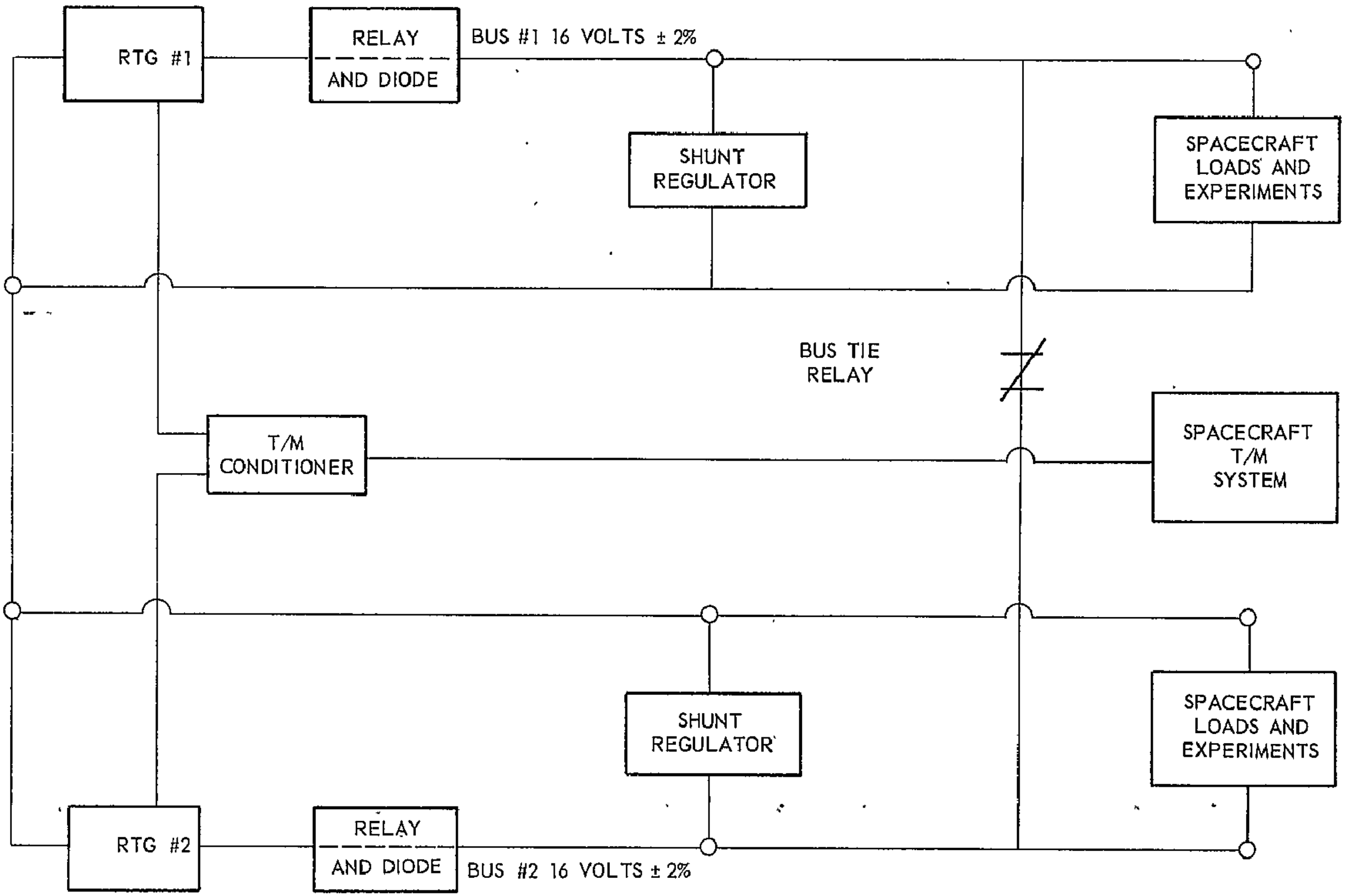

Figure 1. NEW MOONS Power System Block Diagram

mission duration shown in Figure 2, following a maximum one and one-half year period of integration and test. Whenever net bus power is specified herein, a system line loss of 0.2 volts shall be assumed between the RTG output terminals and the bus. The required power profile is shown in Figure 2.

(c) The RTG's shall operate in a parallel (or series) configuration as shown in Figure 1 and the total power output, as measured in this configuration, shall not be less than ___ percent of the sum of the power outputs of the individual RTG's throughout their lifetime.

(d) The maximum net electrical power output per RTG shall not exceed 84 watts at any time.

(e) The power supply output will be clamped at 16 volts ±2 percent volts dc at the spacecraft regulated bus in normal operation. Whenever bus voltage is specified herein, a line drop of 0.2 volts shall be assumed between the output terminals of each RTG system and the bus.

(f) The load as seen by the RTG power supply shall be between 2 and 5 ohms during normal mission operation.

(g) The RTG internal impedance as a function of frequency shall be determined by the RTG supplier and provided to NASA in accordance with 2.1.

(h) The power supply harnesses used during acceptance and qualification tests shall be identical to the spacecraft flight harnesses excluding the heater leads and additional diagnostic instrumentation.

(i) The RTG shall not be adversely affected by a short circuit or an open circuit for a duration of time such that the temperature of the hot shoes shall not exceed ___°C.

(j) The electrical power circuit shall be dc isolated from the RTG housing.

(k) The complete electrical power system shall be dc isolated from the spacecraft structure.

(l) The RTG mountings shall be dc isolated from the spacecraft structure.

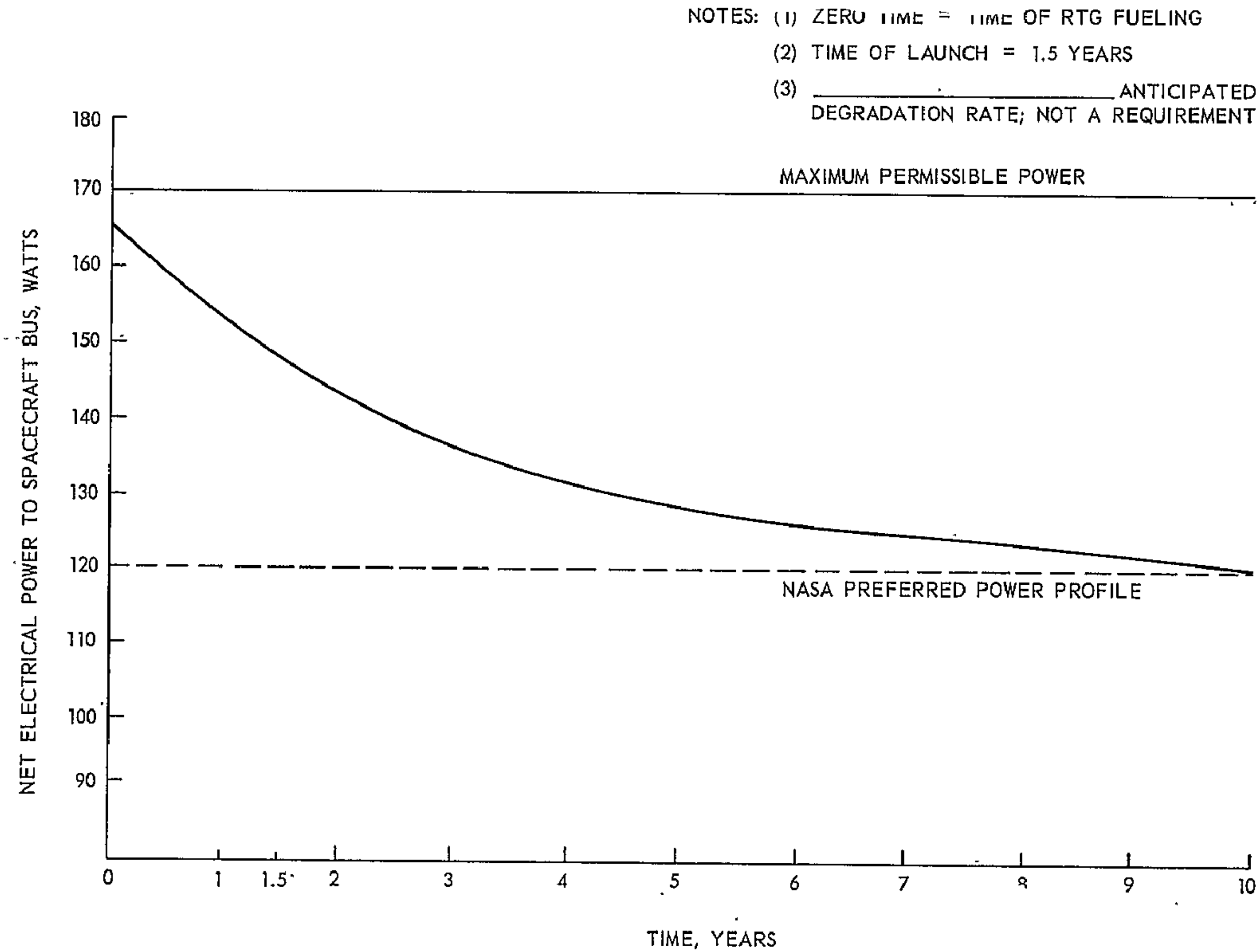

Figure 2. RTG System Power Profile

(m) There shall be no circulating currents between or among the power circuits and the telemetry circuits.

(n) Bus voltage transients up to a maximum of ____volts for periods not exceeding 100 milliseconds, shall not adversely affect the power supply

(o) Wire, lugs, terminals, and connectors shall be keyed, identified and otherwise arranged so as to preclude incorrect system interconnection.

(p) Wires, lugs, terminals, and connectors shall be installed so as to minimize the possibility of mechanical motion during any environmental condition and to minimize ohmic losses in accordance with 2.1 and 2.3.

(q) The RTG's shall be electrically isolated from each other as shown in Figure 1.

(r) Spare pins shall be provided on all electrical connectors. Subject to specific NASA approved exceptions, 20 percent spare pins should be provided.

(s) All relays and other components shall be pre-conditioned in accordance with appropriate specifications listed in 2.1 and shall be derated in accordance with Appendix A.

(t) Shorting devices shall be provided for the RTG's. They shall connect the output leads and also provide a connection to component ground.

(u) Connector and wire requirements shall be consistent with the specifications established in 2.1 and 3.10.

(v) Hipot and meggar tests shall be employed on the RTG's and associated components in accordance with approved levels.

(w) Electric components shall be selected either from NASA preferred parts list or NASA standards manuals, when applicable.

3.4.1 <u>Electrical heater requirements</u>. Electrical heaters used to stimulate the radioisotope fuel shall satisfy the following requirements:

(a) Heaters used to simulate the radioisotope fuel during electrical prototype testing shall be dc electrically powered.

(b) The electrical heat sources shall be physically interchangeable with the radioisotope heat sources with the exception of electrical lead wires.

(c) The electrical heat sources shall satisfy all pertinent interface requirements of this specification.

(d) The electrical heat sources shall be capable of operating, without failure or degradation, under the static and dynamic environmental conditions of 4. In addition, they shall be capable of operating for a minimum period of one year under static endurance test conditions.

3.5 TELEMETRY REQUIREMENTS AND INTERFACES. The power system conditioned telemetry shall be compatible, in every respect, with the spacecraft telemetry in the following areas:

(a) The necessity conditioning equipment shall be housed in a single module, capable of being located within one of the spacecraft bays.

(b) The telemetry output impedance shall be less than the input impedance of the spacecraft telemetry subsystem. The ratio of output to input impedance shall be___.

(c) The minimum type and quantity of monitored parameters shall be as listed in Table 1.

Table 1

Flight Hardware Telemetry Data Point Measurements

Test Point	Item	Range*	Accuracy of Reading	No. of Data Points/RTG
A	RTG Hot Junction Temperature	±150°F	±10°F	3
B	RTG Radiator Temperature	±150°F	±10°F	3

*Deviation from nominal.

(d) Capacitor terminated output impedance shall be provided with an appropriate bleed-off path.

(e) Telemetry conditioning shall not consume more than ___electrical watts.

(f) All external circuitry shall conform to the requirements of 2.1 and 3.10.

(g) Calibration curves shall be provided for each telemetered function.

(h) The RTG telemetry output wires shall not be grounded or connected to the RTG power output leads. The telemetry grounds shall be separate for each function.

(i) All RTG telemetry shall be accurate to within tolerances specified in Table 1.

(j) The hot junction temperature probes shall be located at 120-degree intervals around the RTG. One shall be located at or near the center, one at the radial mid-point, and one near the periphery of the thermopile. Radiator temperature probes shall be adjacent to the hot junction probes.

3.6 NUCLEAR SAFETY CRITERIA AND REQUIREMENTS. The RTG power system shall satisfy the following requirements in addition to those specified in 2.1.

3.6.1 Safety philosophy. The overall safety philosophy for the RTG power supply system shall be that the general population must not be exposed to any undue hazards resulting from normal operation of the system or from any credible accident at any time.

3.6.2 Prelaunch containment. The heat source shall be designed to assure positive containment of the radioisotope fuel at all times during the prelaunch phase of the mission.

3.6.3 Launch vehicle explosion. The system shall be designed to provide containment of the radioisotope fuel when exposed to the shock over-pressure and/or shrapnel environments associated with a launch vehicle explosion.

3.6.4 Launch vehicle fire. The system shall be designed to provide radioisotope fuel containment in the event of a launch vehicle fire.

This shall include the launch pad fireball/afterfire environments as well as the firestream environment associated with launch vehicle fires that may occur during the ascent to orbit phase.

3.6.5 Earth re-entry. The system design shall provide a high degree of assurance that the fuel containment shall re-enter intact under all principal re-entry conditions. These re-entry conditions shall include Earth orbital decay with velocities equivalent to 100 nautical mile circular orbit. Data shall be developed to evaluate the re-entry protection requirements for NASA specified superorbital re-entry modes.

3.6.6 Earth impact. The system shall be designed with the objective of total fuel containment in the event of Earth impact. However, in those cases where the fuel containment may be breached, the heat source design shall be such as to minimize, and localize, the fuel release, should release occur, to the area immediately surrounding the impact point.

3.6.7 Deep and shallow water submersion. The fuel container will be designed to contain the fuel for a period of at least one year at minimum water depths (salt or fresh) or 1000 feet to allow for shallow water recovery.

3.6.8 Earth burial. The design objective will be to provide fuel containment in the event of earth burial. However, in those cases where the capsule integrity may be breached by melting or corrosion the heat source design shall minimize the rate of fuel release from the capsule.

3.6.9 Criticality. Under all normal and credible accident conditions the fuel in the RTG power system shall not constitute a critical mass.

3.6.10 Safety demonstration. The contractor shall plan an analytical and experimental safety program which will demonstrate, with reasonable assurance, satisfaction of the safety criteria specified in 2.1.

3.7 THERMAL REQUIREMENTS AND INTERFACES. The following thermal requirements and interfaces are applicable:*

(a) The maximum permissible total fuel inventory of each RTG shall not exceed 2000 thermal watts (or such other value as may be determined during the study phase) at any time subsequent to delivery.

*See Appendix F for a typical thermal interface document for cylindrical RTG's.

(b) The conduction paths at the RTG - spacecraft interface shall not permit a heat flow greater than___thermal watts per RTG through the interface under mission conditions.

(c) The external surfaces of the spacecraft which view the RTG's shall be assumed to have the properties defined in the NEW MOONS Task V Report, "Spacecraft Analysis and Design", paragraph 2.4.

(d) The RTG and support structure shall be designed such that the temperature at the base of the RTG support structure does not exceed ___°C under normal mission operating conditions. See paragraph 2.3 for additional support structure requirements.

3.8 MATERIALS REQUIREMENTS AND INTERFACES. All RTG power supply system materials shall be capable of sustained operation in the anticipated environments for a minimum of ten years without significant degradation. The materials selection shall consider the effect of environmental conditions on the performance and reliability of the components. Conditions to be considered shall include, but not be limited to, the following:

3.8.1 Temperature and temperature cycling. All materials shall be stable under the cyclic as well as static thermal conditions specified in 3.15. This shall be demonstrated by no significant changes in moduli, the absence of structural relaxation, and no instability as a result of phase transformation, diffusion, precipitation, coalescence, recrystallization or grain growth.

3.8.2 Pressure. The RTG shall be capable of operating throughout ambient pressure range from 20 psi to and including space vacuum.

3.8.3 Humidity. Materials exposed to environmental conditions shall suffer no degradation in properties when subjected to humidity environments as specified in 4.3 Ambient air temperatures will range from -10°F to 120°F with associated relative humidities up to 100 percent.

3.8.4 Mechanical stress and fatigue. All materials and components shall be capable of withstanding the stresses of handling, prelaunch environmental testing, launch, and deployment of the RTG system without permanent degradation.

3.8.5 Corrosion. All component materials shall be either corrosion resistant or protected to resist corrosion under all anticipated environments.

3.8.6 Outgassing, sublimation and vaporization. The use of materials, i.e., polymers and/or inorganics, that outgas, vaporize or otherwise significantly degenerate in the space and temperature environment specified in this specification shall be avoided. Any exposed surface of the RTG system shall have less than 1 percent weight loss per year. Known outgassing materials which are required shall be indicated to NASA. Further, materials such as insulations known to absorb or emit gases shall undergo a suitable "bake out" prior to assembly.

3.8.7 Cold and self-welding. Materials which are required to undergo continuing, periodic or one-time motion shall exhibit no self-welding when in contact at the anticipated mission operating temperatures and pressures during the eight-year mission period.

3.8.8 Nuclear and space radiation. All RTG system materials shall be capable of withstanding nuclear radiation from the fuel source, solar radiation, anticipated solar flares and particulate radiation during the mission lifetime. The sources of spacial radiation will include the Van Allen radiation belts, artificial belts, cosmic radiation and the Jupiter trapped radiation as specified in Appendix C.

3.8.9 Meteroids. All external materials and structures shall be capable of surviving, without significant degradation, the meteroid environment as defined in Appendix B.

3.8.10 Coatings. Thermal control coatings on the RTG's external surfaces shall be chemically stable for a period of three years at the anticipated RTG mission temperatures and space environment. Analysis shall be provided to show the effects on RTG's performance of coating degradation.

3.8.11 Surface finishes. All parts shall receive a suitable protective chemical treatment as specified in 2.1.

3.8.12 Cadmium plating. The use of cadmium or cadmium-plated parts is not allowed.

3.8.13 Electrical connector plating. All external connectors shall receive a double gold plating directly upon the copper or copper alloy base materials with no intermediate silver plate.

3.8.14 Magnetic materials. The use of any magnetic materials or components which generate magnetic fields shall not be used unless specifically approved by NASA. Magnetic materials requirements shall be specified in 2.1 and 3.10.

3.8.15 Dissimilar materials. Dissimilar materials, unless suitably clad, shall not be used in intimate contact, as specified in 2.1.

3.8.16 Generator protection material. The generator outer layer and/or heat source materials shall be selected so as to assure intact re-entry of the radioactive fuel source, as specified in 3.6.

3.8.17 Thermal insulation. The most suitable RTG thermal insulation shall be selected on the basis of current technology and system requirements.

3.8.18 Materials control. All processes such as coatings, finishes, and materials selection shall be reviewed and approved by the contractor's Materials Engineer.

3.9 NUCLEAR RADIATION INTERFACE REQUIREMENTS. The nuclear radiation fluxes emanating from the RTG's shall be compatible with the fuel characteristics, as specified in 2.1. Flux maps shall be provided as shown in Figure 3, a typical example. In addition, shielding will be provided so that the maximum fluxes from two RTG's at the radiation detector shall be $\leq 10\ (\gamma + n)\text{cm}^2\text{-sec}$.

3.10 MAGNETIC INTERFACE REQUIREMENTS. The RTG system shall satisfy the magnetic interface constraints and requirements of 2.1. The magnetic flux density at the magnetometer sensor generated by the spacecraft and its two RTG's shall be less than 0.1 gamma. See NEW MOONS Task III Report, for a discussion of these requirements.

3.11 PHYSICAL AND MECHANICAL REQUIREMENTS AND INTERFACES

3.11.1 Weight. The weights shall be as follows:

(a) The combined weight of the two RTG's including external wiring, necessary mount plates, aerospace nuclear safety devices, T/M sensors, shall not exceed ___pounds.

(b) The total weight of the telemetry conditioning module, if required, shall not exceed ___pounds.

3.11.2 Envelope. The RTG's and associated structures and components shall be compatible with the envelope dimension defined in 2.3.

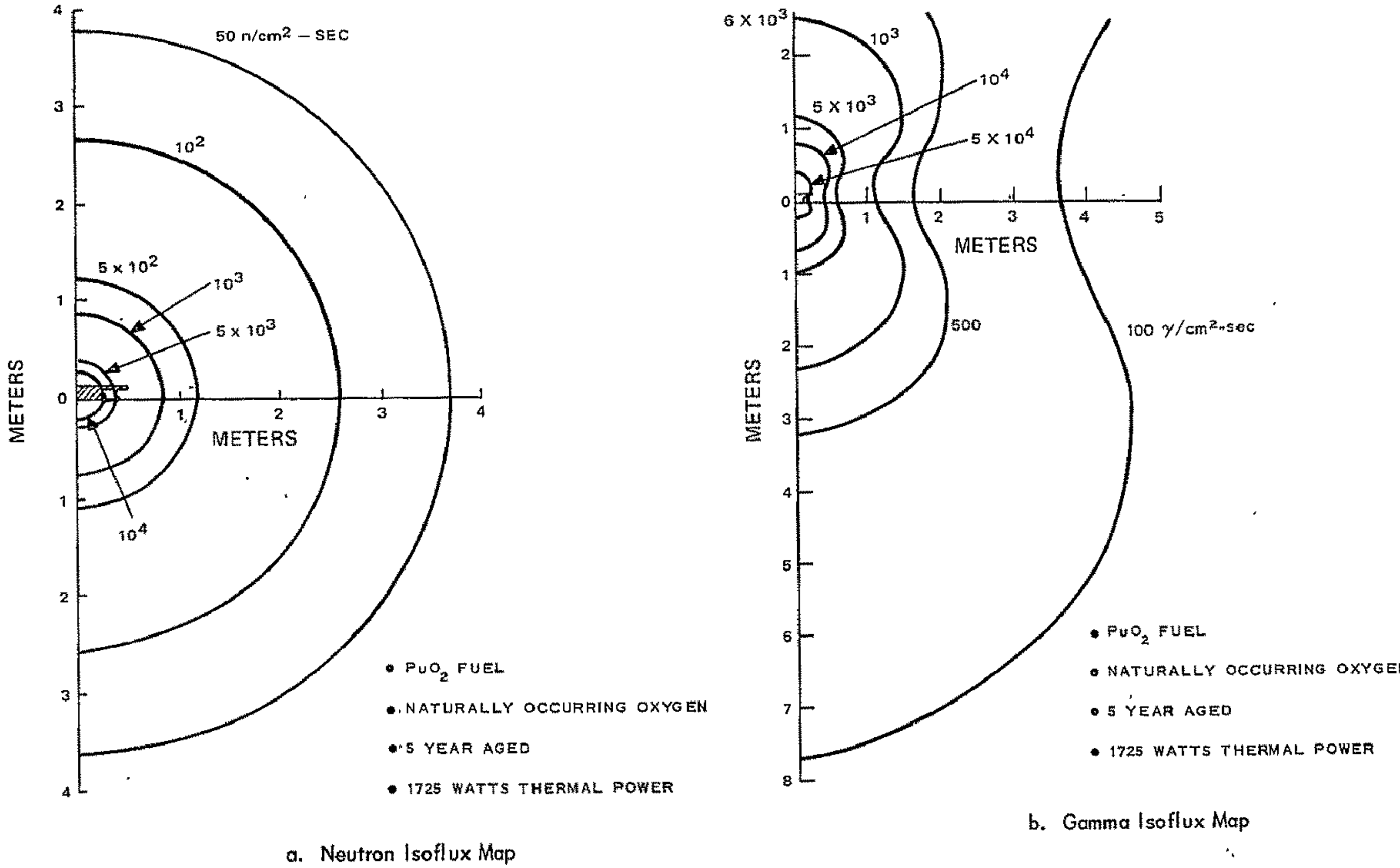

Figure 3. Typical Example of a Neutron and Gamma Isoflux Map

3.11.3 Other physical and mechanical requirements. The following additional requirements apply:

(a) The RTG's shall be capable of mechanical and electrical installation and removal from the spacecraft at any time, without dismantling the spacecraft.

(b) The RTG's shall be designed to minimize the complexity of removing and/or inserting the electrical heaters or fuel source.

(c) Suitable attachment points on the RTG's shall be provided for lifting, handling and installation onto the spacecraft.

(d) All necessary handling tools and fixtures shall accompany the power system.

(e) All individual hardware items shall be suitably marked and identified.

(f) All RTG and module mounting and/or support plates shall be compatible with 2.3.

3.12 RADIOISOTOPE HEAT SOURCE CRITERIA. The primary active radioisotope heat source shall be plutonium-238. All fuel and capsule requirements shall be as specified in 2.1.

3.13 RELIABILITY REQUIREMENTS

3.13.1 RTG system power output reliability. The RTG supplier shall demonstrate a probability of at least ___percent at a ___percent confidence level that the RTG system is capable of supplying a minimum of 120 net electrical watts to the spacecraft bus for a period of 10 years after fueling.

3.13.2 Electrical heat source reliability. To minimize the risk of inadvertent electrical heat source failure or degradation during system tests, and associated schedule delays, the RTG supplier shall demonstrate a performance reliability consistent with 3.4.2 and 4.

3.14 QUALITY ENGINEERING AND QUALITY CONTROL REQUIREMENTS. All quality control criteria, programs and procedures related to the RTG power system and associated equipment shall conform to the requirements of 2.1 and 4.1. The RTG power system supplier shall be responsible for the performance of all inspection requirements as specified herein. Except as otherwise specified, the supplier may utilize his own or any other inspection facilities and services acceptable to NASA, that are covered by an inspection or quality plan as required in 2.1. Inspection and test records shall be maintained in a complete and continuously updated form and, upon request, made available to NASA. NASA, or its designated representative, reserves the right to perform any or all of the inspections set forth herein to assure that the end item conforms to specified requirements.

3.15 ENVIRONMENTAL REQUIREMENTS. The RTG power system shall satisfy the environmental requirements as specified herein and shall be tested, on a subsystem level, in accordance with procedures set forth in 4. In addition, all deliverable models of the RTG power system may be subjected to repetitive environmental tests as a subsystem or part of the spacecraft after delivery to and acceptance by NASA. These tests will include the following with levels and procedures as defined in 2.1.

(a) weight, center of gravity and moment of inertia tests

(b) spin balance tests

(c) magnetic tests

(d) attitude control tests

(e) temperature and humidity tests

(f) vibration tests

(g) acoustic tests

(h) acceleration tests

(i) thermal/solar vacuum tests

(j) plasma environmental tests

(k) pyrotechnic firing shock tests

(l) RFI tests

(m) hot fire motor tests

(n) radiation and radiation mapping tests

(o) meteoroid tests

(p) antenna radiation pattern and command capability tests

(q) life testing

(r) off design testing with limitations noted in paragraph 3.4.

4. SAMPLING, INSPECTION, AND TEST PROCEDURES

4.1 QUALITY CONTROL, RELIABILITY, INSPECTION, AND TEST. Quality and reliability plans shall be established by the RTG supplier at the inception of the RTG program, and shall be in accordance with 2.1 and 3.14 unless specific provisions are waived by NASA. Suitable test procedures shall be developed and implemented by the RTG supplier to assure satisfaction of the reliability requirements as specified in 3.13 and 3.14.

4.2 DEVELOPMENT, QUALIFICATION PERFORMANCE AND OTHER ENGINEERING TESTS. Subsystem component level tests shall include:

4.2.1 Weight, center of gravity and moment of inertia. Procedures for obtaining these measurements on the RTG's and any associated flight subsystems shall be established and performed by the AEC.

4.2.2 Magnetic cleanliness. Magnetic field testing and mapping procedures shall be established and performed by the RTG supplier in accordance with the requirements of 2.1 except as modified herein. The GSFC magnetic test facility can be made available, if required.

4.2.3 Nuclear radiation mapping. Each fueled capsule and RTG shall be tested to measure the surface and surrounding nuclear radiation flux levels and energy spectra mappings. The tests shall be performed as specified in 2.1.

4.2.4 RFI. Tests shall be performed to determine the conducted and radiated RFI generated by the RTG system and to assure conformance with the requirements of 3.4. The test procedure shall be as specified in 2.1.

4.2.5 Electrical heat source development. The RTG supplier, shall perform suitable design studies and development and qualification tests necessary to assure that the electrical heat source satisfies the requirements of 3.

4.2.6 RTG system aerothermodynamics. The RTG supplier shall establish procedures and perform tests, in conjunction with NASA and USAEC, to validate the reentry behavior of the RTG with respect to the requirements of 3.6.

4.2.7 Life and endurance. The RTG supplier shall establish and implement procedures for determining power degradation rates and verifying long term performance in accordance with 3.4 and 3.13. These tests shall be performed on component, subsystem and system levels.

4.2.8 Electrical performance. Electrical performance tests, utilizing specified ground test monitor equipment, shall be developed and conducted by the RTG supplier. Testing shall be sufficient to verify satisfaction of appropriate performance specifications under normal and credible spacecraft fault conditions.

4.2.9 Meteroid vulnerability. The RTG supplier shall establish procedures and perform suitable tests, subject to NASA approval, to establish a reasonable degree of confidence that the RTG power system, when exposed to the space meteroid environment as defined in Appendix B, will perform satisfactorily. The test need be performed on only one generator system. The RTG shall be held in a stable mode and the meteoroid impacted normal to the surface. To incorporate a safety factor, the RTG shall be subjected to an environment 1.5 times the anticipated actual environment. In the event that the RTG supplier determines such a test to be infeasible, written justification of this position shall be presented to NASA. If the test is waived by NASA, the RTG supplier will thoroughly investigate meteroid vulnerability by means of suitable analyses and studies.*

4.3 ENVIRONMENTAL QUALIFICATION AND ACCEPTANCE TESTING. The RTG system shall be tested at the subsystem level under the maximum temperature, humidity, vibration, acoustics, acceleration, thermal/solar vacuum and shock environments anticipated prior to and during launch and

*For a preliminary analysis of a planar RTG's capability to survive the meteroid flux belt, between the 2 and 4 AU regions, see NEW MOONS Task VII Reports.

during the operational phases of its mission lifetime. In those cases where specific information on environmental conditions is not given in 2.1, satisfaction of the requirements shall be in accordance with the test procedures contained herein. All spares and backups required under this program shall be qualified to the same levels as their counterparts. The environmental testing requirements of this section apply only to the RTG's and integral mounting structures. Any required subsystems such as telemetry and power conditioners shall satisfy the environmental requirements of 2.1. All pertinent RTG system input and output parameters shall be monitored with qualified and appropriately calibrated ground test monitor equipment and recorded in permanent test logs before, during and after all environmental tests.

4.3.1 Requirements and procedures.

4.3.1.1 Preparation for test program. The RTG's undergoing prototype level qualification testing shall contain all protective coatings, appendages, and attachments which will be present in flight. RTG's submitted for flight level acceptance testing shall be exact replicas of the qualified prototype.

4.3.1.2 Sequence of testing. Exposure of the engineering, prototype and flight models to the various environments shall be as summarized in Table 3. If an RTG is rejected before, during, or after any exposure, the sequence shall be discontinued, GSFC notified, design defects corrected in accordance with GSFC approved procedures, and the appropriate retest program instituted. In general, complete requalification at prototype levels will be required. A failure during flight model acceptance may necessitate requalification of prototype units.

4.3.1.3 Log and report. A chronological log shall be kept for each RTG system throughout all qualification and acceptance tests. The log shall be maintained in accordance with 2.1 and shall contain all test program data, step-by-step procedures and a record of all pertinent events during the test program. At the conclusion of each test program, a detailed log report containing the reduced data, analysis, technical evaluation, and conclusions shall be prepared. Both the test report and the log report will be provided for review and approval by NASA at least 45 days prior to formal acceptance of each RTG system.

4.3.2 Engineering model test program.

Table 3

RTG Subsystem Environmental Testing Sequence

Model	Paragraph References
ENGINEERING MODELS (Electrically heated)	4.3.2
Temperature/Humidity	4.3.2.1
Vibration	4.3.2.2
Acoustic	4.3.2.3
Acceleration	4.3.2.4
Shock	4.3.2.5
Thermal/Solar Vacuum	4.3.2.6
Life and Endurance	4.3.2.7
PROTOTYPE MODELS (Radioisotope Fueled)	4.3.3
Temperature/Humidity	4.3.3.1
Vibration	4.3.3.2
Acoustic	4.3.3.3
Acceleration	4.3.3.4
Shock	4.3.3.5
Thermal/Solar Vacuum	4.3.3.6
Life and Endurance	4.3.3.7
FLIGHT MODELS (Radioisotope Fueled)	4.3.4
Vibration	4.3.4.1
Acoustic	4.3.4.2
Thermal/Solar Vacuum	4.3.4.3

4.3.2.1 Temperature/humidity testing. Before the RTG's are exposed to temperature/humidity, they shall be visually examined and tested with specified ground test monitor equipment in accordance with the applicable performance specification to assure correct, within-tolerance operation. The RTG's, in inoperative conditions shall then be placed in the humidity chamber and the temperature stabilized at 86°F. The humidity shall then be raised to 95 ± 2 percent RH. These conditions shall prevail for 24 hours, at which time the water flow shall be stopped and the temperature lowered to 77°F. Any free water should be removed from the RTG surfaces. Within two hours after exposure, the RTG shall be visually examined and tested with specified ground test monitor equipment in accordance with the applicable performance specification to determine whether any detectable performance changes have occurred. Any such change that can be interpreted by the RTG supplier and/or NASA as leading to failure or undue degradation during the RTG design life, or any changes in operation, beyond the tolerance specified in the applicable performance specification, shall be cause for rejection.

4.3.2.2 Vibration testing. Before each RTG is exposed to vibration, it shall be visually examined and tested with specified ground test monitor equipment in accordance with the applicable performance specification to assure correct, within-tolerance operation. The RTG shall then be hard-mounted on an electrodynamic shaker at its attachment points to the spacecraft. The RTG shall be attached to the vibration equipment such that it may be vibrated in each of three orthogonal directions, one of which shall be parallel to the spacecraft's thrust axis. The RTG shall be subjected to the sequence of vibration levels as specified in Table 4 while in its normal operating mode.* After exposure, the RTG shall be visually examined and tested with specified ground test monitor equipment in accordance with the applicable performance specification to determine whether any detectable performance changes have occurred. Any such change that can be interpreted by the RTG supplier and/or NASA as leading to failure or undue degradation during the RTG design life, or any change in operation, beyond

*To reduce RTG electrical test time and cost, correlation of RTG electrical performance between space environment and ground environment should be established. Throughout this document "normal operating mode" means test at ground environment thus assuming correlation can be established.

Table 4

Prototype Vibration Levels — A Typical Example

Frequency—Hz	Level
All Axes Sinusoidal*	
5-19	0.4 DA
19-200	±8.0 g
200-2000	±5.0 g
Random†	
20-150	0.023 PSD
150-300	‡
300-2000	0.045 PSD

*Sweeprate — 2 octaves/minute
† 4 min. each axis.
‡ Increasing from 150 Hz at a +3 db/octave rate.

the tolerances specified in the applicable performance specification shall be cause for rejection.

4.3.2.3 Acoustic testing. Before the RTG is exposed to the acoustic environment, it shall be visually examined and tested with specified ground test monitor equipment in accordance with the applicable performance specification to assure correct, within-tolerance operation. The RTG, in its normal operating mode, shall be mounted in the acoustic chamber and exposed to the test levels and sequence specified in 3.15. After exposure, the RTG shall be visually examined and tested with specified ground test monitor equipment in accordance with the applicable performance specification to determine whether any detectable performance changes have occurred. Any such changes that can be interpreted by the RTG supplier and/or NASA as leading to failure or undue degradation during the RTG design life, or any changes in operation beyond the tolerances specified in the applicable performance specification shall be cause for rejection.

4.3.2.4 Acceleration testing. Before the RTG is exposed to acceleration, it shall be visually examined and tested with specified ground test monitor equipment in accordance with the applicable performance specification to assure correct, within-tolerance operation. The RTG shall be rigidly attached to a mounting fixture which shall be capable of attachment to the centrifuge so that the RTG may be accelerated in each of three orthogonal directions, one of which shall be parallel to the spacecraft's thrust direction. The RTG, in its normal operating conditions, shall be subjected to the acceleration levels and sequence shown in Table 5. After exposure to acceleration, the RTG shall be visually examined and tested with specified ground test monitor equipment in accordance with the applicable performance specification to determine whether any detectable perforance changes have occurred. Any out-of tolerance performance, as interpreted by the RTG supplier and/or NASA, shall constitute a failure.

Table 5.

Prototype Acceleration Levels — A Typical Example

Direction	Acceleration	Duration
Spacecraft Thrust	30 g	3 minutes
Lateral	30 g	3 minutes (each axis)

4.3.2.5 Shock testing. The RTG shall then be hard mounted on a shock tester at its interface attachment points. The RTG shall be attached to the shock equipment such that it may be shocked in each of three orthogonal directions, one of which shall be parallel to the spacecraft's thrust axis. The RTG shall be subjected to a 35 g 1 millisecond square wave shock along each of the three orthogonal axes while in its operating mode. After exposure the RTG shall be visually examined and tested with the specified ground test monitoring equipment in accordance with the applicable performance specification to determine whether any detectable performance changes have occurred.

4.3.2.6 Thermal/solar vacuum testing. Before the RTG is exposed to thermal vacuum, it shall be visually examined and tested with specified ground test monitor equipment in accordance with the applicable performance specification to assure correct, within-tolerance operation. Any special structures used to support the RTG in the chamber shall be such that they do not influence the thermal distribution. The RTG shall also be installed in the chamber in such a manner that it is not exposed to abnormal hot or cold surfaces or such that the effects of such sources are minimized. The RTG shall be suitably installed to thermally simulate the actual mounting on the spacecraft.

The test shall be performed in two phases, simulating RTG's radiator temperatures under near-Earth, near Jupiter and deep space solar flux conditions. With the RTG in its normal operating condition, the vacuum chamber shall be evacuated to 10^{-8} torr, or less, at rate not exceeding that of the pressure-time profile of actual flight. The pressure shall be maintained at this level throughout the entire test. The test profile to be followed is that shown in Figure 4 (prototype level cycle). The vacuum chamber shroud and RTG radiator temperatures will be selected so as to have the RTG operate with the same hot and cold junction temperatures expected under the corresponding solar flux conditions in space. While the RTG is operating for extended periods at nominal conditions, performance parameters will be checked at least once every 24 hours using the specified ground test monitor equipment. Once, during each phase of the test, the RTG load voltage shall be varied from 100 to 75 to 50 to 100 percent of the normal operational level. At each level, temperature stabilization will dictate the duration. During these transients, performance parameters will be checked at intervals sufficient to show their effects on thermal profiles and overall performance. Any detectable performance change, during the test, which can be interpreted by the RTG supplier and/or NASA as leading to failure or harmful degradation of the RTG within its design life, or any change in RTG operation beyond the tolerances specified in the applicable performance specification shall be cause for rejection.

4.3.2.7 Life and Endurance testing. Before the RTG is mounted in the vacuum chamber, it shall be visually examined and tested with specified ground test monitor equipment in accordance with the applicable performance specification to assure correct, within-tolerance operation. With the RTG in its normal operating condition at its maximum power point, the test chamber shall be evacuated to a pressure of 10^{-8} torr, or less, at a rate not exceeding that of the

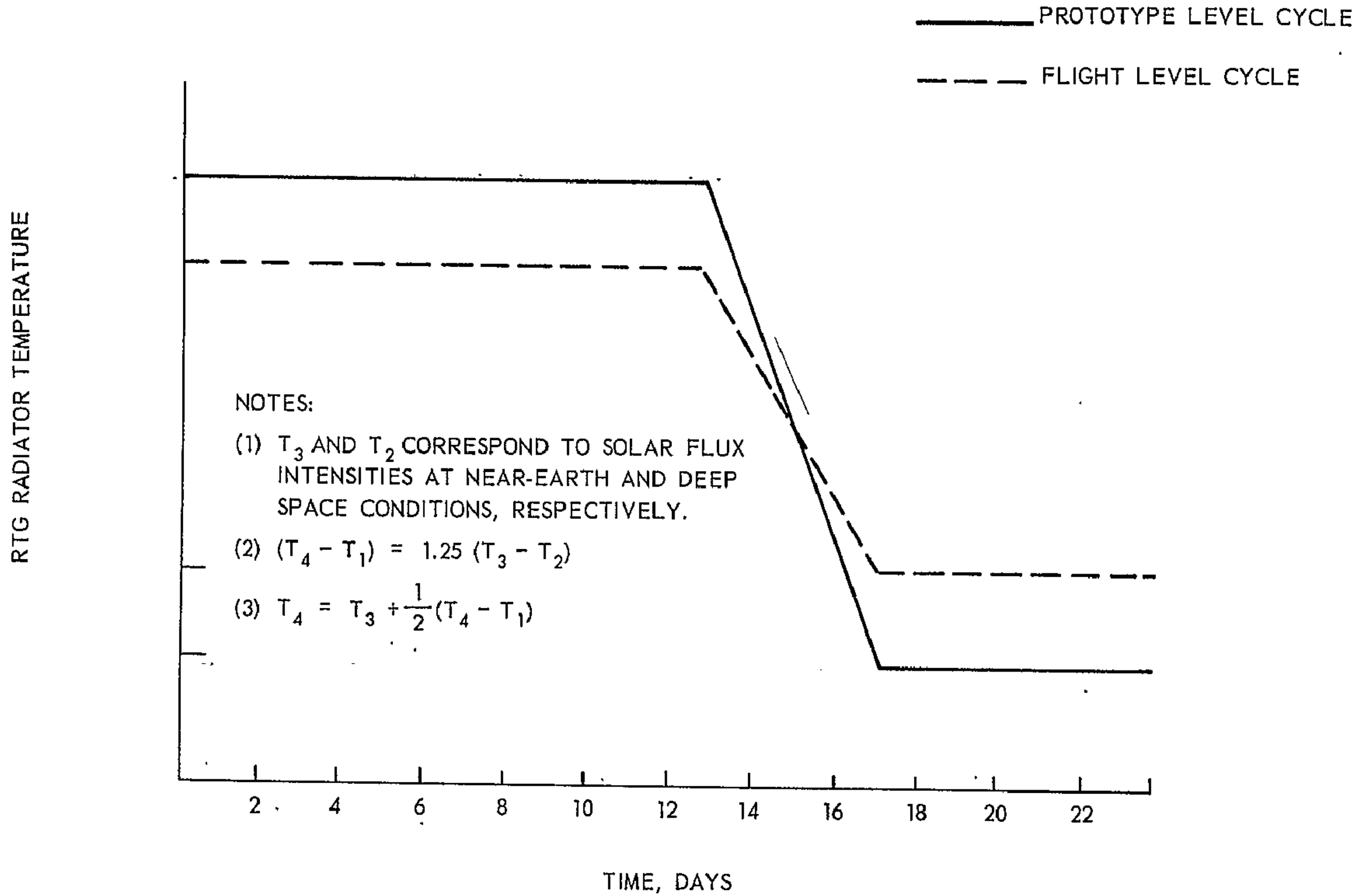

Figure 4. RTG Thermal/Solar Vacuum Radiator Temperature-Time Profile

pressure-time profile of actual flight. The chamber pressure shall be maintained at this level throughout the test. Further, the RTG load voltage, input power and radiator temperature (at the T_4 level on Figure 4) shall be maintained constant throughout the test. The test duration shall be a minimum of one year with longer periods desirable, if possible. The electrical and thermal performance of the RTG shall be checked at least once every 24 hours using the specified ground test monitor equipment. Further, excessive performance degradation beyond the tolerances specified in the applicable performance specification may, at any point in the test, be cause for rejection. This test may be performed either on the prime model or its backup which has an identical prior test history.

4.3.3 Prototype model test program.

4.3.3.1 Temperature/humidity testing. The prototype model shall be subjected to the exposure sequence and levels as specified in 4.3.2.1.

4.3.3.2 Vibration testing. The prototype model shall be subjected to the exposure sequence and levels as specified in 4.3.2.2.

4.3.3.3 Acoustic testing. The prototype model shall be subjected to the exposure sequence and levels as specified in 4.3.2.3.

4.3.3.4 Acceleration testing. The prototype model shall be subjected to the exposure sequence and levels as specified in 4.3.2.4.

4.3.3.5 Shock testing. The prototype model shall be subjected to the exposure sequence and levels as specified in 4.3.2.5.

4.3.3.6 Thermal/solar vacuum testing. The prototype model shall be subjected to the exposure sequence and levels as specified in 4.3.2.5.

4.3.3.7 Life and Endurance testing. The prototype model shall be subjected to the exposure sequence and levels as specified in 4.3.2. This test may be performed either on the prime model or its backup which has an identical prior test history.

4.3.4 Flight model test program.

4.3.4.1 Vibration testing. The flight model shall be subjected to the exposure sequence as outlined in 4.3.2.2 but to the levels specified in Table 6.

Table 6

Flight Vibration Levels — A Typical Example

Frequency—Hz	Level
All Axes Sinusoidal*	
5-19	0.25 DA
19-200	±5.5 g
200-2000	±3.5 g
Random†	
20-150	0.015 PSD
150-300	‡
300-2000	0.03 PSD

*Sweeprate — 2 octaves/minute.
†4 minutes each axis.
‡Increasing from 100 Hz at a +3 db/octave rate.

4.3.4.2 Acoustic testing. The flight model shall be subjected to the exposure sequence and levels as specified in 4.3.2.3.

4.3.4.3 Thermal/solar vacuum testing. The flight model shall be subjected to the exposure sequence and levels as specified in 4.3.2.5.

4.4 ACCEPTANCE REQUIREMENTS. Acceptance criteria shall be as specified in the GSFC Program Specification for the NEW MOONS RTG system.

5. PREPARATION FOR DELIVERY

5.1 FUELED RTG's. Fueled RTG's shall be individually packaged and transported in specially designed shipping containers which satisfy the requirements of 2.2. The shipping container shall be secured to a pallet by means of protective shock mounts and suitable tie-down provisions so as to preclude damage to the RTG's under normal transportation conditions during shipment to NASA, the Integration Contractor or the launch site. The RTG's shall be in a short circuited condition during transit. Packages shall be suitably marked for delivery.

5.2 ELECTRICALLY HEATED RTG's. RTG's with electrical heaters shall be packaged and transported in special protective containers and on suitable pallets which will preclude damage under normal transportation conditions during shipment to NASA or the Integration Contractor. Packages shall be suitably marked for delivery.

5.3 ANCILLARY FLIGHT HARDWARE AND GROUND SUPPORT EQUIPMENT. Miscellaneous system hardware items shall be suitably preserved, packaged and marked for delivery. The packaging shall preclude damage to its contents under normal transportation conditions during shipment to NASA, the Integration Contractor or the launch site.

6. NOTES

6.1 DEFINITIONS. The following definitions are included to facilitate the understanding of certain specialized terms referred to herein.

(a) Spacecraft — The NEW MOONS spacecraft and its collective subsystem including structure, experiments, sensors, controls, antennas, and booms.

(b) RTG — A plutonium-238 radioisotope fueled thermoelectric generator with required diagnostic instrumentation. Certain engineering and development models under this program may contain electrical heaters in lieu of the radioisotope fuel but are similar, in all other respects, to the fueled models.

(c) RTG Power System — The spacecraft electrical power supply system consisting of two RTG's, power and telemetry conditioning package, if required, electrical interconnections and circuitry, and required support and mounting structures.

(d) NASA — The National Aeronautics and Space Administration-Goddard Space Flight Center or its authorized representatives.

(e) AEC — The United States Atomic Energy Commission and/or its contractors.

(f) Integration Contractor — The NASA designed, NEW MOONS Integration and Test Contractor.

(g) Launch Site — The Eastern Test Range located at Cape Kennedy, Florida.

(h) Ground Test Monitor Equipment — A console, to be designed and developed by AEC, which is capable of monitoring all RTG system performance parameters during all ground testing operations.

(i) GSE — All RTG system ground support equipment including shipping containers, handling and mounting devices, temporary nuclear radiation shielding, nuclear radiation monitoring devices, and on-site transportation dollies.

(j) Launch Vehicle — The Atlas SLV-3C/Centaur/TE-364-4(Typical Example).

(k) CRD — Confidential Restricted Data

(l) Supplier — RTG contractor.

6.2 OBJECTIVES OF THE NEW MOONS PROGRAM. The overall objective of the program is to further deep-space investigations centering on an understanding of the origin, operation and evolution of the solar system, the Sun and its interactions with the planets and the galactic medium itself. Specific short and long range scientific objectives of the program include the following studies:

(a) Solar atmosphere — Winds, fields, flare mapping, neutral hydrogen and cosmic ray modulations vs. distance.

(b) Asteroid Belt and interplanetary dust — Astronomy/astrophysics by Jupiter's gravitational field, Asteroid belt engineering parameters.

(c) Jupiter Studies — magnetosphere, trapped matter around the planet, atmospheric studies, decimeter and decameter radiation, and micrometeorites.

(d) Solar Atmosphere — in and out of ecliptic plane.

(e) Galactic Medium — including interaction region with solar atmosphere, cosmic ray density, solar modulation effects.

(f) Outer Planet Studies (Saturn, Neptune, Uranus) — similar to Jupiter studies.

6.3 NEW MOONS SPACECRAFT DEFINITION. The NEW MOONS spacecraft and experiments shall be as described in the GSFC Project Development Plan and Experimenter's Handbook. A preliminary typical spacecraft configuration in the launch condition is shown in Figure 5.

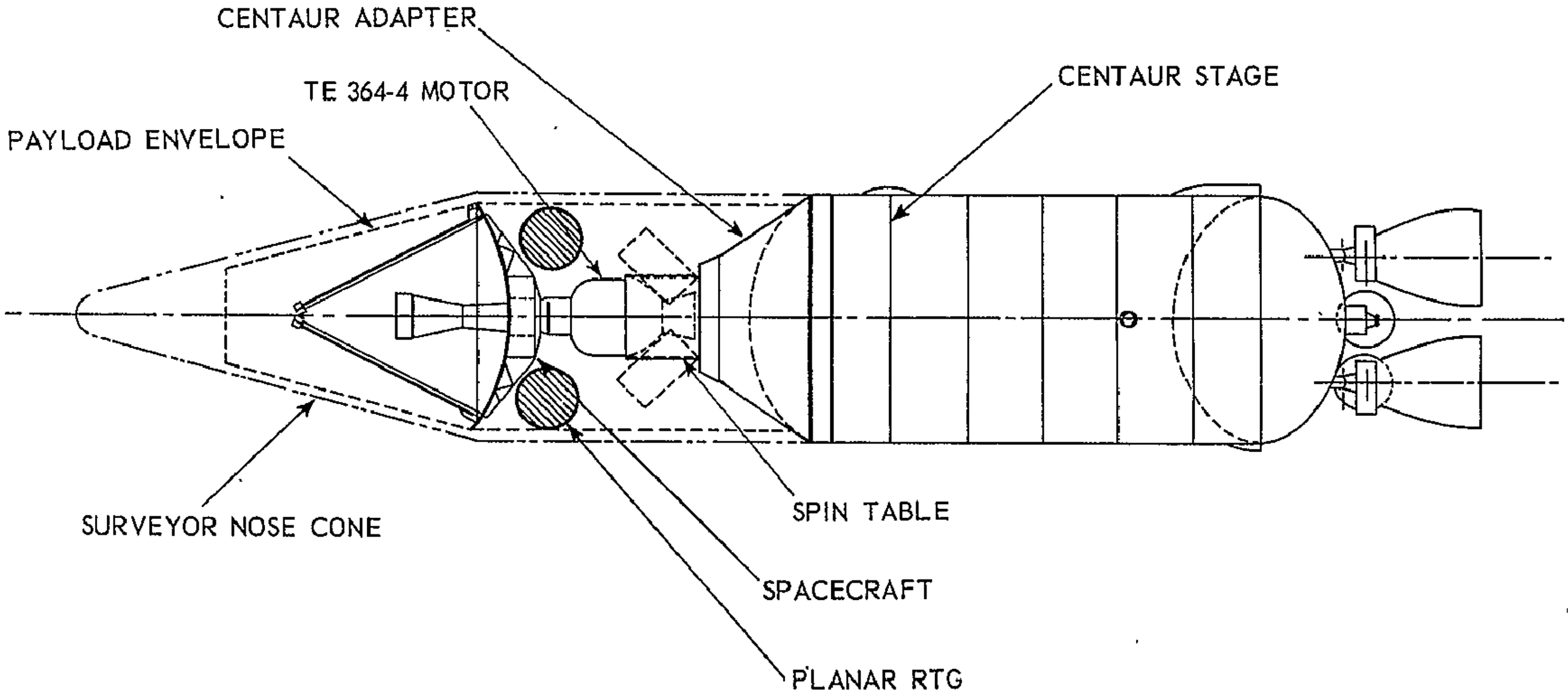

Figure 5. Spacecraft in Launch Configuration

TECHNICAL INTERFACE SPECIFICATION

FOR THE

NEW MOONS RTG POWER SUPPLY SYSTEM

1. SCOPE

This specification describes the technical interface requirements between the NEW MOONS spacecraft and its RTG power system. The spacecraft is to be developed by NASA Goddard Space Flight Center.

2. APPLICABLE DOCUMENTS

The following documents, are hereby incorporated herein by reference and thereby form a part of this document, except as specifically noted herein. In case of conflict between this specification and the documents referenced herein, this specification shall govern. Current versions of each document, as of the date of this specification, shall apply.

2.1 SPECIFICATIONS. The following specifications are applicable:

(a) NPC 200-4, Quality Requirements for Hand Soldering of Electrical Connections, as amended by GSFC S-323-P-9.

(b) S-450-P-3, GSFC Specification for Semiconductor Power Aging.

(c) S-323-P-2A, Workmanship, Marking, Traceability, Age Control and Packaging Requirements for Semiconductors Procured to Electronic Industries Association (EIA) of Commercial Specifications.

(d) S-323-P-8, Identification and Age Requirements for Semiconductors Procured to NASA or Military Specifications.

(e) GSFC-PPL-7, NASA Preferred Parts List

(f) NPC 200-2, Quality Program Provisions for Space System Contractors.

(g) NPC 200-3, Inspection System Provisions for Suppliers of Space Materials, Parts, Components and Services.

(h) GSFC 300-P1A, GSFC Specification-Printed Wiring Boards.

APPENDIX A

DERATING FACTORS FOR COMPONENT PARTS

Part Type	Stress Parameter	Maximum Derating Factor*	Remarks
<u>Capacitors</u>			
Mylar	Rated voltage	0.45	The design shall also provide proper derating of capacitors operated under a-c, pulse, or transient voltage, as specified in the applicable specification. Rated voltage is as prescribed in the applicable control document.
Mica	Rated voltage	0.20	
Fabricated mica solid tantalum	Rated voltage	0.50	
a) $\geq$ 3 ohm/volt	Rated voltage	0.6	
b) < 0.1 ohm/volt	Rated voltage	0.2	
<u>Resistors</u>			
Carbon film	Rated power	0.50	Derating factors apply only for nonstacked resistors. The maximum voltages specified in the applicable control document must not be exceeded.
Metal film	Rated power	0.50	
Power wirewound	Rated power	0.50	
Precision wirewound	Rated power	0.50	
<u>Connectors</u>			
Multipin	Current	0.50	Voltages shall be derated as indicated in the applicable control documents to meet high altitude requirements.
R-F coaxial	Current	0.50	

*Derating Factor is defined as the ratio of the operating stress parameter to the rated stress parameter.

Derating Factors for Component Parts (Continued)

Part Type	Stress Parameter	Maximum Derating Factor*	Remarks
Diodes			
Zener	Rated power	0.50	Derating factors are based on the 25°C ratings. High-current rectifier and zener deratings are also based on the use of heat sinks which limit the temperature rise of the diode case to 6°/watt.
Voltage reference	Rated power	0.50	
Stabistor	Rated power	0.50	
Rectifier	Rated power	0.20	
Computer	Rated power	0.20	
Signal	Rated power	0.20	
Switch	Rated power	0.20	
Transistors			
General purpose	Rated power	0.20	Derating factors are based on the 25° ratings. Power transistor deratings are also based on the use of heat sinks which limit the transistor case temperature rise above ambient to 6°/watt.
High-Speed Switching	Rated power	0.20	
Medium power	Rated power	0.15	
High power	Rated power	0.15	

*Derating Factor is defined as the ratio of the operating stress parameter to the rated stress parameter.

APPENDIX B

METEOROID FLUX AND SPECTRUM FOR THE NEW MOONS MISSION

The meteoroid flux presented is an example of the environmental information necessary to support the RTG Interface Specification. As new data are developed relative to the flux and spacecraft trajectory, it is expected that the information will be provided to the RTG contractor. For a more recent discussion on meteoroid flux and spectrum, see Outer Planets Explorer X-701-69-189.

1. Environment — The most hazardous environment occurs between 2 and 4 AU with an estimated passage time of approximately 120 days.*

2. Mass — Particle mass ranges from 10^{-11} grams to 10^{2} grams.† Range of greatest interest at 2-4 AU is 10^{-4} to 10^{-1} grams.

3. Flux — The calculated worst-case number of impacts is as defined in Table B-1.

4. Density — Mean meteoroid density = 0.44 gm/cm^3. Worst-case asteroid density = 3.5 gm/cm^3.

5. Velocity — The average impact velocity at 2 to 4 AU is approximately 25 km/sec. Estimated maximum velocity ranges are shown in Table B-2.

*The actual duration is about 200 days but for calculations it is roughly equivalent to an exposure of 120 days to the peak flux.

†It is recognized that the upper limit of the mass range may be exceeded by many orders of magnitude so that 10^2 represents an arbitrary cut-off.

Table B-1

Calculated Number of Particle Impacts Through Asteroid Region*

Mass, gm	No. of Impacts/m^2
$>10^{-11}$	300,000
$>10^{-6}$	42
$>10^{-4}$	1
$>10^{-2}$	1.7×10^{-2}
$>10^{-1}$	4.7×10^{-3}

*Calculation based on the Marshall flux equation: $(\Phi)_m = 10^{-10} M^{-0.77}$.

Table B-2

Spacecraft and Meteoroid Velocities

Distance from Sun, AU	S/C Velocity, km/sec	Particle Circular Velocity, km/sec	Escape Velocity, km/sec	Max. Relative Velocity, km/sec
1.0	39.0	30.0	42.1	81.0
2.0	25.5	21.2	30.0	55.5
3.0	18.3	17.3	24.4	42.7
4.0	14.0	15.0	21.1	35.1
5.1	9.0	13.2	18.7	21.0

APPENDIX C

DEFINITION OF PARTICLE FIELDS AND FLUXES

FOR THE NEW MOONS MISSION

The data presented here gives a very brief description of the NEW MOONS environment. For a more thorough discussion and a description of the environment assumed for radiation effects on the NEW MOONS spacecraft systems see Task IIA. For the effects of the environment assumed for spacecraft charge build-up see Task IIB.

1. Environment — Aside from RTG radiation, the NEW MOONS spacecraft will be subjected to a variety of radiation environments. These may include Earth-Trapped radiation (Van Allen Belts which will be traversed in about one hour), Interplanetary Particles, Galactic Cosmic Radiation, Solar Flares, Solar Wind, and Jupiter Radiation Belts.

2. Earth-Trapped Radiation — This radiation environment is defined by Figure C-1. Between the Earth's magnetosphere and several earth radii beyond the Earth's magnetosphere, electron fluxes of 10^{10} electrons/cm^2 (between 1 and 10 kev) and proton fluxes of 10^7 protons/cm^2-sec (2 kev) exist.

3. Interplanetary Particles — These consist of the following contributions:

   (a) Galactic cosmic radiation — mostly protons of several Bev energy with fluxes of 1.5 to 4 protons/cm^2-sec. Approximately ten percent of the total flux is made up of helium and heavier nuclei. Figure C-2 shows the integrated spectra.

   (b) Solar winds — composed of ionized hydrogen gas with fluxes of 10^8 protons/cm^2-sec ($\approx$1 kev) for a quiet sun and 10^{12} protons/cm^2-sec ($\approx$10 kev) for an active sun. Intensity varies as the inverse square of distance from the sun.

   (c) Solar flares — will be encountered with intensities exceeding 10 particles/cm^2-sec and with energies of 10 to 500 mev. Severe and typical flare spectra are shown in Figures C-3 and C-4.

4. Jupiter-Trapped radiation — Jupiter is thought to have a magnetic field from a set of poles whose axis is tilted 10 degrees from the rotational axis. The most probable field strength is about one gauss at three Jupiter radii.

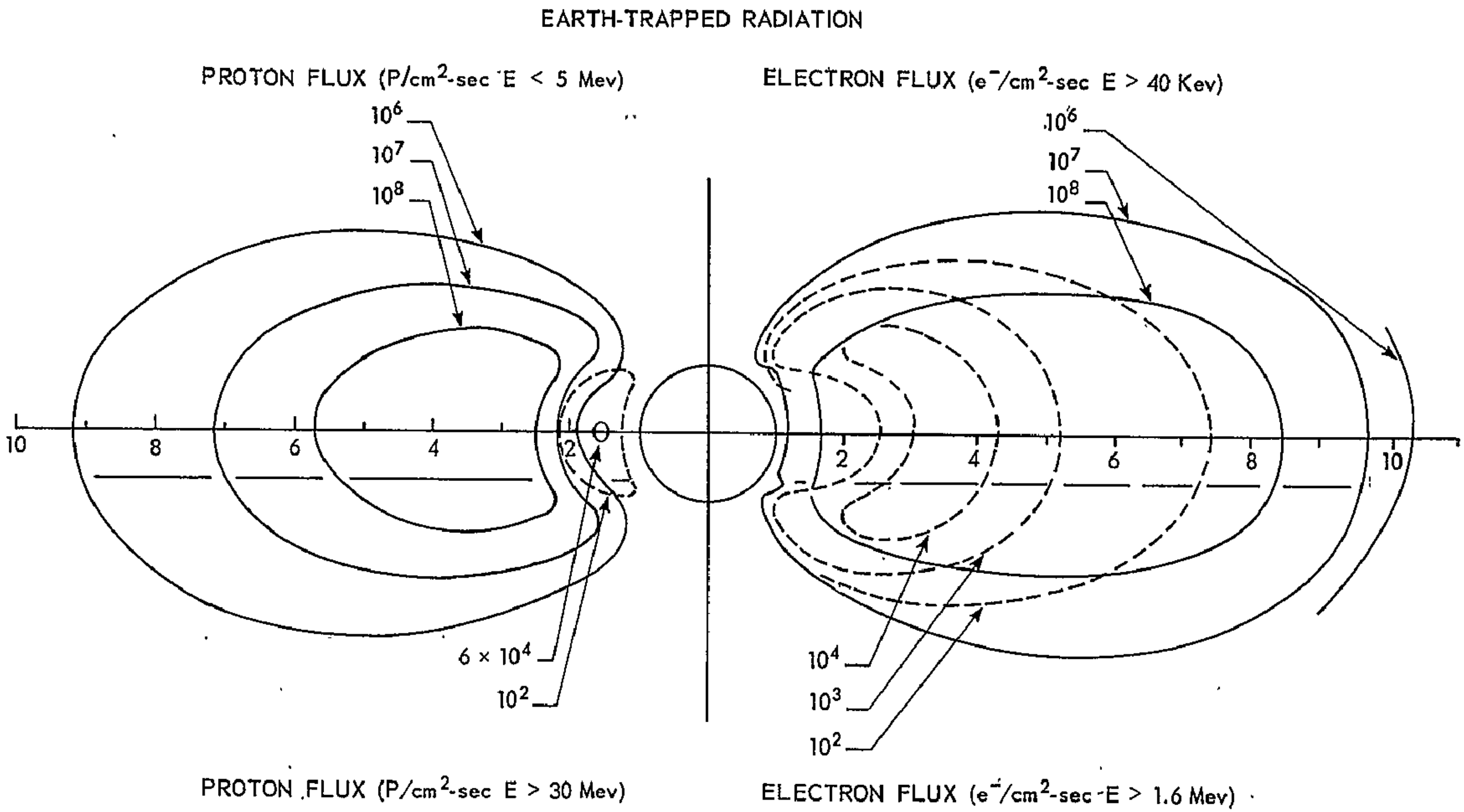

Figure C-1. Radial Distance In Earth Radii R_E (R_E = 3440 n.mi.)

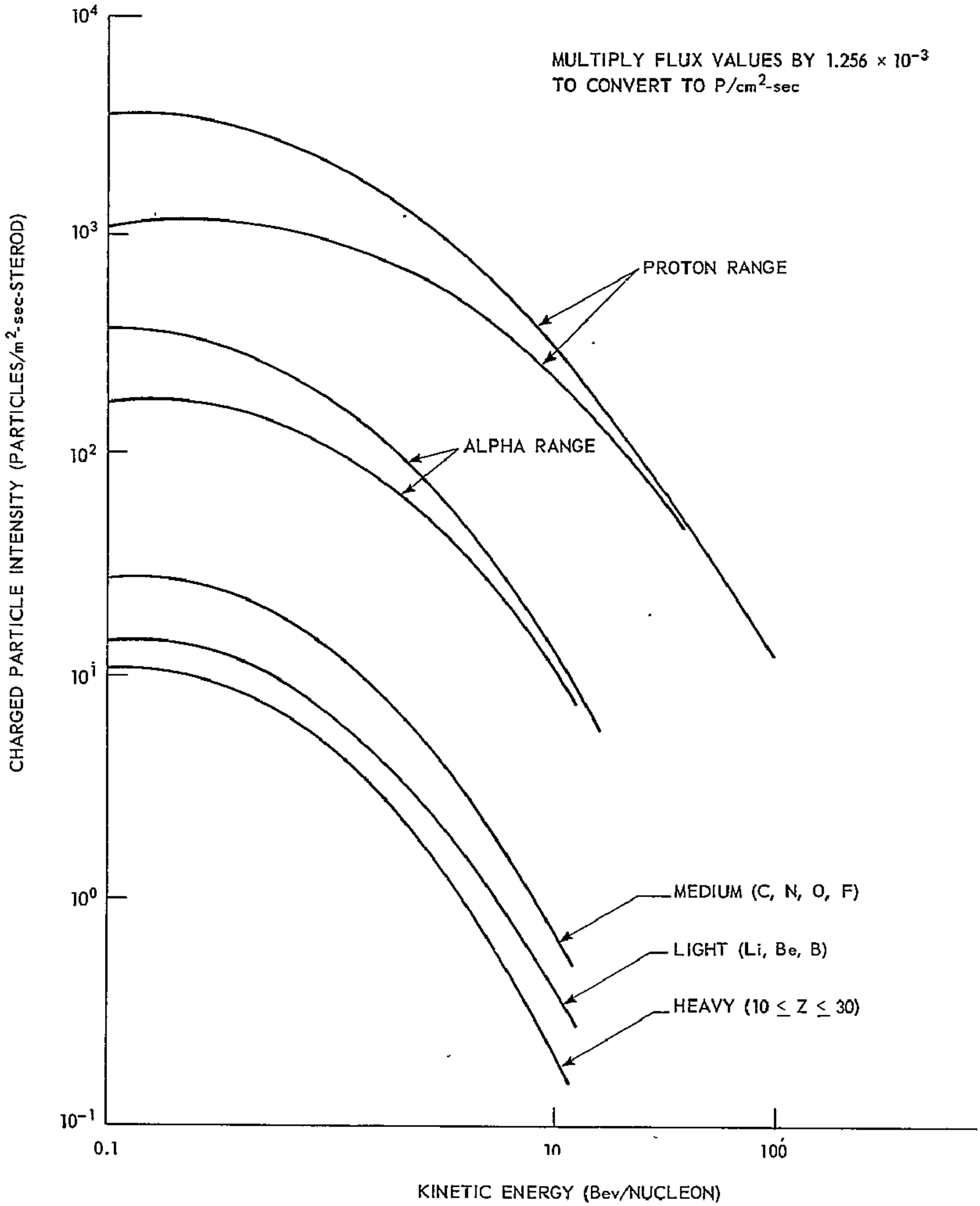

Figure C-2. Integral Spectra of Galactic Cosmic Radiation

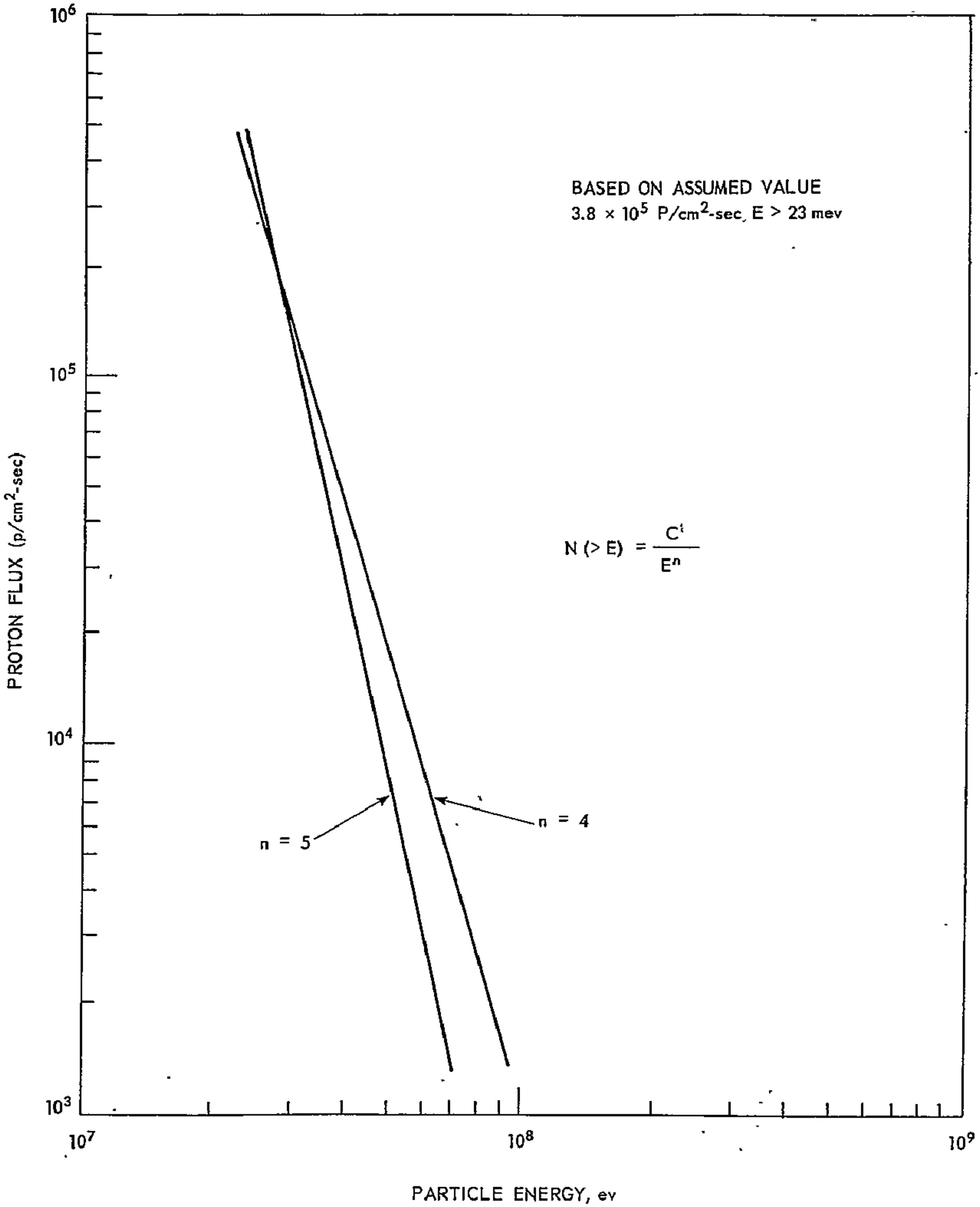

Figure C-3. Energy Spectrum of a Severe Nonrelativistic Flare

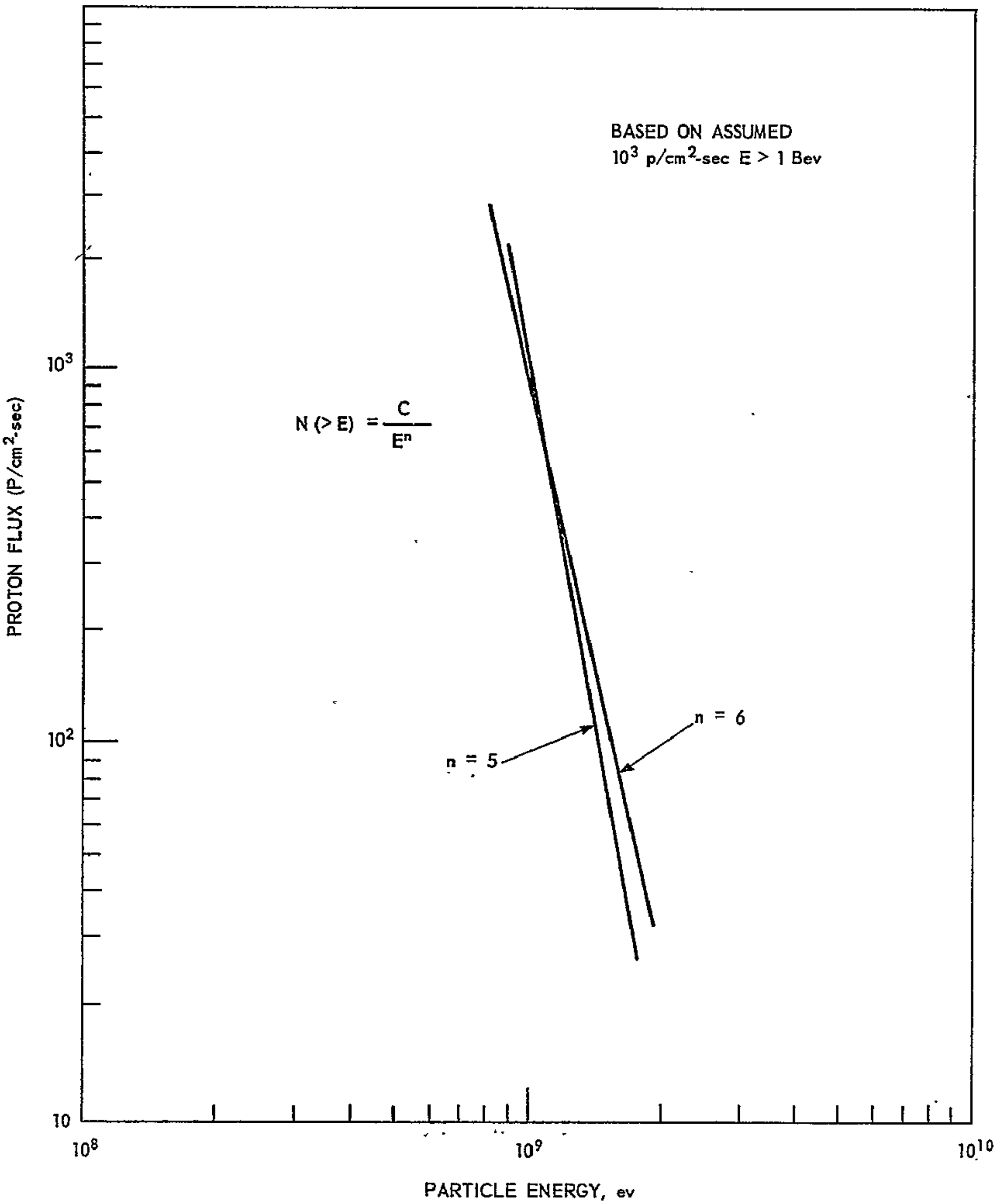

Figure C-4. Energy Spectrum of a Typical Relativistic Flare

To explain this, an average electron density in the vicinity of Jupiter on the order of 10^{-3} electrons/cm^3 over 10 planetary volumes is required. This, compared to an electron density on the order of 10^{-6} electrons/cm^3 in the Earth's field, suggests an electron flux scaling factor of 10^3 for estimating Jupiter's environment. The probable energy range is 1 to 100 mev. This approach is admittedly an approximation, but should be adequate for an intended procurement. Where these requirements appear critical to the procurement, the impacts and assumptions shall be documented and provided to the Technical Officer.

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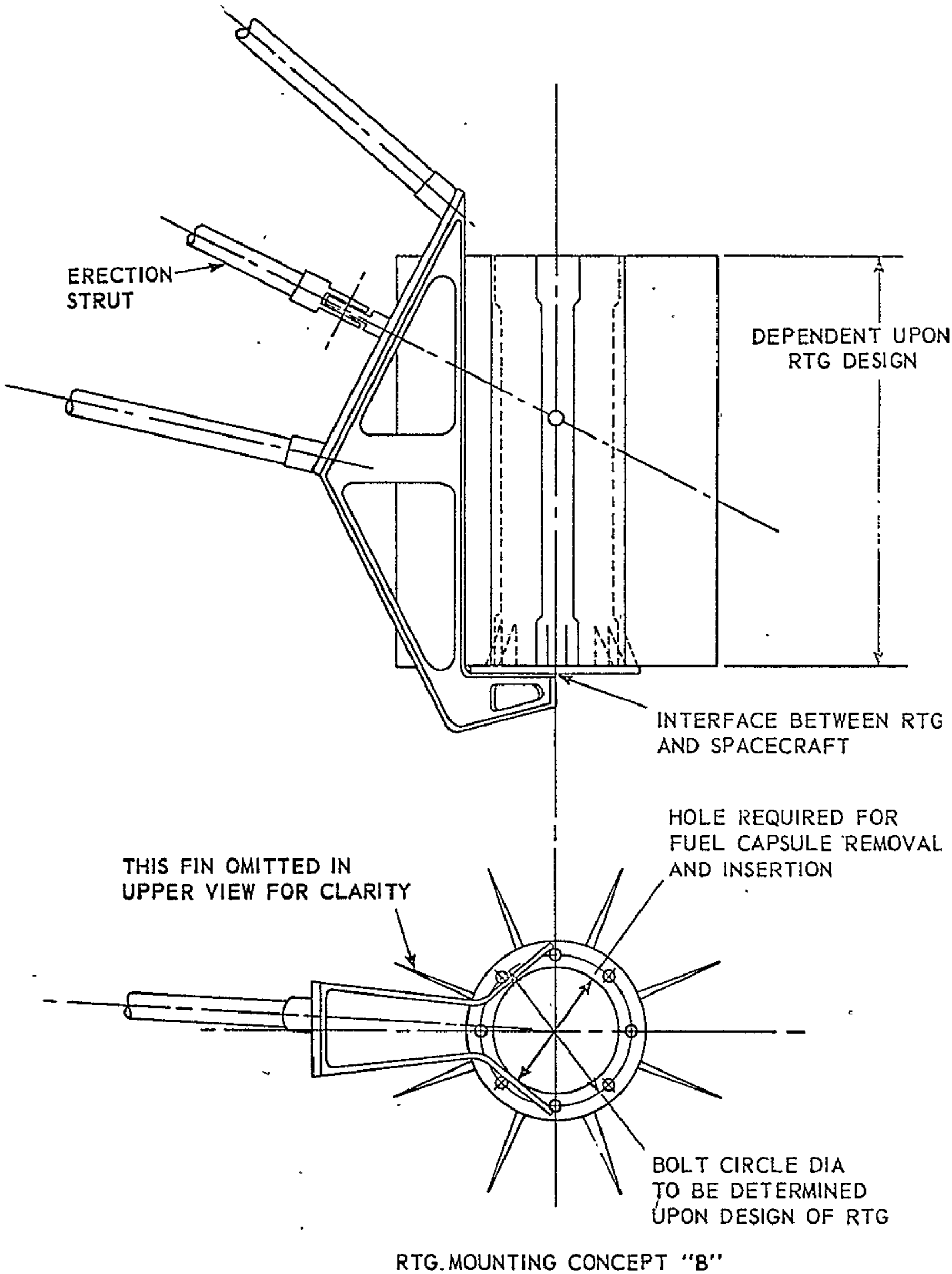
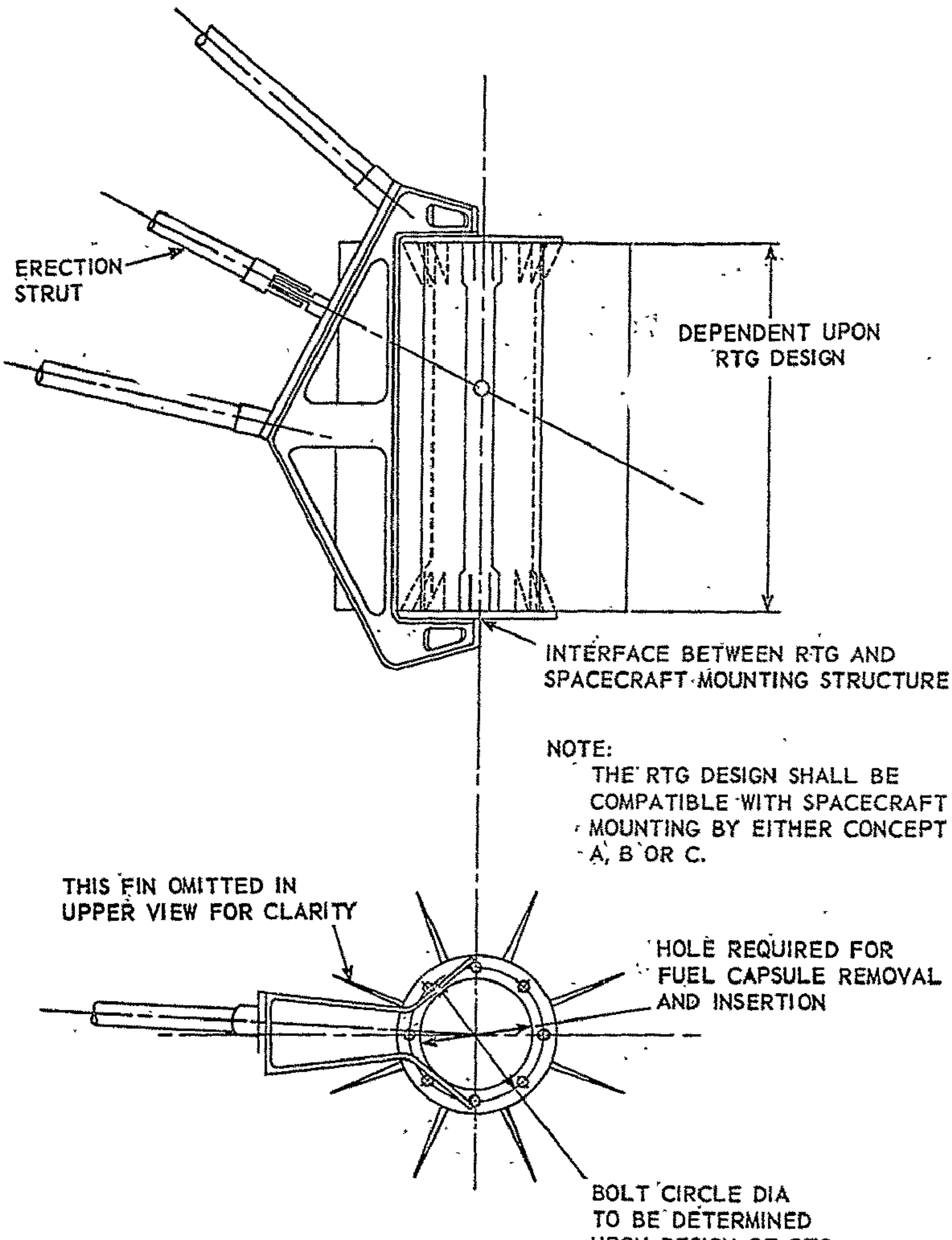


Figure D-1. RTG Power System — Spacecraft Mechanical Interface Drawing (SK-GSFC-701-1) (Sheet 1 of 3)



RTG MOUNTING CONCEPT "A"

Figure D-1. RTG Power System — Spacecraft Mechanical Interface Drawing (SK-GSFC-701-1) (Sheet 2 of 3)

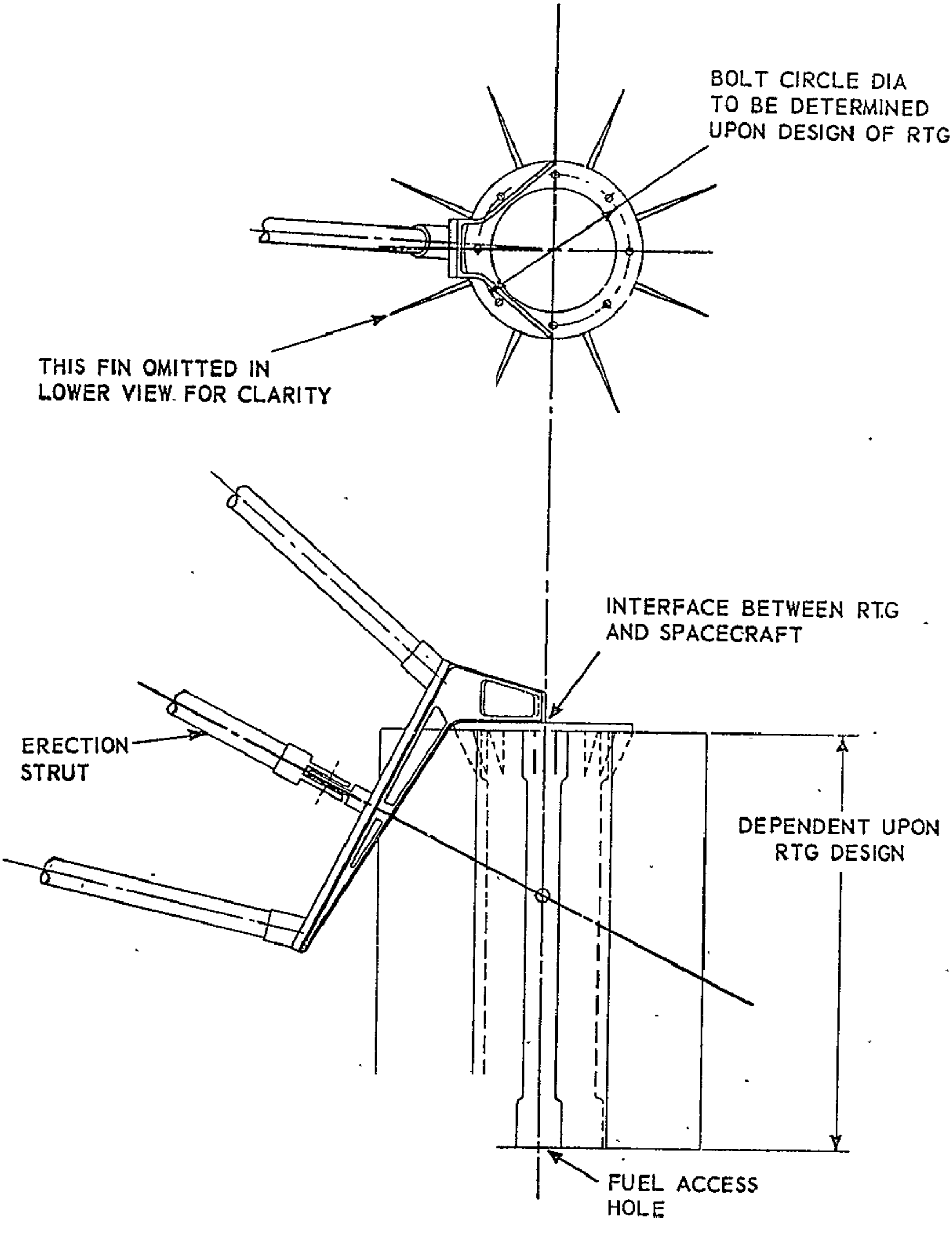


Figure D-1. RTG Power System — Spacecraft Mechanical Interface Drawing (SK-GSFC-701-1) (Sheet 3 of 3)

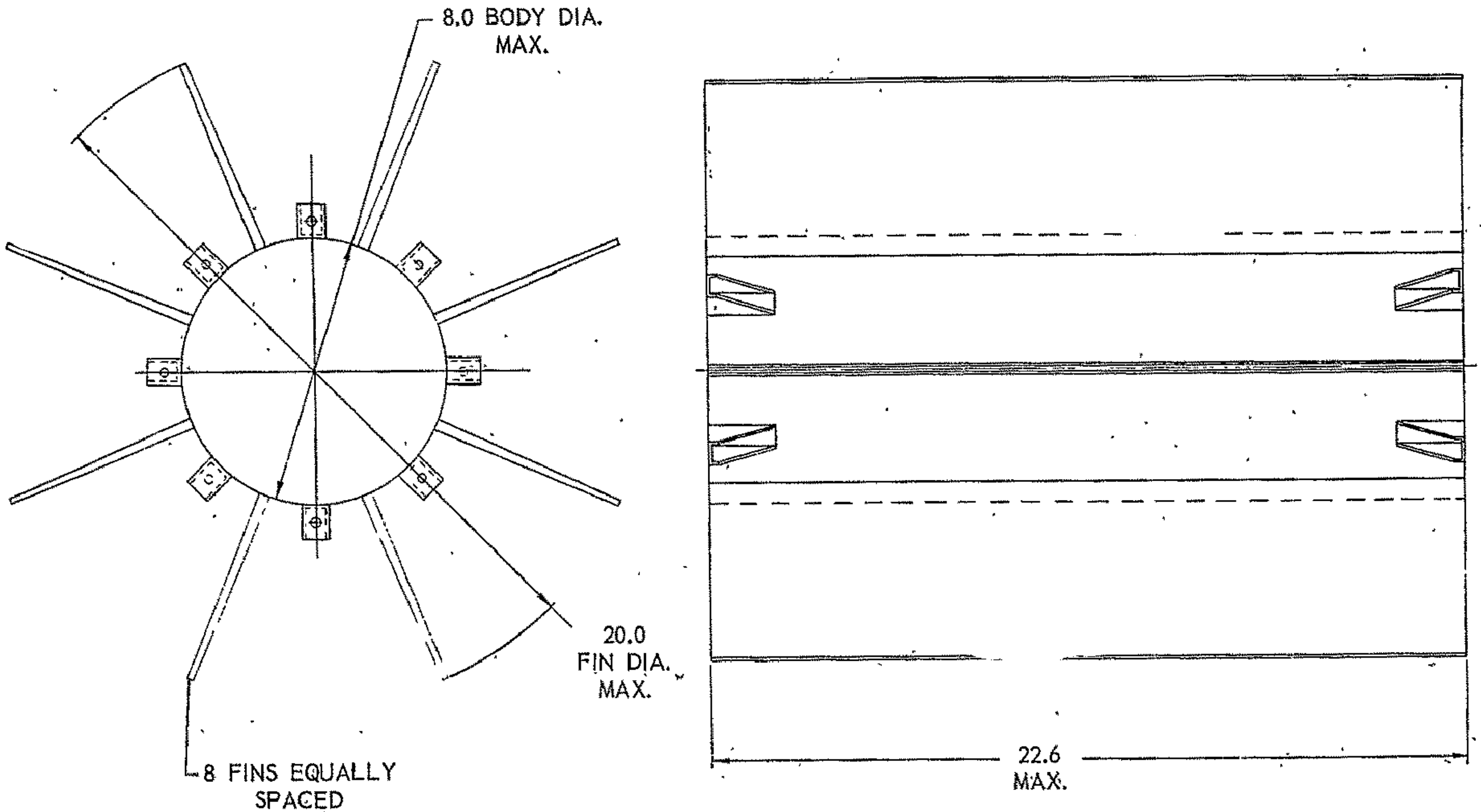


Figure D-2. RTG Power System Envelop Drawing (SK-GSFC-701-2)

APPENDIX E

RTG POWER SYSTEM PRELIMINARY

ELECTRICAL INTERFACE DRAWING — TYPICAI

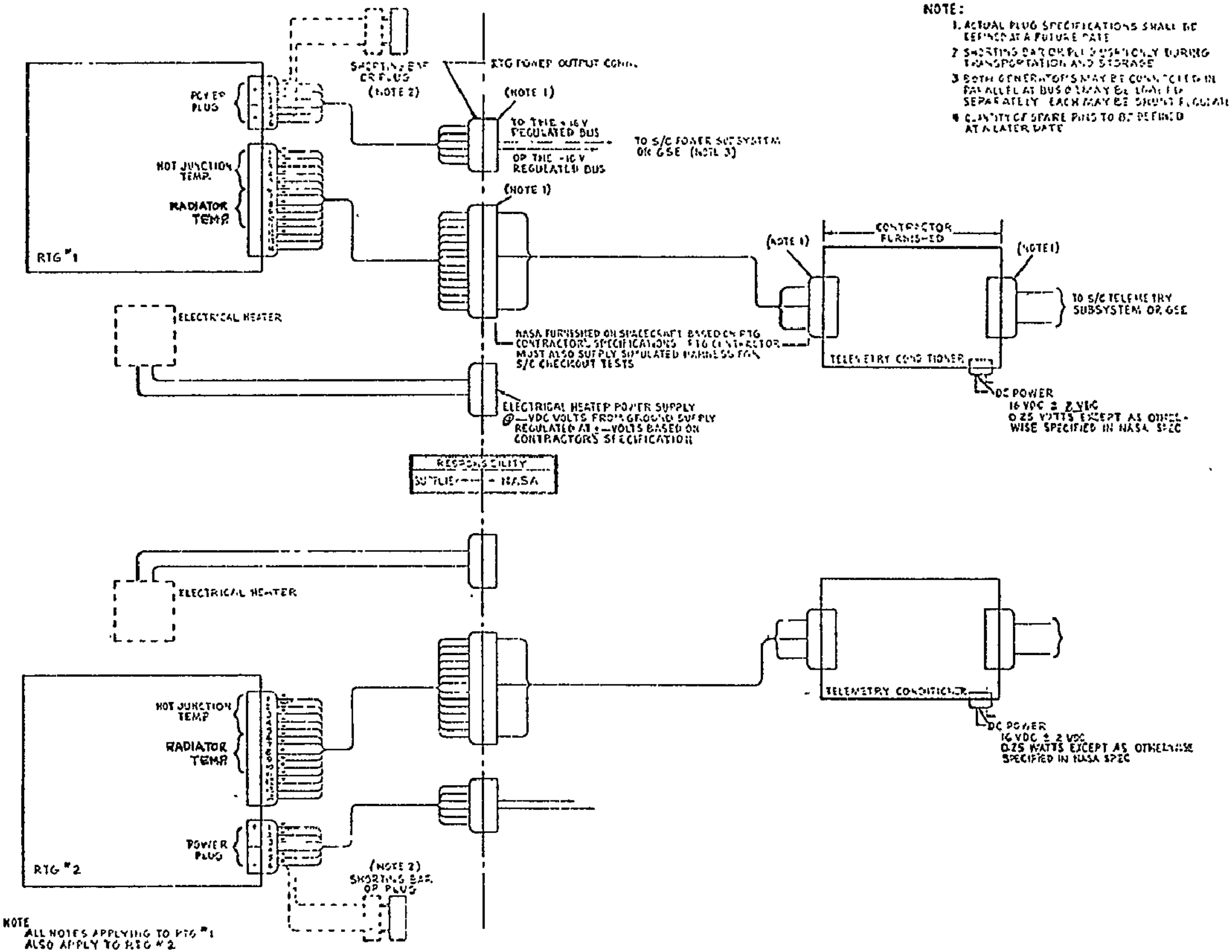


Figure E-1. RTG Power System — Spacecraft Electrical Interface Drawing (SK-GSFC-701-3)

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APPENDIX F

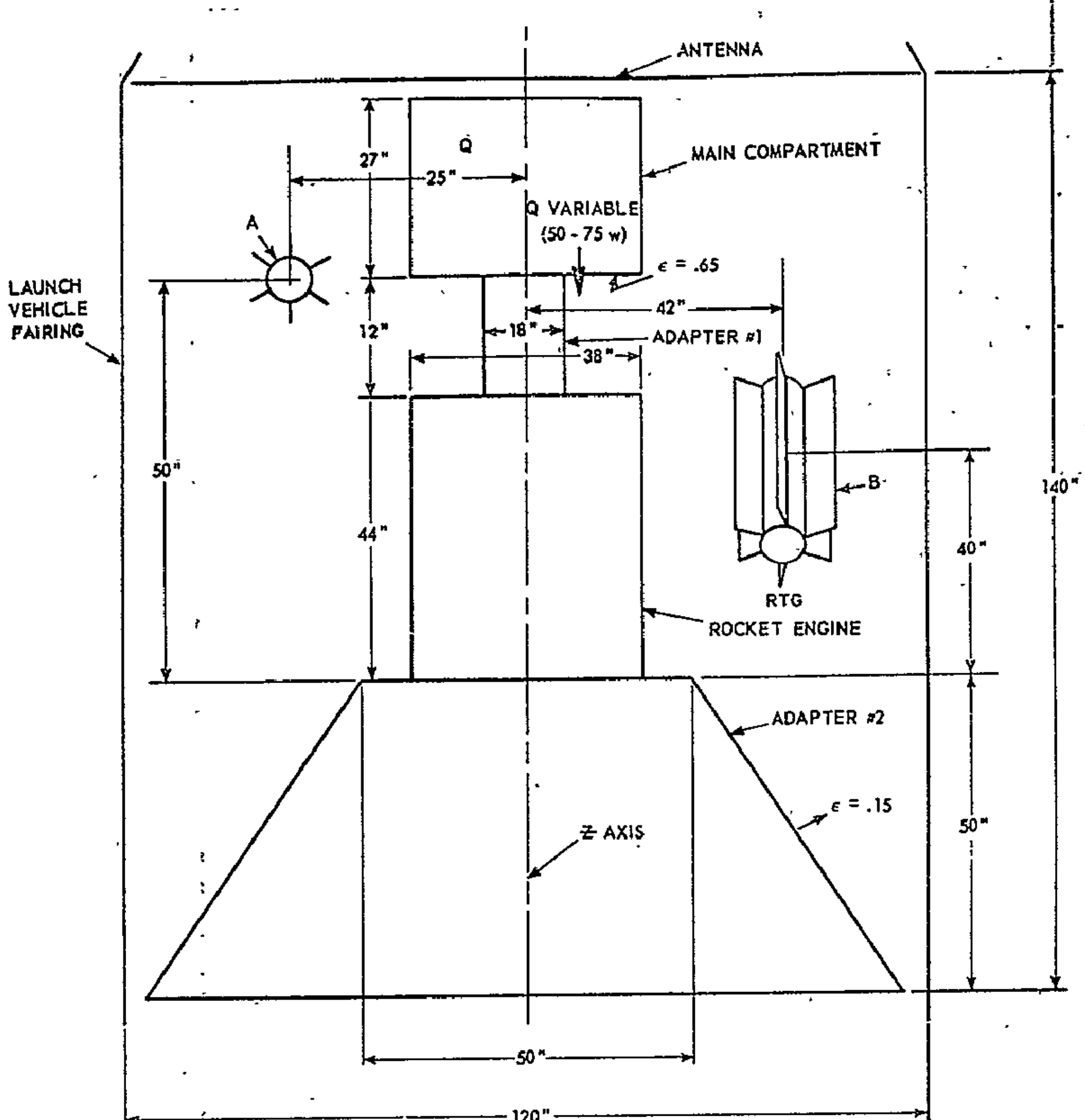
RTG POWER SYSTEM THERMAL INTERFACE REQUIREMENTS — TYPICAL

APPENDIX F

THERMAL REQUIREMENTS AND INTERFACES*

Figures F-1 and F-2 (Appendix F) defines the general thermal models to be used for thermal evaluation purposes. Figure F-1 defines thermal models during the launch phase. RTG may be mounted in either position A or B as shown. Figure F-2 defines the case after launch with the RTG's deployed.

*NEW MOONS Task V Report presents a detailed analysis of the effects of RTG's on various spacecraft elements of the GJP spacecraft.



Mode	Component	ε	Remarks & Typical Material
On Pad-in Fairing	Antenna	.05	Isothermal aluminum disc surface
	Main Compartment	.01 Sides .65 Bottom	Q watts thermal dissipation
	Fairing	.7	Solar heated fiberglass cylinders
	Rocket Engine	.15	Isothermal titanium solid cylinders
	Adapter #1	.15	Isothermal aluminum cylinder
	Adapter #2	.15	Isothermal aluminum conical surface
	Air		Temperature to be calculated
Flight-in fairing* 500 sec	Air	—	Vacuum
	Fairing	.7	Inside surface 125°F
Flight-fairing off*	Space		Space temp at −460°F with 1 solar constant @90° to Z axis
	Rocket motor	.15	Outside surface temp 900°F for 5 min.

*Antenna, main compartment and adapters same as on-pad

Figure F-1. Spacecraft/RTG Thermal Integration Model (Before Deployment) (SK-GSFC-701-4)

Component	Emissivity	Description
Antenna	0.15	Isothermal Solid Frustrum of a Cone
Forward Compartment	0.01	Isothermal Solid Cylinder
Main Compartment	.01	Isothermal Solid Cylinder

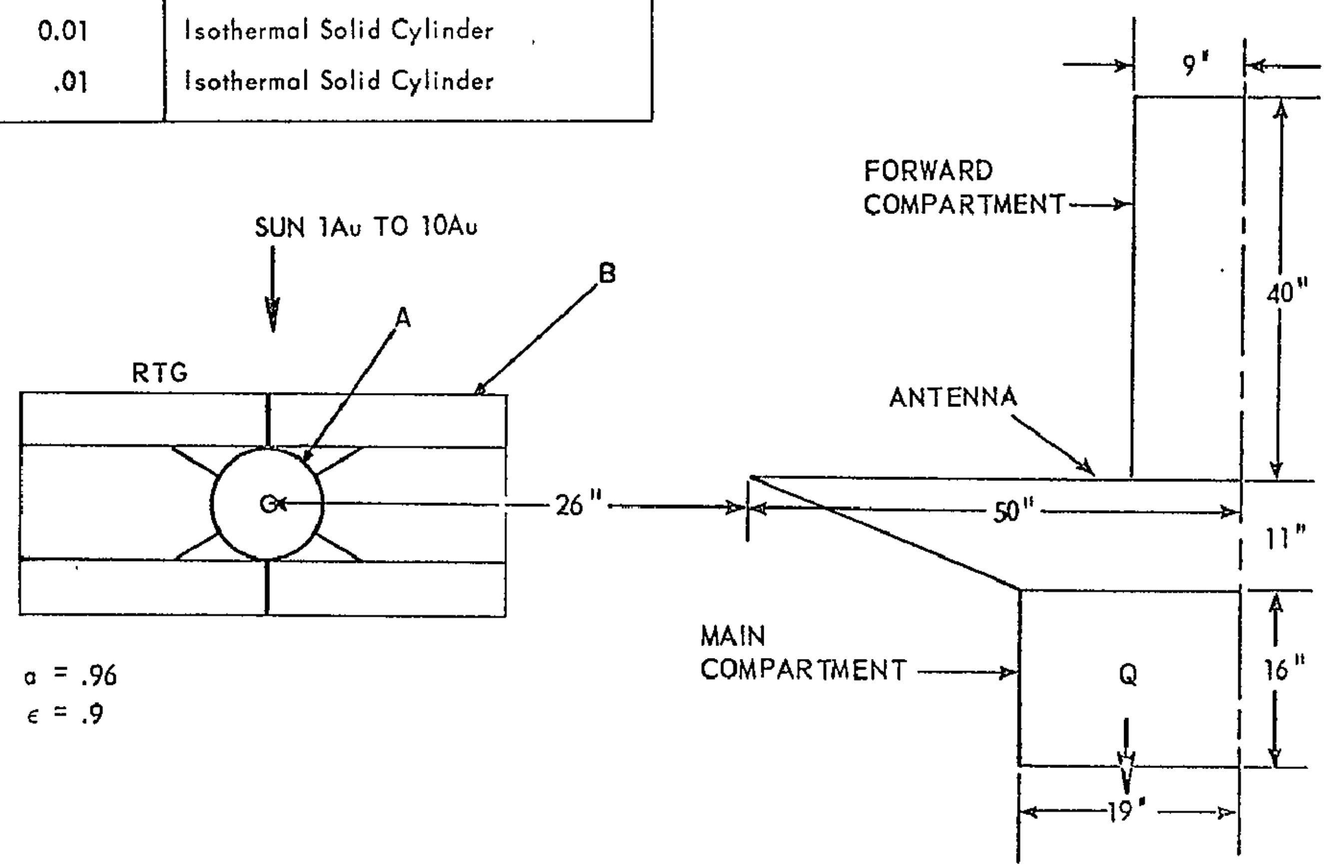


Figure F-2. Spacecraft/RTG Thermal Integration Model (After Deployment) (SK-GSFC-701-5)