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PEGASUS: A MULTI-MEGAWATT NUCLEAR ELECTRIC PROPULSION SYSTEM

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ABSTRACT

A propulsion system (PEGASUS) consisting of an electric thruster driven by a multimegawatt nuclear power system is proposed for a manned Mars mission. Magnetoplasmadynamic and mercury-ion thrusters are considered, based on a mission profile containing a 510-day burn time (for a mission time of approximately 1000 days). Both thrusters are capable of meeting the mission parameters. Electric propulsion systems have significant advantages over chemical systems, because of high specific impulse, lower propellant requirements, and lower system mass.

The power for the PEGASUS system is supplied by a boiling liquid-metal fast reactor. The power system consists of the reactor, reactor shielding, power conditioning subsystems, and heat rejection subsystems. It is capable of providing a maximum of 8.5 megawatts of electrical power of which 6 megawatts is needed for the thruster system, leaving 1.5 megawatts available for inflight mission applications.

INTRODUCTION

With the Space Transportation System (STS), the advent of space station Columbus and the development of expertise at working in space that this will entail, the gateway is open for missions to Mars. The missions are possible with state-of-the-art hydrogen/oxygen propulsion, but would be greatly enhanced by the higher specific impulse of electric propulsion. This paper presents a concept that uses a multi-megawatt nuclear power plant to drive an electric propulsion system. The concept has been named PEGASUS, PowEr GenerAting System for Use in Space, and is intended as a "work horse" for general space transportation needs, both long- and short-haul missions.

The advantages of electric propulsion are well-known in the aerospace community. But this high specific impulse, propellant-saving, potentially cost-saving, and mission-enabling technology has not received serious consideration for spacecraft propulsion, primarily because of the lack of a suitable, light-weight electric power source. The recent efforts of the SP-100 program indicate that a power system capable of producing upwards of 1 megawatt of electric power should be available in the next decade. Of greater interest are efforts in other areas which indicate that a power system with a constant power capability an order of magnitude greater could be available near the turn of the century. With the advances expected in megawatt-class space power systems, the high specific impulse propulsion systems, such as a magnetoplasmadynamic (MPD) or ion propulsion system, powered by a nuclear electric power plant, must be reconsidered as potential propulsion systems. A conceptual drawing of a manned Mars spacecraft powered by the PEGASUS drive is shown in Figure 1. The electrical power for the propulsion system is provided by a nuclear multi-megawatt fast reactor space power system. This power system is capable of meeting both the propulsion system and spacecraft power requirements. The size and mass limitations of the STS are a prime consideration in the design, and the collapsed system will be capable of lower earth orbit insertion by two shuttle missions. Development of this power system could be completed by the mid 1990's and the system available near the turn of the century. Since this is an advanced system concept, some development efforts are still needed in the fuels, heat rejection, and turbo-alternator areas.

MISSION ANALYSIS

Any number of mission profiles may be devised for a manned Mars expedition, depending on the assumptions made concerning desired objectives and available technology. This paper uses a previously developed set of mission parameters (Ref. 1) as a basis for design. However, the propulsion system concept is adaptable to a wide range of mission profiles. The parameters defining this Mars mission consist of a 344 metric ton initial mass vehicle in Low Earth Orbit (LEO) which autonomously raises itself through the radiation belt to Geosynchronous Earth Orbit (GEO), whereupon a crew of three boards. The trip to Mars takes 601

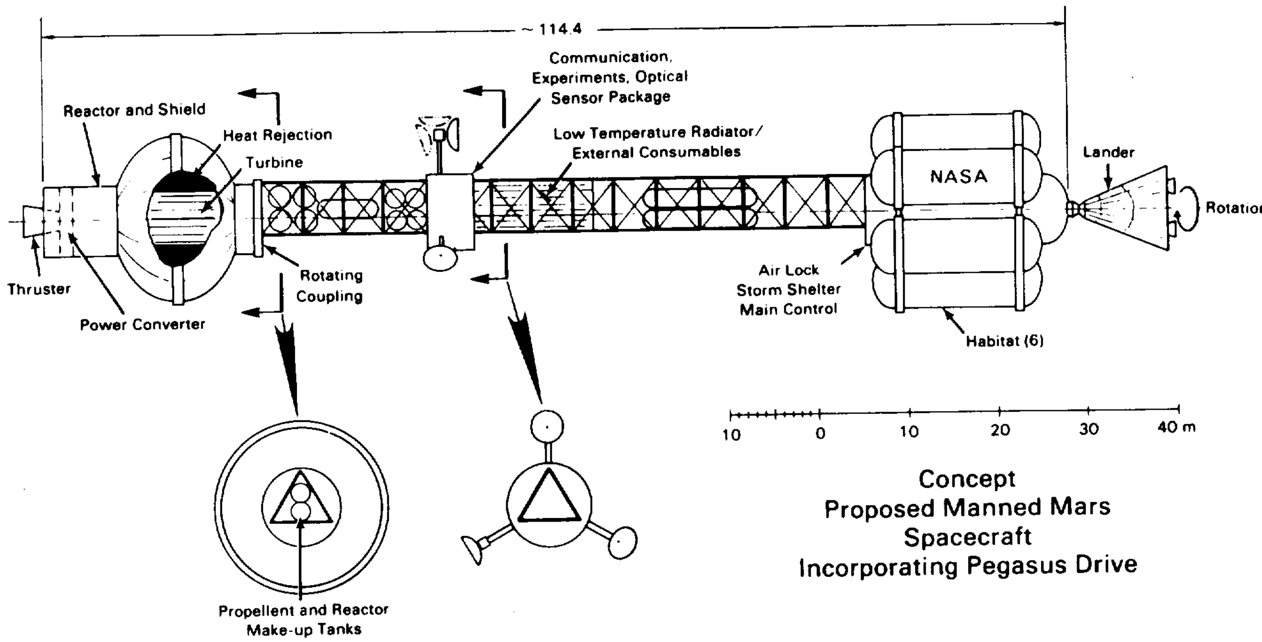


FIGURE 1. CONCEPT OF PROPOSED MANNED MARS SPACECRAFT WITH PEGASUS DRIVE

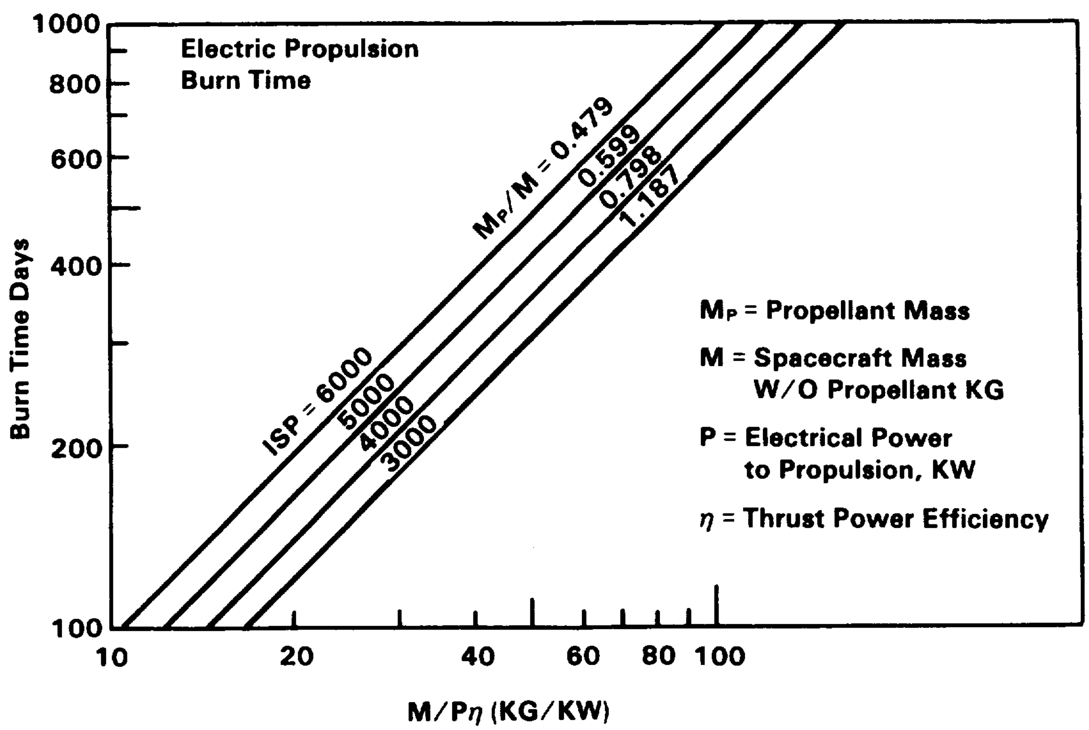


FIGURE 2. ELECTRIC PROPULSION BURN TIME

days, exploration 100 days, and the return trip 268 days. Propellant consumed is 111 metric tons with a 5% reserve. The return mass to GEO is 103 tons, 76 tons having been left on Mars. The allowance for power and propulsion system was 34 tons, including 11 tons for tankage.

Figure 2 shows how vehicle mass M (excluding only propellant) can be traded against power P, overall efficiency, eta, and burn time, t, to provide a desired specific impulse. Improvements to the propulsion system would be reflected in higher values of eta, and have the same effect as reductions in vehicle mass M in reducing the required burn time. Actual trip time for a mission is a function of many variables, but it is clear from Figure 2 that the more power developed by the propulsion system (all other things being approximately equal), the shorter the trip time. The time saving can be accomplished either by the shorter burn times required to produce the same velocities as projected for the chemical-fueled system, or by longer burn times that produce higher velocities in the coast phase. Such trade-offs in power, burn time, and coast time would have to be studied extensively to determine the optimum combination, but it is clear that electric propulsion systems are greatly mission-enhancing in this regard.

Electric propulsion also shows an advantage for this mission in the modest propellant requirements as compared to chemical systems. While the scope of this paper excludes a detailed comparison, the propellant required is characterized by the specific impulse of the propulsion system. Since electric engines have a specific impulse that is 10-20 times that of hydrogen-oxygen chemical systems, the propellant required for electric systems can be from 1/3 to 1/10 that of chemical propellant.

PROPULSION SYSTEM

Electric propulsion systems have not been seriously considered for use with large spacecraft due to the lack of a suitable electric power source to drive them. However, recent efforts to develop megawatt-class space power sources show such systems to be technologically feasible, and a multi-megawatt lightweight nuclear electric propulsion system would enable missions of almost any conceivable duration and scope. The manned Mars mission would be well within the capabilities of such a system.

The electric propulsion could be provided by a magnetoplasmadynamic (MPD) or ion propulsion system. Section 3.1 discusses MPD thrusters as part of such a system. Section 3.2 discusses ion thrusters for this concept. The design of the nuclear power plant that provides the required power is outlined in Section 3.3.

Magnetoplasmadynamic Thrust System

The MPD system is composed of thrusters, propellant tanks, and thermal control subsystems. A schematic of the major components of the MPD thruster system is shown in Figure 3. Three phase ac at 10-20 kV and 1500 Hz is fed through a transformer and rectifier to the MPD thruster. Power control and regulation is confined to adjustments of the ac output from the alternator. The high tension buss runs from near the turbo-alternator past the reactor shield to the transformer and thruster. This 15 meter length of cable is insulated and electrically shielded to reduce EMI, and masses amount to 69 kg.

The transformer mass has been estimated from design equations and weighs only 1911 kg (Ref. 2). (Efficiency of 99% is a reasonable design goal for this delta-delta transformer.) The rectifier is designed to use 3,000-ampere diodes derated to 1,500 amperes. Eight diodes are connected in parallel to handle 12,000 amperes. Six units are required for full wave rectification of three phase AC. The 48 diodes and cooling assembly are estimated to have a total mass of 200 kg, and will dissipate 59.5 kW when rectifying 25,000 ADC. The transformer, rectifier, and high tension buss require 120.8 kW of heat to be removed by a cooling plant associated with the cryogenic superconducting alternator.

The current buss to the MPD thruster is designed to be self-radiating at 373 K; it has a mass of 817 kg, which includes 0.5 cm of insulation between the coaxial conductors and on the outside. An alternative that may reduce overall mass at the cost of complexity is to reduce the cross-section of aluminum, which increases the dissipation and requires active cooling. Also, 3 meters were allowed to electrically connect the MPD thruster, which operates at about 1,600 K, and the rectifier assembly which is designed for 373 K, thus providing thermal isolation.

The MPD thruster assembly consists of seven engines which are used one at a time, for a 510-day burn. Based upon ongoing work at JPL to evaluate thruster lifetime with subscale devices, and the present understanding of cathode physics, thruster lifetime (which is limited by the

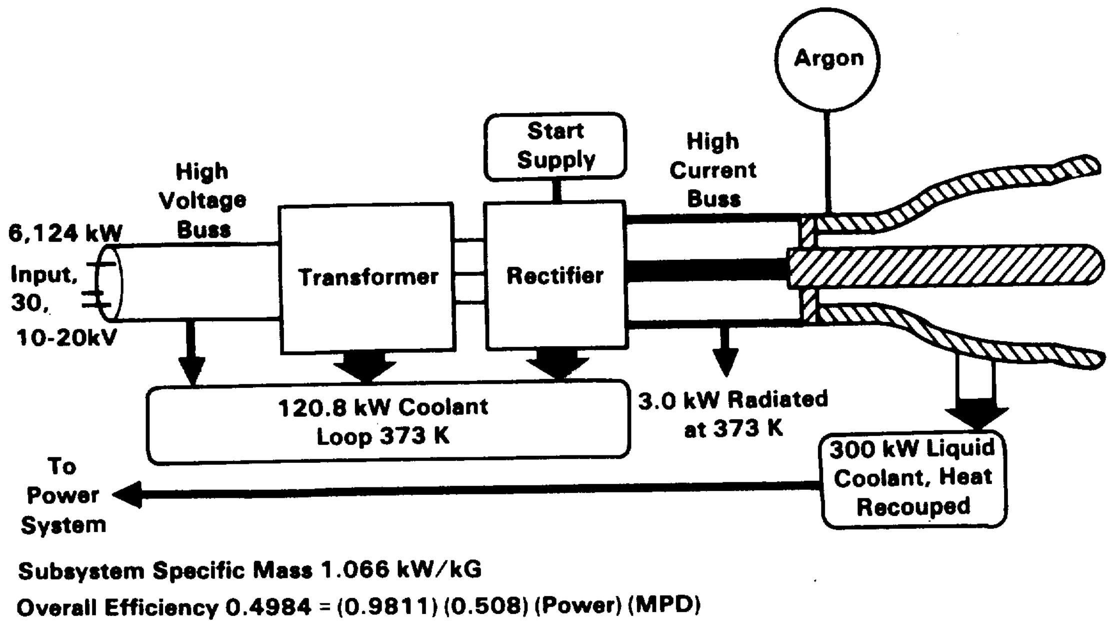


FIGURE 3. MPD THRUSTER SYSTEM COMPONENTS

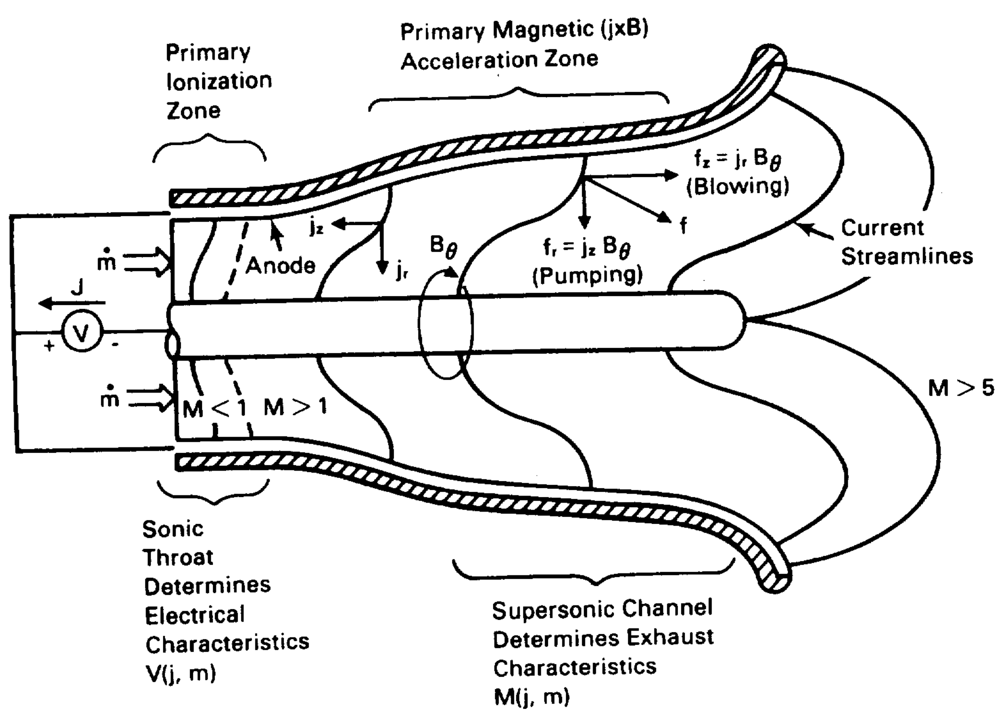


FIGURE 4. MPD THRUSTER

cathode) is estimated to be 2,000 hours. But life test of a multimegawatt MPD thruster can only follow a vigorous development program, neither of which are presently planned in NASA's research program.

Figure 4 shows a schematic of the MPD thruster which accelerates propellant by means of a magnetic body force. Propellant is injected through the rear insulator and is ionized by a high-current, diffuse discharge. The large multi-kiloampere current creates an azimuthal magnetic field which interacts with the current to produce a magnetic body force directly on the ionized gas. Exhaust speeds of 15,000 to 80,000 m/s have been measured on a thrust stand using a variety of propellants (Ref. 3). Thruster conversion efficiency of electrical power to directed kinetic energy has been measured at 35% on a thrust stand and analysis suggests that 50-60% is ultimately possible (Refs. 3,4).

All seven engines are connected in parallel to the high-current buss, but each has a separate propellant valve and a contactor on the cathode current feed. Though performance development may change the electrode shape somewhat, the overall dimensions of a multimegawatt thruster will remain about the same (Ref. 5). Each engine is assumed to have a center-body cathode 3 cm in diameter and 10 cm long, with an anode about 12 cm inside diameter, and of a comparable length. The most massive part of the MPD engine is the anode heat removal system, which must remove approximately 300 kW. An allowance of 700 kg is made for the assembly of seven engines, contactors, and thermal control.

The MPD thruster is started by application of 500 V and a low propellant flow which results in a glow discharge. As the power in this glow discharge is increased to 2-5 kW, the cathode is heated to incandescent temperatures. High temperature results in prodigious thermionic cathode emission and the arc jumps to a low impedance arc mode. The alternator output can then be ramped up, along with the propellant flow rate, to full power. The MPD thruster exhaust velocity and thrust may be throttled down to half of nominal by adjusting the mass flow and power.

A summary of the mass breakdown and power consumption is shown in Table 1. A contingency of 10% of the system mass is included, and 20% on top of that is added for structure. The net propulsion system mass, excluding power and the cooling plant, is 5,144 kg. The electrical system is 98% efficient, while the MPD thruster is assumed to have a 50%

TABLE 1: MANNED MISSION TO MARS PROPULSION SYSTEM SUMMARY

ITEM	MASS	POWER DISSIPATED
MPD THRUSTER ASSEMBLY - 7 engines required for 510 day burn - 2000 hr life per engine assumed - 240 VDC, 25,000 ampere input - 5,000 S specific impulse - 50% thrust efficiency - 2.5 G/S propellant flow rate	700 kg	6,000.0 kW
ENGINE STARTUP AND CONTROL SUBSYSTEM	50 kg	
HIGH CURRENT BUSS - 25,000 amp insulated coaxial conductor - 0.36 M dia by 3 M length - self-radiating at 373 K	817 kg	3.0 kW
RECTIFIER - 48 diodes, 3,000 amp rating each - heat sink requires active cooling at 373 K	200 kg	59.5 kW
TRANSFORMER - 6,123 kW, 1,500 HZ, 3 phase, 10-20 KV input - 99% efficiency - 0.5 M^3 volume	1,911 kg	60.0 kW
HIGH VOLTAGE BUSS - 10-20 KV, 3 phase, 500 A rating - insulated, shielded - self-radiating at 373 K - 10 cm dia. by 50 M long	69 kg	1.3 kW
THERMAL CONTROL		
- liquid coolant to MPD anode, recoups 300 kW at up to 1,600 K	100 kg	
- coolant to diodes and transformer and high voltage buss	50 kg	
Subsystem Total	3,897 kg	6,123.8 kW
Contingency Mass 10%	390 kg	
Structural Mass 20%	857 kg	
Propulsion System Total Mass	5,144 kg	
Specific Mass	0.840 kg/kW	
Power Processing Electrical Efficiency	98.0 %	

conversion efficiency. An interesting point on the waste heat of the MPD thruster is that since the engine operates at a high 1,600 K temperature, the 300 kW of heat rejected can be recouped to the advantage of the power conversion system.

The MPD thruster has been studied extensively with argon propellant, and the range of specific impulse available with this propellant is well suited to this mission. Cryogenic storage of argon has been considered in detail elsewhere, and presents no unusual constraints or problems. A 16,301 kg storage tank has been designed, with a mass of 721 kg (Ref. 6). This tank is designed as Shuttle-compatible, and operates in an environment of 293 K, while allowing only 0.14 of heat energy to leak into the fluid. Additional heat is needed to drive off gas at 0.0025 g/s at 2 atm for the MPD thruster. Seven of these tanks are required to store 111,000 kg of propellant for the mission for a total tankage mass of 5,047 kg.

Potassium, which is the working fluid in the dynamic conversion equipment, has nearly the same atomic mass as, and one more proton than, argon. Since it's ionization potential is about one-fourth of argon, potassium should provide a higher thrust efficiency than argon. Potassium offers a simpler storage problem in that it can be stored in a solid form and then liquified at 62 C. Since the density of solid potassium is comparable to liquid argon, and mercury tanks for ion propulsion have a tank fraction comparable to 4.4% for liquid argon, this study will assume that the tank mass for potassium will be the same as for argon.

There is a concern that the small fraction of the exhaust that flows back toward the spacecraft may plate out on many surfaces after 111 metric tons of potassium are exhausted. Though potassium may be a better propellant from a performance and storage standpoint, contamination of the spacecraft surfaces may require argon propellant.

Ion Thrust System

Thrust system characteristics for the Mars mission trajectory are presented in Table 2 for 100 cm, 50 cm, and 30 cm ion thrusters for both mercury and xenon propellant. The J-series 30 cm ion thruster served as the baseline for this study, since it has undergone extensive design evolution over the past fifteen years (Ref. 7) and has been operated over a full range of high performance levels necessary to support future

TABLE 2: THRUST SYSTEM CHARACTERISTICS

	100 CM ION THRUSTER						50 CM ION THRUSTER						30 CM ION THRUSTER					
	CASE I			CASE II			CASE III			CASE IV			CASE V			CASE VI		
MISSION PARAMETERS:																		
Velocity increment (m/s)	22,600																	
Thrust time (days)	510																	
Final mass (kg)	103,000																	
Specific impulse (S)	4665	5390	6030	5770	6600	7400	4665	5390	6030	5770	6600	7400	4665	5390	6030	5770	6600	7400
Thrust Module:																		
propellant	Hg	Hg	Hg	Xe	Xe	Xe	Hg	Hg	Hg	Xe	Xe	Xe	Hg	Hg	Hg	Xe	Xe	Xe
total accel. voltage (V)	3000	4000	5000	3000	4000	5000	3000	4000	5000	3000	4000	5000	3000	4000	5000	3000	4000	5000
number of thrustors *	6	3	2	5	3	2	21	6	5	20	9	5	57	25	14	54	24	13
total mass (kg)	9977	10250	11018	11105	11960	10236	11104	10581	10820	12595	12298	12701	13295	11857	11762	14752	13493	13478
input power (kw)	2181	2396	2594	2563	2816	3067	2183	2397	2594	2565	2817	3068	2188	2399	2595	2569	2819	3069
Interface Module:																		
propellant mass (kg)	65801	54950	47942	50564	43042	57632	65801	54950	47942	50564	43042	37632	65801	54950	47942	50564	43042	37632
tank mass (kg)**	398	327	282	298	250	217	398	327	282	298	250	217	399	327	282	298	250	217
total mass (kg)	70013	58799	51709	54182	46579	11223	10019	58692	51448	54253	46445	40902	70122	58754	51496	54349	46486	40506
input power (kw)	2192	2408	2607	2576	2830	3082	2194	2409	2607	2578	2831	3083	2200	2411	2608	2582	2833	3084
Transmission Line:																		
mass (kg)	3830	4207	4554	4500	4945	5385	3833	4209	4554	4504	4946	5386	3841	4212	4556	4511	4949	5388
dissipated power (kw)	191	210	228	225	247	269	192	210	228	225	247	269	192	211	228	226	247	269
OVERALL Thrust System:																		
dry mass (kg)	18018	18307	19338	19224	20442	21931	19155	18532	18880	20787	20648	21357	21456	19874	19872	23048	21886	22140
total mass (kg)	23820	73256	67201	69708	63484	59563	84957	73482	66822	71351	63690	58989	87257	74823	67815	73612	64528	59772
input power (kw)	2373	2607	2822	2788	3064	3337	2375	2608	2822	2790	3064	3337	2380	2610	2823	2795	3066	3338
radiator area (m^2)	31	34	37	37	40	44	32	35	37	37	41	44	33	35	38	39	41	44
specific impuse (S)	4665	5390	6030	5770	6600	7400	4665	5390	6030	5770	6600	7400	4665	5390	6030	5770	6600	7400
thrust (N)	68.5	66.1	64.5	65.1	63.4	62.1	68.5	66.1	64.5	65.1	63.4	62.1	68.5	66.1	64.5	65.1	63.4	62.1

* No thruster redundancy included

** Propellant storage was assumed to be noncryogenic pressurized spherical tanks planetary missions. The design-level performance of the 30 cm J-series thruster is shown in Table 3. It has a lifetime design goal of 15,000 hours (demonstrated 10,000 hr and projected in excess of 25,000 hours (Ref. 8)) at the design-level operating conditions.

Figure 5 shows a cross-section of the 30 cm J-series mercury ion thruster. The thruster consists of a cathode, an anode, a cylindrical discharge chamber, an arrangement of magnets in the discharge chamber, and a set of closely-spaced grids downstream of the cathode and propellant injection. Thrust is developed when electrons passing through the grid system form an ion beam accelerated to high velocities by a large electric field between the screen and acceleration grid. Electrons injected into the ion beyond the extraction system neutralize the current to prevent charge buildup on the spacecraft and beam divergence. The thrust levels are not comparable to those of chemical rockets, but the extremely high exhaust velocities permit large characteristic-velocity missions to be accomplished.

Some component development will be required to produce advanced thrusters for planetary missions, particularly for the 50 and 100 cm thrusters. Present magnetic field and chamber designs may require modification to reduce losses. Operation of the accelerator system at voltages above 3,000 V will require the development of advanced grid materials and fabrication procedures (Ref. 9). The present J-series cathodes are designed for operation up to 20 amperes emission current, but the systems presented in Table 2 would require currents ranging from a low of 80 amperes for a 30 cm ion thruster (ISP=4665) to a high of 2800 amperes for a 100 cm ion thruster (ISP=7400).

NUCLEAR ELECTRIC POWER SOURCE

The proposed power source for the electric propulsion system is PEGASUS, an 8.5 MWe boiling liquid-metal, space-based nuclear power system. The system employs a direct Rankine power cycle and is designed to meet the power requirements of 6 MWe for the electric propulsion system with an additional 1.0 to 1.5 MWe available for mission-specific tasks and experiments.

PEGASUS comprises five major subsystems and components. These are a cermet fueled, boiling liquid metal fast reactor; a four-pi contoured

TABLE 3: 30 CM J-SERIES THRUSTER PARAMETERS

PERFORMANCE	DESIGN-LEVEL DEMONSTRATED PERFORMANCE	DEMONSTRATED EXTENDED PERFORMANCE	MARS MISSION CASE V PERFORMANCE REQUIREMENTS
Beam Current, J_B (A)	2.0	7.9	13.3
Beam Ion Production Cost, E_i (W/A)	192	220	150
Specific Impuse, ISP (S)	3000	4880	4665
Thrust, T (N)	0.13	0.51	1.20
Thruster Efficiency, N_t	72.3%	71.2%	69.5%
Thruster Input Power, (W)	2650	17,210	38,300
Propellant	MERCURY	XENON	MERCURY
Lifetime (hours)	25,000	-	12,500

TABLE 4: MANNED MISSION TO MARS OVERALL MASS SUMMARY

			MPD		ION
1.	SPACECRAFT, HABITAT, LANDER, ETC.		143,310 kg		129,640 kg
2.	POWER PLANT SUMMARY		36,500 kg		36,500 kg
	Reactor	3,730		3,730	
	Shield	28,350		28,350	
	Turbine	1,290		1,290	
	Generator	840		840	
	Pumps	590		590	
	MBR	540		540	
	ACS	1,160		1,160	
3.	PROPULSION SYSTEM		121,190 kg		125,310 kg
	Engines, etc.	5,144		19,400	
	Propellant	111,000		101,450	
	Tankage	5,047		4,460	
TOTAL MASS IN GEO			301,000 kg		291,450 kg

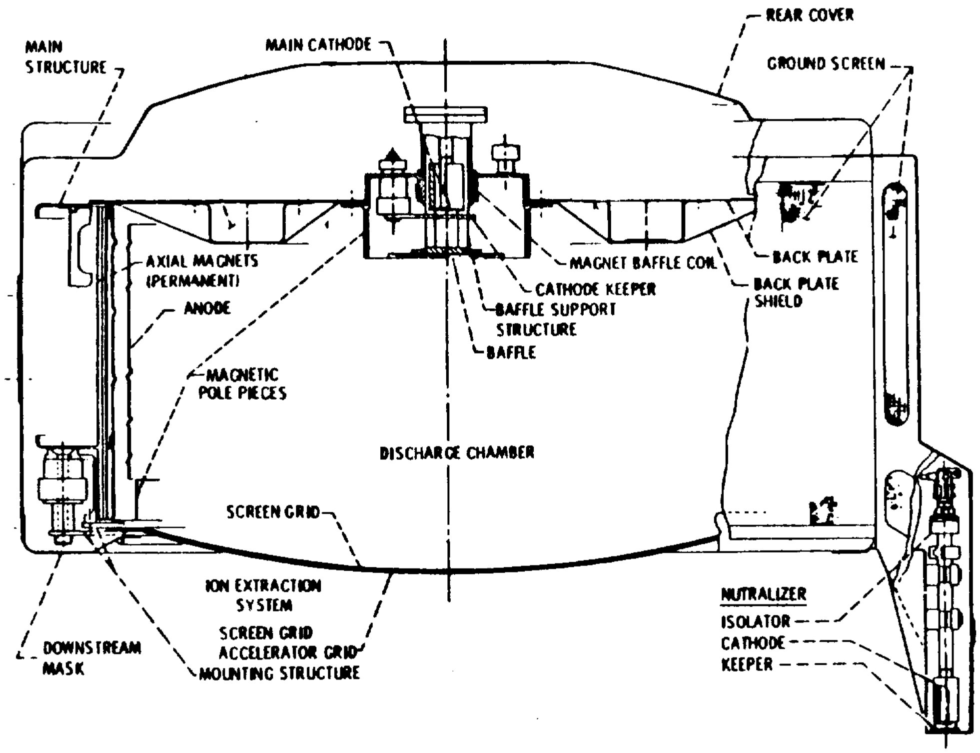


FIGURE 5. MERCURY ION THRUSTER

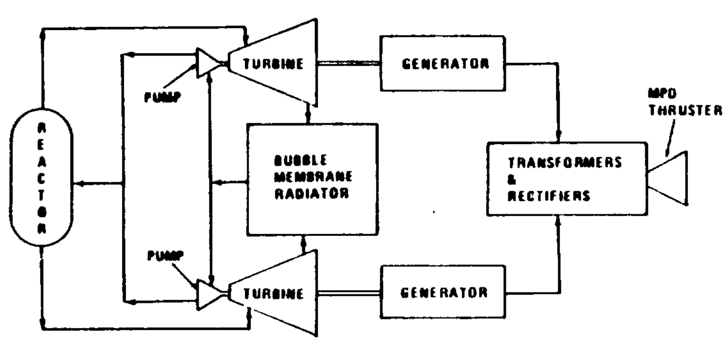


FIGURE 6. PEGASUS SCHEMATIC DIAGRAM

man-rated reactor shield; an axial flow turbine and superconducting alternator for power conversion; a power conditioning subsystem; and a heat rejection and thermal control subsystem. A schematic of the system is shown in Figure 6, and the basic design of these subsystems is discussed in the following sections.

Reactor Subsystem

The reactor selected for the PEGASUS system uses a boiling liquid metal coolant and cermet fuel. The reactor is a right cylinder approximately 53 cm long and 50 cm in diameter. It has a 15 cm diameter exit plenum at each end and a 2.5 cm inlet plenum in the center, forming a double wafer. The reactor is controlled by means of control drums located in the reflector region and is fueled with cermet blocks. Each block is 7.5 cm high and 7.5 cm on a side, with a hexagonal cross-section honeycombed by coolant channels. The coolant channels are 5 mm across the flats with a fueled web 3 mm thick. A preliminary thermal analysis of the cermet fuel, assuming full power operation and stable nucleate boiling, indicates that a peak centerline fuel temperature of 1300 C can be expected for a coolant channel bulk temperature of 1100 C. It should be noted, however, that detailed thermal analysis of this system (or any system involving two-phase flow) will require additional experimental and theoretical work.

Reactor Shielding Subsystem

The PEGASUS system incorporates a four-pi contoured shield to reduce radiation from the reactor to safe levels during full-power operation. The shield is a composite material of LiO, W, and LiH. To maintain the physical integrity of the shield, it is cooled by the reactor inlet coolant. The mass of the shield requires that the major portion of it be delivered to orbit on a separate Shuttle load, to be installed after the system is deployed.

Power Conversion Subsystem

Power conversion is accomplished by means of a pair of turboalternators. The turbine designs incorporate a multi-stage axial flow design which is based on previous development work performed at NASA's Lewis Research Center. This is a saturated potassium vapor turbine which is operating with an inlet pressure of 1.5 MPa and an exit quality of 80% or less. Optimum turbine efficiency is obtained with blade speeds of 300 to

600 m/sec. System shaft speeds will be determined through trade-off studies involving stress analysis of the turbine components at desired operating temperatures and size optimization of the turboalternator system.

Superconducting alternators are chosen to develop the electrical power because of their high power-to-weight ratio. Each superconducting alternator is expected to operate at 1500 Hz and 10 to 20 kva, and be capable of providing a continuous electric power output of 5.0 megawatts. The overall power conversion subsystem is expected to have a specific weight on the order of 0.05 kg/kW and an efficiency of 85%.

Power Conditioning Subsystem

The requirements for the power conditioning system will be set by the specific propulsion unit with which the power system is to be integrated. However, primary power conditioning is expected to be accounted for in the generator portion of the power conversion system.

Heat Rejection Subsystem

Heat rejection for the PEGASUS is accomplished by means of both a high and a low temperature heat rejection subsystem. The high temperature subsystem handles waste heat rejection from the turbines. The low temperature subsystem takes care of waste heat from the alternator and other components requiring cooling.

The high-temperature heat rejection system consists of a bubble membrane radiator, associated pumps, plumbing, and structure. The radiator is sized to reject 21.5 MW of waste heat from the system during full-power operation. This system consists of a 19.9 m diameter thin film spherical envelope, coolant stowage tanks, structural piping, thermal insulating material, and associated pumps. Turbine exhaust vapor enters the radiator and is condensed on the inner surface of the bubble membrane. System rotation causes the condensed fluid to travel along the inner surface of the membrane to the gravity trough at its equator. Once collected, low pressure EM pumps are used to return the coolant fluid to the turbine driven high-pressure reactor feed pumps.

The low-temperature heat rejection system consists of an auxiliary cooling system designed to reject waste heat produced within the alternator and other equipment operating at much lower temperatures. The auxiliary cooling system has its own working fluid, pumps, and radiator.

This system utilizes helium as its working fluid and is composed of a Stirling cycle cryogenic cooler, an auxiliary chiller, a low-temperature radiator, and associated pumps and piping. The cryogenic cooler removes heat from the liquid helium and transfers this heat to the auxiliary coolant in the chiller. A closed loop system is used to pump this coolant through the auxiliary cooling radiator located around the perimeter of the main radiator. Before returning to the chiller, this coolant is used to cool pumps and other components.

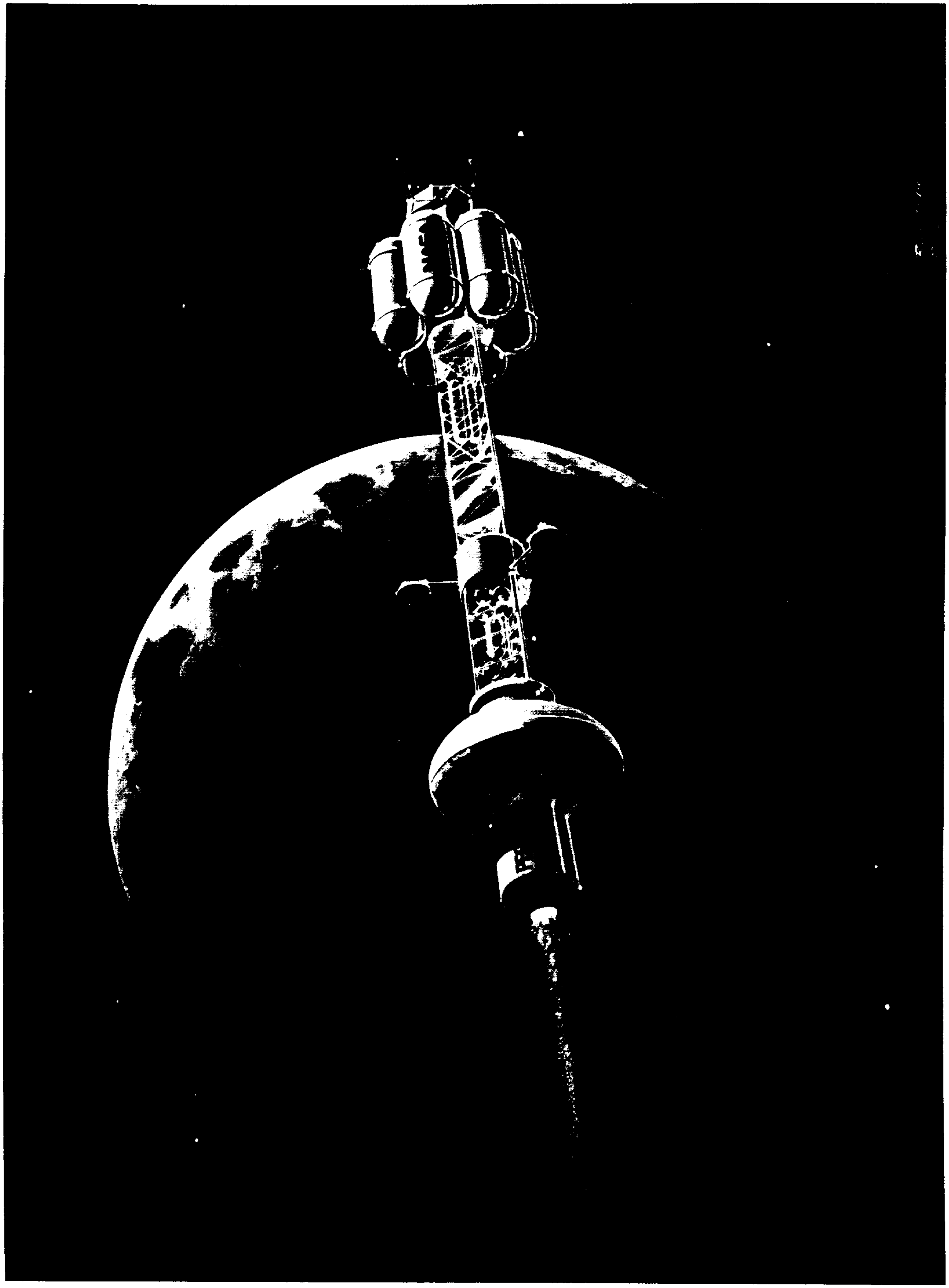
PEGASUS SYSTEM PARAMETERS

Table 4 contains the major parameters that have been examined thus far. All calculations are of a scoping nature and large tolerances exist in all areas. The specific impulse and the thrust of the ion system were matched to that of the MPD system, and the burn times held the same for both systems so as to match the trajectory calculations. The trajectory used may not be an optimum for either the MPD or the ion thruster system.

ARTIST'S CONCEPT

Figure 7 is an artist's sketch of the Pegasus concept.

FIGURE 7



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