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SPACECRAFT AND MISSION DESIGN FOR THE SP-100 FLIGHT EXPERIMENT

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Abstract

The design and performance of a spacecraft employing arcjet nuclear electric propulsion, suitable for use in the SP-100 Space Reactor Power System (SRPS) Flight Experiment, are outlined. The vehicle design is based on a 93 kW_e ammonia arcjet system operating at an experimentally-measured specific impulse of 1031 s and an efficiency of 42.3 percent. The arcjet/gimbal assemblies, power conditioning subsystem, propellant feed system, propulsion system thermal control, spacecraft diagnostic instrumentation, and the telemetry requirements are described. A 100 kW_e SRPS is assumed. The spacecraft mass is baselined at 5675 kg excluding the propellant and propellant feed system. Four mission scenarios are described which are capable of demonstrating the full capability of the SRPS. The missions considered include spacecraft deployment to possible surveillance platform orbits, a spacecraft storage mission and an orbit raising round trip corresponding to possible OTV missions.

NOMENCLATURE

ELV	Expendable Launch Vehicle
EMI	Electromagnetic Interference
ESD	Electrostatic Discharge
HEO	High Earth Orbit
I_{sp}	Specific Impulse, s
JPL	Jet Propulsion Laboratory
KSC	Kennedy Space Center
$M_{f/s}$	Mass of Propellant Feed System
M_p	Propellant Mass
NEP	Nuclear Electric Propulsion
NH_3	Ammonia
NSO	Nuclear Safe Orbit; 28.5° inclination, 925 km altitude
OTV	Orbit Transfer Vehicle
PGM	Power Generation Module
PLF	Payload Faring
PPU	Power Processing Unit
QCM	Quartz Crystal Microbalance
SDI	Strategic Defense Initiative
SOA	State of the Art
SP-100	Space Power at 100 kW_e
SRM	Solid Rocket Motor
SRPS	Space Reactor Power Source
STS	Shuttle Transportation System
UIM	User Interface Module
ΔV	Velocity Increment

Units

A	Amperes
cm	Centimeters
g	Grams
kg	Kilograms
km	Kilometers
kW_e	Kilowatts of Electrical Power
m	Meters
ms	Milliseconds
N	Newtons
nmi	Nautical Miles
Pa	Pascals
psi	Pounds per square inch
s	Seconds
V	Volts
W	Watts of thermal power

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INTRODUCTION

Exploration and intensive study of the planets of our solar system will require high-power, electrically-propelled spacecraft.[1-5] In addition, high-power, lightweight propulsion systems will be needed to transfer high mass payloads from low earth orbit to their operational orbits.[6-10] Nuclear Electric Propulsion (NEP) systems utilizing Space Reactor Power Systems (SRPS) and electric propulsion modules are being studied as options to satisfy these mission needs. Numerous mission studies have been conducted in which NEP was identified as either mission enabling or as the optimal propulsion choice.[1-11] Several studies also considered the integration of power and electric propulsion subsystems into an NEP spacecraft.[1,10,12-17]

The future availability of viable NEP systems requires the simultaneous development of an SRPS and electric propulsion systems. The projected needs of the Strategic Defense Initiative (SDI) indicate unprecedented power level requirements (hundreds of kilowatts to hundreds of megawatts) and an order of magnitude increase in power density to 1.0 kW_e/kg. A program in space power and power conversion has been initiated for the development of the critical technologies required to meet these power needs.[18] The four program elements are: requirements and assessment, multimegawatt prime power, pulsed power conditioning and baseload power. The last element, baseload power, consists of SP-100 and alternative non-nuclear technologies. The nuclear technology assessment phase of the SP-100 program has been completed with selection of an SRPS concept which includes a fast-spectrum, liquid-metal cooled reactor coupled with an out-of-core thermoelectric conversion system.[19] The primary objective of Phase II, which has been initiated, is the 1991 ground test of a 100 kW_e SRPS based on the selected system concept.

The SP-100 Flight Experiment, a flight demonstration of a 100-kW_e class SRPS, has been proposed as an adjunct to the SP-100 program using an electric propulsion module as an active load.[20] The primary purpose of this proposed flight test is the demonstration of space-based nuclear power system operation. The SP-100 Flight Experiment will also demonstrate nuclear electric propulsion for orbit raising and maneuvering.

The Flight Experiment test goal is to operate the SP-100 SRPS for its seven year, full power life. An active power system load is required for up to six months to verify power system compatibility with a payload and satisfy potential users of this compatibility.[18,20] No alternative to electric propulsion has been identified for the active load which meets the Flight Experiment constraints as presently defined. The constraints include a low developmental risk and cost, wide performance throttleability, and scaleability to future SDI power levels well beyond the 100 kW_e range being considered for the flight demonstration. This mission will provide a unique opportunity to examine the control scenarios required for NEP orbit transfer, to examine the maneuvering of an orbiting spacecraft to enhance operations and survivability, and to examine a representative transfer similar to that required for the SDI. Arcjet electric propulsion has been selected as the baseline electric propulsion system for the SP-100 Flight Experiment.[20]

This paper outlines a baseline arcjet NEP spacecraft design for use in the SP-100 Flight Experiment. Detailed descriptions of the arcjet/gimbal assemblies, Power Conditioning Unit (PCU) subsystem, propellant flow subsystem, thermal control subsystem, diagnostics package and telemetry requirements are included. Expected propulsion system performance is described for two experimentally determined arcjet technology levels and two SRPS power levels (30 kW_e and 100 kW_e) with launches from the Kennedy Space Center (KSC) using the Shuttle Transportation System (STS) and the Titan IV expendable launch vehicle (ELV). The missions considered include spacecraft deployment to possible SDI platform orbits, a spacecraft storage mission and an orbit raising round trip. This paper builds on four previous papers[10,15-17] and is aimed at better defining the SP-100 Flight Experiment NEP opportunity by using recently measured values of arcjet performance and providing a more detailed analysis of the spacecraft mission design, options and performance.

SP-100 FLIGHT EXPERIMENT SPACECRAFT CONFIGURATION

A proposed spacecraft configuration for the SP-100 Flight Experiment is shown in Fig. 1. This system is comprised of a 100-kW_e SP-100 SRPS, spacecraft bus, an arcjet propulsion module, and an SRPS radiation/arcjet plume diagnostics package. A 100 kW_e power level was chosen since it is the recommended power for the SRPS flight demonstration.[10,18,20] This spacecraft concept utilizes an end thrust design, through the spacecraft centerline, so that the deployment boom is in compression during thrusting. The SP-100 SRPS consists of the Power Generation Module (PGM) and the User Interface Module (UIM). The PGM consists of the reactor, shield, auxiliary cooling loop, thermoelectric electromagnetic (TEM) pumps, power converters, multiplexers and the heat rejection radiator. The UIM is composed of the separation boom, shunt dissipator and the user interface equipment module. The SRPS will be considered in this paper only to the extent of general performance specifications and major SRPS/payload interactions. The SP-100 SRPS parameters germane to this study are listed in Table 1.[19,21,22] A power system specific mass of 30 kg/kW_e is used in this study since it is the official SP-100 program goal.[21,22]

TABLE 1
Space Reactor Power System Performance Specifications[19,21]

Parameter	Specification
Power Level	100 kW_e
Primary Voltage	200 V_{dc}
Specific Mass	30 kg/kW_e
Secondary Power	300 W
Secondary Voltage	28 V_{dc}
Continuous Load Following	0.1 kW_e/ms
Thermal Flux at User Interface	0.14 W/cm^2
10 Year Radiation Fluence	$< 10^{13}$ $neutrons/cm^2$
at User Interface	$< 5 \times 10^5$ Rads

The arcjet propulsion module is comprised of: three (3) sets of four (4) engines with each set of engines on a single gimballed platform, a PCU system, the propellant feed system, thermal control, a radiation/thruster efflux diagnostics package and associ-

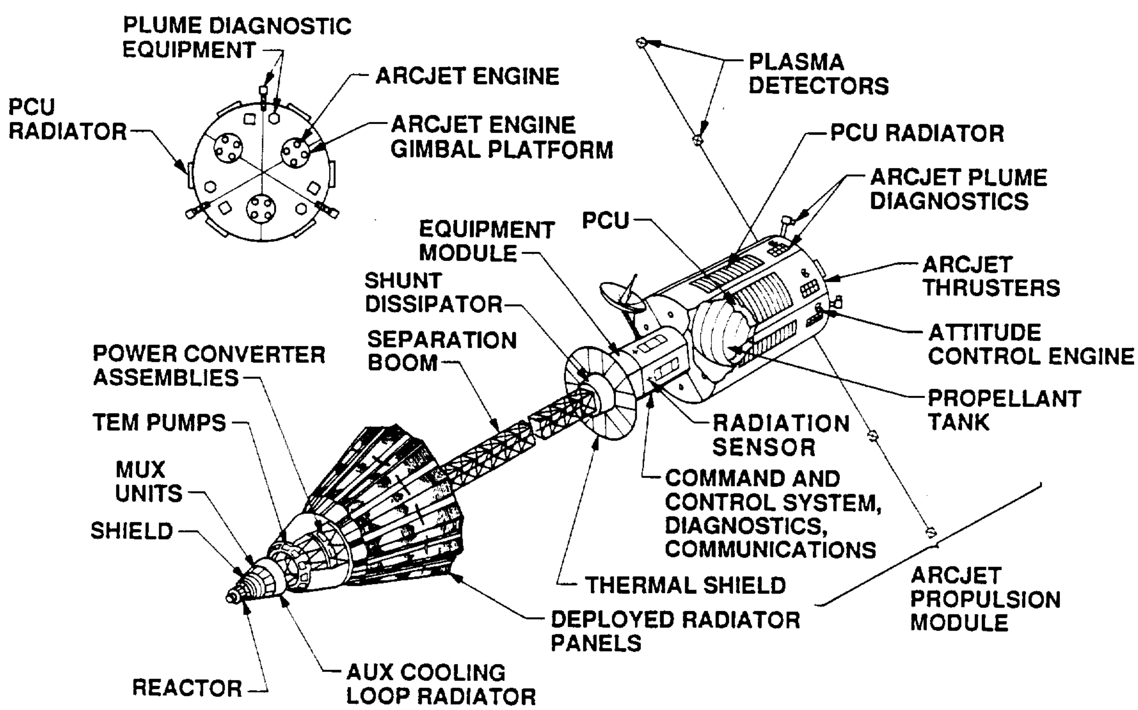

Figure 1. Proposed spacecraft configuration for the SP-100 Flight Experiment.

ated structure. During arcjet system operation, one engine from each platform operates to provide thrust. After 1500 hours of operation, these three engines are turned off and another three (one engine per platform) are turned on. This process repeats after the next 1500 hours of operation to accumulate a total operating time of 4500 hours. At that time the arcjet mission has been completed. A fourth set of three engines is provided as backup. There are two dedicated PCUs per gimballed platform with one serving as a spare. Separate propellant feed lines provide ammonia to each platform. Three thrusters can be operated at maximum power using 93 kW_e of input power when accounting for the 98% efficiency of the PCU system.

The thruster module is enclosed within a 4.4-m outside-diameter, 6-m long cylinder with the propellant tank located on the end nearest the SRPS. The three sets of arcjet engines and gimbals are located on the end of the cylinder opposite the SRPS. The PCU subsystem is located within the cylindrical enclosure between the propellant tank and engine modules. The six PCU low temperature radiators face space on the outer surface of the cylindrical enclosure. The combined thrust of this system is 7.6 N when three engines are operating at full power. The command, data handling and telecommunications functions are part of the spacecraft bus.

A mass summary of the spacecraft components is provided in Table 2. As discussed above, the mass goal for the 100-kW_e SP-100 SRPS is given as 3000 kg.[21,22] The propulsion system is assumed to have a mass of 575 kg excluding propellant, tankage and the feed system. The spacecraft bus, which includes the primary command, control and communications equipment, is assumed to have a mass of 1250 kg. The mass assumed for the diagnostics equipment is 300 kg. An additional 550 kg has been set aside as a contingency.

The SP-100 Flight Experiment spacecraft is shown in its stowed configuration within a Titan IV ELV payload faring (PLF) in Fig. 2. The SP-100 SRPS is located at the top of the ELV. The spacecraft bus attaches to the SP-100 UIM and the arcjet propulsion system. The expendable upper stage and contamination shield are located at the bottom of the Titan IV payload faring. This vehicle configuration also fits in the STS payload bay.

TABLE 2
Projected Mass Summary for the 100 kW_e SRPS SOA Arcjet Flight Experiment Spacecraft

Subsystem	Mass (kg)
SRPS	3000
Spacecraft Bus	1250
Thruster System Diagnostics	300
Arcjet Module	575
Propellant Feed System	*
Contingency	550

*Depends upon propellant load (see Propellant Flow Subsystem section) and launch vehicle mass limit.

The SP-100 Flight Experiment launch and deployment sequences are shown in Figs. 3a and 3b using a Titan IV ELV. In Fig. 3a, the Titan IV lifts off using the SRMs. The stage 1 chemical engine ignites and is followed by SRM burnout and separation. The PLF is then jettisoned. After stage 1 burnout, stage 1 and stage 2 separate; then, stage 2 ignites to continue the vehicle into orbit. Once state 2 burns out, it separates from the SP-100 Flight Experiment spacecraft and upper stage. The upper stage ignites to inject the SP-100 Flight Experiment vehicle into a 300 km by 925 km, 28.5° elliptical orbit.

As shown in Fig. 3b, the upper stage reignites to circularize the elliptical orbit into a 925 km, 28.5° parking orbit. A 925 km, 28.5° circular orbit will be defined as nuclear safe orbit (NSO) in this paper. The upper stage and contamination shield are then jettisoned. This is followed by the deployment of the separation boom, SP-100 radiator, and instrumentation. The SP-100 power system is activated and the spacecraft systems checkout tests are completed. Finally the arcjet NEP system is turned on and the mission spiral is begun.

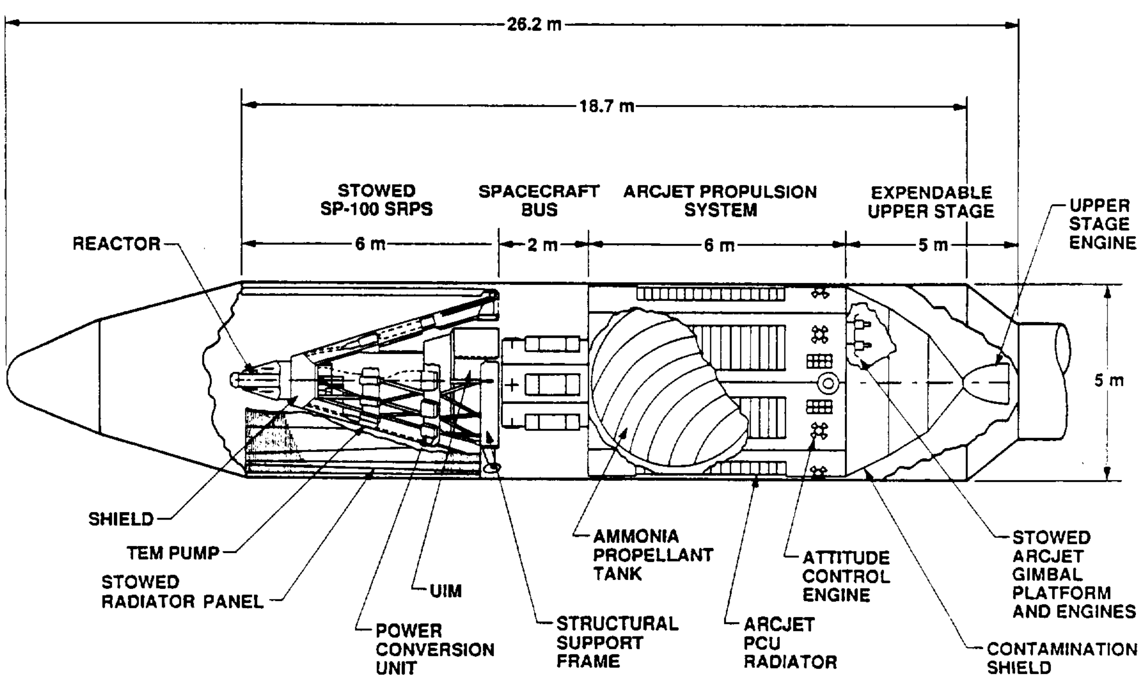

Figure 2. SP-100 Flight Experiment in stowed configuration in a Titan IV payload faring.

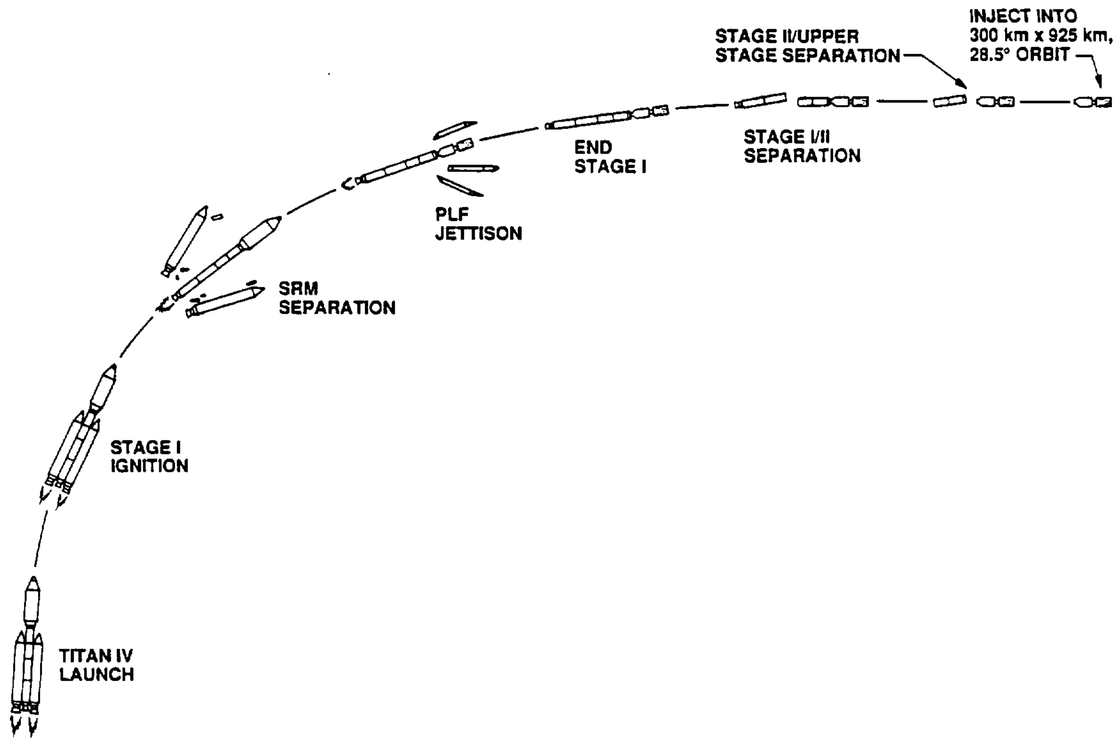

Figure 3a. Titan IV launch sequence.

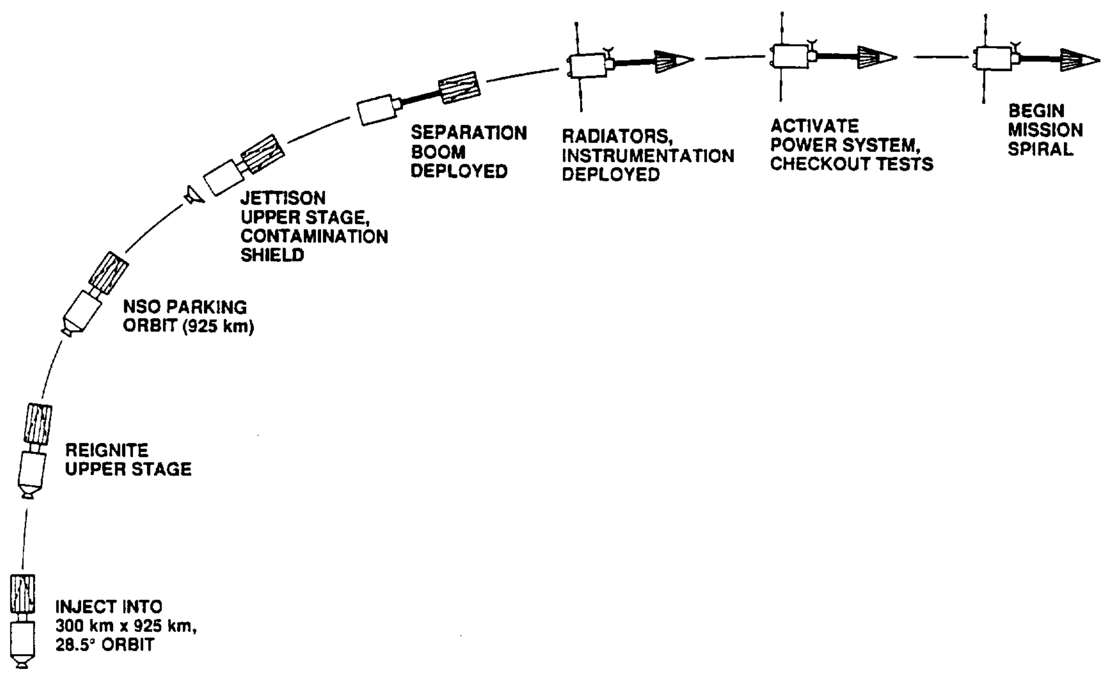

Figure 3b. SP-100 Flight Experiment deployment sequence.

A block diagram of the arcjet SP-100 Flight Experiment vehicle is shown in Fig. 4. It includes all of the primary system components for converting SRPS power into thrust. The power system consists of the SP-100 PGM and UIM and provides both 28V and 200V (primary) outputs. The spacecraft bus contains the navigation and the command, data handling and telecommunications subsystems which receive and process ground commands and control overall system operation. The arcjet PCU subsystem starts and runs the arcjets. The propellant system runs parallel to the power train and includes the tankage, valves, lines, etc. required to provide a constant propellant flow rate to each operating engine. The diagnostic package provides the ability to monitor the reactor radiation-induced environment, to measure the particu-

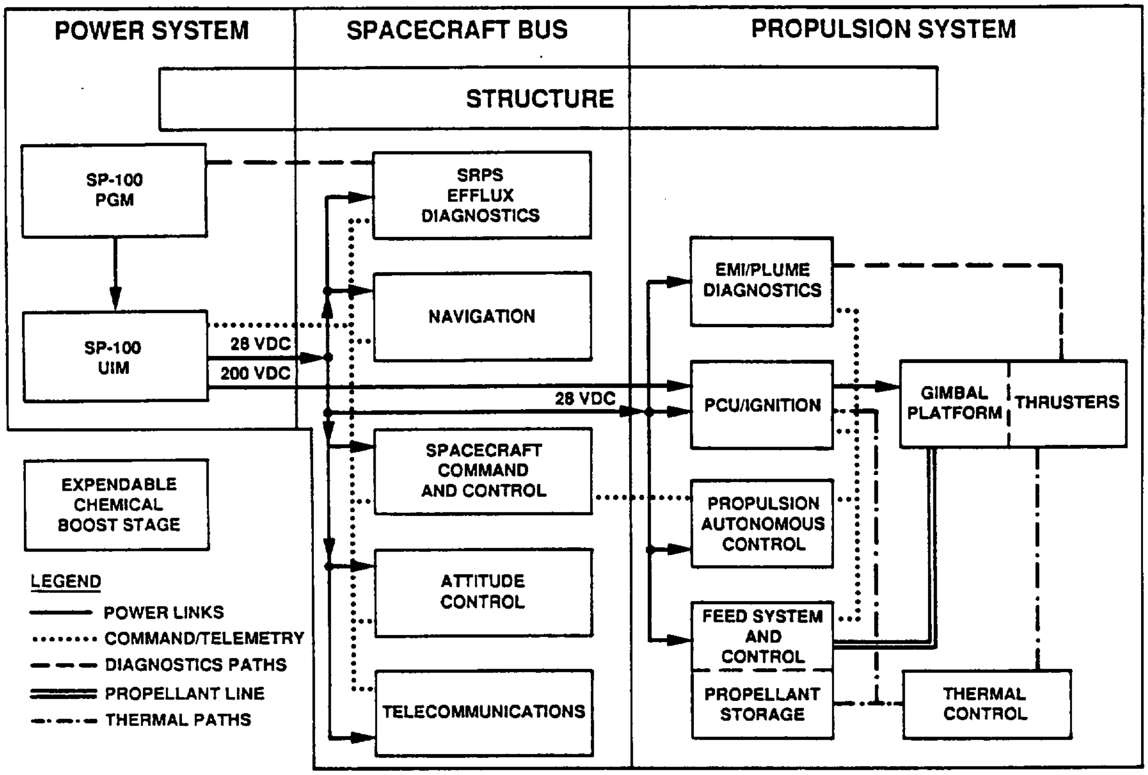

Figure 4. Arcjet NEP system block diagram for the SP-100 Flight Experiment.

late and field emissions from the arcjet thrusters in the vicinity of the electric propulsion module and to examine the spacecraft/space environment interactions. Thermal control allows for the rejection of waste heat from the arcjet and PCUs while the structural members tie all of the subsystems together.

PROPULSION SYSTEM COMPONENTS

Descriptions of the engine/gimbal assemblies, PCU subsystem, propellant handling subsystem, thermal control methodology, diagnostics package and telemetry needs are presented below.

Arcjet Engine/Gimbal Platform

A schematic of a proposed engine/gimbal platform configuration is shown in Fig. 5. Each engine/gimbal platform consists of four 30-kW_e arcjet engines, a heat shield/platform, a high-power, high-current switch, a propellant distribution manifold, and a gimbal mechanism including a set of flexible high-current power leads and propellant lines. Three platforms are used and are located on the aft end of the spacecraft (see Fig. 1) with one engine per platform operating at a time. The arcjet technology level assumed for the SP-100 Flight Experiment spacecraft, as defined in this study, is given in Table 3 and is based on experimentally derived performance data. These performance values were measured while running a new engine design over a 9 hour period, 7 1/2 hours of which was at a power levels between 30.1 kW_e and 30.9 kW_e. This performance level will be defined as State-of-the-Art (SOA) in this paper. The high-power, high-current switch selects the arcjet engine to be operated on that

TABLE 3
Operating Characteristics for an SOA Arcjet Engine*

Parameter	Value
Propellant	NH_3
Engine Input Power, kW_e	30.3 ± 0.2
Specific Impulse, s	1031 ± 35
Engine Efficiency	0.423 ± 0.025
Arc Voltage, V	106 ± 3
Arc Current, A	284 ± 5
Mass Flow Rate, g/s	0.25 ± 0.002
Thrust, N	2.53 ± 0.12
Engine Mass, kg	7
Lifetime,** hours	1500

* Engine run for 9 hours at JPL on July 6, 1988.
**1500 hour lifetime assumed.

platform. As engines reach the end of their useful life a new engine can be switched into operation. Some development of mechanical high-power rotary switches has taken place.[23] However, with the gains made recently in high power electronics, such a switching mechanism should be possible using high power transistors, diodes, etc. and contain no moving parts. The use of a power switch can be avoided if each engine has a dedicated PCU and the associated mass penalty is acceptable. A propellant feed manifold runs parallel to the power switch and distributes propellant to the desired engine. The platform is the primary structural member and serves as a heat shield to protect the main spacecraft structure from the radiated arcjet heat.

Arcjet Power Conditioning Unit (PCU)

There will be two (2) PCUs associated with each engine gimbal platform. One PCU will serve as

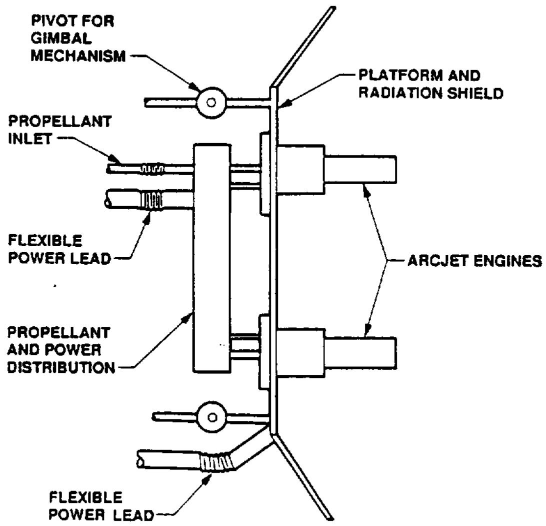

Figure 5. Schematic of proposed arcjet engine/gimbal platform configuration.

a spare. Each PCU consists of a pulsed, low-power, high-voltage "starter" circuit in parallel with a high-power, low-voltage "run" power supply. The "run" power supply is based on a three phase "buck" regulator design which is efficient, reliable and compact.[24,25] The PCU is shown schematically in Fig. 6. The constricted arc in the arcjet has a negative dynamic resistance. A modified current mode feedback, which compares the actual arc current with the desired current, and an improved control algorithm reduce ripple amplitude and provide more positive control of the arc. The PCU specific mass is taken as 0.4 kg/kW_e at an efficiency of 98%. The PCUs are self-radiating, rejecting 0.65 kW_e of power while maintaining the component base plate at a temperature of less than 300 K. The high power and elevated temperature electronic components could be mounted directly to the PCU baseplate which might be a honeycomb panel heat pipe/radiator. This type of light-weight radiator has been investigated and shows promise for use as a low temperature radiator.[26,27]

Propellant Flow Subsystem

The propellant flow system includes the propellant storage tank and a feed system to supply a constant propellant flow to each operating thruster. Ammonia propellant storage and feed systems are a mature technology which have been flown several times.[28-31] A schematic of the proposed ammonia propellant flow system is shown in Fig. 7. The propellant system specifications are summarized in Table 4. Ammonia is stored in a spherical titanium tank at about 150 psia. Titanium was chosen for the tank material due to its low mass and chemical compatibility with ammonia. At 150 psia, ammonia boils at 298 K, implying that a minimum of propellant thermal control is required. An electric heater system provides heat to vaporize the ammonia and maintain the 150-psia tank pressure. Multilayer insulation minimizes the number of heating cycles required to maintain ammonia vapor in the propellant tank. The tank is loaded with the proper mission-dependent propellant mass prior to launch. A space-based propellant refill capability is assumed should future testing or other needs require restart of the arcjet NEP system.

TABLE 4
Propellant System Specifications

Propellant	NH_3
Tank Capacity	13,150 kg
Storage Pressure	150 psi
Internal Tank Diameter	3.5 m
Tank Material	Ti
Flow to Each Platform	0.25 g/s

The feed system consists of the propellant lines, valves, transducers, filters, regulators, heater/vaporizers, flow controllers, structure, etc., required to provide the proper propellant flow rate to the arcjet thrusters. Electronic flow controllers

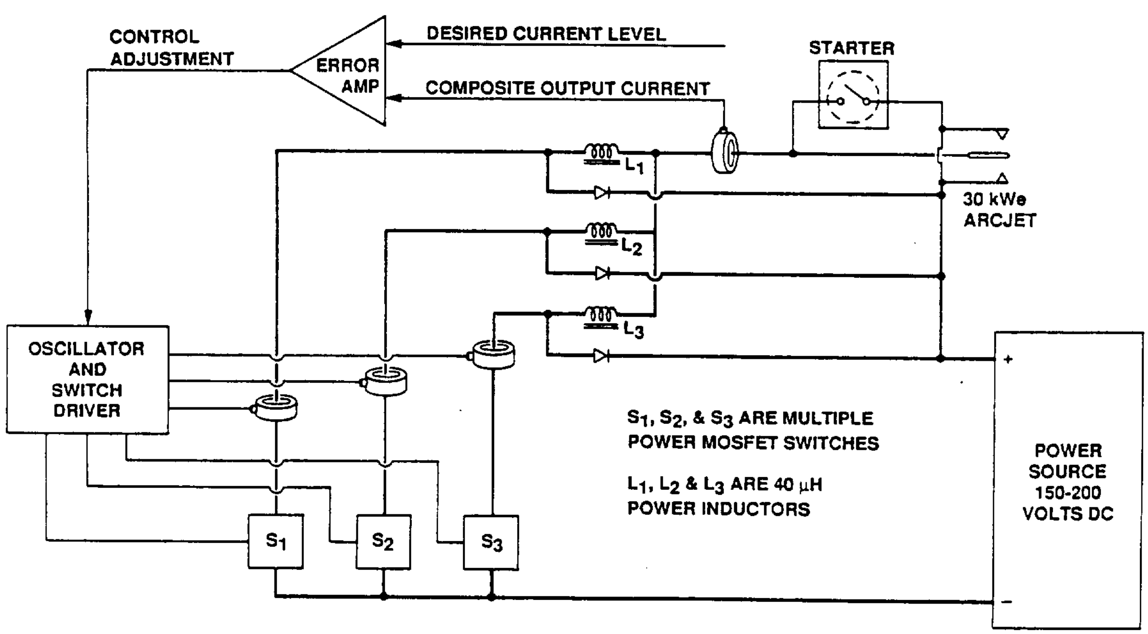

Figure 6. Schematic of a possible arcjet PCU configuration.[26]

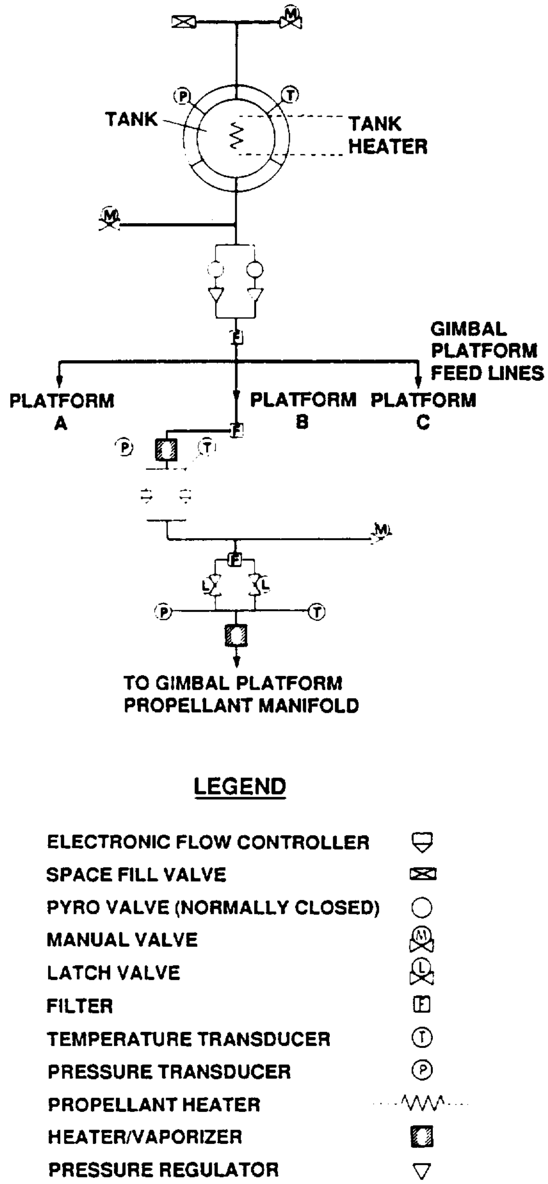

Figure 7. Ammonia (NH_3) feed system schematic.

are needed to throttle the engines and optimize their operation as functions of efficiency and specific impulse. Some development of this type of flow controller has taken place.[32] If the mission design does not require engine throttling as functions of efficiency and specific impulse, then a single flow rate can be provided by a regulator/orifice assembly. The total tankage and feed system mass, $M_{f/s}$, consists of a fixed component independent of propellant load and a variable component dependent on the propellant load, Mp, and is given by,

$$M_{f/s} = 100.0 \text{ kg} + 0.20 \, M_p \qquad (1)$$

This equation includes a 10 percent contingency on all components. This system provides a constant mass flow of 0.25 g/s of ammonia to each operating arcjet thruster for the full mission duration. The maximum tank storage capacity is 13,150 kg of ammonia using a 3.5 m internal diameter tank.

Thermal Control

Thermal control for the arcjet module is achieved by standard engineering techniques. For instance, it is estimated that 10% of the arcjet power input is distributed in the anode electrode, amounting to 3 kWe per engine. This power is readily self-radiated by the anode at 2300 degrees Kelvin. If the surface is treated with a high emissivity coating (emissivity greater than 0.9) the temperature requirement can drop to 1900 degrees Kelvin. The arcjet platform acts as a radiation shield between the spacecraft and the hot arcjets. In addition, conducted heat from the platform to the spacecraft is minimized by using propellant cooling of the interconnecting structures. The thermal control design for the PCUs consists of low temperature radiators located on the outside of the propulsion module. Thermal control of the propellant storage and feed system is accomplished by the straightforward application of multi-layer insulation around the tank in conjunction with an internal tank heater.

Diagnostics Package

A diagnostics package is carried on the SP-100 Flight Experiment to monitor the SRPS-induced radiation environment at and beyond the user interface, to examine the arcjet propulsion system particulate and field emissions and to examine the spacecraft/space environment interactions. Such a diagnostics package will enable future users of both the SP-100 SRPS and arcjet engines to better assess the potential impacts of these systems on their payloads.

SRPS-INDUCED RADIATION ENVIRONMENT The SRPS will be emitting neutrons and gamma rays, the levels of which will have to be evaluated. As shown in Table 1, the design goal for the 10 year total doses of neutrons and gamma rays are less than 10^{13} neutrons/cm^2 and 5 x 10^5 rads, respectively, at the user side of the UIM. Also, the SP-100 SRPS thermal environment is designed to be less than 0.14 W/cm^2 (less than one sun) at the UIM. Instrumentation is included on the SP-100 Flight Experiment spacecraft, as defined in this paper, to evaluate these levels.

PROPULSION SYSTEM DIAGNOSTICS Three primary types of measurements needed to characterize the performance and effects of the arcjet propulsion system. These measurements are summarized in Table 5 and include the monitoring of thruster operation, arcjet dynamics, and arcjet/spacecraft interactions.

Thruster Operation The engine performance will be evaluated and compared to ground test measurements and theoretical models. Measurements of arc current and voltage, mass flow rate and component temperatures will be made. The thrust will be monitored using accelerometers mounted onboard the SP-100 Flight Experiment spacecraft. These measurements will allow verification of ground test experiments and models.

Arcjet Dynamics Measurements of the components of an arcjet plume could enable a deeper understanding of thruster operation, leading to improved arcjet design. Space-based measurements eliminate ground test facility effects and act to verify the ground test measurements. Measurements of plasma density, species concentrations, temperature distributions and plume spatial extent could provide the desired information on arcjet dynamics. This information would provide a better understanding of arcjet physics.

Arcjet/Spacecraft Interactions A small portion of the exhaust plume will extend back behind the thruster nozzle exit plane, due to gas dynamic expansion, and will impinge on the arcjet module

TABLE 5
Propulsion System Diagnostic Instrumentation

NEED	MEASUREMENTS	INSTRUMENTS
THRUSTER OPERATION	ARC CURRENT ARC VOLTAGE MASS FLOW RATE TEMPERATURES	VOLT METER AMMETER FLOW CONTROLLER THERMOCOUPLES
ARCJET DYNAMICS	ELECTRON DENSITY ION DENSITY TEMPERATURE DISTRIBUTIONS PARTICLE SPECIES	FARADAY PROBES LANGMUIR PROBES MASS SPECTROMETER VIDEO CAMERA
ARCJET/ SPACECRAFT INTERACTIONS	PARTICLE DEPOSITION PARTICLE SPECIES SPACECRAFT CHARGING EMI TEMPERATURES	QCM SOLAR CELL WITNESS PLATES MASS SPECTROMETER LANGMUIR PROBE ANTENNAS INFRARED MONITORS

and SRPS. Particulate contamination is expected to be minimal since the gas is rarified and the volatile contaminant density is very low.[33] The primary particulate contaminants are expected to be hydrogen, nitrogen, tungsten, boron and thorium. Of these, the metals and boron pose the greatest potential hazard since they will condense on most surfaces they contact. For a six-month mission, the maximum expected tungsten loss from all engines totals less than 30 g based on erosion data from previous arcjet tests.[34-37] Previous work has shown that only a very small fraction of this tungsten loss would reside in the plume backflow.[33] All of this material would have to be focused to one area to cause a significant problem.

The Electromagnetic Interference (EMI) characteristics of arcjet thrusters are not well known but the engines are expected to radiate electromagnetic energy since they produce a plasma.[38] The effects of EMI on such spacecraft systems as communications, guidance, navigation and power control electronics must be examined. Since the SP-100 Flight Experiment onboard spacecraft power is almost two orders of magnitude greater than that of present-day spacecraft, EMI guidelines will require extensive revision. Thermal radiation from arcjet thrusters can also present a problem since up to 10% of the engine input power is radiated away by the nozzle alone.[39,40] The gimbal platforms serve as heat shields to reduce radiative heating of the upstream spacecraft components.

SPACECRAFT/ENVIRONMENT INTERACTIONS No spacecraft of this size with so many different materials exposed to the space environment and with as high an onboard power level has ever been flown. As a result, the potential for spacecraft/space environment interactions is high. Possible effects such as spacecraft frame charging, differential charging of neighboring spacecraft surfaces, electrostatic discharge (ESD), parasitic power drain to the space plasma, and the long term effects of the SRPS radiation environment and propulsion system effluents on overall spacecraft integrity will need to be carefully monitored. Previous space experiments have shown that spacecraft charging and its related effects can be reduced by electric thruster operation.[41-46]

Telemetry Needs

S-band and X-band communications capabilities will meet the telemetry needs of the SP-100 arcjet propulsion module. Those needs can be divided into two categories: 1) housekeeping and 2) engineering data transmission. Housekeeping pertains to the propulsion module health and includes engine operation (arc voltages and currents, etc.), propellant storage and feed status (flow rate, tank pressure, etc.), and various critical temperatures throughout the module, such as at the PCU baseplate and arcjet anode. Engineering data refers to information gathered from the diagnostic monitoring of arcjet effluents. These data, such as camera outputs, plasma probe currents and voltages, in general will require greater resolution than data gathered on housekeeping status and, therefore, will require higher storage density.

ARCJET NEP PERFORMANCE

The following analysis is based on the well-known orbital mechanics equations for electric propulsion transfers[47] and on the propellant feed subsystem characterization given above. Launches from Kennedy Space Center (KSC) using the STS launch vehicle and Titan IV ELV are assessed for four proposed Flight Experiment scenarios. The analysis assumes two different SP-100 SRPS power levels; 100 kW_e and 30 kW_e, and two different arcjet/PCU technology levels; baseline and State-of-the-Art (SOA). It is assumed that only one arcjet operates on a spacecraft with a 30 kW_e SRPS and up to three arcjets can operate simultaneously on a spacecraft with a 100 kW_e SRPS for either arcjet technology.

ARCJET PROPULSION SYSTEM PARAMETERS

The two arcjet system technology levels used for this mission analysis are presented in Table 6. The baseline system parameters are derived from a recent 573-hour long duration test of an arcjet engine.[34,48] The baseline values shown in Table 6 represent averaged arcjet engine performance over the 573 hour duration test at 25.1 kW_e and provide an effective lower bound for arcjet performance. A baseline engine/PCU requires 27.9 kW_e of input power when accounting for the 90 percent efficiency of the PCU. Therefore, a system of three engines requires 83.7 kW_e.

As mentioned previously, the SOA arcjet technology level in Table 5 (see Table 3) also represents measured arcjet performance. These performance values were measured while running a new engine design over a 9 hour period, 7 1/2 hours of which was at a power levels between 30.1 kW_e and 30.9 kW_e. The engine incorporates a bell-shaped nozzle which has shown potential engine efficiency improvements of up to 20 percent.[49-51] In addition, improved

TABLE 6
Arcjet Performance Characteristics Used for this Study+,[34,48]

Parameter Technology Level	Value Baseline	SOA+
Propellant	NH_3	NH_3
Input Pwr Per Thruster (kWe)	25.1	30.3 ± 0.2
Thruster Efficiency	0.39	0.423 ± 0.025
Specific Impulse (s)	867	1031 ± 35
Thrust Per Engine (N)	2.3	2.53 ± 0.12
Thruster Lifetime (hours*)	573	1500
PPU Efficiency	0.90	0.98
System Specific Mass Per Engine** (kg/kW_e)	2.0	1.6

+ Engine run for 9 hours at JPL on July 6, 1988.
* 573 hour lifetime measured, 1500 hour lifetime assumed.
**Excludes SRPS, spacecraft bus propellant, tankage and feed system.

propellant cooling helps recover some of the conducted power loss through the cathode. Such cooling also preheats the propellant gas and should enable a small increase in overall engine efficiency. This new engine design is described in detail in Reference 52. A 1500-hour lifetime is assumed for this engine. Finally, a high-temperature, high-emissivity coating could be applied to the outer nozzle surface to improve its radiative cooling properties. This reduces the nozzle temperature and should enhance the thruster durability.[52] An SOA arcjet/PCU requires 30.9 kW_e of input power with a three engine system needing 92.7 kW_e.

CONSTRAINTS AND ASSUMPTIONS

Due to safety concerns, the SRPS can not be operated until the spacecraft has reached a 925 km (500 nmi) NSO. An expendable chemical upper stage will boost the NEP flight demonstration spacecraft to NSO from STS orbit or Titan IV separation orbit. It is further assumed that the upper launch mass limit for the STS is 23,182 kg,[53] that 4,100 kg of Airborne Support Equipment (ASE) is needed, and that a single, dedicated shuttle launch from KSC is required for the Flight Experiment. It is also assumed that the upper launch mass limit for the Titan IV ELV is 17,700 kg,[53,54] that 3300 kg of ASE type equipment is needed and a that dedicated Titan IV ELV is required. The orbit and launch vehicle assumptions are summarized in Table 7. An expendable chemical upper stage (Isp = 300 s) used to orbit raise to NSO corresponding to a ΔV of 338 m/s, weighs 2380 kg and has a dry to fueled mass ratio of 0.15. The chemical upper stage does not perform any part of required plane changes.

TABLE 7
Launch Vehicle and Orbit Assumptions[53,54]

Parameter	Launch Vehicle STS	Titan IV
Payload (kg)	23,182	17,700
ASE mass (kg)	4,100	3,300
Altitude (km)	300	165
Inclination (degrees)	28.5	28.5
NSO altitude (km)	925	925
NSO inclination	28.5	28.5

A mass summary for the different SP-100 Flight Experiment spacecraft configurations is given in Table 8 as a function SRPS power level and arcjet system technology level. The specific mass for the 30 kW_e SRPS is assumed to be 65 kg/kW_e[22] and for the 100 kW_e SRPS, 30 kg/kW_e.[21,22] The spacecraft bus is assumed to have a mass of 1100 kg on a spacecraft powered by a 30 kW_e SRPS and 1250 kg on a spacecraft powered by a 100 kW_e SRPS. A diagnostics package with a mass of 300 kg is included for all spacecraft configurations. Contingencies of 265 kg and 550 kg are included for the 30 kW_e and 100 kW_e spacecraft, respectively.

TABLE 8
SP-100 Flight Experiment Spacecraft Mass Summary

Quantity	Based on 30 kW_e SRPS	Based on 100 kW_e SRPS
SRPS	1950 kg	3000 kg
SRPS Specific Mass	65 kg/kW_e	30 kg/kW_e
Spacecraft Bus	1100 kg	1250 kg
Diagnostics	300 kg	300 kg
Contingency	265 kg	550 kg
Propulsion System*		
Baseline	720 kg	1800 kg
SOA	288 kg	575 kg

*Excludes propellant, tankage and feed system. Includes engines and spares for 4500 hours of propulsion system operation.

The propulsion system mass is also given in Table 8 for the two different arcjet technology levels assuming that the propulsion system must operate for a total of 4500 hours. The values in Table 8 do not include the propellant, tankage and feed system masses which are given by Eq. 1 and also depend on the launch vehicle mass limits. The baseline system has a mass of 720 kg when the available spacecraft power is 30 kW_e. Since the baseline engine has a lifetime of 573 hours, 8 baseline arcjet engines are required and an additional 4 are included as spares in the mass value. When the spacecraft power is 100 kW_e, the baseline propulsion system mass increases to 1800 kg. This value includes 30 engines, 6 of which are spares. Using SOA arcjet technology, a propulsion system based on a total of 6 engines (3 of which are spares) has a mass of 288 kg on a spacecraft with 30 kW_e on board. Finally, the propulsion system mass is 575 kg for a spacecraft with 100 kW_e of onboard SRPS power and SOA arcjet technology, as discussed in the "SP-100 Flight Experiment Spacecraft Configuration" section above.

MISSION SCENARIOS AND RESULTS

Four missions are examined which could be used to demonstrate SRPS operation. The first two missions involve power system deployment to possible SDI platform orbits of 3,000 and 10,000 km. An advantage of these orbits is that they contain a minimum of man-made orbital debris, reducing the chance of a collision.[55] The third mission involves a spacecraft storage demonstration to very high orbits. The final mission examines an orbit raising round trip to and from NSO.

3000 km Orbit

A 3,000 km circular orbit, with a final inclination between 55° and 85°, has been identified as a potential SDI platform orbit.[56] As a result, this orbital altitude was chosen for this study so that the mission would address the control scenarios required for a low-altitude, high-inclination change, low thrust mission.[57] The orbital analysis is done such that the entire available propellant load is consumed to reach the highest inclination possible for each of the arcjet technologies described in Table 6, the launch vehicle characterizations summarized in Table 7 and the spacecraft power levels as shown in Table 8. The results of this analysis are summarized in Table 9. If the transfer time is greater than 180 days, the propulsion system has

TABLE 9
SP-100 Flight Experiment Performance from NSO to a 3000 km Final Altitude

Launch Vehicle	SRPS Power (kW_e)	Arcjet Technology	Trip Time* (days)	Final Inclination (degrees)
STS	100	baseline	114	58.0
STS	100	SOA	142	72.0
STS	30	baseline	412	68.5
STS	30	SOA	500	85.5
Titan IV	100	baseline	66	48.5
Titan IV	100	SOA	88	60.5
Titan IV	30	baseline	267	59.5
Titan IV	30	SOA	334	73.5

*Propulsion system designed for total trip time when greater than 180 days.

been resized with respect to the values discussed in Table 8 to account for the larger number of engines required. For example, an SP-100 Flight Experiment vehicle using the baseline arcjet system enables a 100 kW_e SRPS to be delivered to a 58° final inclination in 114 days at an orbital altitude of 3,000 km using the STS as a launch vehicle. If the vehicle used SOA arcjet technology, a 100 kW_e SRPS, and was launched in the STS, it would be capable of achieving a 3,000 km, 72° final orbit in 142 days. A Titan IV launch of a vehicle based on the SOA arcjet technology and a 100 kW_e SRPS would achieve a 60.5° inclination, 3000 km orbit in 88 days.

10,000 km Orbit

A 10,000 km circular orbit was chosen as the target altitude for an arcjet NEP spacecraft throttling demonstration and is compared to a non-throttled case. Again, the analysis is done such that the entire available propellant load is consumed to reach the greatest orbital inclination possible for each of the characterizations and levels described in Tables 6 through 8. Only the 100 kW_e SRPS is considered in this case. The non-throttled cases are summarized in Table 10. The baseline arcjet technology with an STS launch provides a total ΔV capability of 5559 m/s corresponding to a 10,000 km, 59.5° final orbit with a 115 day trip time. The SOA arcjet technology with an STS launch enables a non-throttled total ΔV of 7856 m/s corresponding to a final orbit of 10,000 km at 77.0° and a trip time of 142 days. A 10,000 km, 62.5° final orbit could be achieved in 88 days with a spacecraft based on the SOA arcjet system and Titan IV launch for a ΔV of 5965 m/s.

TABLE 10
SP-100 Flight Experiment Performance from NSO to a 10,000 km Final Orbit, Unthrottled

Launch Vehicle	SRPS* Power (kW_e)	Arcjet Technology	Trip Time (days)	Final Inclination (degrees)	ΔV (m/s)
STS	100	baseline	115	59.5	5559
STS	100	SOA	142	77.0	7856
Titan IV	100	baseline	65	46.5	3843
Titan IV	100	SOA	88	62.5	965

The cases for which the propulsion system is throttled are summarized in Table 11. Again, only the 100 kW_e SRPS is considered. As above, the increased propulsion system mass was accounted for if the total trip time was greater than 180 days. The calculations were conducted as follows: with three arcjets operating at full power, the Flight Experiment spacecraft is raised from a 925 km, 28.5° orbit to a 10,000 km, 28.5° orbit corresponding to a ΔV of 1,827 m/s. From this orbit, a vehicle using SOA arcjets is moved to a 10,000 km, 38.5° orbit, a ΔV of 1,567 m/s, with one arcjet operating at full power. The next leg is accomplished using two SOA arcjets operating at full power and results in a final orbit of 10,000 km, at 48.5° for an additional ΔV of 1,325 m/s. The final leg is completed with three SOA arcjets operating at full power until all the available propellant is consumed. This results in final orbits of 10,000 km at 54.5° assuming a Titan IV launch and 10,000 km at 70.5° assuming an STS launch corresponding to ΔVs for the final legs of 1,187 and 3,120 m/s, respectively. A similar methodology was followed when considering the baseline arcjet technology. Throttling of the engines provides a demonstration of the SRPS load-following capability in splitting power between the user and power system shunt and demonstrates the flexibility of both the arcjet NEP system and the SP-100 SRPS.

Spacecraft Storage Mission

The third mission demonstrates low thrust control scenarios to very high orbits. A spacecraft storage mission from NSO to an altitude of 107,580 km with a return to 35,860 km was selected. The first leg of the trip has a ΔV of 6,211 m/s and the return leg a ΔV of 1,204 m/s. The results for this scenario are summarized in Table 12 for the different launch vehicles, SRPS power levels and arcjet technology levels. For example, the baseline arcjet system could not reach 107,580 km with a 100 kW_e SRPS, but

TABLE 11
Summary of Arcjet Throttling Orbital Analysis, NSO to a 10,000 km Final Orbit

Launch System	Arcjet Technology	Operating Arcjets	Power (kW_e)	Initial Orbit Alt., Incl. (km, degrees)	Final Orbit Alt., Incl. (km, degrees)	Trip Time (days)	Total ΔV (m/s)
STS	baseline	3	83.7	925, 28.5	10,000, 28.5	60	
		1	27.9	10,000, 28.5	10,000, 33.5	41	
		2	55.8	10,000, 33.5	10,000, 38.5	19	
		3	83.7	10,000, 38.5	10,000, 51.5	28	5559
	SOA	3	92.7	925, 28.5	10,000, 28.5	57	
		1	30.9	10,000, 28.5	10,000, 38.5	76	
		2	61.8	10,000, 38.5	10,000, 48.5	33	
		3	92.7	10,000, 48.5	10,000, 70.5	37	7839
Titan IV	baseline	3	83.7	925, 28.5	10,000, 28.5	45	
		1	27.9	10,000, 28.5	10,000, 31.5	19	
		2	55.8	10,000, 31.5	10,000, 33.5	6	
		3	83.7	10,000, 33.5	10,000, 38.5	9	3809
	SOA	3	92.7	925, 28.5	10,000, 28.5	43	
		1	30.9	10,000, 28.5	10,000, 38.5	58	
		2	61.8	10,000, 38.5	10,000, 48.5	25	
		3	92.7	10,000, 48.5	10,000, 54.5	9	5906

TABLE 12
SP-100 Flight Experiment Performance for a Spacecraft Storage Mission

Launch Vehicle	SRPS Power (kW_e)	Arcjet Tech.	Trip-time NSO-3GSO+ (days)	Trip-time 3GSO-NSO+ (days)	Residual Mass (kg)
STS	100	baseline	130	*	
STS	100	SOA	126	16	1357
STS	30	baseline	391	46	634
STS	30	SOA	387	52	2941
Titan IV	100	baseline	*	*	
Titan IV	100	SOA	95	*	
Titan IV	30	Baseline	295	*	
Titan IV	30	SOA	292	38	928

*Transfer not possible.
+Propulsion system designed for total trip time when greater than 180 days.

could achieve 107,580 km assuming a 30 kW_e SRPS, and return to 35,860 km assuming an STS launch. The SOA arcjet propulsion system propels the spacecraft to 107,580 km and then return to 35,860 km in all cases except for a 100 kW_e baseline system launched with a Titan IV. In each case where a spacecraft could complete the storage mission there was some residual propellant left over, indicating a greater ΔV capability. Again, the propulsion system is resized if trip times greater are than 180 days.

Round Trip

The final mission considered is a round-trip mission from NSO to some high earth orbit (HEO) and back to NSO to simulate an Orbit Transfer Vehicle (OTV) mission. This mission provides an opportunity to examine the control scenarios required for a round trip-type OTV mission.[57] No plane changes are considered. The round-trip mission results are summarized in Table 13. As before, the propulsion system mass is increased to account for trip times greater than 180 days. For example, a spacecraft launched using the STS with a 100 kW_e SRPS and an SOA arcjet system achieves a HEO of 27,000 km at 28.5° in 97 days and return to NSO in 53 days. If a Titan IV launch vehicle is used to inject a spacecraft with a 30 kW_e SRPS and a baseline arcjet system onboard, the spacecraft will reach a HEO of 12,300 km at 28.5° in 171 days and return to NSO in 96 days.

TABLE 13
Analysis for Roundtrip OTV Mission

Launch Vehicle	SRPS Power (kW_e)	Arcjet Tech.	Trip-time NSO-HEO (days)	HEO (kg)	Trip-time HEO-NSO (days)
STS	100	baseline	76	12,400	40
STS	100	SOA	97	27,000	53
STS	30	baseline	285	22,000	151
STS	30	SOA	353	58,000	190
Titan IV	100	baseline	39	6,300	22
Titan IV	100	SOA	55	12,800	32
Titan IV	30	baseline	171	12,300	96
Titan IV	30	SOA	222	26,500	126

*Propulsion system designed for total trip time when greater than 180 days.

CONCLUSIONS

The design and performance characteristics of an arcjet NEP spacecraft suitable for conducting the SP-100 Flight Experiment have been presented. The simplicity of arcjet thrusters and their relatively advanced state of development (the arcjet performance values used in the analysis were experimentally determined) allow them to meet the SP-100 Flight Experiment constraint of low developmental risk. In addition, arcjets can be scaled up in power into the 100s of kilowatts regime and beyond, making them compatible with future SDI power levels. As a result, arcjets are particularly well-suited for the SP-100 Flight Experiment.

A proposed Flight Experiment vehicle has been outlined and consists of a 100 kW_e SRPS, a spacecraft bus, a radiation/arcjet efflux diagnostics package, and an arcjet propulsion module, in an end thrust configuration. The propulsion module consists of three 30-kW_e ammonia arcjets, operating at a specific impulse of 1031 s and an efficiency of 42.3 percent. A total system thrust of 7.6 N is generated with three engines operating at full power. The baseline vehicle mass is 5675 kg excluding the propellant, tankage and feed system.

Orbital analysis was conducted to evaluate the SP-100 Flight Experiment vehicle performance. A single dedicated STS or Titan IV launch was assumed from KSC. A number of candidate missions were proposed with no attempt to recommend one over another. The intent was to present options, any one of which might be representative of future mission deployment requirements. The analysis showed that this vehicle is capable of mission ΔVs of 6,000 to 7,900 m/s. A propulsion system throttling demonstration would verify the SRPS load-following capabilities.

Four specific missions were examined which included power system deployment to possible surveillance platform orbits, a spacecraft storage mission and a round-trip OTV mission. Analysis has shown that the vehicle could reach a 3,000 km, 72° inclination final orbit in 142 days with an STS launch. A 10,000 km, 62.5° final orbit could be achieved in 88 days with a Titan IV launch. A spacecraft storage mission with power system deployment to a high altitude was also examined. The up leg required 126 days while the return required 16 days following an STS launch. The final mission, a round-trip OTV-type demonstration, achieves a HEO of 27,000 km at 28.5° in 97 days with return to NSO in 53 days assuming an STS launch.

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REFERENCES

1. J. F. Mondt, "Multimission NEP System for Outer Planet Exploration Missions," AIAA Paper 81-0698, April 1981.

2. R. P. Nagorski and R. J. Boain, "An Evaluation of Nuclear Electric Propulsion for Planetary Exploration Missions," AIAA Paper 81-0705, April 1981.

3. P. W. Garrison, "Advanced Propulsion for Future Planetary Spacecraft," Journal of Spacecraft and Rockets, Vol. 19, 1982, pp.534-538.

4. K. T. Nock and P. W. Garrison, "Nuclear Electric Propulsion Mission To Neptune," AIAA Paper 82-1870, November 1982.

5. W. M. Phillips, "Nuclear Electric Power System for Solar System Exploration," AIAA Paper 1337R, Journal of Spacecraft Rockets, Vol. 17, 1980, pp. 348-353.

6. E. F. Rutkowski and M. H. Kaplan, "A Nuclear Electric Transfer for Nuclear Waste Disposal," AIAA Paper 81-0706, April 1981.

7. W. D. Deininger and R. J. Vondra, "Electric Propulsion for Constellation Deployment and Spacecraft Maneuvering," AIAA Paper 88-2833, July 1988.

8. R. J. Vondra, K. Nock and R. M. Jones, "A Review of Electric Propulsion Systems And Mission Applications," Proceedings of the 17th International Electric Propulsion Conference, Tokyo, Japan, IEPC Paper 84-82, July 1984, pp. 600-613.

9. C. Selph and D. Perkins, "An Analysis of Electromagnetic Thrusters for Orbit Raising," Proceedings of the 17th International Electric Propulsion Conference, Tokyo, Japan, IEPC Paper 84-80, July 1984, pp. 580-589.

10. W. D. Deininger and R. J. Vondra, "Development of an Arcjet Nuclear Electric Propulsion System for a 1993 Flight Demonstration," AIAA Paper 86-1510, June 1986.

11. G. Aston, "Ion Propulsion Technology Requirements for Planetary Mission Applications," AIAA Paper 85-2000, Oct. 1985.

12. P. W. Garrison and K. T. Nock, "Nuclear Electric Propulsion (NEP) Spacecraft for the Outer Planet Orbiter Mission," AIAA Paper 82-1276, June 1982.

13. E. V. Pawlik and W. M. Phillips, "A Nuclear Electric Propulsion Vehicle for Planetary Exploration," AIAA Paper 76-1041, Journal of Spacecraft and Rockets, Vol. 14, 1977, pp. 518-524.

14. D. Q. King and L. K. Rudolph, "100 kWe MPD Thruster System Design," AIAA Paper 82-1897, Journal of Spacecraft and Rockets, Vol. 21, 1984, pp. 563-572.

15. W. D. Deininger and R. J. Vondra, "Design and Performance of an Arcjet Nuclear Electric Propulsion System for a Mid-1990s Reference Mission," (accepted for publication in Journal of Spacecraft and Rockets) AIAA Paper 87-1037, May 1987.

16. W. D. Deininger and R. J. Vondra, "Spacecraft and Mission Design for the Space Nuclear Power Source Reference Mission," (accepted for publication in Journal of Spacecraft and Rockets) AIAA Paper 87-2026, July 1988.

17. W. D. Deininger and R. J. Vondra, "Arcjet Nuclear Electric Propulsion (NEP) System for the SP-100 Reference Mission," JPL D-4739, March 4, 1988.

18. R. Wiley, "SDIO Space Power and Power Conversion," Nuclear Working Group/MHD Working Group Meeting, 15 April 1986, Dept. of Energy, Germantown, Maryland.

19. General Electric Corporation, "SP-100 Ground Engineering System (GES) Baseline System Definition and Characterization Study - Thermoelectric Power Conversion Study," Final Report, Vol. 1, Document 85SDS 4268, August 1985.

20. W. M. Zelinsky, "SP-100 Reference Mission Integration Study," Final Report, Aerospace Report No. TOR-0086A (2052-20)-2, Vols. 1-3, November 1986.

21. "Technical Specification for the SP-100 Space Reactor Power System (SRPS)", Revision 7, Approved: V. C. Truscello and E. J. Wahlquist, September 18, 1987.

22. J. F. Mondt, Jet Propulsion Laboratory, Private Communication, September 1, 1988.

23. R. H. Dawe, J. C. Arnett, E. R. Bunker, Jr., F. L. Lane, and J. C. Lewis, "High Voltage, High Current Rotary Switch Development - FY73," JPL Pub. 900-640, August 29, 1973.

24. R. J. Cassady, E. J. Britt, and R. D. Meya, "Performance Testing of a Lightweight 30 kW Arcjet Power Conditioning Unit," AIAA Paper 87-1085, May 1987.

25. S. Wong and E. J. Britt, "High Power Arcjet Power Conditioner," AIAA Paper 88-3101, July 1988.

26. A. Basiulus and C. J. Camarda, "Design, Fabrication and Test of Liquid Metal Heat-Pipe Sandwich Panels," AIAA Paper 82-0903, June 1982.

27. H. J. Tanzer, "High Capacity Honeycomb Panel Heat Pipes for Space Radiators," AIAA Paper 83-1430, June 1983.

28. T. K. Pugmire, "Flight Prototype Ammonia Storage and Feed System," Final Report, Avco Report No. AVSSD-0100-67-RR, January 1967.

29. General Electric Corp., "ATS F & G Prototype Ammonia Feed System Program Description, "G.E. Report SPPS-8-100, March 1968.

30. W. F. Krieve, F. L. Merritt, and R. Grabbi, "Zero Gravity Ammonia Propellant System," AIAA Paper 70-1151, September 1970.

31. B. Palaszewski, "Hydrogen-, Ammonia-, and Xenon-Propellant-Feed Systems," JPL Interoffice Memorandum (Internal Document), March 11, 1986.

32. L. C. Pless, "Vacuum Rated Flow Controllers for Inert Gas Ion Engines," AIAA Paper 87-1078, May 1987.

33. W.D. Deininger, "Electric Propulsion Produced Environments and Possible Interactions with the SP-100 Power System," AIAA Paper 85-2046, October 1985.

34. T. J. Pivirotto, D. Q. King and W. D. Deininger, "Long Duration Test of a 30-kW Class Thermal Arcjet Engine," AIAA Paper 87-1947, July 1987.

35. AVCO Corporation, "Thirty Kilowatt Plasmajet Engine Development/Third Year Development Program," First Quarterly Progress Report, RAD SR-63-207, NASA CR-85344, September 1963.

36. R. R. John, J. F. Conners, and S. Bennett, "Thirty Day Endurance Test of a 30 kW Arcjet Engine," AIAA Paper 63-274, June 1963.

37. R. R. John, S. Bennett, and J. F. Conners, "Arcjet Engine Performance: Experiment and Theory," AIAA Journal, Vol. 1, 1963, pp. 2517-2525.

38. J. S. Sovey, L. M. Zana, and S. C. Knowles, "Electromagnetic Emission Experiences Using Electric Propulsion Systems - A Survey," AIAA Paper 87-2028, July 1987.

39. L. L. Price and W. K. McGregor, "Spectral Characteristics of Low Density Arc Heated Nitrogen Plasma," AEDC-TR-77-23, March 1977.

40. A. C. Ducati, H. Humpal, J. Meltzer, E. Muehlberger, J. P. Todd and H. Waltzer, "1 kW Arcjet Engine System Performance Test," Journal of Spacecraft and Rockets, Vol. 1, 1964, pp. 327-332.

41. R. O. Bartlett, S. E. DeForest, and R. Gold stein, "Spacecraft Charging Control Demonstration at Geosynchronous Altitude," AIAA Paper 75-359, March 1975.

42. C. K. Purvis, R. O. Bartlett, and S. E. DeForest, "Active Control of Spacecraft Charging on ATS-5 and ATS-6," Proceedings of the Spacecraft Charging Technology Conference, edited by C. P. Pike and R. R. Lovell, NASA TM X-73537, 1977, pp. 107-120.

43. W. R. Kerslake and S. Domitz, "Neutralization Tests on the SERT-II Spacecraft," AIAA Paper 79-2064, November 1979.

44. R. C. Olson, "Modification of Spacecraft Potentials by Plasma Emission," Journal of Spacecraft and Rockets, Vol. 18, No. 5, Sept.-Oct. 1981, pp. 462-469.

45. R. C. Olson, "Modification of Spacecraft Potentials by Thermal Electron Emission on ATS-5," Journal of Spacecraft and Rockets, Vol. 18, No. 6, Nov.-Dec. 1981, pp. 527-532.

46. K. Kuriki, N. Kawashima, M. Sasaki, M. Yanagisawa, and T. Obayashi, "Space Experiment with Particle Accelerators (SEPAC) Performed in Spacelab First," AIAA Paper 85-1996, October 1985.

47. R. M. Jones, "A Comparison of Potential Electric Propulsion Systems for Orbit Transfer," AIAA Paper 82-1871, November 1982.

48. T. J. Pivirotto, D. Q. King, W. D. Deininger and J. R. Brophy, "The Design and Operating Characteristics of a 30 kW Thermal Arcjet Engine for Space Propulsion," AIAA Paper 86-1508, June 1986.

49. J. R. Brophy, T. J. Pivirotto and D. Q. King, "Investigation of Arcjet Nozzle Performance," AIAA Paper 85-2016, October 1985.

50. W. D. Deininger, T. J. Pivirotto and J. R. Brophy, "The Design and Operating Characteristics of an Advanced 30-kW Ammonia Arcjet Engine," AIAA Paper 87-1082, May 1987.

51. W. D. Deininger and T. J. Pivirotto, "Detailed Operating Characteristics of a 30 kW Ammonia Arcjet Engine with a Contoured Nozzle, "Proceedings for the SPIE Symposium on Innovative Science an Technology-Propulsion, No. 872, pp. 108-118, January 1988.

52. W. D. Deininger, A. Chopra, D. Q. King and T. J. Pivirotto, "Thermal Design Improvements for 30 kW Arcjet Thrusters," IEPC Paper 88-073, October 1988.

53. R. W. Klemetson, Jet Propulsion Laboratory, Private Communication, August 1988.

54. Titan IV Users Handbook.

55. R. Manvi, "Recommendations on SP-100 Survival in the Debris and Meteoroid Environment," JPL Internal Document IOM 354-SP-100-87-001 to C. Shinbrot, May 18, 1987.

56. Martin Marietta, Denver Aerospace, "Space Power Architecture Study," OTV Section, Technical Report MCR-86-613, Task 1; Requirements Definition, September 2, 1986.

57. R. E. Oglevie, T. R. Egan and J. P. Penn, "Autonomous Flight Control for Low Thrust Orbit Transfer Vehicles," AIAA Paper 88-2838, July 1988.