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MISSION OPTIONS FOR AN ELECTRIC PROPULSION DEMONSTRATION FLIGHT TEST

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ABSTRACT

Several mission options are discussed for an electric propulsion space test which provides operational and performance data for ion and arcjet propulsion systems and testing of APSA arrays and a Super power system. The results of these top-level studies are considered preliminary. Ion propulsion system design and architecture for the purposes of performing orbit raising missions for payloads in the range of 2400-2700 kg are described. Focus has been placed on a design which can be characterized by simplicity, reliability, and performance. Systems of this design are suitable for an electric propulsion precursor flight which would provide proof of principle data necessary for more ambitious and complex missions.

INTRODUCTION

Many possible JPL and military missions benefit from the propellant savings obtained using ion or arcjet electric propulsion. These missions include Asteroid Rendezvous,[1,2] Luner Base Development,[3] Lunar Get Away Special,[4,5] Comet Nucleus Sample Return (CNSR),[6] SP-100,[7] and emplacement of global positioning satellites and communications satellites. The benefits to these missions are increased if satellites and other payloads are developed which make use of the electric power that is required to operate the arcjets or ion engines.

Despite having significant advantages and benefits, electric propulsion systems are still not employed because of high development costs and the perceived technical risk to the first users of electric propulsion. These concerns can be mitigated by an electric propulsion flight demonstration which tests both arcjet and ion propulsion systems.

A flight demonstration of electric propulsion will provide the information necessary to quantify propulsion system operating data such as plume-solar array interactions, plume-communications interactions, spacecraft and sensor charging and contamination, EMI, navigation, attitude control of large structures such as solar arrays, autonomous computer operation of the space test, engine operating life, and overall system operation and integration. The results of an electric propulsion demonstration flight may be reduced development costs and reduced technical risk to the first users of electric propulsion.

This paper discusses three options for such a demonstration flight. In all options a multi-engine ion and arcjet propulsion system are tested for significant fractions of their expected operating life. In addition, the mission options provide for testing of APSA solar arrays and the SUPER power system. The final mission option provides significant science return by delivering a satellite with a diagnostics payload to a polar lunar orbit.

SYSTEM OVERVIEW

The payload system overview is depicted in Figure 1. The payload system and respective subsystem masses are listed in Table 1. The spacecraft subsystem mass allowances are representative of existing flight hardware except for the power and electric propulsion subsystems. The overall spacecraft design is simply a brief first-cut estimate. The mission analysis was performed assuming that the payload system would be stowed in a Delta II 7920 launch vehicle and placed in a 300 km circular orbit.[8] For all mission options, this results in a launch margin of 25% or more. A brief discussion of some of the major subsystems follows.

Figure 1. Overview of Spacecraft

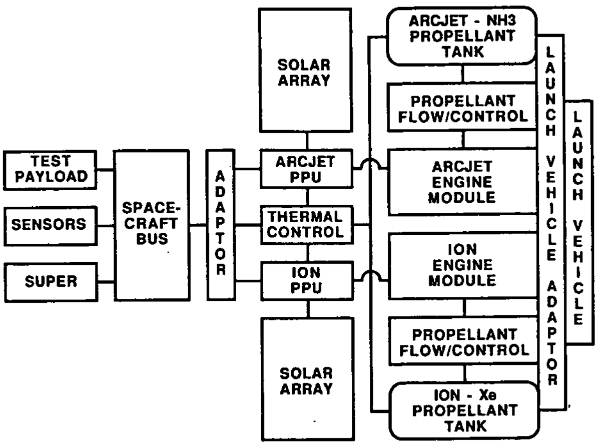


TABLE I. PAYLOAD SUBSYSTEMS AND SUBSYSTEM MASSES

SUBSYSTEM ELEMENT	MASS kg
Super power system demo	250
Sensors	150
Spacecraft Bus	600
APSA solar array	300
Contingency	225
Ion propulsion module:	
4 engines per module	105
Propellant flow control	35
25% ion system contingency	35
Ion system PPU (4 ea @ 4 kg/kw)	80
Arcjet engine module:	
2 engines per module	44
Propellant flow control	23
50% arcjet system contingency	33
Arcjet system PPU (2 ea @ 2 kg/kw)	120

Total system dry mass excluding propellant, tanks, and adapters	2000
Xenon ion propellant tank	0.22Mxe
Ammonia arcjet propellant tank	0.15MNH3
E.P. Payload Adaptor	0.03Mpayload
Launch vehicle adaptor	0.03(Spacecraft mass)

ION PROPULSION SYSTEM

The ion propulsion module consists of a 4-engine array combined with the required gimbals, xenon propellant flow and distribution, and power processors (PPU). Each xenon ion engine has one PPU for power conditioning. PPU specific masses were estimated using previous studies,[9] and allowing for new component development and system design.[10] Only three of the ion engines would be used during any mission and only two engines would be operated at any time. The fourth engine would serve as a spare.

Ion propulsion systems are being developed by NASA[11-14] with near term goals of developing engines with operating powers of 5-10 kW and operating life times of 10000-15000 hours.[15,16] However, for a 1994 launch time frame it is expeditious to flight qualify the engines for an operating life of no greater than 4380 hours (6 months). The missions described in this paper require ion engine operating life times of no greater than 4286 hours; this limited operating life greatly reduces required life test duration and better allows for technology development for all aspects of the ion propulsion system.

The ion engines in these systems can be operated at constant or variable power, but in the interest of increased system simplification and reliability the ion engine propellant flow rates, discharge current, and beam current are configured to be operated at constant values. Ion propulsion system operation at constant discharge current, beam current and propellant flow rates may simplify engine life testing because the engines would be operated at only a single operating condition as opposed to a spectrum of power and discharge chamber conditions.[1] Operation at constant power may also simplify PPU design. If required, the ion engines may be throttled over a limited power range by still maintaining constant propellant flows, engine discharge current and beam current and varying the beam voltage alone. However, the solar array has been sized such that a constant value of 10 kW power is provided to the ion engine PPUs (5 kW each PPU), so that engine throttling should not be necessary.

The ion propulsion system performance parameters are listed in Table II. Performance parameters listed are for J-series thrusters that have already been developed but require life time enhancement or additional life testing. Ring-cusp ion engines presently under development by NASA[16] may also be used, with slightly different performance parameters than those listed in Table II.

TABLE II. ION PROPULSION SYSTEM PERFORMANCE PARAMETERS

THRUST Nt	Isp sec	ENGINE EFF %	BEAM CUR Amps	BEAM VOLTAGE Volts	ENERGY COST ev/ion	DISCHARGE CURR Amps	DISCHARGE VOLTG Volts	TOTAL SYSTEM EFF %
0.17	3504	0.70	3.33	1100	180	26	28	0.63

The propellant tank, control and distribution masses were obtained using previous studies[11,17] and by adding the combined weights of all flow components.[11] A representative propellant flow control system is depicted in Figure 2. The flow system is designed to provide constant mass flows to the engine discharge plasma (main), engine cathode, and neutralizer cathode. The advantages this propellant flow control system has over a variable flow system are

(1) operation of the high voltage isolators at high pressure, thus minimizing the chance of HV breakdown across the isolator.

(2) the use of fixed flow orifices instead of propellant flow controllers; this is important as there are no space qualified flow controllers,[18] and their removal from the propellant flow system results in the flow system having only two moving parts, the pressure regulators and the latch valves.

ARCJET PROPULSION SYSTEM

Arcjet propulsion systems for stationkeeping and primary propulsion are under development by NASA,[19] the Air Force[20] and SDI.[7] Operating powers are baselined at 30 kW with operating life goals of 1500 hours. In this study, however, the arcjets are configured to be operated at a constant power of 25 kW. A reduction of 16% in the arcjet operating power requirement provides some engine technology development margin, reduces the total solar array mass, and permits partial degradation of the solar arrays without impacting arcjet engine operating power. Since there are no batteries in the launch payload the arcjet must endure many on-off cycles as it transits through the earth's shadow to higher orbits.

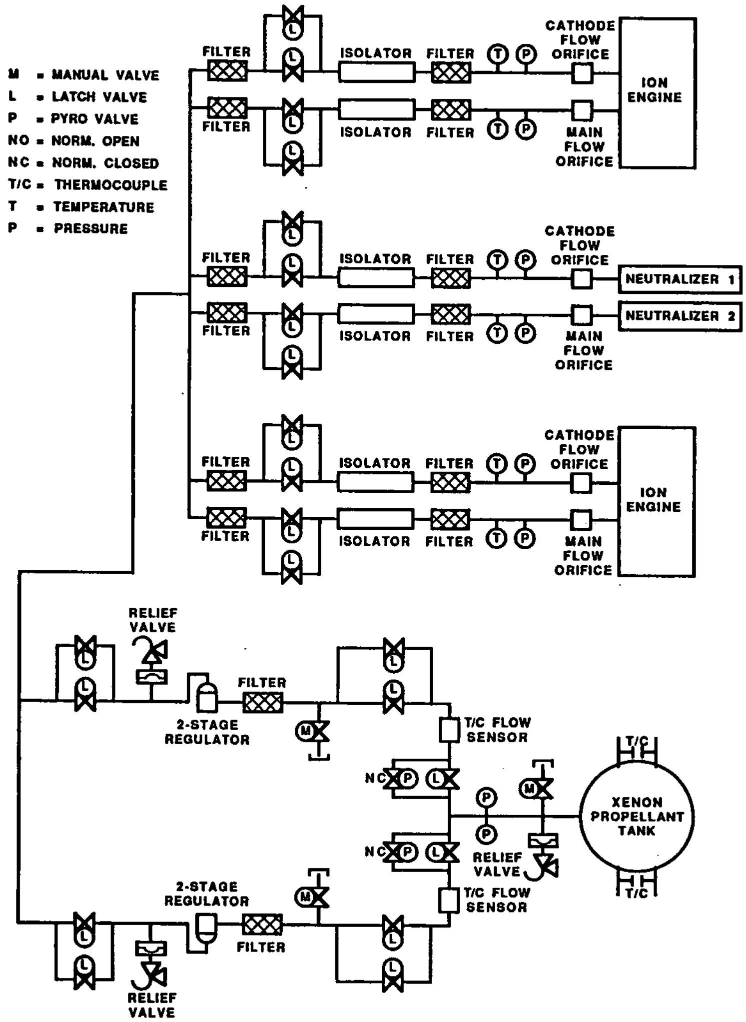
APSA SOLAR ARRAY

The Advanced Photovoltaic Solar Array (APSA) program is being developed at JPL under NASA OAST sponsorship.[21] This program has a design goal of developing an 8-12 kW array with greater than 130 W/kg (7.7 kg/kW) BOL specific mass and 100 W/kg (10 kg/kW) EOL specific mass after 10 years at GEO. The array design consists of thin (63 um) silicon cell modules and a canister deployed continuous longeron lattice mast system. Because of the thin silicon cell design array degradation should not exceed 12% at an altitude of 4000 km.[22] The BOL specific mass of 7.7 kg/kW, which includes a 10% contingency, does not include components outboard of the solar drive unit. Because of uncertainties in the final APSA system mass a conservative value of 10kg/kW is used as the APSA BOL specific mass.

DIAGNOSTICS PACKAGE

The diagnostics package would contain instruments to assess the effects of the electric propulsion system on sensors, instrumentation, and spacecraft operations. Science payloads for the mission to a polar lunar orbit would include instrumentation to provide a geological mapping of the lunar surface. The launch and payload contingencies may also permit the inclusion of some science instruments in the other missions if the impacts of these instruments on spacecraft mass and system integration are low.

Figure 2. Xe-ION Engine Propellant Feed System Schematic



SUPER DEMO

The solar powered Super power system is being developed with a goal of achieving survivability under severe radiation and kinetic environments. The Super demo package would consist of a portion of a Super solar array and the associated power subsystems. This demo package would serve as a test payload only and would not supply power to the ion or arcjet power processors. The demo package would permit testing of the Super power system in orbits that range from 300-35744 km.

MISSION ANALYSIS

There are many possible missions suitable for an electric propulsion flight test. The three mission options presented here utilize arcjets and ion engines for orbit transfer and were selected because they enable the study of a mix of desirable tests, including electric propulsion system operation, plume and EMI interactions, thermal control, solar array degradation at orbits between 300-35744 km, and lunar science. These missions can be summarized as

I. Orbit transfer to GEO altitude.
II. Orbit transfer to 20000 km and return to 300 km for shuttle retrieval
III. Orbit transfer to lunar orbit

A first order mission analysis of these three mission options was performed using the following assumptions:

(1) Payload is inserted into a 300 km circular orbit using a Delta II 7920 launch vehicle.
(2) Launch payload capability for this orbit altitude is 4830 kg.
(3) Maximum allowable payload for the electric propulsion space test is 3623 kg, which allows a 25% launch mass margin.
(4) Orbit inclination is 28.5 degrees with no plane changes.
(5) Arcjet is operated at a constant power of 25 kW, and ion engines are operated at a constant power of 5 kW each engine, with only two engines operated at any time.
(6) Thrusting occurs only in sunlight.

RESULTS AND DISCUSSION

I. ORBIT TRANSFER TO GEO ALTITUDE

The objective of this mission option is to demonstrate electric propulsion orbit transfer to a GEO altitude of 35744 km. In this mission option a 30kW BOL solar array is used to power a 25kW arcjet at constant power to an altitude of 2000 km; the arcjet is then turned off and an ion engine pair is used to power the spacecraft to an altitude of 35744 km. The arcjets are tested first to make use of the solar array power before significant solar array degradation occurs. The expected APSA array degradation during operation of the arcjet to a 2000 km orbit is not expected to be severe, due to the radiation environment and the thin silicon cell design, and hence it is anticipated that the arcjet can be operated at constant power. Table III lists the results of this mission analysis.

The mission option discussed here enables the testing of the small Super power system at altitudes from 300-35744 km, which represents more than the full range of altitudes required for survivability of the Super power system. In addition, data can be obtained on APSA array degradation in the peak radiation region of 10000 km.[5,22] The arcjet burn time of 309 hours represents a significant fraction of the expected operating life of 1500 hours. A spare arcjet engine provides redundancy.

The ion propulsion system would be used to raise the spacecraft to a 35744 km orbit. It is expected that at GEO solar array power will be a minimum of 12 kW. Two ion engines will be operated simultaneously at a constant 5 kW for a total of 10 kW. A total of three engines would be used for the mission, with one engine serving as a spare. The required operating life for each of the three ion engines is only 2317 hours. If one or both of the ion engines in the first ion engine pair that are used operate without failure for the duration of the mission, life times of 3476 hours, or 79% of the flight-qualified life, can be demonstrated with a high degree of redundancy.

TABLE III. ORBIT TRANSFER TO GEO ALTITUDE

ENGINE TYPE	INITIAL ORBIT KM	FINAL ORBIT KM	WET SPACECRAFT MASS KG	PROPELLANT USED KG	BURN TIME HOURS	TRIP TIME DAYS
NH3 ARCJET	300	2000	2652	250	309	19
XENON ION	2000	35744	2402	267	6952	338

II. VAN ALLEN BELT TRANSFER AND RETURN TO LEO

This mission option employs an arcjet to boost the payload to a 3900 km orbit, and an ion propulsion system to raise the spacecraft to a 20000 km orbit and return the spacecraft to 300 km for retrieval by the space shuttle. The aspects of this mission option which differ significantly from the first option discussed are the longer propulsion system burn times and return of the spacecraft and propulsion systems for physical inspection. The results of the mission analysis are shown in Table IV.

The arcjet burn time of 674 hours represents 45% of the full arcjet operating life of 1500 hours. Also significant is the large number of orbits, estimated to be 478, which requires the arcjet to start and stop hundreds of times, a significant engine and system issue.

With three ion engines used (two at any time) for the mission, the required operating life to perform this ambitious mission is only 3829 hours, or 87% of the flight-qualified operating life. If one of the ion engines used does not fail for the duration of the mission, lifetimes of 5744 hours, or 131% of the flight-qualified operating life can be demonstrated.

TABLE IV. VAN ALLEN BELT TRANSFER AND RETURN TO LEO

ENGINE TYPE	INITIAL ORBIT KM	FINAL ORBIT KM	WET SPACECRAFT MASS KG	PROPELLANT USED KG	BURN TIME HOURS	TRIP TIME DAYS
NH3 ARCJET	300	3900	3205	545	674	40
XENON ION	3900	20000	2181	162	4222	204
XENON ION	20000	300	2154	280	7265	381

The return to LEO for shuttle retrieval adds a significant new dimension to the space qualification test. The Super power system, other test payloads, sensors, the APSA array, and the electric propulsion systems can be inspected to determine the exact causes for failures which may occur. Plume interactions, which may be evident from sensor readouts, can be noted by observing film deposits and evidence for arcing. Finally, this mission demonstrates the ability for electric propulsion to deliver heavy payloads to half-GEO orbits.

III. TRANSFER TO LUNAR ORBIT

This final mission option again employs an arcjet to boost the satellite to a 3900 km orbit. The arcjet is then turned off and the ion propulsion system is used to thrust the spacecraft to a 100 km polar lunar orbit. The ion propulsion system is used to increase orbit altitude and for polar lunar orbit insertion. Virtually all of the lunar surface can be studied to obtain a thorough lunar geological survey. Table V summarizes the results of this mission analysis.

The arcjet burn time is almost identical to the burn time presented in the previous option. By utilizing three ion engines (two at any time) the required burn time per engine is 4286

TABLE V. ORBIT TRANSFER TO LUNAR POLAR ORBIT

ENGINE TYPE	INITIAL ORBIT KM	FINAL ORBIT KM	WET SPACECRAFT MASS KG	PROPELLANT USED KG	BURN TIME HOURS	TRIP TIME DAYS
NH3 ARCJET	300	3900	3280	556	689	41
XENON ION	2000	LUNAR POLAR	2724	495	12857	603

hours, or 98% of the expected space-qualified operating life of the engines. However, if one of the ion engines used does not fail for the duration of the mission, operating times of 6429 hours can be demonstrated even though the engines are flight qualified for only 4380 hours.

CONCLUSIONS

Three options for an electric propulsion demonstration test were studied. In all options both arcjet and ion propulsion systems are operated for significant fractions of their expected space-qualified operating life. These missions also permit testing of the APSA array, a small Super power system, and other payloads in orbital altitudes where solar array degradation is expected to be significant. Both ion and arcjet systems are operated at constant power and propellant flow rates to simplify system design and life testing. The second option presented enables the return of the spacecraft to a shuttle-retrievable orbit to permit physical inspection of all systems. The third option presented enables significant lunar science return by placing the spacecraft in a 100 km polar lunar orbit.

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