NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Memorandum 33-647

Nuclear Electric Propulsion Stage Requirements and Description

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PREFACE

The work described in this report was performed by the Propulsion Division of the Jet Propulsion Laboratory.

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FIGURES

I. INTRODUCTION

A. SCOPE

This document establishes preliminary performance, operation, design, reliability, test, and scheduling requirements for developing a nuclear electric propulsion (NEP) stage. The NEP stage is designed for both planetary and geocentric missions. The mission requirements, flight trajectory, and operations of the NEP stage are quite different for the planetary as opposed to the geocentric mission, and therefore stringent requirements are placed on the versatility and capability of the stage. Sufficient NEP baseline data of various subsystems is provided so that a complete, integrated NEP stage meeting these requirements can be developed.

This document is intended as a guideline for the implementation of a program to develop a NEP stage. Therefore, as more information becomes available, the NEP baseline data should be updated and reissued as required, with the expectation that fewer updates will be needed as the development proceeds.

B. GENERAL DESCRIPTION

The NEP stage is composed of avionics, power, thrust, structure, and propellant subsystems as shown in Fig. 1. The power subsystem consists of a liquid-metal-cooled, fast-spectrum thermionic reactor, a heat-rejection subsystem using heat pipes, a lithium hydride (LiH) neutron shield, electrical cabling, a low dc voltage to high ac voltage power inverter, and a power level control subsystem. The thrust subsystem consists of power conditioners, thrusters, and electrical cabling. The power subsystem supplies 114.1 kWe to the thrust subsystem for at least 20,000 hours. The complete power distribution for the 120-kWe NEP stage is shown in Fig. 2.

C. STAGE VS MODULE

A stage is considered to be an autonomous vehicle, complete with its own avionics subsystem. The avionics subsystem provides spacecraft guidance, control, communication, and power. The stage is attached to the payload and is the active member with control of the flight. The payload is dependent on the stage. A propulsion module, on the other hand, is dependent upon the payload for flight command, control, and communications. The propulsion module could include certain control and engineering functions, but they must be integrated with the payload to complete the system.

For the stage to perform a planetary mission it must possess refinements which may not be required for geocentric missions. Some of these refinements could be required because of the following:

(1) Longer communication distances require more varied and complex operating modes and allocation of bit rates in planetary missions.

(2) Attitude sensors are more varied in planetary missions. Their field of view may vary from mission to mission and even during the course of a single mission.

(3) Data storage, retrieval, and transmission are more complex operations on a planetary mission.

(4) Lifetime and reliability of components may be more critical on planetary missions because retrieval, inspection or repair are not possible.

D. SIDE THRUST VS AXIAL THRUST

While this report deals primarily with the side thrust configuration, it is perhaps appropriate to mention an alternative configuration, the axial thrust system. The current side thrust configuration is in a more advanced state of development since the bulk of the work of the past several years has been on this system (Refs. 1-3). Earlier studies examined the axial thrust or conical configuration in detail (Ref. 4), and recently, because of interest in NEP for geocentric tug applications (Ref. 5), there was new examination of the axial thrust NEP stage. This work is described in detail in Refs. 6, 7, and 8.

The flight configuration for the side thrust NEP system is shown in Fig. 3. This system has several advantages: principally, (1) there is good separation between the high- and low-temperature components of the spacecraft; (2) at least 3 π view angle can easily be obtained for the science; (3) interference between the thruster ion beam and the science is minimized; and (4) thermal control of the thrusters is easily achieved since the back of the thrusters is open to space. Unfortunately, the side thrust configuration has certain drawbacks, the major ones being difficulty in accommodating variable mass payloads (e.g., probes) and difficulty in accommodating growth to higher power levels owing to problems in packaging the larger systems for shuttle launch. Even with folding, the higher-power-level (240-400 kWe) side thrust systems are too long for packaging in the shuttle bay.

Conversely, the axial thrust configuration (Fig. 4) solves the variable payload problem since the center of mass moves only along the thrust axis, and because of its stubby conical shape even the higher-power-level (240 kWe) system will fit in the shuttle. But, while the axial thrust configuration solves several problems it complicates others. Thermal control and radiation shielding problems must be resolved. These are discussed in detail in Ref. 8.

Because of the better definition available, the remainder of this report deals primarily with the side thrust propulsion system/stage. It is important to note, however, that in many cases the requirements are configuration-independent and the descriptions are applicable to both the axial and side thrust systems.

II. REFERENCE MISSIONS

The NEP stage will be designed for various classes of planetary missions, represented by the following five:

A: Mars Rover, 1984.

B: Halley Rendezvous, 1983.

C: Uranus/Neptune Flyby (with probe), 1986.

D: Mars Surface Sample Return, 1990.

E: Ganymede Orbiter/Lander, 1990.

A detailed description of these mission objectives and trajectories is attached as Appendix A. These missions are typical of the high-energy unmanned planetary exploration missions under consideration to be conducted in the last two decades of this century (Ref. 9). The NEP stage as defined in this report provides the necessary versatility and capability to accomplish these missions. In addition, the NEP stage will be designed for each satellite placement orbit and/or retrieval mission in combination with the Shuttle (Ref. 10) or Shuttle/Tug vehicles (Ref. 11).

III. NEP STAGE REQUIREMENTS AND DESCRIPTION

A. FUNCTIONAL CHARACTERISTICS

An in-flight configuration of a side thrust NEP stage is shown in Fig. 3. Detailed descriptions of the side thrust stage and the engineering considerations that have led to the present concept are described elsewhere (Refs. 1 and 12). A schematic temperature diagram for the power and thrust subsystems is given in Fig. 4. When integrated with the Shuttle/Centaur, the side thrust NEP stage is folded during launch, while the axial thrust NEP stage is not folded (Fig. 5).

1. System Interfaces

The thrust, power, avionics, propellant, and structure subsystems all interface with each other. In order to separate the design and development of the various subsystems yet insure that the designs will be compatible and will result in the "best" configuration, it is necessary to identify all of the interfaces between each of the subsystems and to set requirements which both subsystems must adhere to. Tables 1, 2, and 3 list the major interfaces that exist between the subsystems. In the first two columns the major components involved are identified. If one of the columns is blank the interaction is presumed to be with the entire subsystem in general rather than a specific component.

2. Thrust Subsystem

The thruster specific impulse is between 4000 and 5000 seconds. The lower value provides larger values of thrust but requires more thrusters per kWe, more propellant, and larger propulsion system specific mass, and it leads to lower optimum power levels. The higher specific impulse (5000 s) allows lower propellant fractions and higher payloads but requires longer mission times. The lower specific mass and lower propellant fractions coupled with mission analysis considerations lead to higher optimum power levels.

The choice of the number of power conditioners and the power capacity of each power conditioner is primarily dependent on the thruster diameter and the number of thrusters. The output voltage of the power conditioners is determined from the specific impulse.

A typical thrust subsystem is comprised of the following elements:

(1) Eighteen 30-cm-diameter mercury bombardment ion thrusters (14 operating, 4 standby).

(2) A thruster array structure (TAS) for mounting the ion thrusters.

(3) Eight gimbal actuators.

(4) Two translator actuators with carriage and translator rods.

(5) Thirty-six power conditioning units (PCUs), 3.8-kW input power each; two PCUs required for each thruster.

(6) Two propellant storage tanks, capacity 1000-4500 kg of mercury each, depending on the mission.

(7) One propellant feed and manifold with associated valves.

3. Power Subsystem

The power subsystem provides 114 kWe of electrical power to the thrust subsystem and 1 kWe to the avionics subsystem when the reactor is operating (see Fig. 2). During the launch phase or any period when the reactor is dormant, the power for the avionics subsystem is provided along the same interface by the NEP stage batteries.

A typical power subsystem is comprised of the following elements:

(1) One thermionic reactor with 162 thermionic fuel elements (TFEs), 18 control drums, and 9 control drive motors.

(2) A heat rejection subsystem including two EM pumps, NaK headers, and 2690 heat pipes arranged in a planar or cylindrical radiator assembly.

(3) A LiH neutron shield clad in stainless steel.

(4) Propellant, propellant tank, and structure serving an additional function as gamma radiation shielding.

(5) Twenty-seven power converters each connected to a series string of 6 TFEs forming a 23-Vdc, 6-kWe TFE power source.

4. Avionics Subsystem

The avionics subsystem provides guidance, control, communications, and data processing. A typical avionics subsystem is comprised of the following elements:

(1) Control for NEP stage/science interface.

(2) Control for NEP/Centaur/Shuttle interfaces.

(3) Control of deployment mechanisms.

(4) Thrust vector control (TVC).

(5) Propellant feed control.

(6) Flight data system (FDS).

(7) Computer command subsystem (CCS).

(8) Attitude control subsystem (ACS).

5. Propellant Subsystem

The propellant subsystem provides propellant to the thrust subsystem in the form of a regulated flow of liquid mercury. The propellant serves a dual function. In addition to its role as the working fluid for the thrusters, it serves as additional shielding for neutrons and gamma radiation. As such, the tank(s) must be configured to satisfy both propellant feed requirements and shielding requirements.

6. Structure Subsystem

The structure is treated as a single subsystem in order to provide a consistent structural design philosophy for the entire NEP stage. The structure subsystem provides the basic structural support for the entire NEP stage. It also includes launch vehicle adapters, flight structure, and deployment mechanism.

B. REQUIREMENTS

1. Mass Requirements

The NEP-stage mass breakdown is as follows:

Power subsystem			2400 kg
Reactor		1200	
Radiator		460	
Shield		640	
Batteries		100	
AgZn (2 kW-h)	27		
NiCd (1.7 kW-h)	73		
Thrust subsystem			780
Thruster		250	
Power conditioner		430	
Cables		100	
Avionics subsystem			480
Structure subsystem			540
Miscellaneous structures for power subsystem		270	
Miscellaneous structures for thrust subsystem		90	
Miscellaneous structures for avionics subsystem		60	
Stage adapter		70	
Deployment mechanism		50	
NEP stage			4200

2. Specific Mass Requirements

Power subsystem	20.0 kg/kWe
Thrust subsystem	6.5
Avionics subsystem	4.0
Structure subsystem	4.5
NEP stage	35.0

3. Thermal and Electrical Power Requirements

The power balance for the nuclear thermionic propulsion system is shown in Fig. 2.

4. Power Subsystem Performance Requirements

Power level	120 kWe
Voltage (at I_s = 4000 s)	1500 Vac
Efficiency (end of life)	8%
Lifetime (effective full power hours)	~20,000 EFPH
Total useful life	~6 yr
Specific mass for reactor	<12 kg/kWe
Output voltage from reactor	≥23 V
Heat rejection area	31 m^2
Shutdown to near zero EFPH during coast phase	

5. Thrust Subsystem Performance Requirements

Specific impulse (variable) of thrusters	4000 to 5000 s
Propulsion efficiency	75%
Power available to spacecraft	
- thrusting	1 kWe
- coasting	1 kWe
- launch	400 We
Thruster lifetime	20,000 hr
Nonjettison mode	Yes
Escape and capture mode	mission-dependent

Dimensions (stowed, including payload)	<8.02 m, length 4.58 m, diam
Thruster restart capability	Yes
Thruster type	30-cm Hg ion
Number of thrusters	18
Thrust vector control	gimbal and translator
Heat rejection area	20.4 m^2

6. <u>Avionics Subsystem Performance Requirements</u>

Attitude control subsystem

Flight command subsystem

Flight telemetry subsystem

Video/lighting subsystem

Docking subsystem

7. <u>Propellant Subsystem Performance Requirements</u>

Propellant	Mercury or xenon
Propellant tank and feed system	~3% of mercury mass ~10% of xenon mass

8. <u>Nuclear Radiation</u>

The reactor nuclear shield will be designed so that the LiH neutron shield will reduce the integrated neutron flux to less than 10^{12} nvt and the mercury propellant and tungsten gamma shield (if required) will reduce the gamma dose to less than 10^6 rad to the science instruments.

9. <u>Thermal Control</u>

There are two critical thermal control areas which should be considered for the NEP stage. First, in the thrust subsystem, the mercury vaporizer in each thruster should be kept at a temperature not exceeding 250°C. This is necessary to assure adequate control margin. Second, the surface temperature of the power conditioner radiator should be less than 100°C.

10. Reliability

The reliability requirement for the NEP propulsion system is to be 0.900. This is the probability that the NEP stage will be operating at 120 kWe at the end of 20,000 h at full power. In doing the reliability analysis (including failure mode and effect analysis, criticality analysis, and reliability prediction), sources and causes of unreliability or single failure modes leading to catastrophic mission failure will be eliminated by design modification.

C. TRANSPORTABILITY

The major subassemblies when protected by packing are to be transportable in any attitude by land, sea, or air without degradation The complete spacecraft assembly is to be transportable in special containers, either vertically or horizontally after final checking.

D. ACCEPTANCE

The stage will be accepted on the basis of successful functional tests, together with a review of the production and test history of those components in the stage.

E. CLEANLINESS

A plan will be developed to prevent and control deterioration of parts, components, subsystems, and systems by corrosion and contamination. Emphasis will be placed on (1) particulate contamination interface with moving surface or propellant channels and orifices, (2) propellant contamination, and (3) influence of contaminations on material properties during processing, shipping, and operations. The plan will specify fabrication, assembly, cleaning processes and facilities, inspection and marking, sealing and packaging, certification and surveillance, and shipping and site operations.

F. ELECTROMAGNETIC INTERFERENCE

The spacecraft will be designed and tested to satisfy the requirements of military specification MIL-E-6051-D. Undesired interactions and malfunctions of all electronic and electrical components are to be eliminated.

G. INTERCHANGEABILITY

All qualification hardware will be interchangeable down to the component level.

H. RELIABILITY, QUALITY ASSURANCE, AND SAFETY

The reliability and safety goals are to be achieved through inherent conservative design whenever possible. The quality assurance for high reliability and safety is to be achieved through rigorous screening processes at both the manufacturing and assembly sites. In particular, maximum effort should be placed on a design which eliminates all credible single failure modes which could jeopardize the completion of the mission or endanger the launch crew or the general public.

The nature of NEP is such that reliability must be obtained primarily through consistent, thorough, rigorous design and test practices. Design practice standards covering principles, philosophy, procedures, and criteria will govern the requirements of stage hardware. These design practice standards are to be prepared by the contractors and approved by NASA representatives.

Reliability, safety, and quality assurance program plans will be developed by the prime and principal subcontractors and approved by NASA.

I. RADIATION SHIELDING

Radiation shielding provision will be made to protect components from radiation degradation of reliability, performance, or endurance beyond the limits specified (see Section J-3).

J. ENVIRONMENTS

The spacecraft is to be capable of meeting all performance and endurance requirements after being subjected to all ground, launch, flight, and self-induced environments. The environmental conditions expected are as follows:

1. Launch

a. Shock. A single shock pulse (5-200 g, 0.7/0.2 msec terminal peak saw tooth pulses) shall be applied to the subsystem, five times in each of three orthogonal directions.

b. Vibration. The subsystem shall be subjected to sinusoidal vibration of from 5 to 30 Hz at 0.75 g (rms), 30-250 Hz at 7.5 g (rms), 250-1000 Hz at 9.0 g (rms), and 1000 to 2000 Hz at 6.0 g (rms). Sweep rate shall be 2 octaves/min. Each thermionic fuel element should be acceptance-tested for vibration as follows:

(1) The thermionic fuel element shall be subjected to a sinusoidal sweep test. The frequency shall be swept at a rate of one octave/min from 5 to 200 Hz and from 200 to 5 Hz. The input levels shall be 0.125 g (rms) or as low as possible, 0.25 g (rms) and 0.5 g (rms) in each of two perpendicular axes.

(2) The fuel element shall be subjected to three random vibration tests over a frequency interval of 200 to 2000 Hz for 1 min in each of two perpendicular directions. The input levels shall be 0.0022 g^2/Hz, 0.0044 g^2/Hz, and 0.0088 g^2/Hz.

2. Thermal/vacuum

The following thermal/vacuum conditions are expected:

Radiative sink:	-55 to +20°C 10^{-5} to 10^{-8} torr
Solar:	the NEP stage should be capable of surviving the solar incidence at 0.3 AU (Encke) and 0.7 AU (Halley)

Deep space:	heat must be provided to maintain a minimum temperature of -55°C on the PCU, especially during coast phase of the mission when the thermionic reactor is shut down

3. Radiation

The total radiation dose to the critical spacecraft components including thrust subsystem power conditioners must be less than 10^6 rad (gamma) and 10^{12} nvt (neutron). Electron and proton particle flux to be experienced by the spacecraft (mission-dependent) during transit through the geomagnetic radiation (Van Allen) belts may be converted to an equivalent dose (rad or nvt).

4. Meteoroid

The detailed models of meteoroid environment for interplanetary space and near a planet are given in Appendix B.

5. Electromagnetic Radiation

The ranges of intensity, flux, and temperature associated with electromagnetic radiation depend on the mission and the relative position of the spacecraft to the planet.

IV. NEP STAGE DEVELOPMENT PLAN AND TEST PROGRAM

A preliminary development schedule for the NEP system is given in Fig. 6.

A. SYSTEM DEVELOPMENT PLAN

1. Propulsion System Technology Program Definitions

a. Basic Technology.

(1) Development of analytical tools for prediction of component performance under range of parameter variations.

(2) Characterization of basic component materials properties under operating conditions.

(3) Determination of a wide range of future applications that direct the basic parameters and test conditions.

(4) Development of fabrication processes.

b. Component Technology.

(1) Development of components and/or elements of components for anticipated functional requirements of prototype device.

(2) Identification of major component failure mechanisms and the modification or redesign of components as necessary to eliminate or reduce effects to acceptable level.

(3) Demonstration of performance over a range of possible operating conditions.

(4) Complete specification of materials and fabrication procedures.

(5) Component qualification for range of anticipated parameter variations and full lifetime.

(6) Definition of interfaces and interactions with other components.

c. Technology Verification Tests.

(1) Complete specification of operating conditions and interfaces for all components.

(2) Complete fabrication drawings and specifications issued.

(3) Component fabrication and operations as specified. Environmental tests at conditions beyond design requirements and recheck operations as specified.

(4) Verification of any component performance uncertainty.

2. NEP Technology Tests (to be completed prior to development)

a. Basic Technology.

(1) Clean critical reactor experiments.

(2) Reactor design (mechanical, nuclear, thermal, and hydraulic detailed design).

(3) TFE (emitter dimensional stability, carbide fuel performance stability, fission product management, insulator irradiation stability for >20,000 hr).

(4) Heat pipe (operational characteristics, ΔT, ΔP, and fabrication procedures).

(5) Thrusters (high specific impulse, high power, and 20,000-hr life).

(6) Power conditioner (radiation hardening, high temperature, and beryllium structure).

b. Component Technology.

(1) Electromagnetic pump windings.

(2) Transmission line joints.

(3) Thermal insulation.

(4) Heat pipe radiator.

(5) Control drum environmental test.

(6) TREX vessel hydraulic test.

(7) TFE-vessel joint development.

(8) Reactor C&I breadboard tests and irradiation tests.

(9) Mechanical qualification testing.

(10) Mockup critical experiments.

(11) TFE (cesium envelope integrity, flight-configured electrical lead and Cs reservoir, reflection blocks,

fabrication cost reduction, shock and vibration tests, operation in liquid-metal-cooled fast reactor).

(12) Power conditioner components.

(13) Control system breadboard tests.

(14) Thruster life tests.

(15) Thruster/breadboard PCU tests.

(16) Thruster-translator/gimbal actuators.

c. Technology Verification Tests.

(1) Reactor vessel joint fabrication and test.

(2) Control drive environmental test.

(3) Reflector elements.

(4) Prototype vessel hydraulic tests.

(5) Expansion compensators.

(6) Nuclear shields.

(7) Instrumentation and controls.

(8) Electromagnetic pump.

(9) Radiator.

(10) Piping.

(11) Power conditioner.

(12) Thruster.

(13) Thrust vector control.

B. MAJOR TEST PROGRAM

1. TREX (thermionic reactor experiment)

a. Test Article. A fast thermionic reactor containing 162 thermionic fuel elements and delivering 162 kWe at 23 Vdc. The reactor vessel and reflector are prototypic. The reflector control drives are not necessarily prototypic.

b. Test Objectives.

(1) Measure dynamic control characteristics of the reactor, both prompt, delayed, and static reactivity effects.

(2) Measure performance on an all-TFE core including controllability and electrical power distribution.

(3) Measure reactor control margin at design temperature.

(4) Determine core thermal flux profile.

c. Test Facility. Modify the existing SPERT IV building at NRTS, Idaho, to perform reactor test of a 162-TFE thermionic reactor. Incorporate digital-computer-controlled TFE electrical loads.

2. HRS (Heat Rejection Subsystem)

a. Test Article. The heat rejection subsystem removes the waste heat from the reactor and rejects it to space. This test article includes flight prototype components that have passed type approval testing. This includes a full heat-pipe radiator, one dual EM pump, two expansion reservoirs, and piping, all arranged in flight configuration. The heat input will be by an electric heat source into the primary loop piping, and the heat will be radiated to a cold wall within a vacuum chamber.

b. Test Objectives.

(1) Confirm performance of prototype components.

(2) Confirm performance of assembly as to temperatures and heat rejected.

(3) Determine the interface temperatures and heat loads between radiator, piping, structure, reactor, and shield.

(4) After all performance checks, conduct a 20,000-hr life test.

(5) Post-test examine and analyze all components for any possible life-limiting conditions.

c. Test Facility. Use existing vacuum chamber, control room, and assembly buildings at JPL. Use I^2R heating for the heat source and water-cooled cold wall for heat rejection load.

3. TSM (Thrust Subsystem Module)

a. Test Article. This is a module of the thruster power conditioner array and consists of six thrusters with six power conditioners mounted on a partial (approximately one-half) thrust vector control (TVC) mechanism. The TVC electronics and computer are included. One propellant tank with its associated feed system is also included. All components are prototype and type-approval (TA) tested prior to use. The simulators required are 3000-V, 40-kWe power source, command control sequence hardware, attitude control logic, and flight trajectory data.

b. Test Objectives.

(1) Verify thruster and power conditioner performance in flight configuration.

(2) Verify thermal vacuum operations of power conditioner, switches, actuators, electronics, propellant tank, and propellant feed system.

(3) Finalize automatic control routines.

(4) Finalize preflight checkout requirements.

(5) Determine and eliminate any undesirable subsystem interactions.

(6) Perform a 20,000-hr endurance test on as many components as possible.

c. Test Facility. Use an existing vacuum chamber and facility at JPL.

4. GPR (Ground Prototype Reactor) 1 and 2

a. Test Article. All-TFE, prototype thermionic reactor vessel, reflector, fuel elements, and control drives; test will incorporate flight prototype instrument and control package for evaluation.

b. Test Objectives.

(1) Confirm performance of prototype reactor.

(2) Confirm thermal environmental factors of reactor components.

(3) Evaluate performance of flight instrument and control package when operated integrally with a reactor test.

(4) Evaluate performance for startup and shutdown power profiles and for operation at part load.

(5) Measure electrical output spatial distribution for TFEs.

c. Test Facility. Use the facility prepared for TREX. Remove the TREX reactor and install GPR reactor and instrument package.

5. GPPS (Ground Prototype Propulsion System)

a. Test Article. All flight components configured as a prototype propulsion system; complete power subsystem with a non-nuclear reactor core. The core would contain 20 to 30 fueled flight-type TFEs. The power subsystem includes prototype shield, EM pumps, piping, NaK, and radiator. The thrust subsystem includes the complete thrust vector control, 6 active thrusters, 12 dummy thrusters, 6 active power conditioners, and 12 dummy conditioners. The propulsion system has a complete flight structure with joints and the complete control system. Also included is the propellant subsystem with active feedlines to the 6 active thrusters. The avionics subsystem simulator is a complete subsystem with the payload mass simulated. The electric power source simulator is included to provide power to the 6 active power conditioners.

b. Test Objectives.

(1) Design, fabricate, and assemble the complete propulsion system.

(2) Check out mechanical, electrical, and control interface with all other subsystems.

(3) Fold propulsion system into launch configuration.

(4) Check to see that structure and components can withstand launch environment under extreme conditions.

(5) Unfold into flight configuration after environmental tests and perform system operational tests.

(6) Perform startup and shutdown tests with commands through the net spacecraft, etc.

(7) Failure modes and effects testing on entire system.

c. Test Facility. Modify an existing facility and vacuum system at Edwards Test Station, California. The environmental test equipment is available there.

6. NEP Stage PTV (proof test vehicle)

a. Test Article. Complete flight NEP stage assembled, checked out, and type-approval (TA) tested, and operational tests performed on all components. The reactor is complete except that the TFEs are simulated with rejects and/or non-nuclear fuel.

b. Test Objectives.

(1) Establish all assembly, checkout, and acceptance test procedures.

(2) Make sure all systems are operational after TA test of a complete propulsion system, including unfolding.

(3) Operate the heat rejection subsystem at low temperature to insure no leaks.

(4) All components, except TFEs, are useful spares for the flight system.

c. Test Facility. Use the facility at Edwards Test Station, California, as built for the GPPS test above. Unfolding test to be done at LeRC, SPF.

7. FTV (Flight Test Vehicle)

a. Test Article. The full NEP stage to be used for the flight acceptance test, including all the subsystems with a dummy payload for docking and rendezvous checkout.

b. Test Objectives.

(1) Perform flight acceptance (FA) environment test in the launch configuration with an actual payload attached.

(2) Perform operational checkout on the complete thrust subsystem, propellant feed subsystem, control subsystem, guidance, attitude control, communication, etc.

(3) Circulate the NaK and perform a zero power critical experiment with the reactor prior to launch.

(4) After all things are checked, the stage is flight ready.

c. Test Facility. Space shuttle to earth orbit with tracking, data acquisition, and flight operations controlled from JPL DSN.

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Table 1. Avionics/thrust subsystem interfaces

Subsystem: Avionics	Subsystem: Thrust	Interface type	Interface definition	Capacity requirement
Structure	Structure	Mechanical		
Flight data system (FDS)		Electrical	From each PC analog data lines	4
Computer controller sequence (CCS)		Electrical	1. To each PC command lines	5
			2. To each PC analog REF's	2
			3. To TVC electronics digital REFs	2
			4. Storage	3000 10-bit words
Attitude control system (ACS)		Electrical	To TVC electronics	10 signal channels

Table 2. Avionics/power subsystem interfaces

Subsystem: Avionics	Subsystem: Power	Interface type	Interface definition	Capacity requirement
FDS		Electrical	1. Voltage from thermionic fuel element (TFE)	27
			2. Current from TFE	27
			3. Neutron level from reactor	3
			4. Average neutron level	1
			5. Coolant temperature	10
			6. Coolant pressure	3
			7. Control drum position	9
			8. Reactor cesium temperature	27
			9. Contact closure	100
			10. Other	100
CCS		Electrical	1. To drum stepping control	2
			2. To cesium temperature control	2
			3. Storage	3000 10-bit words
	Power	Electrical	To avionics	1000 W
	Reactor	Radiation	Dose rate at avionics	10^6 γ and 10^{12} nvt

Table 3. Power/thrust subsystem interfaces

Subsystem		Interface type	Interface definition	Capacity requirement
Power	Thrust			
Structure	Structure	Mechanical		
Power		Electrical	1. To power conditioners for thrusting	114.1 kWe HV-AC
			2. To power subsystem	4.98 kWe
	Thrust	Electrical	Power demand on AC bus	0 to 114.1 kWe

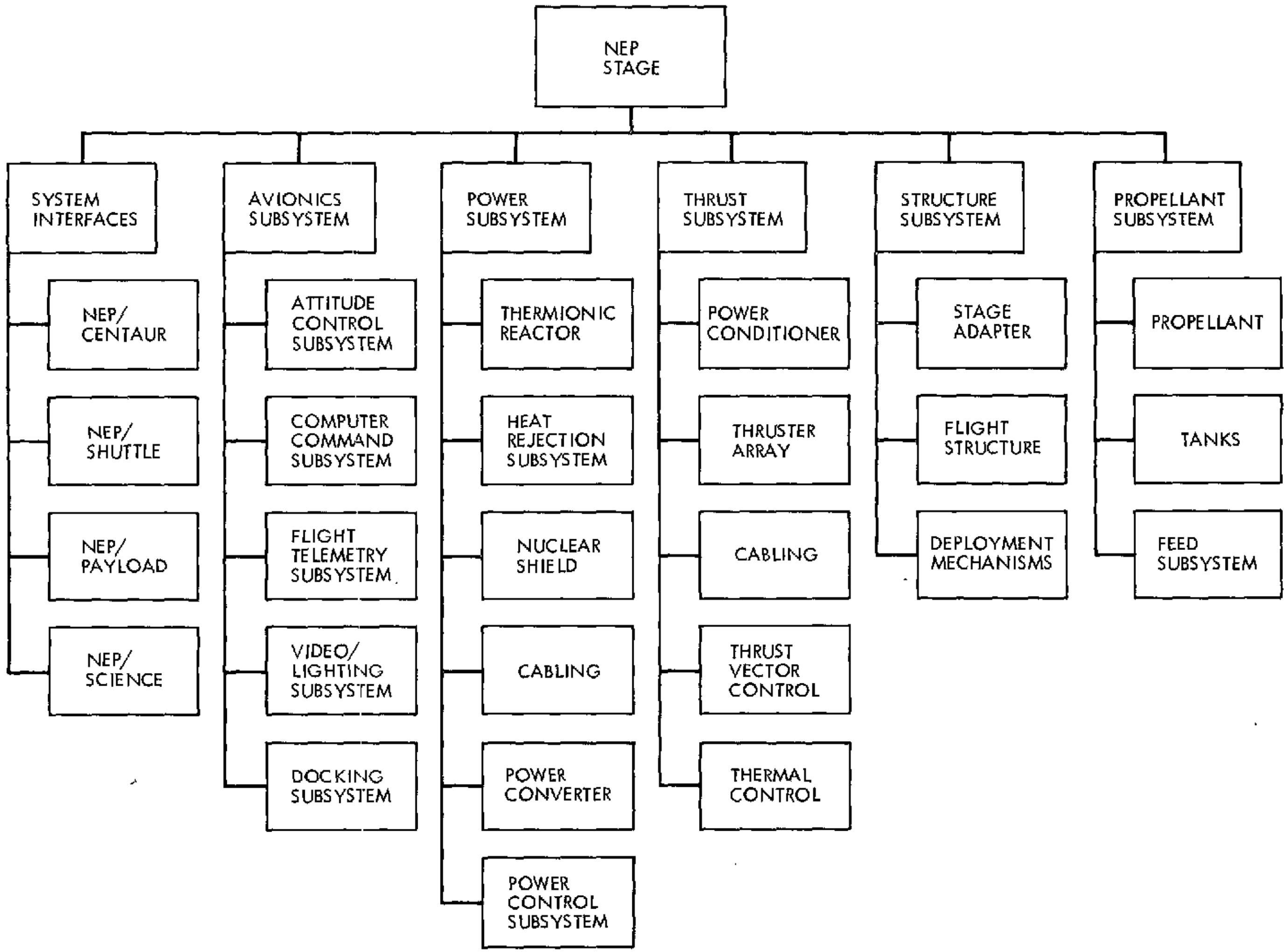

Fig. 1. NEP stage subsystems

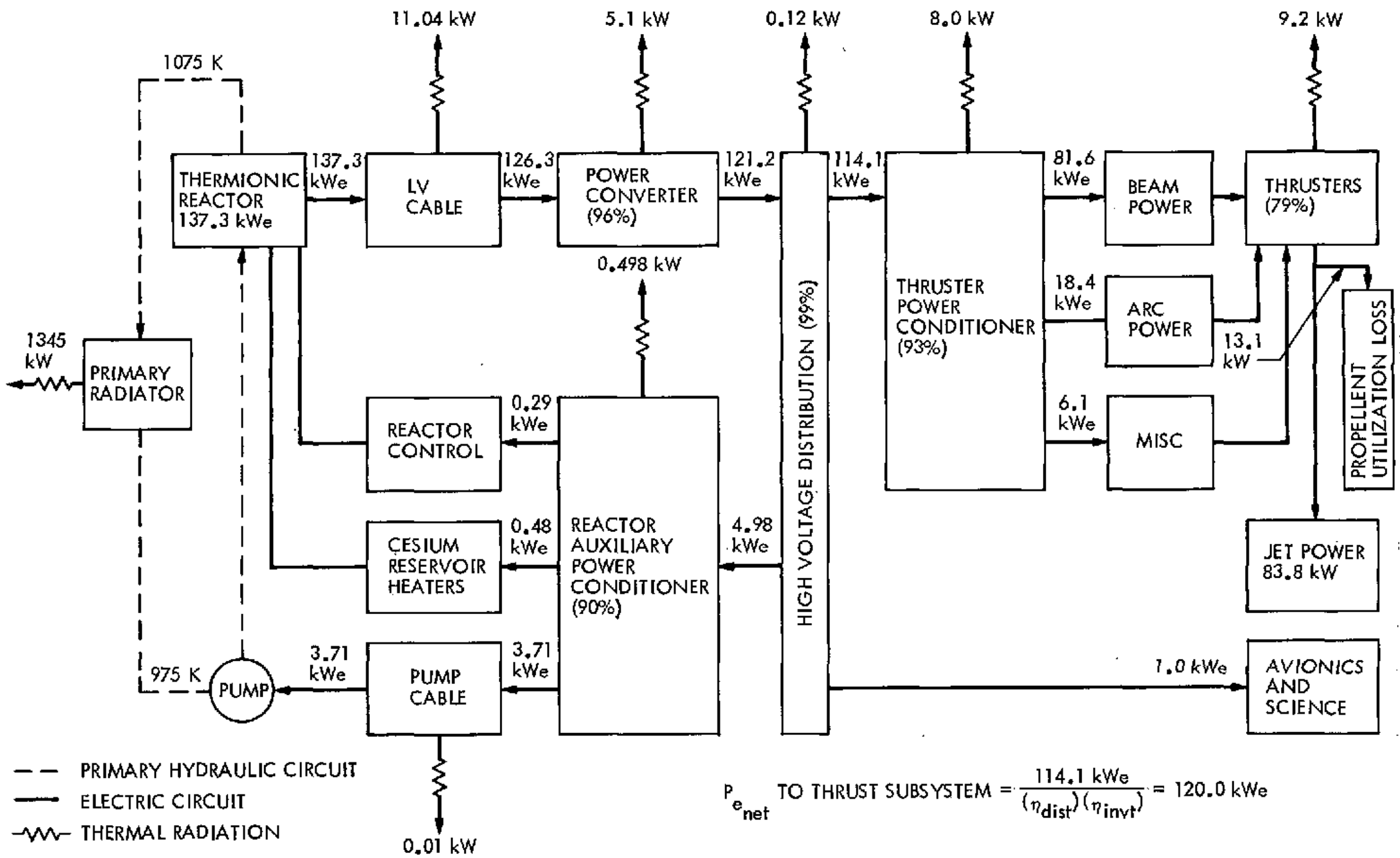

Fig. 2. NEP stage power distribution

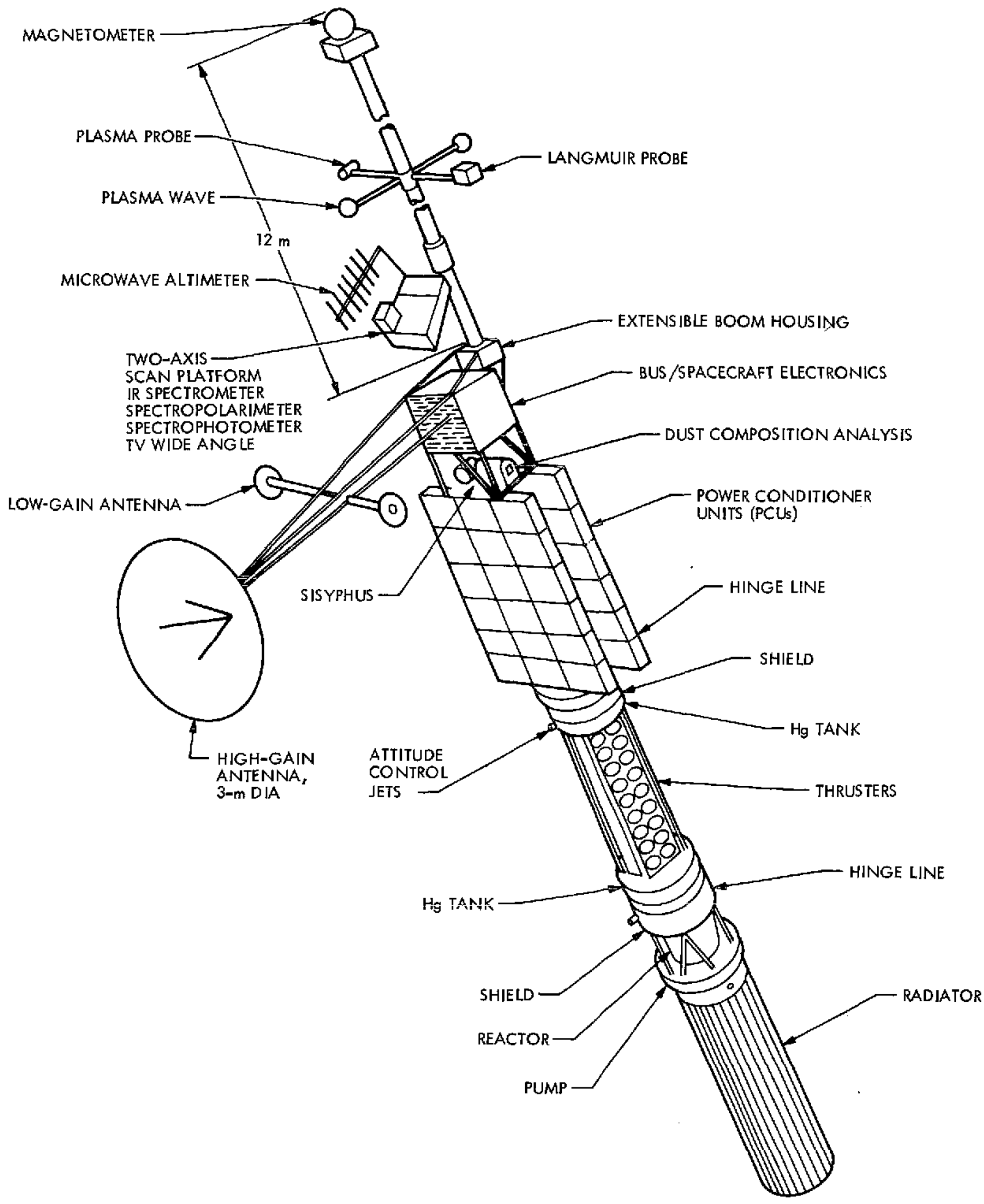

Fig. 3. NEP stage side thrust conceptual arrangement

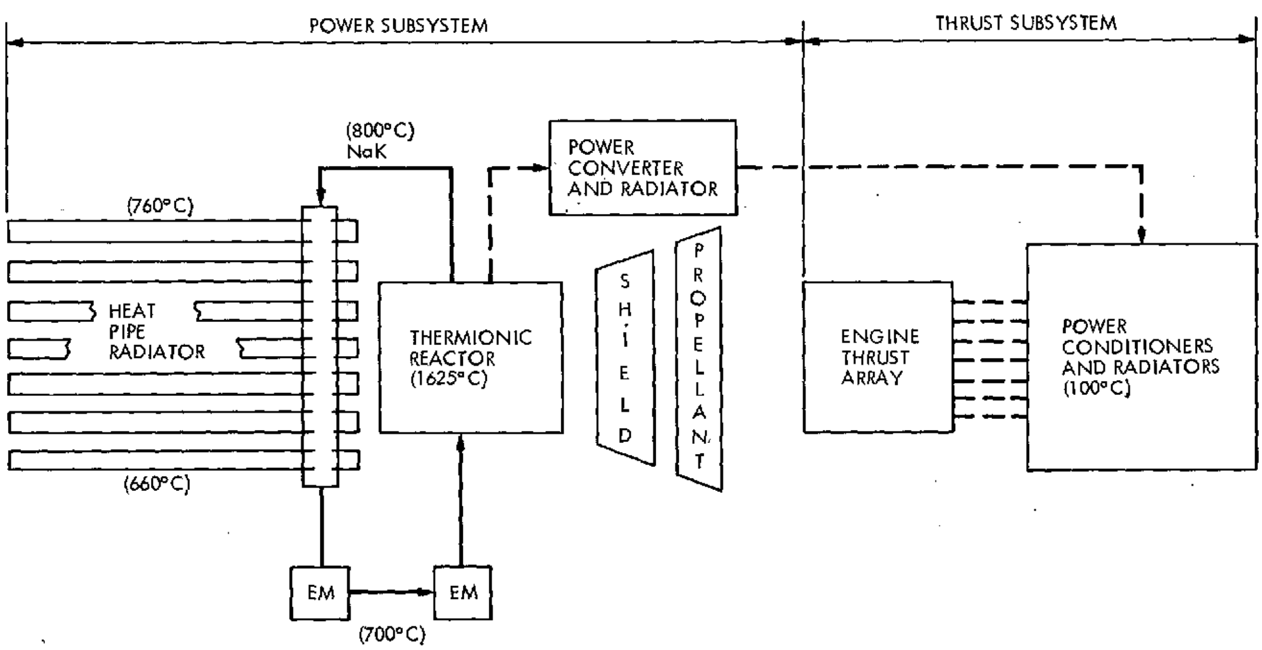

Fig. 4. Power and thrust subsystem temperature schematic

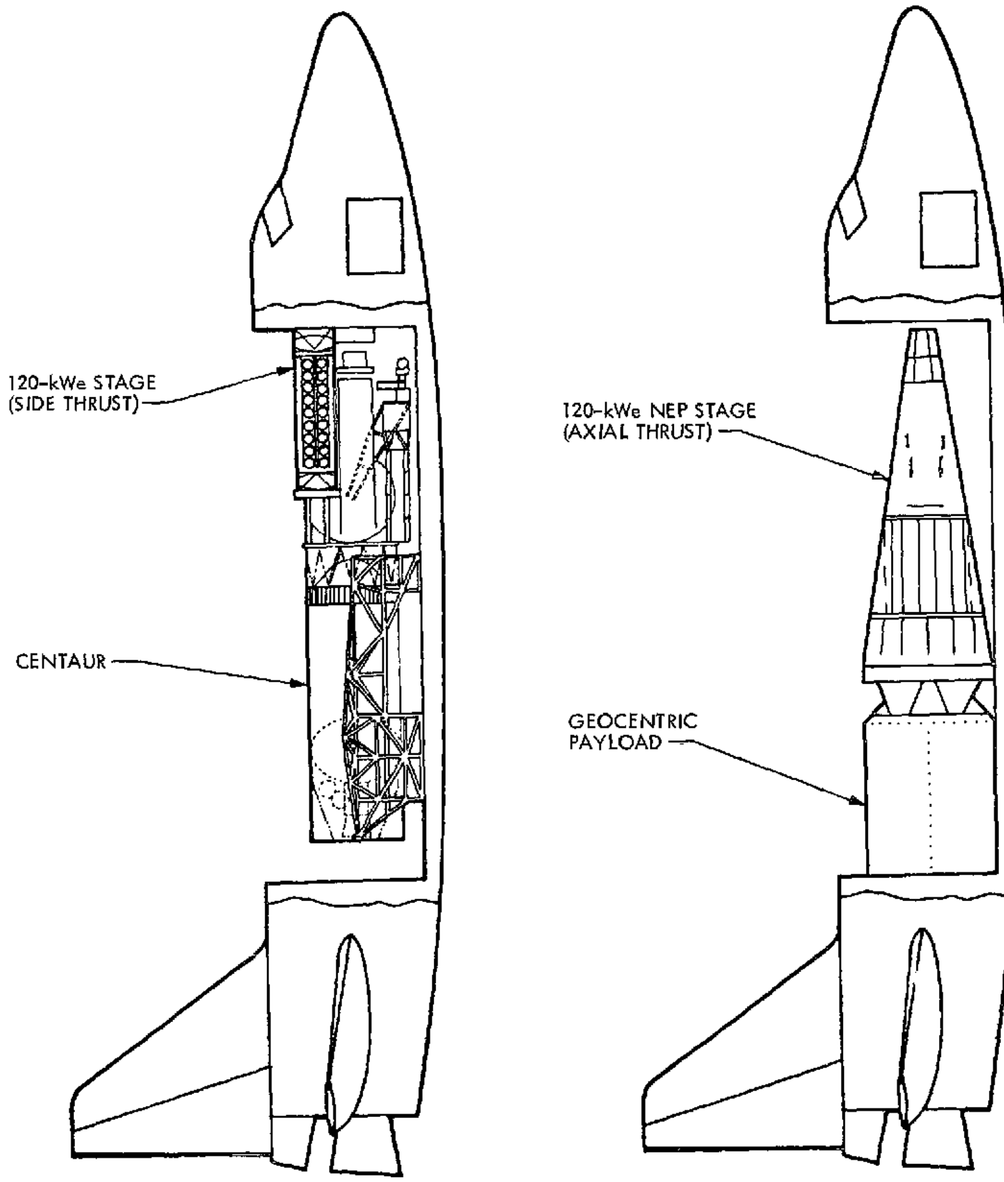

Fig. 5. NEP stage shuttle integration

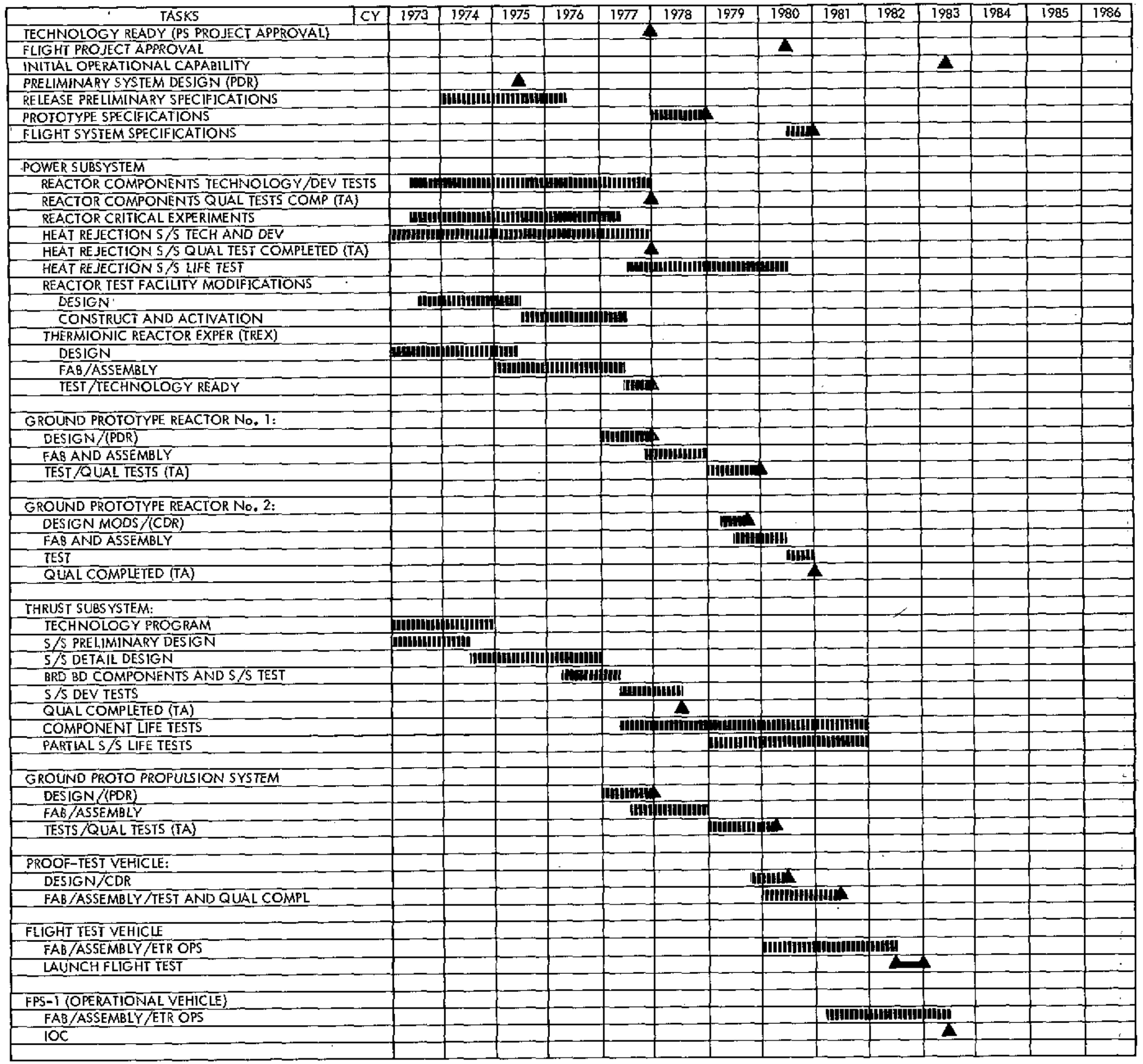

Fig. 6. Preliminary development schedule: 120-kW NEP system

APPENDIX A

REFERENCE MISSIONS

I. MISSION A, MARS ROVER, 1984

The science objectives for this mission are listed below:

(1) To measure the distribution of the surface chemical composition and, if possible, the surface isotopic composition.

(2) To determine, as functions of depth, the density, rigidity, and nature of the principal petrologic phases.

(3) To determine the present thermal regime within Mars and obtain information on past thermal conditions.

(4) To define the nature of major geologic provinces and larger surface features and the processes that formed them.

(5) If possible, to determine the dates of major geologic events.

(6) To observe short- and long-term variations in the atmosphere and obtain evidence of past atmospheric conditions.

(7) To determine present and past environmental conditions that may have affected the development of life.

(8) To determine if there is or ever has been life on Mars.

(9) If life has existed, what form(s) did it assume.

To meet these objectives the science package will include an orbiting science complement, an entry science package and a landed roving vehicle with a science payload (a Rover as a science experiment).

This mission considers the NEP unit as a "propulsion module" and requires that the spacecraft perform two-way information flow required for the operation of the "propulsion module." The propulsion module will supply 1 kW to the spacecraft during the mission and will not be jettisoned.

The NEP system can deliver a great deal more net mass in Mars orbit than that specified in the spacecraft definition. As a nominal mission, a trajectory with a heliocentric flight time of 250 days is selected.

The spacecraft definition defines the NEP system and net spacecraft as an integrated system. As such, the NEP system is also placed into Mars orbit and provides a convenient source of a great deal of electrical energy.

It is not necessary to provide a retropropulsion system for a Mars orbit insertion maneuver for the NEP option. In fact the size of such a system would be prohibitive for injection of the large mass (~6400 kg) of the NEP system plus the orbiter/lander payload. A more efficient method of achieving a Mars orbit is by a low-thrust spiral capture phase at Mars. With this method, arrival at Mars is essentially at parabolic speeds (VHP = 0), and the NEP system continually thrusts in such a direction as to achieve the desired Mars orbit. Approximate calculations can be made of the propulsion requirements for such a spiral capture maneuver although a detailed trajectory analysis has not been attempted.

The length of time required for such a maneuver is mainly a function of the thrust acceleration and the energy of the desired orbit and is approximately 15.6 days for the nominal NEP mission. This maneuver takes place over a number of revolutions about Mars, each succeeding orbit being nearer to the desired final orbit. It should be possible, using this spiral capture strategy, to achieve any desired orientation of the final orbit with little additional propellant consumption.

An ecliptic projection of the path of the spacecraft is shown in Fig. A-1. The spacecraft is launched due east from ETR using the Shuttle/Centaur launch vehicle combination. Following injection the NEP system is turned on and thrusts for a period of approximately 76 days. At this time the thrust is terminated and the spacecraft travels in a ballistic coast phase for another 86 days, after which the low-thrust engines are turned on again and the spacecraft thrusts for the remainder of the time to Mars.

The final 15.6 days of thrusting are for the capture spiral phase. After the desired orbit is achieved, the nuclear-electric propulsion system is turned off and orbiter and lander operations are started. Final orbit is achieved when Mars is near perihelion. The communications distance at this time is in excess of 200 million km and is increasing at a rate of approximately 1 million km per day. The mission profile, Table A-1, summarizes the trajectory and spacecraft parameters for the baseline mission.

The maximum performance realized by the NEP option is shown in Fig. A-2 as a function of heliocentric flight time. Note that the total mass in orbit includes the NEP system and is greater by approximately 3600 kg. The NEP system reaches its greatest performance for flight times in excess of 300 days and is capable of delivering approximately 5500 kg of net payload into the specified Mars orbit. The NEP system is inherently capable of delivering greater payloads than this but is limited by the Shuttle/Centaur injection capability.

II. MISSION B, HALLEY RENDEZVOUS, 1983

The Halley rendezvous mission provides an opportunity to observe a spectacular comet during this unique (to our generation) apparition. The Encke slow flyby 1979 mission will be the only flight with a chance of impacting the science design. The Halley rendezvous mission is so difficult a mission relative to the propulsion system performance, targeting, and temperature control that there will be little resemblance to the Encke slow flyby spacecraft. Halley is not only a fast comet but its retrograde motion about the sun complicates the relative velocity problem.

The primary science objective of the Halley rendezvous mission is to map the source of the visible emanations. This includes mass, dimensions, dynamics, surface characteristics, composition and age of the cometary nucleus. The composition, concentration, and velocity of the neutral gases, ions, and solid particles in the coma and tail are also to be mapped, especially in relation to their interaction with the solar wind. The "newness" and visible activity of the Halley comet make it of particular scientific interest.

The Halley rendezvous mission is so difficult that only the NEP could do the targeting job. The spacecraft must be ejected out to about 3.5 AU, where it makes a large plane change, reverses direction for retrograde motion, eventually rendezvousing with the comet back near the earth's orbit about 55 days before Halley perihelion. The spacecraft then follows for at least 100 days (see Fig. A-3). Observations are needed at least every other day, and at earth/comet conjunction, data must be stored to allow the continuous progression.

During rendezvous the spacecraft should circumnavigate the tail, halo, and the inner and outer coma, special care being taken to identify the interfaces between the comet and solar wind both before and after perihelion. If a nucleus exists it should be approached as closely as possible after perihelion. With two spacecraft in the vicinity, one of them should emphasize the outer or downstream portions of the emanations, the other the inner or upstream portions.

The spacecraft needed to support the Halley rendezvous science mission can be derived from the Mariner Jupiter/Saturn swingby spacecraft. The NEP module provides the power, but the spacecraft must double its timing and sequencing capability in order to control the low-thrust maneuvers. The temperature control must also be modified to allow for ~0.5 AU Halley perihelion. Otherwise data rates are adequate and the data processing capacity should be adequate for the baseline payload. There is sufficient capacity with NEP to deliver two spacecraft to the rendezvous condition. The second spacecraft would have to be somewhat simpler with limited exploration capacity and would therefore be ideal for attempting the first approach to the nucleus. Performance data is given in Table A-2.

III. MISSION C, URANUS/NEPTUNE FLYBY (WITH PROBE), 1986

The general mission objectives for each planet are to develop an understanding of the comparative properties of (a) each planet, (b) their satellites, and (c) the interplanetary medium as a function of distance from the sun.

The first objective, to develop an understanding of the comparative properties of Uranus and Neptune, can be achieved by measuring:

(1) Major atmospheric constituents (H, He, NH_3, CH_4) for Uranus and Neptune.

(2) General cloud structure and composition and nature of optical scattering, for Uranus and Neptune.

(3) Existence of CH_4 clouds, for Neptune.

(4) General circulation (dynamics) of the atmosphere, for Uranus.

(5) Radiation balance, compared to Jupiter and Saturn, for Uranus and Neptune.

(6) Scale heights and vertical temperature structure, for Uranus and Neptune.

(7) Size and therefore the density of Neptune.

The second objective, to develop an understanding of the satellites of Uranus and Neptune as compared to those of Jupiter and Saturn, can be attained by measuring:

(1) Mass, size, and density.

(2) Spin.

(3) Existence of atmosphere.

(4) Surface markings, structure, and composition.

The third objective, to develop an understanding of particles and fields associated with Uranus and Neptune as compared with those at Jupiter and Saturn, of the outer portions of the interplanetary medium, of its interaction with the interstellar medium, and of the nature of the interstellar medium, can be attained by measuring:

(1) Planetary ionospheres, magnetic fields, trapped radiation, thermal and nonthermal radio emission.

(2) X-ray emission from interaction of charged particles with atmospheres.

(3) Interaction of planetary fields with interplanetary/interstellar plasma, including removal of atmosphere and plasma waves.

(4) Cosmic dust in orbit around planet and in interplanetary and interstellar media.

(5) Satellite magnetic fields and magnetospheres and their interaction with the plasma.

(6) Properties of the solar wind as a function of distance from the sun, the transition from solar wind to interstellar medium, and the properties of the interstellar medium, including magnetic

field, density of neutral and charged particles, and cosmic rays as modulated by solar activity.

To meet these objectives, the science package will include both a flyby science complement and a probe (as a science experiment). The flyby science will put the following requirements on the mission:

(1) Flybys should be 2-5 planetary radii from the centers of Uranus and Neptune.

(2) Earth occultations by both Uranus and Neptune are needed.

(3) A close encounter (10, 000 km) of Triton is needed.

(4) A close encounter (10, 000 km) of Ariel, Titania, or Oberon is needed.

Additionally, it would be desirable to attempt a sun occultation by Uranus and an earth occultation by Ariel, Titania, or Oberon. The mission considers the NEP unit as a "propulsion module" and requires that the spacecraft perform the two-way information flow required for the operation of the "propulsion module." The propulsion module will supply 1 kWe to the spacecraft during the mission.

Table A-3 summarizes the trajectory and spacecraft characteristics for the baseline NEP Uranus/Neptune mission.

Maximum performance data for the Uranus/Neptune mission is shown in Fig. A-4. In this figure, maximized net spacecraft mass is shown as a function of flight time to Neptune for various values of thruster specific impulse. The net mass requirement of 1000 kg based on the other sections of this definition indicates a mission with a flight time to Neptune of 5 to 5-1/2 years. However, the science considerations indicate a minimum periapsis distance of two planet radii. As shown in Fig. A-4, this requirement dictates a minimum mission time of around 6.25 years.

The maximum net mass capability at this flight time is in excess of 2000 kg, using the 5000-sec specific impulse thrust system. The excess performance is best utilized by using a higher than optimum launch energy. The baseline mission selected corresponds to a nominal 6.70-year mission, with a Uranus periapsis distance of 2.5 planet radii.

The baseline mission selected for the NEP system has a nominal flight time to Uranus that ranges from 1518 to 1558 days. The resulting total mission time to Neptune is thus approximately 6.7 years. The spacecraft propulsion parameters of 120 kW thruster PCU input power and 5000 sec specific impulse dictate a maximum propellant loading of 1820 kg, corresponding to a maximum propulsion time under full power of 6720 hours.

Following low-thrust cutoff, the spacecraft continues on a ballistic path during the remainder of the mission. The propulsion system is not jettisoned but is available to provide trajectory corrections during the remainder of the mission. In addition, the NEP system is designed to provide attitude control during the coast phase of the mission. In this mode the propulsion system would be operated at a greatly reduced power. The performance degradation resulting from experiencing a small net thrust acceleration during what would normally be considered a coasting phase of flight is very small. Ideally, separate small thrusters operating on a short duty cycle which provided attitude control functions exclusively during coast phases may be preferred from the standpoint of fuel economy. Depending upon specific impulse and duty cycle, attitude control should be possible with less than 100 kg additional fuel expenditure.

IV. MISSION D, MARS SURFACE SAMPLE RETURN, 1990

The objectives of this mission are:

(1) To return to earth a selected sample of the Martian soil, preserving the scientific information in the sample to the greatest extent possible.

(2) To obtain additional information about the Mars surface and terrain.

A. SURFACE SAMPLES

The first objective is attained by collecting a minimum of three samples from the surface as follows:

(1) A sample primarily of biological interest consisting of the top 1 cm of soil. The minimum sample quantity of 100 g; however, 1 kg is desirable.

(2) A sample primarily of geological interest consisting of a soil core extending from the surface to a depth of 30 cm minimum, 60 cm desirable, 100 cm highly desirable. The minimum sample quantity is 100 g; however, 1 kg is desirable.

(3) A sample of rock chips. The minimum sample quantity is 100 g; however, 1 kg is desirable.

B. MARTIAN SCIENCE

The second objective is attained by:

(1) Imaging experiments performed by the roving vehicle and an orbiting spacecraft.

(2) Surface experiments which can be performed with the mechanical soil manipulators. The propulsion module will supply 1 kWe to the mission.

The propulsion module will not be jettisoned. There is no need for spacecraft propulsion modules.

A Shuttle/Tug-launched 120-kWe NEP system is more than capable of accomplishing the Mars surface sample return mission. Both the capture and escape phases at Mars in addition to the capture phase upon return to earth are accomplished with the NEP system using spiral phases of flight. The NEP system is not separated from the rest of the spacecraft during the mission except for the lander at Mars. Consequently, the NEP system returns to earth orbit at the end of the mission and could be available for subsequent use.

The mission starts with a launch from ETR at a nominal launch azimuth of 90 deg (due east). The electric propulsion thrusters are turned on soon after spacecraft injection and thrust for the first 100 days of the mission. A coast period of 198 days follows, after which the thrusters are turned on for the remainder of the earth-Mars heliocentric phase and Mars capture phase. Capture is into the same 1000-km altitude circular orbit as for the SEP option. Allowance has been made for a 3177-kg lander. The remaining mass in orbit includes the NEP system, an orbiter payload and various auxiliary systems. This mass, not including the NEP system, is 1543 kg. Of this amount, 500 represents contingency or mass which is jettisoned prior to Mars departure. A net mass of 1043 kg is returned to earth and includes 293 kg available for auxiliary support systems for the NEP and spacecraft systems or for contingency.

The departure spiral phase from Mars requires less time and a lower propellant requirement than for capture because of the increased thrust acceleration resulting from the lower spacecraft mass. A 300-day flight time is used for the heliocentric return trajectory to earth. The total return flight time including the escape and spiral capture phases is 373 dyas. During the heliocentric phase, a coast period of 193 days occurs during the middle of the Mars-earth transfer phase.

It is assumed that capture is into a circular orbit at synchronous altitude upon return to earth. Because of the high altitude of this orbit a relatively short time of 27.4 days is required for capture, involving only about 12 spiral turns. The resulting total mission time is 1056 days or slightly less than three years.

A detailed trajectory and spacecraft parameter breakdown is shown in Table A-4.

V. MISSION E, GANYMEDE/LANDER

In general, mission objectives are (a) to develop an understanding of Ganymede, the largest satellite in the solar system, as a step toward understanding the origin and history of satellite systems and (b) to obtain further information about Jupiter.

The first objective can be achieved by measuring the following properties of Ganymede:

(1) The diameter and shape.

(2) Position of the axis of rotation and the rotation rate.

(3) Overall density and the radial distribution of density (differentiation).

(4) Thermal regime and the structure of the interior.

(5) Large- and small-scale surface features, particularly those characteristic of internal activity, impact, and erosion.

(6) Nature and composition (mineralogical, elemental, isotopic) of the surface material, including the presence of ice (water and ammonia), and their variations across the surface.

(7) Presence and nature of any organic materials.

(8) Presence and nature of the atmosphere.

(9) Magnetic field and trapped radiation.

(10) Interaction of Ganymede with the Jupiter magnetosphere.

The second object, to obtain more information about Jupiter, can be accomplished by:

(1) Continued imaging observations of Jovian atmospheric phenomena.

(2) Magnetic-field and energetic-particle measurements at Ganymede for correlation with earth-based observations of Jovian radio-emission.

The mission will consist of two launches and will be launched by shuttle. The NEP propulsion module will supply 1 kWe to the spacecraft during the mission and will not be jettisoned.

The NEP system is ideally suited for a high-energy mission of this type. Employing the basic Shuttle/Centaur D1-T launch vehicle, the NEP spacecraft is capable of delivering more than adequate payload into a 100-km Ganymede orbit.

Launch is nominally at an azimuth of 90 deg from AFETR. The electric propulsion phase is initiated shortly after injection and continues for approximately 342 days. A coast phase of 590 days follows and the engines are then turned on and the thrust phase is continued for the remainder of the mission.

The NEP system is used for the capture phase by means of spiral trajectories around both Jupiter and Ganymede. In Table A-5 the electric propulsion phases have been divided into three parts: a heliocentric phase, a Jupiter spiral phase, and a Ganymede capture phase. There is not a clear demarcation between these propulsion phases; rather, they merge with one another with no discernible boundary. For the purposes of this study, however, the performance has been separated to conform with the trajectory analysis employed.

The capture spiral phase at Jupiter lasts approximately 126 days and involves approximately nine revolutions of Jupiter to arrive into an orbit with the same parameters as Ganymede. The Ganymede capture phase commences at the end of the Jupiter capture phase and takes approximately 29 days for capture into a 150-km-altitude circular orbit around Ganymede. The Ganymede capture phase requires nearly 100 revolutions of the satellite and suggests that meaningful science observations could be made during this time. The science pointing requirements during this capture phase are easy to meet since the spacecraft must thrust approximately in a circumferential direction during a majority of the capture phase.

The baseline mission has a maximum net orbiter/lander payload in orbit of 2538 kg. Of this amount 1815 kg is required for the orbiter/lander, while the remaining 723 kg can be considered as either propellant reserve or contingency. The NEP spacecraft also is capable of supplying virtually unlimited power to science and engineering subsystems when in Ganymede orbit.

An area of uncertainty for all NEP missions is the guidance and navigation requirements during a low-thrust spiral phase portion of the mission. Employment of NEP will almost certainly involve spiral phases to utilize the full performance capability of NEP.

The parameters most affecting performance are the thruster specific impulse and flight time. Figure A-5 presents data illustrating the variation of performance with these two variables. This data, supplied by Illinois Institute of Technology Research Institute, is conservative since a more exact trajectory simulation gives about 20% greater performance. Note that the flight time used in Fig. A-5 is the heliocentric flight time and does not include the time required for the spiral capture phases.

Even greater performance is possible by going to the Tug or NRP injection vehicles. However, the performance of the basic Shuttle/Centaur D1-T was deemed sufficient.

Table A-1. Mission profile: Mars Rover (1984)

Launch and injection parameters	
Launch vehicle	Shuttle/Centaur NEP (120)
Injection energy	$C_3 = 0.1\ km^2/s^2$
Declination of launch asymptote	-7.4 deg
Right ascension of launch asymptote	193.7 deg
Launch azimuth	90 ±20 deg
Launch period	Minimum of 30 days
Nominal launch date	2/14/84

Spacecraft parameters, NEP (120)	
Launch vehicle capability at $C_3 = 0.1\ km^2/s^2$	10500 kg
Injected spacecraft mass	7594 kg
Propulsion system mass (120 kW at 30 kg/kW)	3600 kg
Propulsion stage inert mass	N/A
NEP propellant consumed	1171 kg
NEP propellant reserve	117 kg
NEP propellant tankage mass	39 kg
Thruster specific impulse	5000 s
Thruster efficiency	$\eta = 0.75$
Maximum thruster PCU input/power	120 kW
Auxiliary spacecraft power	1 kW available
Nominal propulsion time	164 days
Anticipated maximum propulsion time	180 days
Orbit insertion using low-thrust spiral capture maneuver	
Coast phase from 76 days to 162 days during nominal trajectory	

Table A-1 (contd)

Orbiter parameters	
Total mass prior to spiral capture phase	6525 kg
Propellant consumed during spiral capture phase	101 kg
Duration of spiral capture phase	15.6 days
Total dry mass in orbit	6424 kg
Net (orbiter + lander) mass in orbit	2668 kg

Trajectory and arrival parameters	
Heliocentric transfer time	250 days
Total mission duration	266 days
Arrival date at end of spiral capture phase	11/5/84
Arrival energy (start of spiral capture phase)	0.0 km^2/s^2 (V_{HP} = 0.0 km/s)
Orbit parameters	
Periapsis	altitude 1500 km, R_p = 4893 km
Apoapsis	R_a = 35937 km
Period	24.6 hr
Minimum orbital inclination without plane change	7.4 deg

Table A-2. Halley's comet rendezvous performance table (Centaur/120-kW NEP, 5000 I_{sp})

Parameter launch date	5/83
Launch period, days	20
C_3, km^2/sec^2	9.4
Required ejected mass, kg	8400
Low-thrust propellant, kg	4300
Low-thrust duration, days	600
Transit time, years	2.6
Required mass at rendezvous,[a] kg	500
Deliverable mass at rendezvous,[a] kg	1200
Transit time maximum mass, years	2.6

[a]Without propulsion modules/stages.

Table A-3. Mission profile: Uranus/Neptune flyby (1986)

Launch and injection parameters	
Launch vehicle	Shuttle/Centaur/ NEP(120)
Injection energy	$C_3 = 21.2\ km^2s^2$ ($V_{HL} = 4.6$ km/s)
Declination of injection asymptote	Less than 30 deg
Launch azimuth	80-100 deg (estimated)
Launch period	Minimum of at least 40 days
Spacecraft parameters	
Dry mass	4655 kg
Propulsion system mass (α = 30 kg/kW)	3600 kg
Propellant mass	1820 kg
Propellant tankage mass (0.03 Mp)	55 kg
Initial injected mass	6475 kg
Launch vehicle capability at V_{HL}	4.6 km/s: 6475 kg
Thruster specific impulse	5000 sec
Thruster efficiency	$\eta = 0.75$
Auxiliary spacecraft power	$\Delta P = 1000$ W
Thruster PCU input power	120 kW
Propulsion stage not jettisoned after completion of low-thrust propulsion mode	
Maximum propulsion time	280 days (6720 hr)
Trajectory and arrival parameters	
Total mission duration (Earth-Uranus-Neptune)	6.7 years
Launch date	11/27/85-1/6/86
Arrival date at Uranus	3/4/90
Arrival excess velocity at Uranus	$V_{HP} = 20.3$ km/s
Flight time, Earth-Uranus	1518-1558 days

Table A-3 (contd)

Trajectory and arrival parameters	
Uranus periapsis distance (from center)	2.5 R_u (planet radii)
Arrival date at Neptune	8/23/92
Flight time, Uranus-Neptune	903 days
Ecliptic departure right ascension at Uranus	289.6 deg
Ecliptic departure declination at Uranus	2.83 deg
Arrival excess velocity at Neptune	V_{HP} = 21.2 km/sec

Table A-4. Uranus/Neptune (1860) mission profile: Shuttle/Tug/NEP (120)

Launch and injection parameters	
Launch vehicle	Shuttle/Tug/NEP (120)
Injection energy	$C_3 = 1\ km^2/s^2$
Declination of launch asymptote	8.1 deg
Launch azimuth	90 ±20 deg
Launch period	Minimum of 30 days
Nominal launch date	7/20/90
Trajectory and arrival parameters, Earth-Mars	
Nominal transfer time	350 days
Nominal arrival date at Mars	7/5/91
Arrival excess velocity	$V_{HP} = 0$ km/s
Spiral capture into 1000-km circular orbit at Mars	
Length of Mars capture spiral phase	86.4 days
Approximate number of turns	242
Date of end of spiral capture	9/29/91
Orbital period	147.4 min
Communications distance at start of spiral phase	343×10^6 km
Departure parameters from Mars	
Nominal departure date	6/2/92
Total wait time in Mars orbit	246 days
Spiral departure to C_3	$0\ km^2/s^2$
Length of spiral departure phase	45.7 days
Approximate number of spiral turns	144
Nominal start of heliocentric phase	7/18/92
Mars-earth Heliocentric transfer time	300 days
Communications distance at start of heliocentric phase	235×10^6 km

Table A-4 (contd)

Arrival parameters at earth	
Nominal heliocentric arrival date	5/14/93
Spiral capture into 24-hr synchronous orbit at earth	
Length of spiral capture phase	27.4 days
Approximate number of spiral turns	12
Date at end of capture phase	6/10/93
Total mission time	1056 days
Propulsion and power parameters	
Thruster input power	120 kW
Auxiliary power	1 kW available
Propulsion system specific mass	30 kg/kW
Specific impulse	5000 s
Thruster efficiency	0.754
Earth-Mars heliocentric propulsion time	152.0 days
Mars capture spiral propulsion time	86.4 days
Mars departure spiral propulsion time	45.7 days
Mars-earth heliocentric propulsion time	106.5 days
Earth capture spiral propulsion time	27.4 days
Total NEP propulsion time	417.9 days (10,000 hr)
Spacecraft masses, kg	
Initial injected mass	11400
NEP propellant mass, earth-Mars	989
NEP propellant, Mars capture spiral	562
Total initial mass in orbit	9850
Separated lander mass	3177
Mass in orbit less lander	6673

Table A-4 (contd)

Spacecraft masses, kg	
Net orbiter mass	1000
Contingency or jettisoned orbiter mass	500
Spacecraft mass at Mars departure	6173
Propellant mass, departure spiral	297
Propellant mass, Mars-earth transfer	107
Propellant mass, earth capture spiral	178
Total propellant consumed	2717
Propellant contingency	273
Total propellant loaded	2990
Propellant tankage mass	90
Mass in earth orbit	5006
Propulsion system mass	3600
Net payload mass returned	750
Payload contingency	293

Table A-5. Mission profile: Ganymede-Orbiter/Lander (1990)

Launch and injection parameters	
Launch vehicle	Shuttle/Centaur D1-T
Injection energy	0.0 km^2/s^2
Declination of launch asymptote	-0.3 deg
Right ascension of launch asymptote	4.7 deg
Launch azimuth	90 ±20 deg
Launch period	Minimum of 30 days
Nominal launch date	6/14/90

Spacecraft parameters	
Launch vehicle injection capability	10500 kg
Injected spacecraft mass	10487
Propulsion system mass (120 kW at 30 kg/kW)	3600
NEP propellant mass, heliocentric	3190
NEP propellant mass, Jupiter spiral	820
NEP propellant mass, Ganymede spiral	188
NEP propellant reserve[a]	723
Total loaded propellant including reserve	4921
Propellant tankage mass	151
Net mass in Ganymede orbit	1815
Total propulsion time	645 days (15500 hr)
Thruster specific impulse	5000 s
Thruster efficiency	0.754

[a]This propellant reserve represents a performance contingency that can be traded one to one with Orbiter/Lander mass. The maximum net mass in orbit is thus 2538 kg.

Table A-5 (contd)

Trajectory parameters	
Nominal flight time	1000 days
Nominal launch date	6/14/90
Nominal arrival date	3/10/93
Coast phase	490 days
Arrival excess velocity	$V_{HP} = 0.0$ km/s
Communication distance at arrival	677×10^6 km
Jupiter spiral parameters	
Mass at start of Jupiter spiral phase	7172 kg
Length of spiral phase at Jupiter	126 days
Approximate spiral turns to capture	9
Propellant consumed during spiral phase	820 kg
Ganymede spiral parameters	
Mass at start of Ganymede spiral phase	6352 kg
Length of spiral phase at capture	29 days
Approximate spiral turns to capture	99
Propellant consumed during spiral phase	188 kg
Gross mass in 100-km orbit at Ganymede	6164 kg
Net Orbiter/Lander mass	1815 kg
Total heliocentric/spiral flight time	1155 days

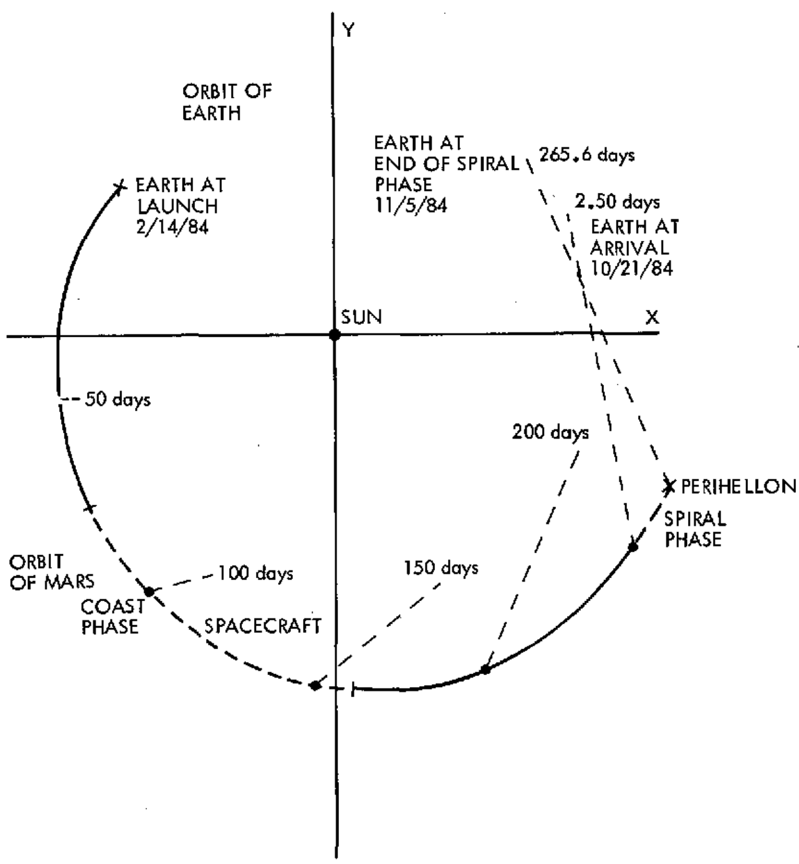

Figure A-1. Ecliptic projection: Mars Orbiter/Rover Mission

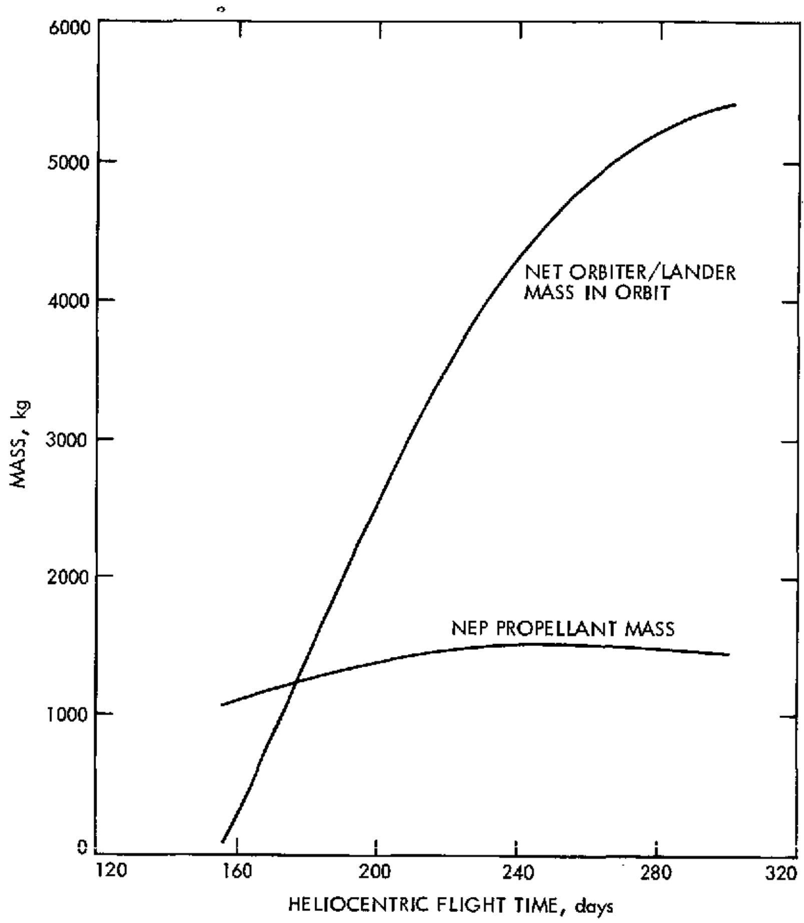

Figure A-2. Mars 1984 Orbiter/Rover maximum performance: Shuttle/Centaur/NEP

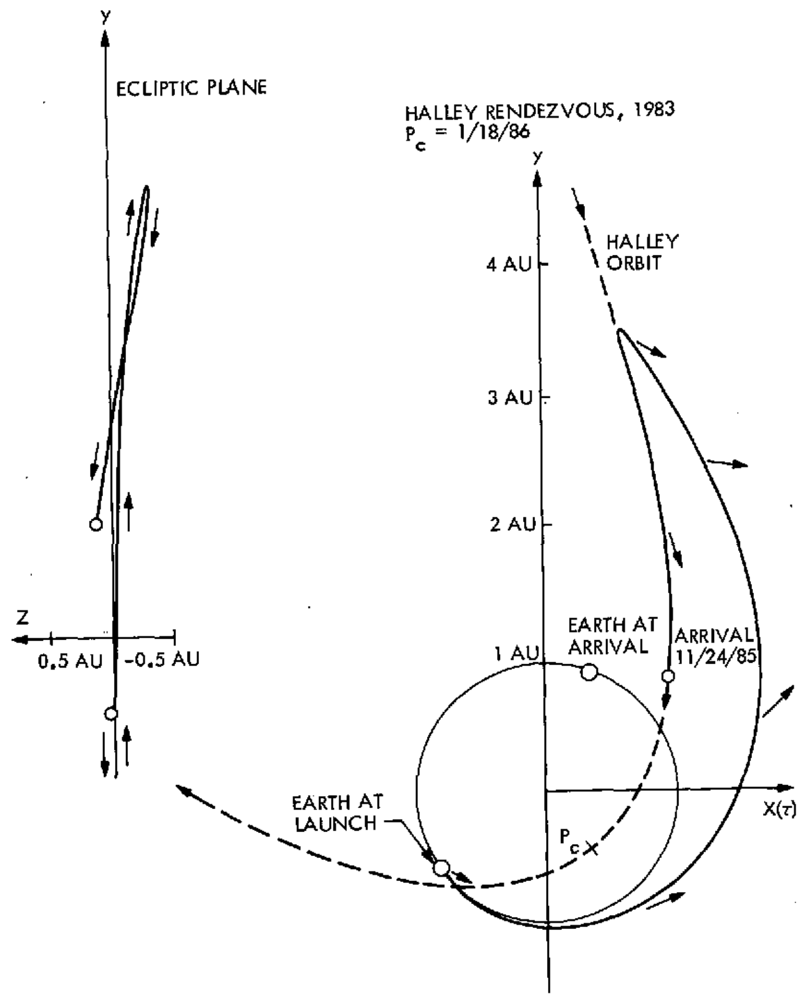

Figure A-3. Halley rendezvous, 1983

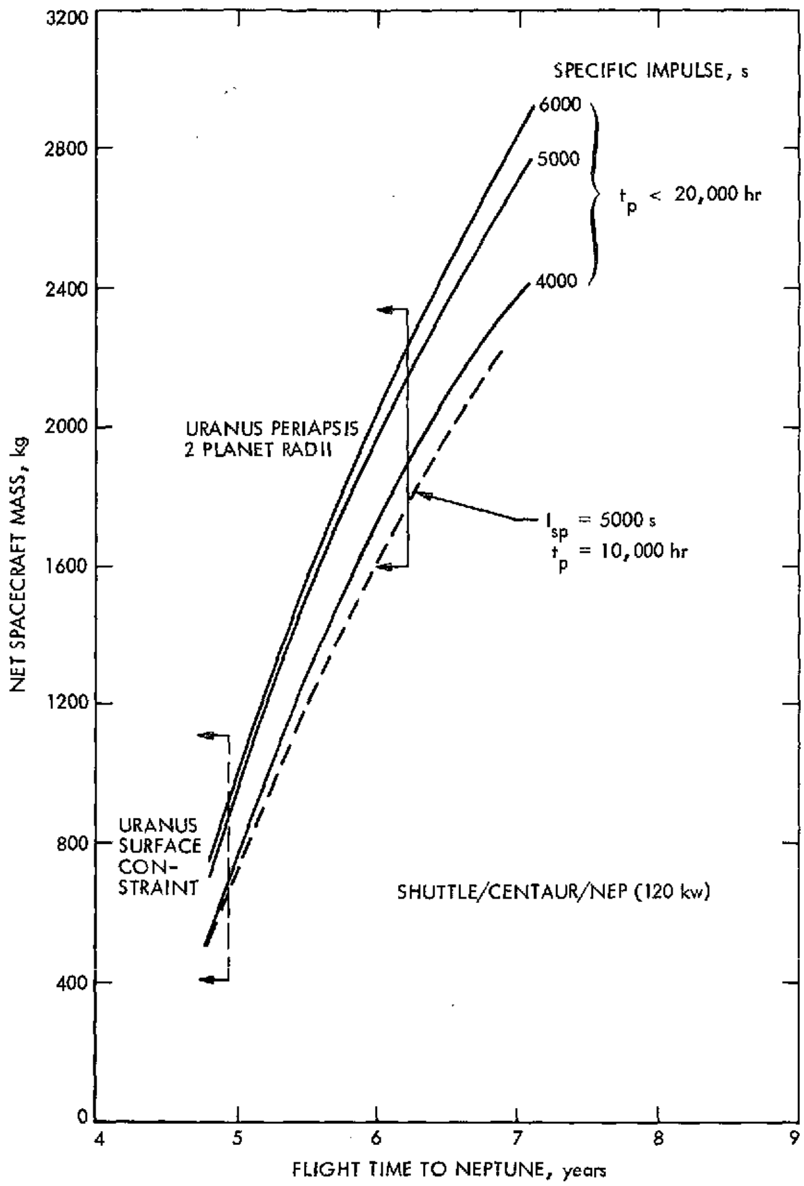

Figure A-4. Nuclear electric performance for Uranus/Neptune Mission, 1985 - 1986 launch

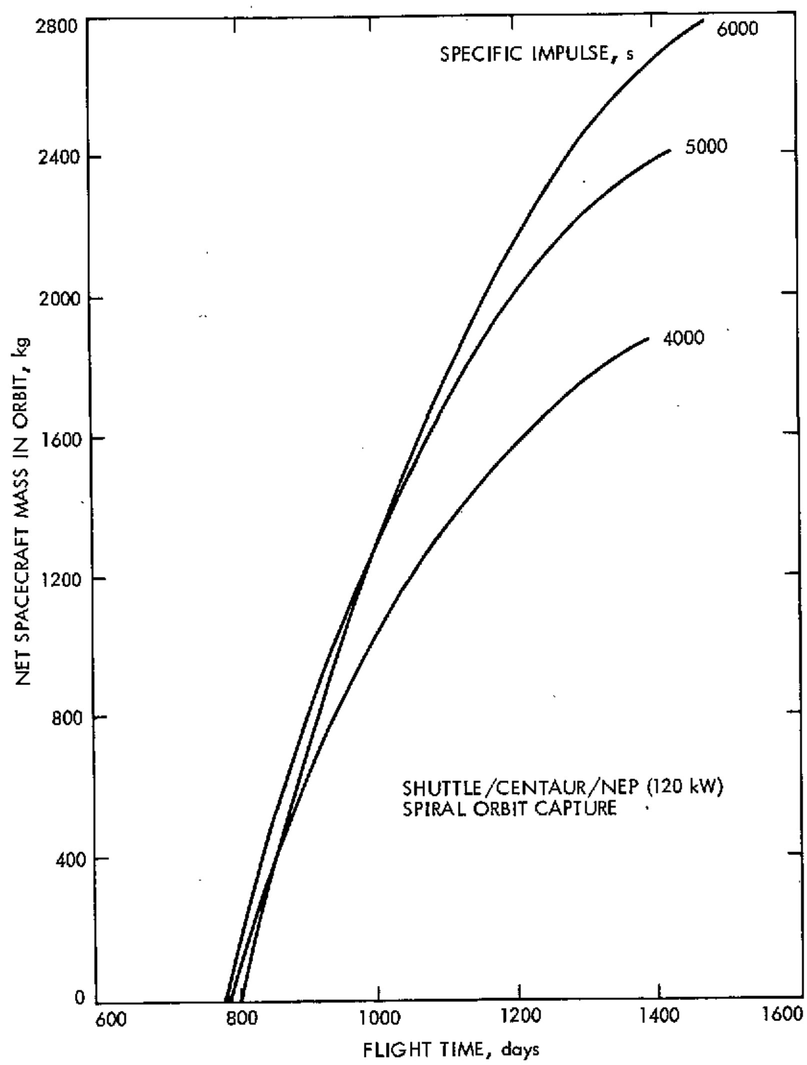

Figure A-5. Nuclear electric performance for Ganymede Orbiter/Lander Mission, 1990 launch

APPENDIX B

INTERPLANETARY METEOROID MODELS[1]

I. INTRODUCTION

The nominal meteoroid environment for interplanetary space and near a planet should be determined from the detailed models presented in the following sections. Application of the models requires numerical integration along the spacecraft trajectory.

II. COMETARY METEOROIDS

A. SPATIAL DENSITY AND FLUX MODEL

The spatial density of cometary meteoroids is expressed mathematically as follows:

$$\log_{10} S_c = -18.173 - 1.213 \log_{10} m - 1.5 \log_{10} S - 0.869 \sin \beta$$

for $10^{-6} \leq m \leq 10^{2}$

$$\log_{10} S_c = - 18.142 - 1.584 \log_{10} m - 0.063 (\log_{10} m)^2$$

$$- 1.5 \log_{10} S - 0.869 \sin \beta$$

for $10^{-12} \leq m \leq 10^{-6}$

Spatial density is related to flux on a randomly tumbling surface by

$$F_c = \frac{1}{4} S_c \bar{V}$$

---

[1]MJS natural space environmental estimates, JPL.

where

S_c = number density of cometary meteoroids of mass m or greater (m^{-3})

m = mass of the meteoroid (g)

F_c = number of cometary meteoroids of mass m or greater ($m^{-2}\ s^{-1}$)

S = distance from the sun in astronomical units

$\overline{V}$ = relative velocity of meteoroids to the spacecraft (m/s)

β = heliocentric latitude

B. MASS DENSITY

The mass density is assumed to be 0.5 g/cm^3 for all cometary meteoroids.

C. RELATIVE VELOCITY

The average relative velocity of cometary meteoroids to a spacecraft is expressed as follows:

$$\overline{V}(S, \sigma, \theta) = S^{-1/2}\ \overline{U}_c(\sigma, \theta)$$

where

$\overline{U}_c$ = cometary velocity parameter, a function of σ and θ, (10^3 m $AU^{1/2}/s$)

σ = ratio of the spacecraft's heliocentric speed at S to the speed required for a circular orbit of radius S

θ = angle between the spacecraft velocity vector and the surface of an imaginary sphere of radius S (deg)

Values for $\overline{U}_c$ can be found from the expression

$$U_c = 31.29 \times 10^3\ (1.30 - 1.9235\ \sigma \cos\theta + \sigma^2)^{1/2}$$

D. DIRECTIONALITY

For most types of spacecraft transfer orbits, cometary meteoroids may be considered omnidirectional; thus, the flux on any part of an oriented spacecraft is considered to be the same as the flux on the surfaces of a randomly tumbling spacecraft.

III. ASTEROIDAL METEOROIDS

A. SPATIAL DENSITY AND FLUX MODEL

The spatial density of asteroidal meteoroids is expressed mathematically as follows:

$$\log_{10} S_a = -15.79 - 0.84 \log_{10} m + f(S) + g(S) \cos \lambda + h(\beta)$$

for $10^{-9} \leq m \leq 10^{2}$

$$\log_{10} S_a = -8.23 + f(S) + g(S) \cos \lambda + h(\beta)$$

for $10^{-12} < m < 10^{-9}$

Spatial density is related to flux on a randomly tumbling surface by

$$F_a = \frac{1}{4} S_a \overline{V}$$

where

S_a = number density of asteroidal meteoroids of mass m or greater (m^{-3})

m = mass of the meteoroid (g)

F_a = number of asteroidal meteoroids of mass m or greater ($m^{-2}\, s^{-1}$)

$f(S)$ = $\log_{10}[S_a/S_a(S = 2.5)]$ (values are given in Fig. B-1)

$g(S)$ = coefficient of $\cos \lambda$ (values are given in Fig. B-2)

λ = heliocentric longitude of the spacecraft

$h(\beta) = \log_{10}[S_a(\beta)/S_a(\beta = 0)]$ (values are given in Fig. B-3)

β = heliocentric latitude

B. MASS DENSITY

The mass density is assumed to be 3.5 g/cm^3 for asteroidal meteoroids.

C. RELATIVE VELOCITY

The average relative velocity of asteroidal meteoroids is

$$\overline{V}(S, \sigma, \theta) = S^{-1/2}\, \overline{U}_a(S, \sigma, \theta)$$

where

$\overline{U}_a$ = asteroidal velocity parameter, a function of σ and θ (10^3 m AU$^{1/2}$/s)

(For distances from the Sun other than those given $\overline{U}_a$ is found by linear interpolation or extrapolation.) Values for $\overline{U}_a$ can be found from the following expressions:

At $S = 1.7$ AU,

$$\overline{U}_a = 30.05 \times 10^3 (1.2292 - 2.1334\, \sigma \cos\theta + \sigma^2)^{1/2}$$

At $S = 2.5$ AU,

$$\overline{U}_a = 29.84 \times 10^3 (1.0391 - 1.9887\, \sigma \cos\theta + \sigma^2)^{1/2}$$

At $S = 4.0$ AU,

$$\overline{U}_a = 29.93 \times 10^3 (0.9593 - 1.923\, \sigma \cos\theta + \sigma^2)^{1/2}$$

D. DIRECTIONALITY

Since asteroids are in near-circular orbits around the Sun, they may be considered nearly monodirectional relative to the spacecraft for most types of spacecraft orbits. The approximate direction of the asteroids relative to the spacecraft can be predicted by the vector relationship between the spacecraft velocity vector and a circular orbit velocity vector.

A surface whose normal vector is pointing in the direction of a truly monodirectional flux will encounter exactly 4 times the flux of a randomly tumbling surface. If it is pointed 60 deg from this direction, the surface will encounter exactly the same flux as the randomly tumbling surface. Orbits of the asteroids are sufficiently noncircular to reduce the flux in the direction of maximum flux to about 3 times the flux on a randomly tumbling surface; the flux 90 deg away from this direction will be approximately the same as on the randomly tumbling surface.

III. MODIFICATION IN ENVIRONMENT NEAR A PLANET

The cometary environment at various distances from the Sun is described elsewhere in this appendix. The decrease in meteoroid flux which results from planetary shielding is obtained by the factor

$$\eta = \frac{1}{2} + \frac{1}{2} (1 - R_{eq}^2/R^2)^{1/2}$$

and the increase in meteoroid flux which results from the gravitational effect of a planet is obtained by the factor

$$G = 1 + 0.76 \frac{R_p V_p^2 R_{eq}}{V_e^2 R}$$

where

R_{eq} = planet radius

R = distance of spacecraft from the center of the planet

G = gravitational increase in flux factor

R_p = distance from the sun in AU

V_p = escape velocity from the planet surface

V_e = escape velocity from earth

Escape velocities for the planets are given elsewhere in this document in the appropriate sections describing planetary gravitational parameters. For the large planets, the planet's escape velocity must be used as the average meteoroid velocity relative to the planet. The average velocity relative to a spacecraft near the large planets is found by

$$\overline{V} = \left(\frac{R_{eq} V_p^2}{R} + V_s^2\right)^{1/2}$$

Within the volume centered on Jupiter of radius 500 R_{eq} and at latitudes within ±30 deg, asteroidal-like meteoroids of mass density $3.5 \times 2^{\pm 1}$ g/cm^3 and relative velocity $v = \left[v_s^2 + (R_{eq}/R)\ (40\ \text{km/sec})^2\right]^{1/2}$ have a concentration of particles of mass greater than m given by

$$\log N_m = -31 \pm 1 - (0.8 \pm 0.1)(\log m - 18.5)$$

where v_s is the spacecraft speed, N_m is in m^{-3}, and m is in grams ranging from 10^{-9} to 10^{20}. The corresponding isotropic meteoroid flux is $N_m v/4$. The observed small satellites of Jupiter form the large-mass end of the distribution.

IV. UNCERTAINTIES IN THE MODELS

Because data is scant for meteoroids in most of the mass range of interest in the asteroid belt and data interpretation unsure, exact uncertainties cannot be established. Models that are believed to have reasonable upper and lower limits are presented. The assumption that mass velocity and radial distributions are independent could lead to variation from actual conditions. The effect of possible clustering of meteoroids can be ignored in engineering design because clustering would statistically reduce the overall hazard to space vehicle missions.

A. COMETARY METEOROIDS

A factor of approximately 7 in uncertainty exists for the cometary flux near earth. The free-space cometary environment at 1 AU is not much greater than the uncertainty in the near-earth environment.

Various analyses lead to variations of particle population with distance from the Sun between S^0 and S^{-3}. If $S^{-1.5}$ is adopted as the nominal value and S^0 and S^{-3} are taken to be the extremes of possible radial distribution, an uncertainty factor of 4 results for the region between 0.4 and 2.5 AU. Combination of the uncertainty factor of 4 for radial distribution and the uncertainty factor of 7 for the cometary flux near earth (derived from the mass uncertainty) gives an extreme uncertainty in cometary flux of 28 at 0.4 and 2.5 AU. This uncertainty lessens as the region of interest approaches 1 AU.

B. ASTEROIDAL METEOROIDS

The ratio of the upper limit to the nominal asteroidal flux is 26 $m^{0.17}$ ($m \geq 10^{-9}$ gram). The lower limit is set by the cometary environment.

C. METEOROID VELOCITY

The average relative meteoroid velocity is uncertain by about ±25% for a spacecraft in near-circular orbit around the Sun and only a few percent for spacecraft with very elliptical transfer orbits.

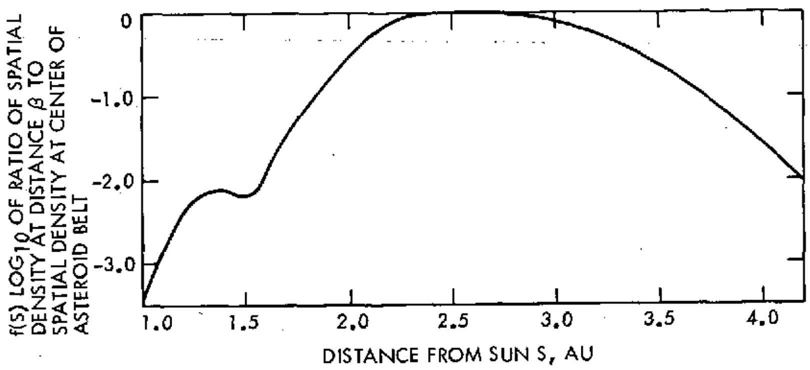

Figure B-1. Asteroid radial distribution

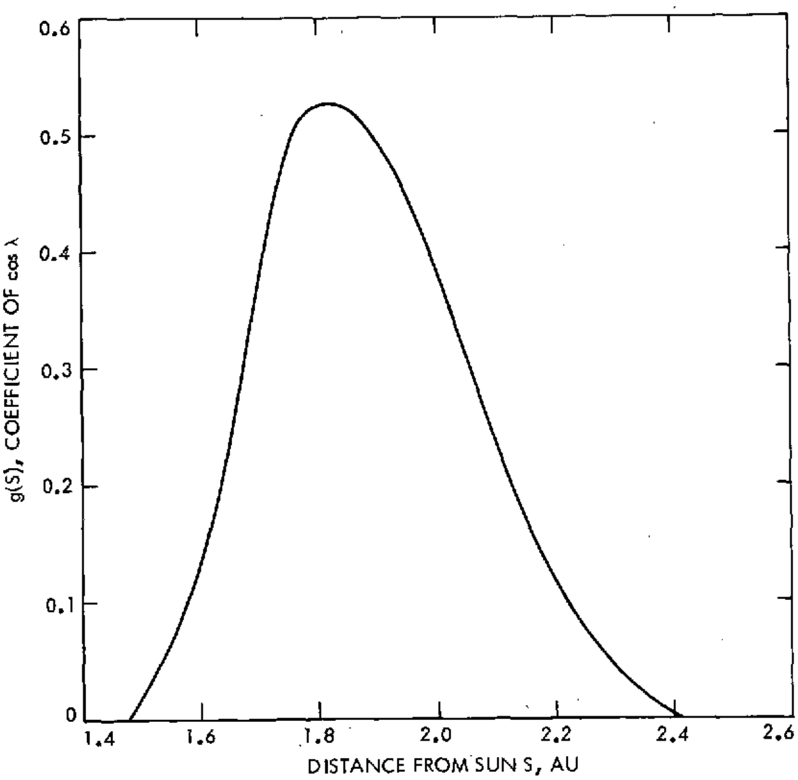

Figure B-2. Asymmetry of the asteroid belt with heliocentric longitude

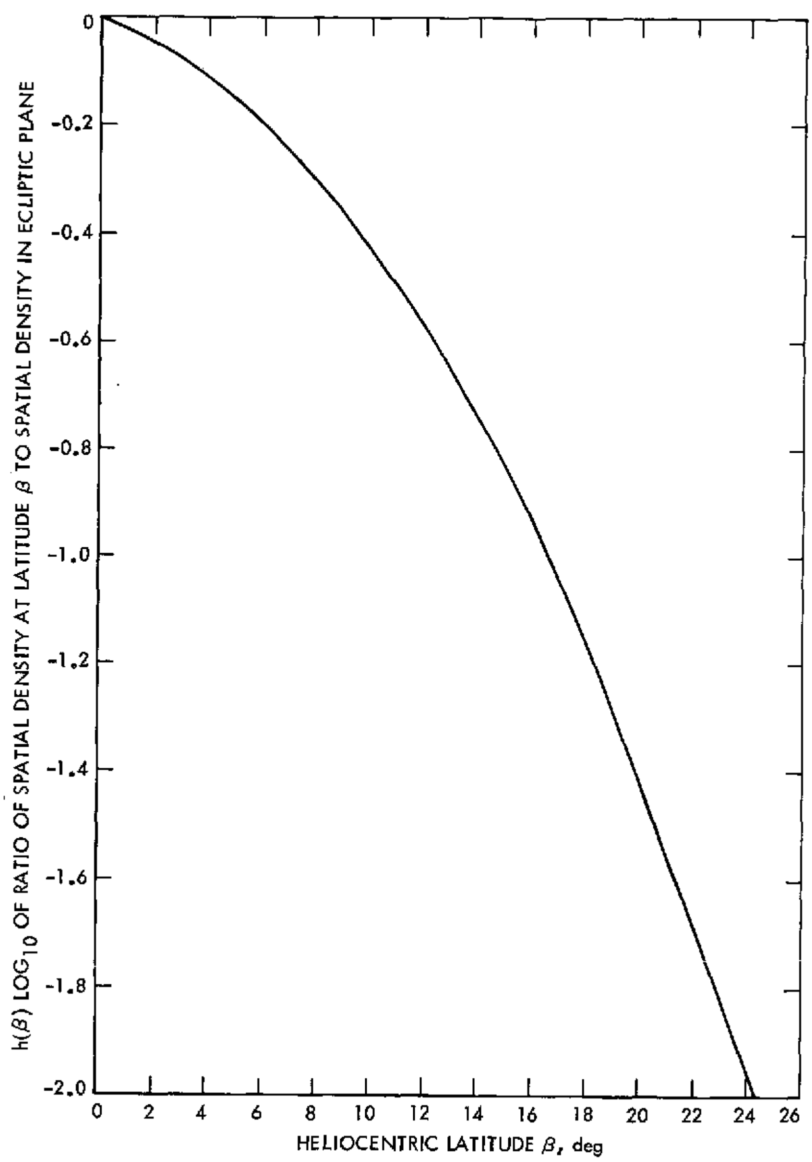

Figure B-3. Latitudinal distribution of asteroidal particles

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