N87-17789

POWER SYSTEM TECHNOLOGIES FOR THE MANNED MARS MISSION

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ABSTRACT

The high impulse of electric propulsion makes it an attractive option for manned interplanetary missions such as a manned mission to Mars. This option is, however, dependent on the availability of high energy sources for propulsive power in addition to that required for the manned interplanetary transit vehicle. Two power system technologies are presented here. They are the nuclear and solar technology options. The ion thruster technology for the interplanetary transit vehicle is described for a typical mission.

The power management and distribution system components required for such a mission must be further developed beyond today's technology status. High voltage-high current technology advancements must be achieved. They are described in this paper. In addition, large amounts of waste heat must be rejected to the space environment by the thermal management system. Advanced concepts such as the liquid droplet radiator are discussed as possible candidates for the manned Mars mission. These thermal management technologies have great potential for significant weight reductions over the more conventional systems.

POWER/ENERGY SYSTEMS

Nuclear Power System (megawatt class)

Reference 1 describes a nuclear power system for a manned interplanetary spacecraft utilizing electric propulsion. This power system consists of a lithium cooled-uranium nitride fueled reactor. The characteristics for this power system are given in Table 1. This system represents technology which should be operational in the year 2000. A typical spacecraft configuration utilizing this power system is shown in Figure 1. The 300 foot separation distance between the power plant and the manned modules is required for crew radiation isolation.

A schematic of the nuclear power/energy conversion system is given in Figure 2. On this Figure are shown the various states throughout the system. Table 2 shows the weight breakdown of this system.

TABLE 1

Nuclear Turboelectric Indirect Rankine System

Lithium Cooled Reactor

(Reference 1)

Reactor Power	34.3 MW_{THER} @ 1510 K Outlet Temperature
Net Electric Power	$5MW_e$
Turbine Inlet Temperature	1450^o K
Turbine Efficiency	80%
Number of Stages	5
Inlet Pressure	209 PSI
Exit Pressure	27.5 PSE
Condenser Temperature	1100^o (Exit Quality 88%)
Cycle Efficiency	16.8%
Overall Efficiency	14.6%

TABLE 2

Nuclear Power System Weights (KG)

Reactor	3900 KG
Shield	30000 (Man Rated)*
Reactor Pump	1750
Boiler	2400
Turbogenerator	3300
Feed Pump	1250
Condenser	3000
Radiator	8700 (535 M^2)
Power Conditioner	2000
Miscellaneous (Structure, Etc.)	2900
Total Weight in LEO	59200 KG

*Man Rated: 13 rem total integrated mission dose

This nuclear power system concept results in a specific weight of 12 KG/KW_e.

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FIGURE 1: SPACECRAFT CONFIGURATION

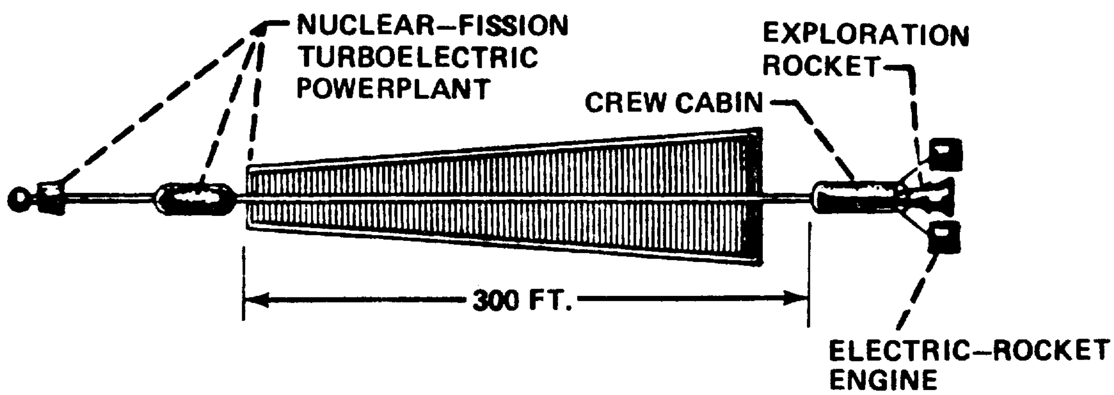
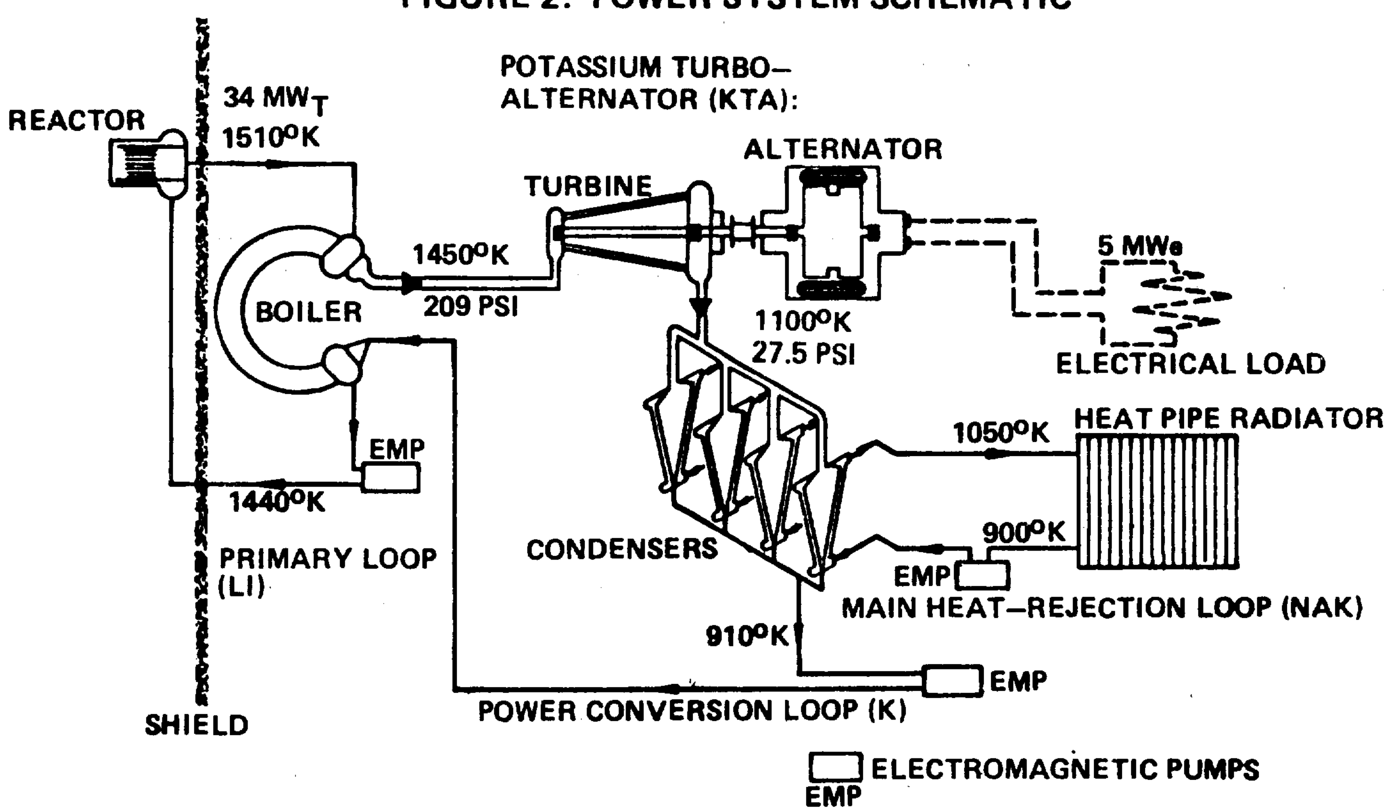


FIGURE 2: POWER SYSTEM SCHEMATIC



Technology advances projected for the nuclear power systems for the early part of the 21st century indicate a possibility of reducing this weight by a significant amount. Reactor designs employing cermet fuels, boiling liquid metal cooled reactors, improved turbo-alternator machinery, and advanced heat rejection concepts offer potential for substantial weight savings over the technologies which make up the design shown here. Projections of achieving specific weights of 7 KG/KW_e for manned nuclear systems by the first part of the century have been made, and appear attainable.

<u>Other Power Options</u>

The possibility of using other non-nuclear power sources for the manned Mars mission was also addressed.

<u>Solar Photovoltaic Power Systems</u>

Photovoltaic system energy densities of 300 Watts/KG, at 1 AU, are projected to become available at the early part of the 21st century. Such systems, when <u>sized for low Mars orbit</u> (1.52 AU) and with regenerative fuel cell storage for shadow period operation, appear to indicate specific weights on the order of 25-30 KG/KW, depending on the system hardware parameters assumed. However, special mission designs, wherein partial or no electric propulsion thrusting would take place during the Earth or Mars shadow phases could significantly reduce this figure by eliminating the weight penalty associated with providing shadow period operation. Technology advances beyond the 300 Watts/KG and special trajectory design could possibly make solar photovoltaic systems weight competitive with nuclear systems. This would come about, however, at a penalty in transit trip time. Also, the enormously large areas of megawatt-size photovoltaic systems would pose an additional problem.

<u>Solar Thermal Dynamic Systems</u>

These systems were also investigated for application to the manned Mars mission. However, even with optimistic projections for techonolgy advances, these systems do not seem weight competitive for those missions/orbits where extended shadow period operation is required. If the weight penalty associated with shadow period operation (storage) can be eliminated (or reduced), these systems may be competitive for the Mars mission.

ION THRUSTER PROPULSION SYSTEM

The interplanetary phase (trajectory) of the manned Mars mission has not yet been designed. However, in order to carry out a preliminary evaluation of possible thruster technologies, a representative mission scenario has been selected (Ref. 2). One possible thruster system which shows considerable potential for application to this mission is an outgrowth of the 30-cm ion thruster technology (J-series mercury ion thrusters), developed by the Lewis Research Center. This thruster has been investigated for application to that performance range required for future interplanetary missions, such as the manned Mars mission. In addition, the technical maturity of this technology makes it a prime candidate for this mission. The interplanetary trajectory analysis for the manned Mars mission carried out in Ref.2 used this technology to derive thruster performance data. A cross section of the 30-cm mercury ion thruster is shown in Figure 3.

The design level performance of the 30-cm J-series thruster is shown in Table 3. This shows the extended performance (demonstrated) and the required performance if this thruster were to be used for the manned Mars mission.

TABLE 3

30-cm J-Series Thruster Parameters

	Design Performance*	Extended Performance*	Manned Mars Mission Requirements
Beam Current	2.0 A	7.9	13.3
Beam Ion Prod. Cost	192 W/A	220	150
Specific Impulse	3000 sec	4880	4665
Thrust, N	0.13 N	0.15	1.20
Thruster Effic.	72.3%	71.2	69.5
Thruster Input Pow.	2650 W	17210	38300
Propellant	Mercury	Xenon	Mercury
Lifetime	25000 hrs	-	12.500

*(Demonstrated by Test)

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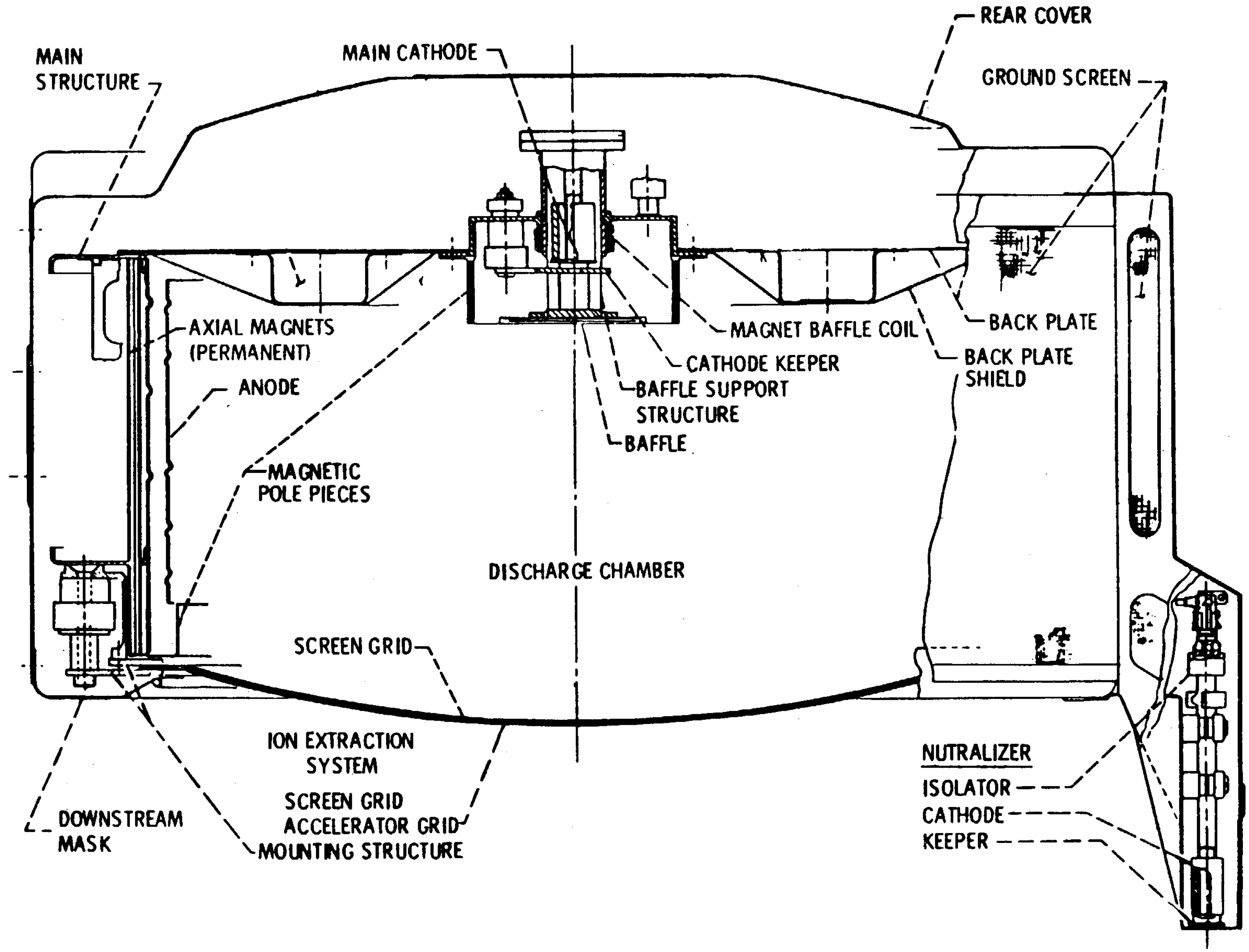
FIGURE 3. J–SERIES 30 cm MERCURY ION THRUSTER



In carrying out the analyses for the manned Mars mission, thruster system characteristics were also developed for 50-cm and 100-cm ion thrusters. Both mercury and xenon propellants were considered at specific impulses associated with 3000, 4000, and 5000 total volts across the accelerator grids. Operation of the J-series thruster with xenon and other inert gases has been demonstrated (Ref.3). The performance is comparable to that documented with mercury propellant.

There are critical technology advances which must be made before this technology can be incorporated into a flight system. The major developments required are:

Discharge Chamber: The present magnetic field and chamber design of the J-series thruster may require minor modifications to reduce losses to the level used in the analyses. However, recent research at Lewis (Ref.4) has demonstrated a 30-cm xenon ion thruster operating at losses below this level.

Accelerator System: Operation of the J-series accelerator system at 3000 v total has been demonstrated. However, higher voltages will require technology levels beyond those of present J-series accelerator grid materials and fabrication procedures (Ref.5). Further, although thrusters of up to 150-cm dia have been successfully designed and operated (Ref.6), dished-grid optics of the J-series type have not been fabricated for thrusters larger than 30 cm.

Hollow Cathode: The present J-series cathodes are designed for operation up to 20 A emission current. However, recent tests at Lewis demonstrating an extended performance of the J-series thruster (Table 3) have operated a modified cathode design at up to 50 A emission current. The systems analyzed would require cathode operation ranging from a low of 80 A for the 30-cm ion thruster scenario (Isp = 4665) to a high of 2800 A for the 100-cm ion thuster (Isp = 7400). These levels may be achieved by operating a larger cathode, or by operating a multiple cathode arrangement. This is a technology that also requires demonstration.

The thruster technology required for the Mars mission scenario at 4665 seconds specific impulse is considered near-term.

Power Conditioning Subsystem

The AC Power Management and Control approach used in this study is that presented in Ref.5. This includes the methodology employed and assumptions made to define the thrust system characteristics in this analysis.

System Parameters

Figures 4 to 9 present thrust system parameters as a function of specific impulse. Figure 4 presents total number of thrusters in the propulsion system as a function of specific impulse for the 30-, 50-, and 100-cm thrusters, for both mercury and xenon propellants. Because the ratio of net to total accelerator voltage in fixed at .9 to maintain a high specific impulse, the mercury propellant systems require less total thrusters to attain the mission specified thrust, at a specific impulse. Because the mission parameters are fixed, increasing the thruster diameter decreases the total number of thrusters required to accomplish the mission. Aside from the near-term technology capability of the 30-cm thruster, the 100-cm thruster technology would represent a significant reduction in system complexity through a reduction in component count.

The thrust system characteristics are relatively insensitive to thruster size (for the specified mission profile): figures 5-8 present thrust system parameters for the 100-cm thruster technology. Figure 5 presents overall propulsion system total mass (including propellant) as a function of specific impulse. The system mass decreases strongly with increasing specific impulse due to the reduction in propellant mass. The propulsion system dry mass is relatively insensitive to changes in specific impulse, as indicated in Figure 6. This is due to the competing facts that the power supply and thermal system masses increase with specific impulse, while the thruster, tankage, and support structure masses decrease with increasing specific impulse. Figure 7 presents the total payload mass (power and nonpower) as a function of specific impulse. The payload mass (the final return mass less the propulsion system dry mass), is relatively insensitive to changes in specific impulse for the region investigated.

Propulsion system total efficiency vs. specific impulse is presented in Figure 8. The system efficiency, about 67%, is relatively insensitive to specific impulse and propellant type. This reflects the near-constant

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FIGURE 4. TOTAL NUMBER OF THRUSTERS VS. SPECIFIC IMPULSE
CASE I–VI

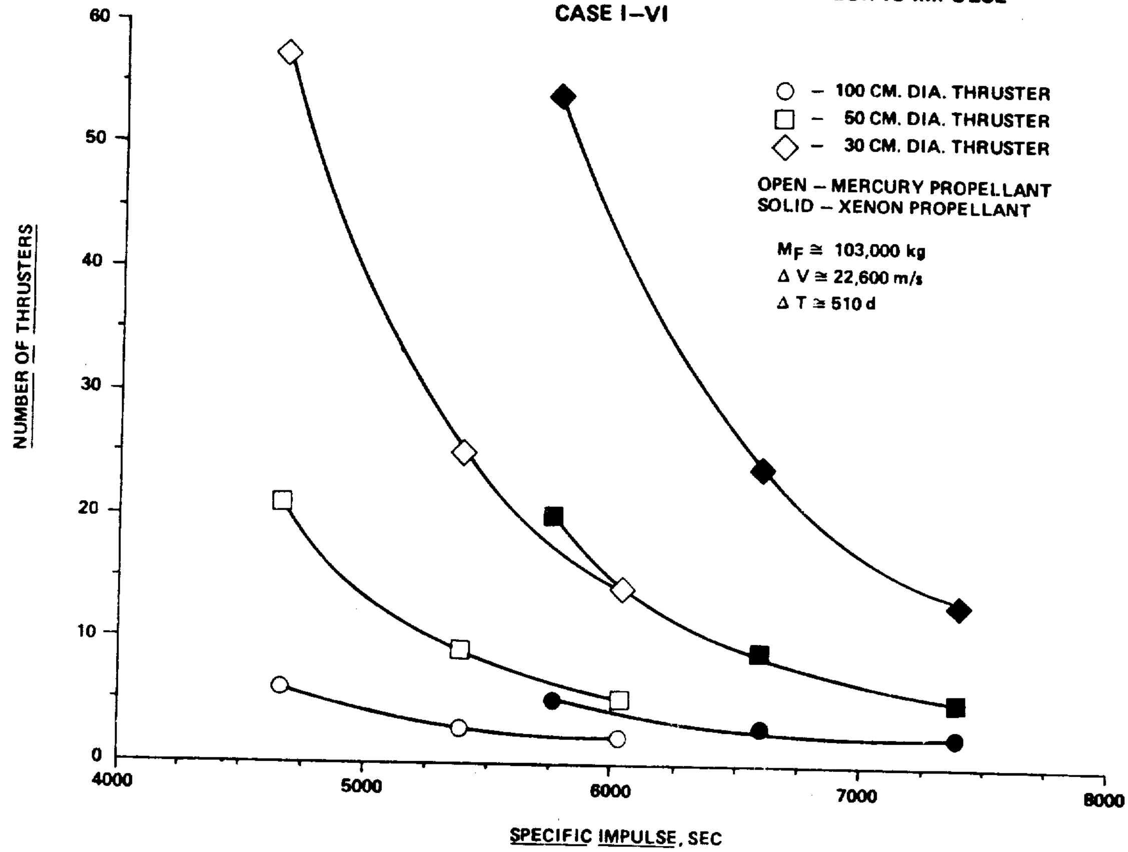

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FIGURE 5. TOTAL PROPULSION SYSTEM MASS VS. SPECIFIC IMPULSE
CASE I–II

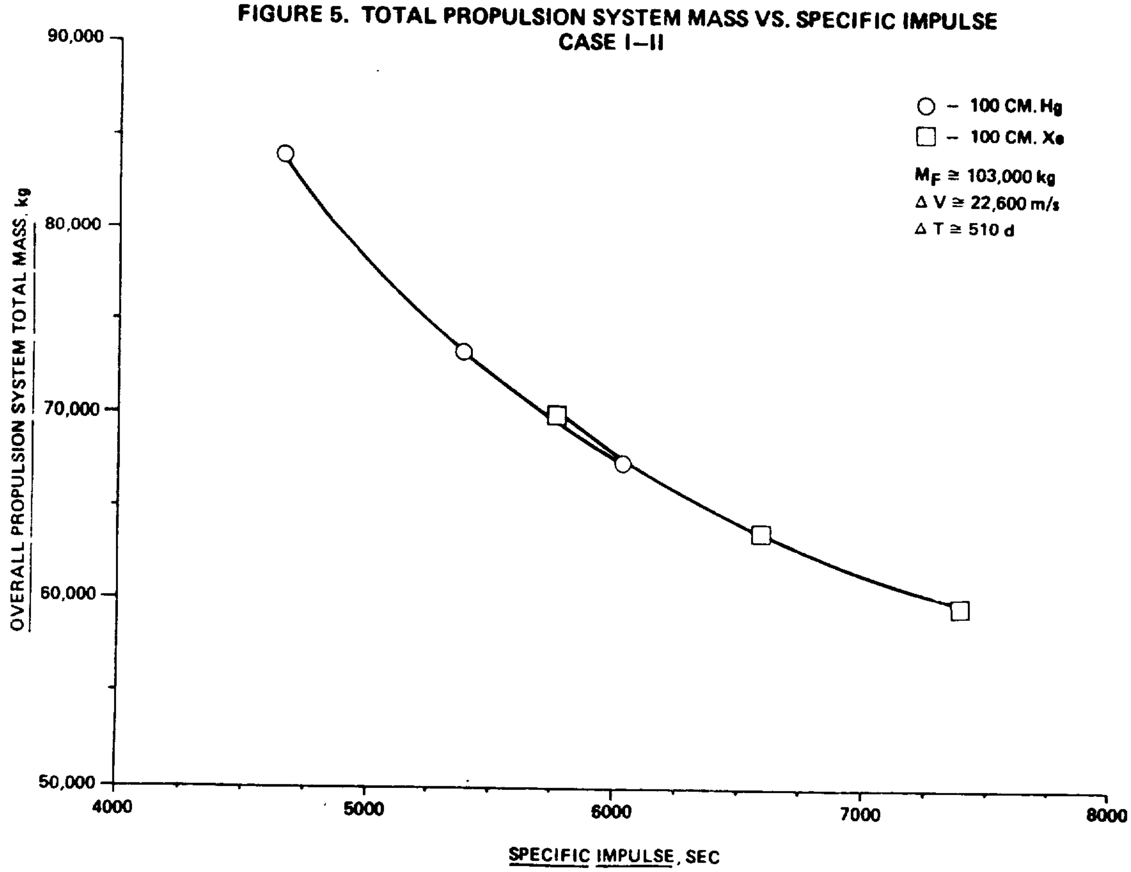

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FIGURE 6. PROPULSION SYSTEM DRY MASS VS. SPECIFIC IMPULSE
CASE I–II

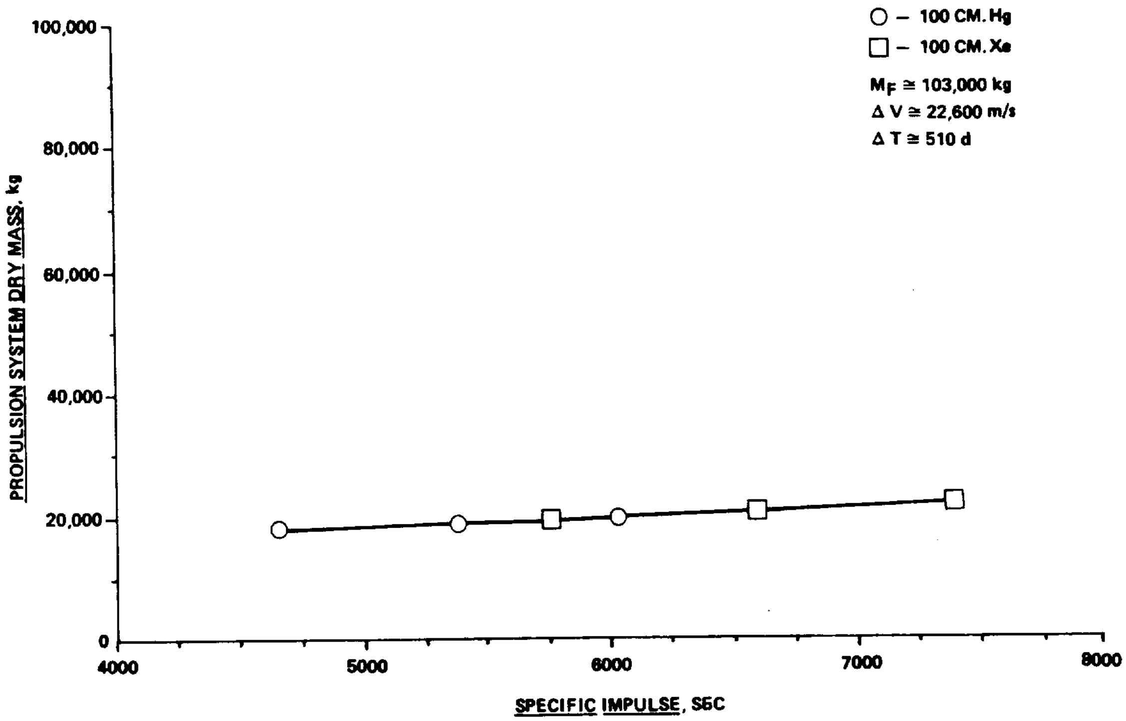

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FIGURE 7. TOTAL PAYLOAD MASS VS. SPECIFIC IMPULSE
CASE I–II

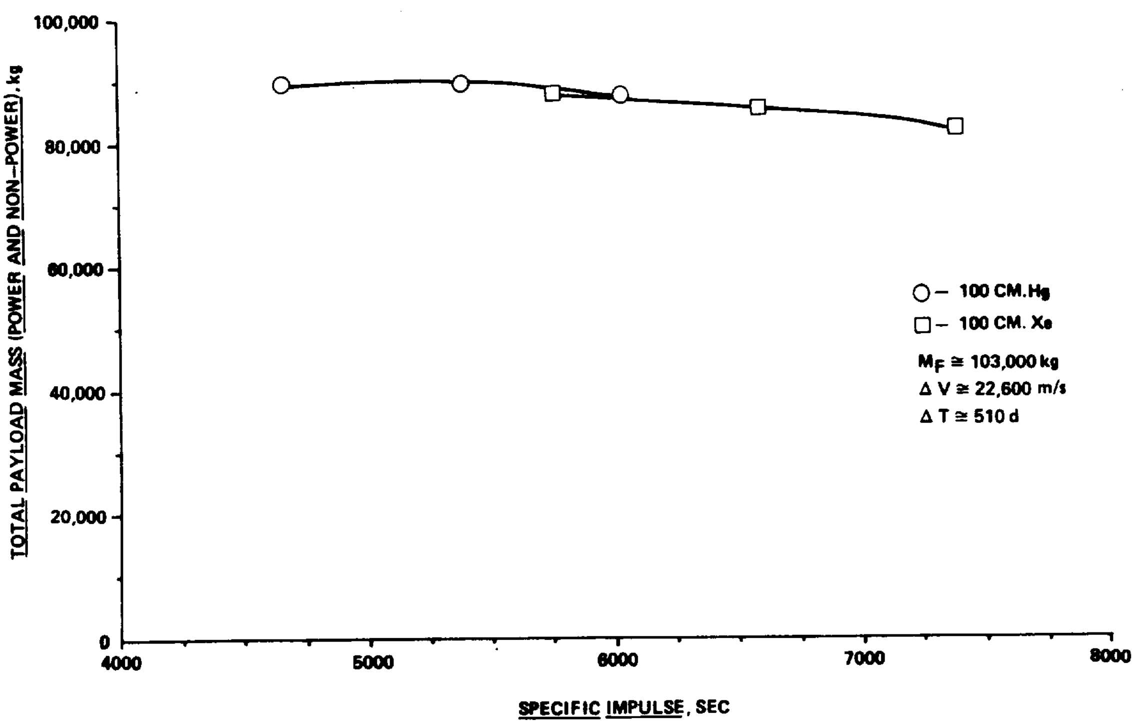


FIGURE 8. PROPULSION SYSTEM TOTAL EFFICIENCY VS. SPECIFIC IMPULSE
CASE I–II

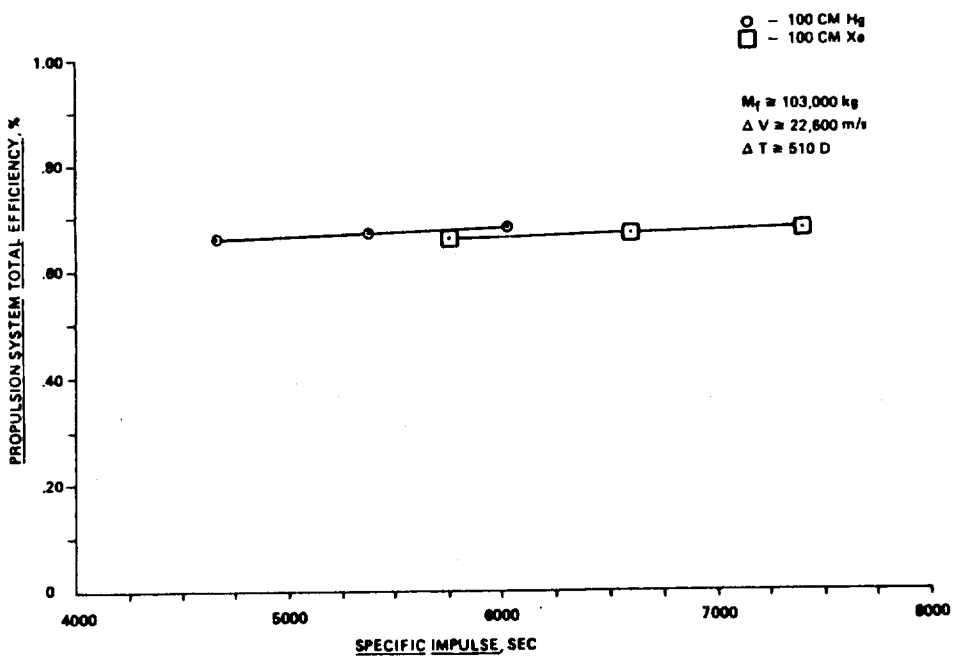
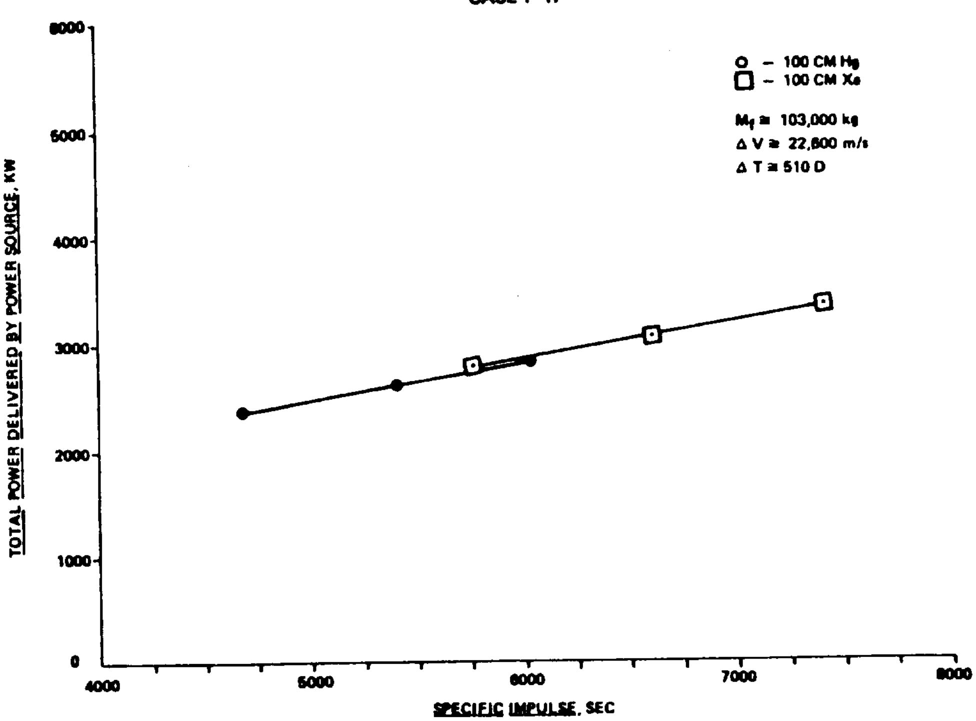


FIGURE 9. TOTAL POWER VS. SPECIFIC IMPULSE
CASE I–II



thrust-to-power ratio for a wide range of specific impulse. Figure 9 presents the system power as a function of specific impulse. Because the mission parameters were fixed, the system power increased by approximately 30% from 2370 KW to 3340 KW as the specific impulse was varied from 4665-7400 seconds.

POWER MANAGEMENT AND DISTRIBUTION TECHNOLOGY

The multimegawatt power requirements of a manned Mars mission will require the development of high power-high voltage-high current power conditioning, transmission and distribution systems. Power requirements have been increasing at a rapid pace since the early 70's when space power requirements could be fulfilled by systems of 1 KW or less. Figure 10 shows the increase which space power systems have experienced over the years. The manned Mars mission will require power electronic components in the multimegawatt range to be ready by the turn of the century.

The basic power requirement for the manned Mars mission is to the ion thruster with the beam supply being the predominant user at 3000, 4000, or 5000 volts, depending upon the thruster concept used. Since the mission and spacecraft design have not as yet been determined, the exact power requirements are not known. However, Table 4 was constructed to give an estimate of the mission power requirements.

TABLE 4

MARS MISSION POWER REQUIREMENTS

Housekeeping Power	100 KW
Low Voltage Thruster Power	200 KW
High Voltage Thruster Power	3000 KW

The mass of the power management and distribution system will depend upon the actual spacecraft and mission design and thruster concept used. For the concentrated loads, such as the ion thruster beam supply, a projection of present high frequency technology to the early 2000's results in a specific weight of approximately 1 KG/KW. The same projection for the low voltage power supplies for the ion thruster may be in the 5 KG/KW range. Due to the smaller power range and higher complexity, the housekeeping power, with provision for emergency operation and some storage capacity, will be closer to a specific weight of 50 KG/KW. Transmission line weights for a 150 meter line at 3000 volts are expected to be on the order of 3500 KG. It is estimated that the losses incurred in the power management and distribution system are on the order of 500 KW.

TABLE 5

POWER MANAGEMENT AND DISTRIBUTION SYSTEM WEIGHT

Housekeeping Power	5000 KG
Low Voltage Ion Thruster Power	1000
High Voltage Ion Thruster Power	3000
Transmission Line	3500
TOTAL WEIGHT	12500 KG

As indicated in Figure 10, about 4 years are required to advance the state of the art in power management and distribution system components by one decade of power. At the present time - 1985, the power technology level for these components is less than 100 KW. To arrive at the required level of 3000 KW or greater will require approximately 6-8, years provided that these power levels will not require "relatively" more development than the lower power levels. Circuit, systems development, and testing will require approximately 3-4 years additional. Thus, the feasibility of the power management and distribution systems technologies could be established by the turn of the century to support the manned Mars mission.

ADVANCED THERMAL CONTROL

Multimegawatt space power/energy systems will require the control of multimegawatt quantities of waste heat which must be rejected (radiated) to the space environment. Even the most optimistic estimates of the more conventional space radiator concepts such as those proposed for the nuclear power system described in a preceding section of this paper, represent a sizable fraction of the total power system weight (see Table 2).

FIGURE 10. POWER ELECTRONIC COMPONENT DEVELOPMENT AT LEWIS RESEARCH CENTER

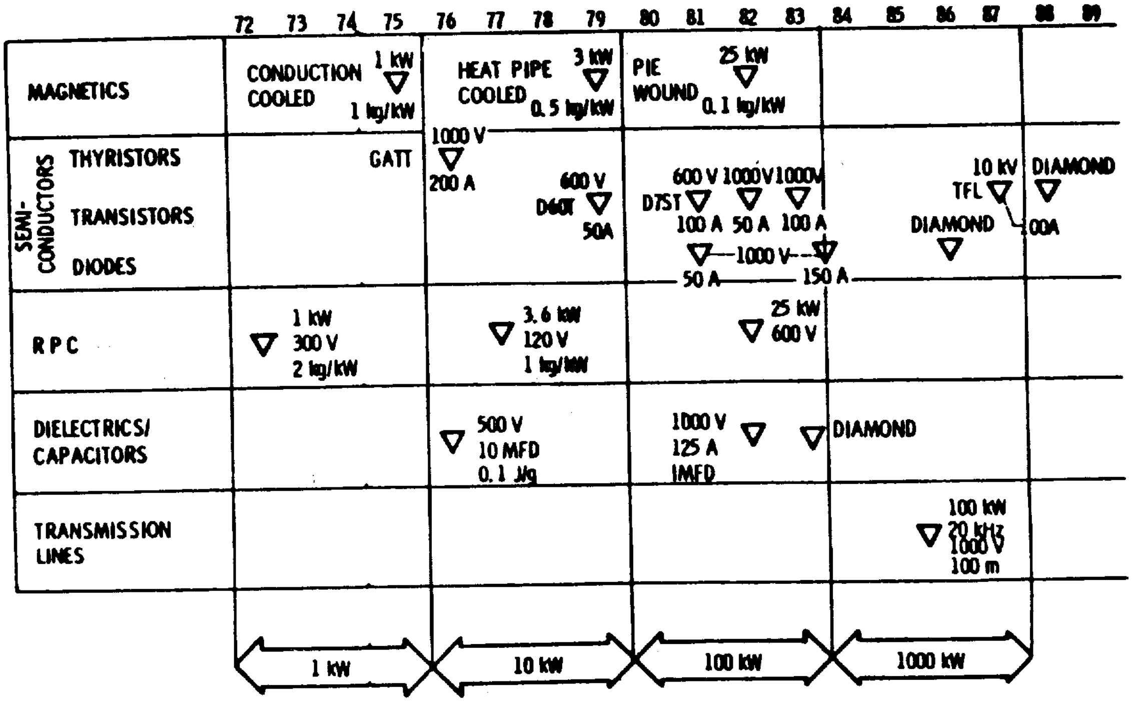
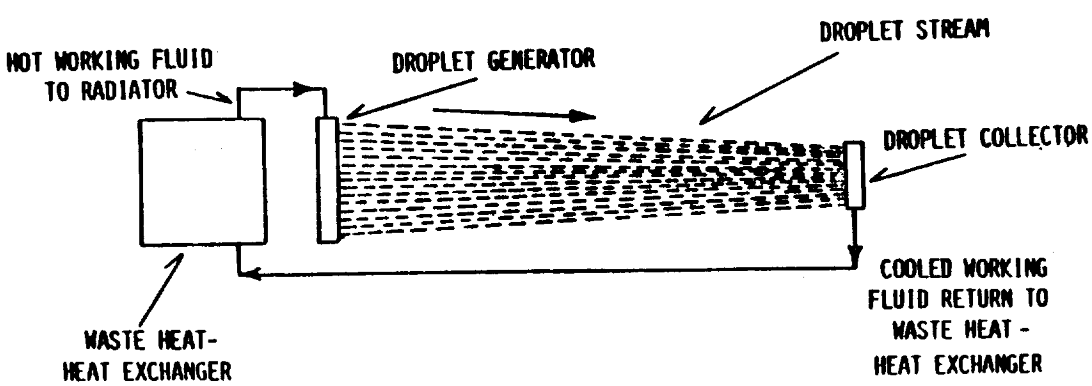


FIGURE 11. LIQUID DROPLET RADIATOR CONCEPT

RADIATIVE "FINS" AND "HEAT PIPES" OF CONVENTIONAL RADIATORS
REPLACED BY MULTIPLE STREAMS OF UNIFORM LIQUID DROPLETS



There are a number of advanced heat rejection concepts, under investigation at the Lewis Research Center, which show potential for considerable reductions in radiator system weights in addition to providing greater operational flexibility. These devices are, at present, in the concept feasibility assessment stage and still require extensive development before they can be incorporated into prototype or flight systems.

One such device is the liquid droplet radiator being jointly studied by the Lewis Research Center (NASA) and the Rocket Propulsion Laboratory (Air Force) through an interdependency agreement. A schematic of such a device is shown in Figure 11.

Design approaches to this concept are being investigated at this time. The design of the droplet generator, selection of a fluid compatible with its application and the space environment (operating temperature), lossless collection of the drops, and coalescence of the drops into a pumpable fluid stream for recirculation through the return loop, are areas which will require significant development.

As a result of studies presently underway, it was possible to arrive at preliminary estimates of the parameters which characterize a liquid droplet radiator system. A liquid droplet radiator which would meet the requirements of the nuclear power system described in the first section of this paper would be:

TABLE 6

LIQUID DROPLET RADIATOR PARAMETERS

Fluid	Tin (SN)
Density	6600 KG/M^3
Droplet Sheet Emissivity	0.5
Droplet Sheet Flow Rate	765 KG/SEC
Droplet Sheet Area (One Side)	289 M^2
Droplet Temperature (Generator)	1050^o K
Droplet Temperature (Collector)	900^o

Preliminary design estimates, (Phase I, Report, Ref.7), indicate that such a radiator could be constructed for 1.72 KG/M^2 of droplet sheet area, (both sides). This estimate includes the hardware items shown in Table 7 below.

TABLE 7

Liquid Droplet Radiator Components
Droplet Sheet
Fluid Inventory
Droplet Generator and Plenum Chamber
Droplet Collector
Return Loop and Pumps
Astro Mast, Boom, Miscellaneous Structure

One design concept of such a radiator is shown in Figure 12. This concept, if sized to meet the heat rejection loads of the nuclear power system of Section I of this paper, would result in a radiator mass of approximately 1000 KG which represents a considerable weight savings over that given in Table 2 (8700 KG). It is felt that at this point in its development stage, no problems have surfaced which indicate that this concept is unfeasible.

Another concept, somewhat similar in principle to the liquid droplet radiator, is the liquid belt radiator which is also being investigated for space power systems applications. One concept of such a device is shown in Figure 13. The primary difference between this concept and the liquid droplet radiator is that the free flowing droplet stream is replaced by a fluid belt upon which the radiating fluid is retained by surface tension forces. The circular shape of the belt is maintained by centrifugal forces as the belt is driven through the plenum chamber where it "picks up " the hot fluid.

The development status of this concept is not as advanced as that of the liquid droplet radiator. Areas which require further investigation are those related to the belt properties, fluid properties, seals, and the operational aspects of this device. The liquid belt radiator may show some operational advantages over the liquid droplet radiator from the

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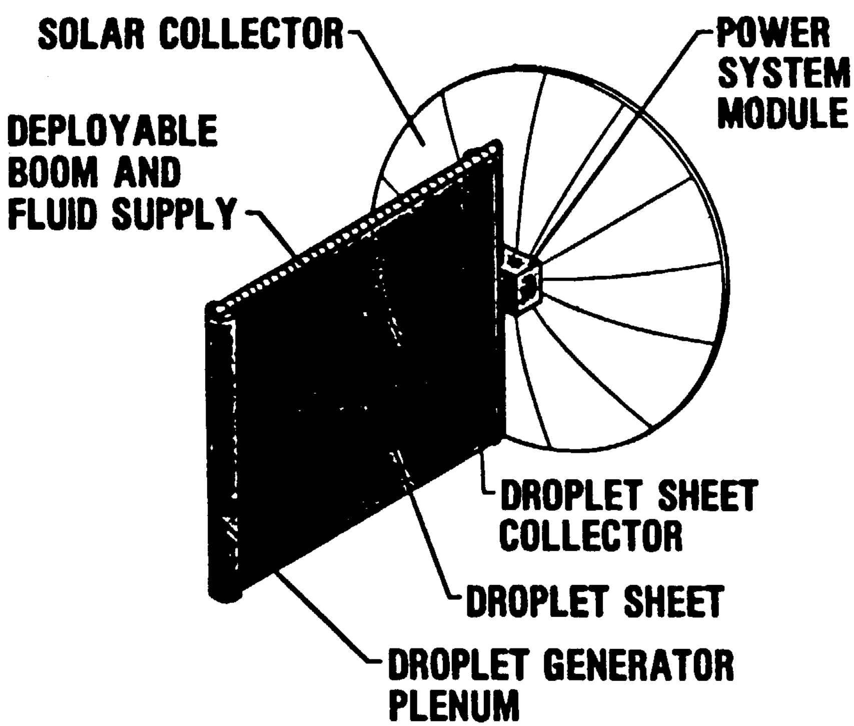

FIGURE 12. LIQUID DROPLET RADIATOR

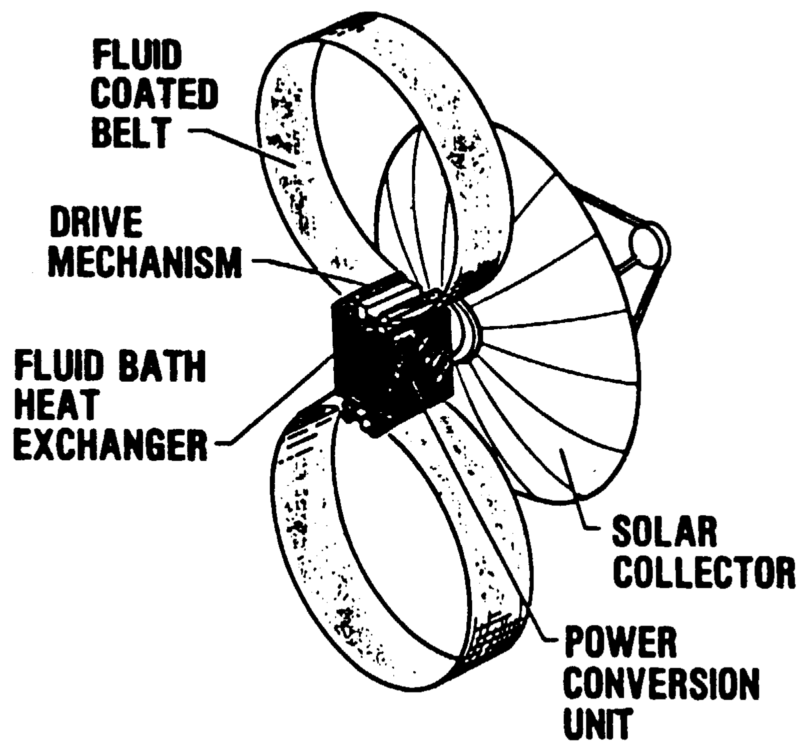

FIGURE 13. LIQUID BELT RADIATOR

viewpoint of management of the radiating fluid. An early projection indicates that such a device can be built at possibly 1.10-1.25% of the weight of a liquid droplet radiator.

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