CONF-920104--29

PNL-SA--20132

DE92 006729

JAN 24 1992

MARS MISSION PERFORMANCE ENHANCEMENT WITH HYBRID NUCLEAR PROPULSION

J. E. Dagle
D. R. Segna

January 1992

Presented at the
9th Symposium on Space Nuclear
Power Systems
January 12-16, 1992
Albuquerque, New Mexico

Work supported by
the U.S. Department of Energy
under Contract DE-AC06-76RLO 1830

Pacific Northwest Laboratory
Richland, Washington 99352

MARS MISSION PERFORMANCE ENHANCEMENT WITH HYBRID NUCLEAR PROPULSION

Jeffery E. Dagle and Kent E. Noffsinger
Pacific Northwest Laboratory
P.O. Box 999 M/S K5-19
Richland, WA 99352
(509) 375-3629

Donald R. Segna
U. S. Department of Energy
Field Office, Richland
P.O. Box 550 M/S A5-90
Richland, WA 99352
(509) 376-8989

Abstract

Nuclear electric propulsion (NEP), compared with chemical and nuclear thermal propulsion (NTP), can effectively deliver the same mass to Mars using much less propellant, consequently requiring less mass delivered to Earth orbit. The lower thrust of NEP requires a spiral trajectory near planetary bodies, which significantly increases the travel time. Although the total travel time is long, the portion of the flight time spent during interplanetary transfer is shorter, because the vehicle is thrusting for much longer periods of time. This has led to the supposition that NEP, although very attractive for cargo missions, is not suitable for piloted missions to Mars. However, with the application of a hybrid approach to propulsion, the benefits of NEP can be utilized while drastically reducing the overall travel time required. Development of a dual-mode system, which utilizes high-thrust NTP to propel the spacecraft from the planetary gravitational influence and low-thrust NEP to accelerate in interplanetary space, eliminates the spiral trajectory and results in a much faster transit time than could be obtained by either NEP or NTP alone. This results in a mission profile with a lower initial mass in low Earth orbit. In addition, the propulsion system would have the capability to provide electrical power for mission applications.

INTRODUCTION

For a piloted Mars vehicle, NEP can reduce both the propellant mass and the interplanetary trip time. With the much higher specific impulse, less propellant is consumed as compared to NTP. In interplanetary heliocentric space, the high-efficiency propulsion system allows continuous acceleration over much of the trip, allowing the vehicle to attain much higher speeds, thus reducing the travel time. However, low-thrust propulsion requires spiral trajectory escape and capture at Earth and Mars, which causes the overall trip time to become longer than that of an NTP vehicle.

There are methods proposed to reduce the manned portion of the trip time. One such approach (Hack et al. 1991) uses a combination of a "crew taxi" for Earth departure, the Deimos orbit at Mars, and an Earth Crew Capture Vehicle (ECCV) upon Earth arrival. With a crew taxi, the electric propulsion vehicle starts in low Earth orbit and spirals out to a high Earth orbit. The vehicle then pauses in this orbit while the crew launches in a separate vehicle to rendezvous with the Mars transporter. The vehicle can then perform the last remaining spirals to attain escape velocity and begin the heliocentric portion of the journey. While this approach eliminates the excessive manned portion of the spiral-out time, the crew taxi has an initial mass in low Earth orbit of 57,000 kg, which must be included in the total mass of the electric propulsion vehicle for comparison studies.

Using the Deimos altitude for a final Mars orbit allows fewer spirals to be taken for capture compared with using a low Mars orbit. However, the Mars excursion vehicle must then be capable of delivering the astronauts and cargo to and from the surface and a higher altitude, which adds mass and capability requirements to the ascent/descent vehicle. Although a significant mass penalty would be associated with a more robust Mars excursion vehicle, this added mass would be carried to Mars using electric propulsion, thus minimizing the penalty associated with the initial mass in low Earth orbit.

Returning to Earth via an ECCV saves both propellant mass and time, thereby eliminating a low-thrust spiral capture by performing an Earth flyby maneuver while sending the ECCV into the atmosphere for an Apollo type of capsule landing. The mass associated with the ECCV is 7000 kg, which is far less than the propellant needed to bring the vehicle into orbit. In addition, a hidden cost of returning the crew from low Earth orbit to the surface is saved using the ECCV.

Additional benefits of electric propulsion for use with Mars piloted missions include decreased variation with opportunity, lengthened Earth launch windows, longer Mars stay windows, and mission aborts (Doherty 1991). Because of decreased variation in propellant consumption, the vehicle can be designed for multiple mission opportunities without major changes in vehicle design. Impulsive thrust propulsion methods, which include chemical and NTP systems, have very wide variations in total propellant requirements for different mission opportunities. Buden and Bartine (1991) found that NTP propulsion requires 600 to 700 tonnes in low Earth orbit for favorable year missions, and as much as 900 tonnes for the worst years. This represents about a 30% to 50% change in total vehicle mass, which would require substantial design changes based on opportunity. This could become a major cost element for an eventual Mars transportation infrastructure. Hack et al. (1991) report a 90-day Earth launch window is available for a 2014 opportunity without changing the total propellant requirements. The Mars stay time can be reduced without changing the propellant requirements, thereby providing continuous ability to leave Mars up to the planned departure date. Electric propulsion also provides mission abort options and flexibility. Hack et al. (1991) have shown that a 10-MWe vehicle downgraded to 5 MWe has a continuous abort capability to return to Earth with minimal time and propellant penalty. To safely return the crew to Earth in the event of a power system failure at any point in the mission would require about an 8% additional mass in low Earth orbit for the piloted vehicle.

Hybrid Mission

The hybrid Mars mission described in this paper is both hybrid propulsion and a combination of mission profiles and techniques that appear to offer benefit. Hybrid propulsion--the use of both low-thrust electric propulsion and high-thrust rockets--can be used to eliminate the trip time associated with near-planetary body spiral trajectories while employing the benefit of high-efficiency electric propulsion for the interplanetary portion of the mission. The system proposed for this application is nuclear electric, consisting of a boiling potassium fast reactor with a direct rankine conversion system coupled with a solid core nuclear thermal rocket. Although it is possible to couple these systems by utilizing the same nuclear heat source for both systems, separate reactors are preferred if tens of thermal megawatts are required for electrical generation (Kirk et al. 1991). Integration issues, which drive the cost, mass, and complexity of a hybrid system, tend to add additional mass over that saved by a common nuclear reactor and common shield. Although an integrated system could be developed and used, this analysis assumes separate systems with consequent mass and performance numbers.

Hybrid attributes common to the NEP Mars mission profile include the following:

- Utilize split mission in which the cargo to perform the mission, as well as the return propellant for the crew vehicle, are carried to Mars on a separate vehicle.
- Use an ECCV for the crew return to Earth, minimizing the capture propellant required to bring the vehicle into orbit.

Hybrid attributes differing significantly from the NEP Mars mission profile are these:

- Eliminate the "crew taxi" to bring the humans into high Earth orbit to rendezvous with the vehicle after it has completed most of its outbound spirals.
- Use low Mars orbit as the destination orbit rather than Deimos altitude, thereby greatly reducing the excursion vehicle requirements.

The hybrid mission profile, which uses the best attributes of using electric propulsion while eliminating many of the major drawbacks, shows promise for human Mars expeditions.

Power and Propulsion System

The advanced power system is a derivative of the Rotating Multimegawatt Boiling Liquid Metal (RMBLR) design developed as part of the U.S. Department of Energy multimegawatt program (Coomes et al. 1991). The 10-MWe power source uses a boiling liquid-metal reactor with a direct rankine cycle. The five major components are a cermet-fueled, boiling metal fast reactor; a shadow shield; two radial flow turbines with two superconductive alternators; power-conditioning equipment; and a heat rejection thermal control system. The mass breakdown of the power system is given in Table 1.

TABLE 1. Power System Component Mass Breakdown

Power System		Mass (kg)
Reactor		3,620
UN Fuel	860	
Molybdenum	1,520	
Vessel and Reflector	1,240	
Support Structure		350
Potassium Inventory		200
Shadow Shield		5,100
Turbines and Alternators		3,120
Fabric Heat Pipe Radiator		1,700
Auxiliary Cooling		500
Power System Total		14,590

From Coomes et al. (1991)

The fast flux reactor uses boiling alkali metal coolant and cermet fuel. The cermet fuel is composed of a refractory-metal matrix, such as tungsten, with 93% enriched uranium nitride (UN) as the fuel material. A cermet fuel was selected because of its superior thermal characteristics and capability to withstand the high stresses imposed by enhanced fission gas release and fuel swelling. Also, the enclosed coolant channel configuration eliminates crossflow instabilities that occur during boiling in open lattice cores.

The Ljungström turbine provides high power density, will not destabilize the vehicle during load changes, has good thermal dynamic characteristics, and is compatible with a low exit quality working fluid. The Ljungström turbine has no stators, but instead uses two sets of counterrotating blades for each stage. Staging occurs in the radial direction, resulting in a series of concentric blade rings attached to each disk. The disks, connected to a counterrotating alternator, rotate in opposing direction. Using this design, a very compact machine with balanced rotational inertia results. This is important for space applications, because excessive vehicle torque cannot be tolerated. In a radial flow turbine, the working fluid velocity is perpendicular to the axis of rotation and parallel to the acceleration, with uniform blade speed along the entire length of the blade. This allows better optimization of the flow and work coefficients, resulting in fewer stages required than for a comparable axial flow machine. Through careful thermal expansion design, the Ljungström turbine has superior ability to handle rapid thermal transients and load ramping compared to equivalent axial flow machines. The exit quality of the turbine can be as low as 70%, exhibiting the same erosion resistance of an axial flow machine having an exit quality of 88%. In addition, less insulation is required because the higher temperatures are in the center of the machine, with the outer stages at the condenser temperature. Overall Ljungström turbine machine efficiency is 1% to 2% higher than the axial design.

The electric propulsion system consists of magnetoplasmadynamic (MPD) thrusters, power conditioning, and thermal control systems. Three-phase AC is fed through transformers and rectifiers to the thruster. The MPD thruster system has a specific impulse of 5000 s, a power input-to-thrust generated efficiency of 50%, and a mass of 8460 kg for a 10-MWe system (Coomes et al. 1991).

The nuclear thermal propulsion system, used for high-thrust maneuvers near planetary bodies, uses a NERVA derivative technology, solid-core, hydrogen rocket. The specific impulse is 925 s, with a 334,000-N (75,000 lbf) thrust and a mass of 4500 kg (Clark et al. 1991). This specific impulse corresponds to a chamber pressure of 2700 K achieved using composite fuels developed in the NERVA program. The mass includes a man-rated shield for the rocket, which, if shared with the nuclear electric reactor shielding, would represent the increased shielding requirements due to the reactors in close proximity, which would include scattering effects and neutron flux linkages between the reactors.

RESULTS

The hybrid mission starts with a 500-km circular Earth orbit and uses the nuclear thermal rocket to achieve escape velocity. The heliocentric transfer is performed with electric propulsion. The Mars capture, from escape velocity to a 500-km circular orbit, was also performed using the thermal propulsion system. The mass for the outbound vehicle is shown in Table 2.

TABLE 2. Piloted Vehicle Mass Breakdown

Piloted Hybrid Vehicle	Mass (kg)
Habitat Module and ECCV	53,300
Nuclear Electric Power System	14,590
Electric Propulsion System, PMAD	8,460
Nuclear Thermal Rocket	4,500
Hydrogen Propellant	97,313
Propellant Tanks	14,597
Initial Mass in Low Earth Orbit	192,760

The return propellant and tanks are carried to low Mars orbit using an electric propulsion cargo vehicle along with the Mars excursion vehicle and all of the surface cargo and supplies needed to perform the mission. The return propellant and tank mass for the piloted vehicle is 41,407 kg. On the Earth return leg of the mission, the vehicle uses electric propulsion to slow to escape velocity upon intercepting Earth. The crew returns to Earth using the ECCV while the vehicle performs a parabolic flyby of the planet. This allows the vehicle to perform a small delta-velocity maneuver to capture the vehicle in a highly eccentric orbit of Earth, if so desired.

Gravity Losses

By using the electric power reactor as a heating source for the thermal propulsion, some degradation in the specific impulse and thrust may result. The specific impulse is determined primarily by the temperature at which the propellant gas is heated. A conventional nuclear thermal rocket is designed to operate at 2700 K during the burn. A nuclear electric reactor, on the other hand, is much cooler at 1500 K. The thrust is determined by the power produced by the nuclear reaction. For typical thermal rockets producing 334,000 N (75,000 lbf) thrust, 1500 MW of thermal power is required. To produce 10 MW of electric power, a nuclear electric reactor produces roughly 32 MW of thermal power.

The primary difference between nuclear electric and nuclear thermal reactor designs is the operating lifetime of the reactor, trading high temperature and power output versus life. While the nuclear electric reactor must operate continuously for many years, the nuclear thermal reactor is at full power for only a few hours. By designing a long-life reactor that could tolerate brief excursions of high power and high temperature operation, it should be possible to utilize a common reactor to both produce multimegawatt electrical power during normal operation and generate significant thrust during brief periods of extreme operating conditions. However, problems associated with the control of the reactor during the transition between these two operating regimes, neutronic and thermal transfer difficulties associated with designing a reactor to be cooled by liquid potassium during one mode of operation and gaseous hydrogen during another mode, and the fact that neither mode could be designed to provide optimum performance all lead to the conclusion that hybrid propulsion should retain independent reactors for the perspective functions. However, if shielding or "cross talk" difficulties associated with two reactors in close proximity becomes a serious design issue, a common reactor to provide both functions should be pursued.

One of the primary problems associated with using reduced thrust is the gravity loss associated with thrusting against a strong gravitational well near a planetary body. If it is desired to use one long burn rather than a series of shorter perigee burns to achieve escape or capture near a planetary body, sufficient thrust must be applied to the vehicle to minimize the amount of time that the thrust is applied. Any thrust applied when the vehicle is not at perigee results in nonoptimum use of fuel (Madsen et al. 1991).

To investigate the effect of gravity losses for reduced levels of thrust and specific impulse, we used a 100,000-kg vehicle in a 500 km circular Mars orbit thrusting until escape velocity was reached. We varied the thrust from 334,000 N (75,000 lbf) to 10,438 N (2,350 lbf) and the specific impulse between 925 s and 650 s. The results are presented in Figure 1. The specific impulse clearly has a significant impact on the amount of propellant consumed, but does not have a significant impact on the gravity loss associated with reducing thrust. Also, for thrust levels above 100,000 N (22,500 lbf), the gravity loss impact is minimal. Therefore, in a truly hybrid propulsion design where the same reactor will be used for both electrical generation and thermal propellant heating, more attention should be given to achieving high specific impulse rather than producing high thrust. This indicates that increased temperature excursions during the thrust mode are more crucial than the ability to throttle the reactor to very high power burst-mode operation.

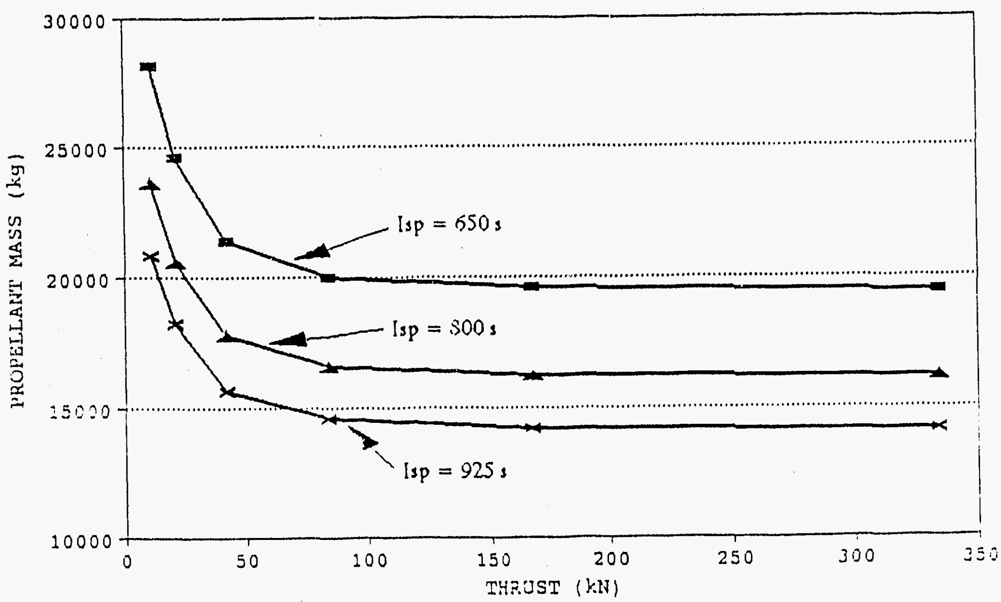

FIGURE 1. Propellant Mass for 100,000-kg Vehicle Thrusting from 500-km Orbit to Escape Velocity

CONCLUSIONS

The nuclear hybrid propulsion scheme is a viable alternative for short-duration Mars missions. By utilizing the high specific impulse and small propellant masses of electric propulsion to achieve rapid heliocentric transit, and nuclear thermal propulsion for impulsive maneuvers near planetary bodies to eliminate the very long spiral time associated with electric propulsion alone, a mission that is both mass effective and of short duration is possible. Using the same reactor to provide both electrical power production and thermal heating of gaseous propellant for high thrust, does not appear to offer the significant mass savings needed to warrant development of the technology. However, reducing the thrust for the thermal propulsion would not seriously impact the propulsion mass required for Mars capture and escape implying that an electric power reactor could be used for both functions, provided that high temperature operation could be tolerated for the short durations necessary to produce the specific impulse needed for a viable thermal propulsion system.

Acknowledgments

This work was performed at the Pacific Northwest Laboratory and supported by the U.S. Department of Energy under Contract Number DE-AC06-76RLO 1830. Pacific Northwest Laboratory is operated for the U.S. Department of Energy by Battelle Memorial Institute.

References

Buden, D., and D. Bartine (1991) "Technology Needs for Manned Mars Mission," AIAA 91-3461, American Institute of Aeronautics and Astronautics, New York.

Clark, J. S., S. Howe, P. McDaniel, and M. Stanley (1991) "Nuclear Thermal Propulsion Technology: Summary of FY 1991 Interagency Panel Planning," AIAA 91-3631, American Institute of Aeronautics and Astronautics, New York.

Coomes, E. P., J.E. Dagle, and J. A. Bamberger (1991) "PEGASUS: An Integrated Power and Propulsion System for the Space Exploration Initiative," AIAA 91-3611, American Institute of Aeronautics and Astronautics, New York.

Doherty, M. P. (1991) "Blazing the Trailway: Nuclear Electric Propulsion and Its Technology Program Plans," AIAA 91-3441, American Institute of Aeronautics and Astronautics, New York.

Hack, K. J., J. A. George, and L. A. Dudzinski (1991) "Nuclear Electric Propulsion Mission Performance for Fast Piloted Mars Missions," AIAA 91-3488, American Institute of Aeronautics and Astronautics, New York.

Kirk, W. L. et al. (1991) *An Investigation of Dual-Mode Operation of a Nuclear-Thermal Rocket Engine,* LA-12102, Los Alamos National Laboratory, Los Alamos, New Mexico.

Madsen, W. W., T. S. Olson, and A. S. Siahpush (1991) "Optimization of Perigee Burns for Manned Interplanetary Missions," AIAA 91-3402, American Institute of Aeronautics and Astronautics, New York.

END

DATE FILMED

3/18/92