IN-20
136206
P.12

NASA Contractor Report 190788

Combined High and Low Thrust Propulsion for Fast Piloted Mars Missions

James H. Gilland and Steve R. Oleson
Sverdrup Technology, Inc.
Lewis Research Center
Brook Park, Ohio

November 1992

NASA

COMBINED HIGH AND LOW THRUST PROPULSION FOR FAST PILOTED MARS MISSIONS

JAMES H. GILLAND, STEVEN R. OLESON
SVERDRUP TECHNOLOGY, INC.
NASA LEWIS RESEARCH CENTER GROUP
BROOK PARK, OHIO
(216) 977 - 7093

ABSTRACT

The mission benefits of using both high thrust nuclear thermal propulsion (NTP) and low acceleration, high specific impulse nuclear electric propulsion (NEP) to reduce piloted trip times to Mars with reasonable initial mass are assessed. Recent updates in mission design, such as the Earth fly-by return, are assessed for their impact on previous studies. In addition, the Synthesis Commission split mission to Mars in 2014 is also assessed using combined propulsion. Results show an 80 to 100 day reduction in trip time over the reference NTP or NEP systems and missions, with comparable or reduced vehicle initial masses. The impacts of the mission and system analyses upon technology planning and design are discussed.

INTRODUCTION

The human exploration of Mars introduces conflicting requirements upon space transportation technology. The Martian environment requires massive support equipment such as habitats for survival on the surface and Mars excursion vehicles (MEV's) for transport from orbit to surface and return. The interplanetary transportation of crew, in turn, imposes fast trip time restrictions due to concerns of exposure to galactic cosmic radiation and zero gravity. Simultaneously, the economics and physical limitations of Earth-to-orbit launch limit the amount of vehicle and propellant mass that can be reasonably used for these missions. Thus, space propulsion systems are faced with the challenge of transporting large payload masses to another planet quickly, and with an acceptably low initial launch mass. These mission level constraints have impact upon propulsion system specific impulse, power, mass, and efficiency. In the case of high thrust, impulsive technologies, the principle performance emphasis is upon specific impulse and power requirements. In the case of low thrust technologies such as nuclear electric propulsion, the system figures of merit are power, specific impulse, specific mass (ratio of propulsion system mass to electric power output, kg/kWe), and thrust efficiency (ratio of output thrust power to input electrical power). In either case, the combination of mission demands drives systems to high power, high specific impulse and low mass; while maintaining suitable reliability and operating lifetimes.

High thrust systems such as chemical or nuclear thermal rockets depend upon high propellant temperatures to produce high specific impulse. Increased specific impulse thus implies higher material temperatures, affecting the lifetime, safety margins and load bearing capabilities of materials. Newer materials may be required in order to meet mission objectives. Current chemical propulsion systems are also limited by the energy release per unit mass of the combustion reaction. The highest specific impulses that might be expected from such systems are between 450 and 500 seconds[1]. Nuclear thermal propulsion state of the art is considered to be the NERVA systems of 20 years ago, which attained specific impulses up to 825 to 870 seconds, at thrust levels of 330 kN in ground tests[2,3]. These systems consisted of pyrolitic graphite coated UC_2 pellets mixed in a graphite matrix. The core was built from hexagonal cylindrical fuel elements extruded from the graphite/pellet material. The concept is based upon transferring heat from the solid core into the hydrogen propellant/coolant flowing axially through longitudinal passages in the core. Ultimate propellant temperature, and therefore specific impulse, are limited by temperature limits of the solid core materials. Specific impulses of 900 to 1000 seconds may ultimately be achieved through the use of new core materials, such as UC_2/carbon composite fuels, or from untested core designs, such as the particle bed reactor concepts. The ultimate capability for a "nuclear thermal rocket" would be the gas core rocket, which is based upon a fissioning uranium plasma at temperatures of 10^4 or 10^5 K, and specific impulses of 2000 to 7000 seconds[3,4]. Obviously, the gas core concept is highly speculative, and is probably not a feasible candidate for early human planetary exploration.

Low thrust systems include both solar and nuclear electric propulsion concepts. Electric propulsion systems are inherently low acceleration, requiring operation over a significant portion of the mission time. In the context of highest performance, the nuclear electric propulsion (NEP) systems are the clear choice, because of their ability to maintain a constant power output regardless of distance from the Sun. NEP systems also have no performance penalty due to power system degradation in the Van Allen Radiation Belts, as do most photovoltaic array power systems[5]. The NEP system consists of a nuclear reactor delivering heat to a dynamic (mechanical, magnetohydrodynamic) or static (thermoelectric, thermionic, electrochemical) power conversion system which converts it to electricity. Due to the inefficiencies of power conversion, these systems also require a heat rejection system to eliminate waste heat through radiation to space. The electric power produced is conditioned and transmitted to electric thrusters, which convert the electrical energy into thrust energy by accelerating a plasma propellant through the application of electromagnetic fields. Current state-of-the art NEP technology is the SP-100 reactor: a UN fuel pin lithium cooled fast spectrum reactor. The reactor is currently sized at 2.5 MWt, with a 1350 K temperature at the power conversion interface[6]. MWe power levels assumed in this paper could be obtained from a scaled up, "Growth" SP-100 reactor system, coupled with either potassium rankine or brayton power conversion and heat pipe radiator[7]. Thrust would be produced either by ion or magnetoplasmadynamic (MPD) thrusters[8].

NEP systems thus require significant dry mass, resulting in a propulsion system which has a mass comparable to the payload and/or propellant; however, specific impulse of 5000 to 10000 seconds reduce propellant mass, resulting in low vehicle intial mass. Vehicle acceleration depends upon both the amount of power that can be generated as well as the mass of the system, commonly expressed as the specific mass. The NEP vehicle's acceleration in turn determines the trip times achievable. In order to achieve trip times for planetary missions, the NEP systems must be lightweight, efficient, and produce 10 MWe of power or more for extended periods of time. By the laws of thermodynamics, higher efficiencies can be achieved by operating at higher peak temperatures and lower heat rejection temperatures. Conversely, increased heat rejection temperatures also minimize system mass. The net result is that higher system performance is best achieved by increasing peak cycle temperatures in the reactor and power conversion systems. Unfortunately, the need for higher powers and temperatures, long lifetime, and high efficiency drive NEP technologies to advanced materials and concepts, as with the high thrust systems.

From an overall mission standpoint, each form of propulsion has benefits for specific portions of a mission. The high thrust systems allow for rapid escape from the relatively high gravity fields of planets, whereas the low acceleration NEP systems require slow spiral escape trajectories. In heliocentric space, the thermally limited specific impulse of the high thrust systems represents a significant propellant requirement for high energy planetary missions, to the point where propellants and tanks make up most of the vehicle mass. Conversely, the order of magnitude higher specific impulse of NEP systems, in conjunction with the relatively low gravity field of the sun in interplanetary space, allows NEP vehicles to perform the planetary transfer portion of a mission with propellant requirements much lower than with the high thrust systems. While the "ideal" propulsion system would allow for high thrust-to-weight and high specific impulse, at the expense of a long and uncertain technology development program, we currently have two propulsion systems under development that could be combined to achieve a similar result.

The work presented herein represents an update of previous work[9]. The aims of this ongoing study are to examine the potential benefits, at a mission analysis level, of the use of combined high and low thrust propulsion systems for short trip time missions. The analysis is performed parametrically, based on propulsion system projections obtained from the literature and related systems and mission analyses of Nuclear Thermal and Nuclear Electric Propulsion systems. In particular, this paper presents an assessment of the use of an Earth fly-by/Earth Crew Capture Vehicle (ECCV) maneuver for crew return, rather than the full propulsive vehicle capture mission mode previously considered for the 2018, All-Up mission. In addition, a "split/sprint" mission mode is considered, using mission payloads and descriptions developed in the NASA assessment of the Synthesis Group report[10,11]. The bases of comparison used throughout this study are initial mass in Low Earth Orbit (IMLEO) and piloted trip time. The combined system performance is compared to Nuclear Thermal Propulsion (NTP) and NEP systems.

SYSTEM ASSUMPTIONS

Parametric system characteristics used for the combined propulsion analysis are given, followed by system assumptions for the reference cases.

Combined Nuclear Thermal Propulsion

A NERVA - based engine using hydrogen propellant was assumed: 333 kN thrust, at specific impulses of 850 and 925 seconds. Engine mass for each was assumed to be 10 MT, including external disk shields for crew shielding and a 10% contingency. Two engines were arbitrarily assumed for each vehicle. Hydrogen tankage fraction was 0.15.

Combined Nuclear Electric Propulsion

System parameters of 10 kg/kWe, 5 MWe were assumed. These values represent conservative projections of NEP system characteristics using SP-100 reactor technology scaled to 25 MWt in conjunction with Rankine dynamic power conversion. A power level of 5 MWe was selected for its commonality with Mars and Lunar cargo vehicle requirements, as determined in past NEP studies[7,12]. A less complete assessment of 5 kg/kWe system performance was also made. Ion propulsion was used, operating at 5000 seconds and 70% efficiency (electric-to-thrust conversion efficiency). The tankage fraction for the argon propellant was taken to be 0.1.

Reference Nuclear Thermal Propulsion

The reference engine for the "All-Up" 2018 mission was a 925 second, 333 kN engine, weighing 20.2 MT with disk shield. Tankage fractions of 0.13 to 0.15 were used, depending upon mission assumptions. A single engine was used for the entire mission, with 3 perigee burns at Earth departure to reduce g-loss effects[13]. For the 2014 split sprint mission, two 333 kN engines, massing a total of 12.4 MT, were assumed. Average tankage fraction for this mission was 0.168[11].

Reference Nuclear Electric Propulsion

A "Growth" SP-100 25 MWt reactor with potassium rankine power conversion was assumed for the 2014 split/sprint mission. System life was 2 years, with 50% redundancy on power conversion and distribution components. Multiple 5 MWe power modules were baselined, allowing 10 and 15 MWe piloted vehicles. Argon ion thrusters, operating at 5000 seconds specific impulse were used, with a tankage fraction of 0.1. Propulsion system specific mass was 7.3

kg/kWe at either 10 or 15 MWe.[14]

ANALYSIS APPROACH

Trajectory analysis was performed using the QT2 trajectory code. This tool is based upon the CHEBYTOP low thrust trajectory analysis routines, with the addition of external optimization drivers for trajectory and system optimization. The approach used in these analyses was aimed at determining minimum initial vehicle mass for a given trip time. Free variables in the optimization were outbound leg time, launch date, and excess hyperbolic velocities at departure and arrival. Due to the large parameter space, iteration was required to determine minimum initial mass values. The code does not assess gravity losses for planetary departure; all high thrust calculations are based on impulsive ΔV's. The possibility of abort trajectories, either powered or free return, were also not taken into account in the analysis, but could be considered in future studies. Instead, these assessments are intended as scoping studies to determine the potential benefits of the combined propulsion approach.

MISSION ASSUMPTIONS

Two missions were considered. The first is an "All-Up" mission, with crew and cargo on a single vehicle. Previous mission analysis of the combined propulsion option for the year 2018 using propulsive capture at Earth is compared to recent results for an Earth fly-by/ECCV scenario, as well as to NTP calculations for the same opportunity and mission profile. The second mission is based upon the Synthesis Group first Mars mission in 2014, using a split crew/cargo mission scenario. Both missions are described in detail below.

All-Up 2018 Mission

The reference mission is an opposition-class, 30 day stay time mission, based on the 1989 NASA "90-Day Study". The mission time frame of interest was from 2010 to 2025, with 2016 identified as a reference case and 2018 and 2025 identified as the easy and difficult opportunities, respectively. For examination of the effects of the ECCV return mission profile, the 2018 opportunity was selected for comparison with the previous, propulsive return results. The ground rules for these studies are summarized from previous work, with the addition of the ECCV ground rules used for this study.

Reference mission parameters for the all-NTP cases are those used in the 1990 NTP Workshop[13]. Departure is from a 407 km Low Earth Orbit (LEO). Increases in departure ΔV due to gravity losses were minimized through the use of a 3 burn, perigee-kick maneuver. The trajectory parameters were optimized for the desired trip time, including Venus swingbys. The Mars arrival and parking orbit was set at an elliptical orbit of 250 km X 1 sol. From this orbit, the aerobraked Mars Excursion Vehicle (MEV) descended to the surface. The crew return to the Mars Transfer Vehicle (MTV) via an ascent vehicle after a 30 day stay on the surface. Earth return was by ECCV, with the rest of the vehicle flying by Earth. An ECCV velocity limit of 9.4 km/s was imposed. For the all-propulsive vehicle return, the Earth return orbit was a 500 X 1 sol elliptical orbit.

For the low thrust systems, certain aspects of the mission profile were adapted to the needs of these propulsion systems. The low thrust vehicles also start at LEO, requiring spiral times on the order of months to escape the Earth. Extended exposure to the Van Allen Radiation Belts during this spiral phase prevents having the crew on-board at this time. Instead, the crew rendezvous with the vehicle using a chemical propulsion system (perhaps a Lunar Transfer Vehicle) at some point beyond the radiation belts. The low thrust vehicle then accelerates to escape and on to Mars.

Because of the time required to spiral in and out at Mars, a high circular orbit was selected to limit this period. For the study, a Deimos altitude of 20077 km was selected for a Mars parking orbit; this orbit is also close to the Areosynchronous orbit of 20430 km. The impact of this orbit on the MEV and ascent vehicle is expected to be small due to the use of aerobraking and the small ascent vehicle mass. Spiral times to and from this orbit were on the order of days. Recent NEP mission analyses of this mission included the effects of both splitting the crew and cargo onto separate vehicles, as well as utilizing the Earth fly-by maneuver baselined for the high thrust systems. In addition, the 7.3 kg/kWe system was used. For the All-NEP systems, trip time reductions on the order of 100 days were obtained through these mission design features.

In all cases, mission payloads are essentially equal. The outbound payload includes the MTV habitat, plus the MEV and associated scientific equipment, and an ECCV. These masses are shown below.[7,9]

Component	Mass (MT)
MTV Habitat	40.3
MEV(left at Mars)	84.0
ECCV	7.0

Table 1. Piloted Mars Mission Payload Masses for the 2018 All-Up Mars Mission.

The combined propulsion system mission assumed is similar to that of the high thrust mission. The vehicle is assumed to start from LEO with a high thrust burn; the NEP system is then turned on for the heliocentric portion of the trip. In the case of a "single burn" combined mission, the high thrust system and tankage are jettisoned after this initial impulse. The NEP system then serves as the sole propulsion system, following the mission scenario outlined for the low thrust system. In the case of the "multiburn" combined mission, the high thrust system is used for Mars capture and escape as well. In previous studies, two high thrust stages were assumed for the vehicle: a more massive, higher thrust stage for Earth departure and Mars capture, and a single 333 kN NTP engine for Mars escape and Earth capture[9]. This was deemed overly conservative, given the lifetime capabilities projected for NTP engines, so a single stage with two engines was assumed in the current study. The Mars orbit is the same 250 km X 1 sol elliptical orbit used for the high thrust mission. The ECCV velocity limit for the Earth return fly-by maneuver is the same as for the other missions, 9.4 km/s. For this study, the Earth return maneuvers are always performed by the NEP system; the NTP stage is jettisoned after Mars departure.

Split/Sprint Mission

The Synthesis Group recently proposed a reference first Mars mission, with a nominal launch in the year 2014. The mission profile is a Mars opposition mission, with 100 day stay time. A split mission is baselined, with the crew and a single MEV on one vehicle, and three additional MEV's on the cargo mission. The cargo mission is launched in the previous opportunity, 2012. The reference propulsion system was NTP. All-NTP mission profiles and performance data are taken from recent NASA studies of this mission scenario[11]. The nominal mission lasts 540 days, including stay time, with an option for a 528 day powered abort at Mars. In the reference mission, the outbound trip time was fixed at 150 days, for crew health and safety reasons. Earth departure is from a 407 km Low Earth Orbit. Mars arrival and departure orbit is an elliptical 250 km X 1 Sol orbit, as in the All-Up study. The Earth return ECCV velocity limit is 9.4 km/s.

The NEP split/sprint mission is similar, including the split mission and Earth fly-by scenarios. The results presented herein are also taken from recent NASA studies[15,16,17]. As in previous low thrust mission studies, the crew rendezvous with the MTV after it has spiralled through the Van Allen Belts. The crew taxi mass is estimated to be 57 MT. Departure and arrival orbits are the same as stated for the All-Up mission. The ECCV velocity limit is also the same. In the NASA studies, the NEP system assumptions of 7.3 kg/kWe, 10 - 15 MWe, could not accomplish a 150 day outbound leg; however, total NEP round trip times were comparable to the NTP reference mission, depending on power level.[15,16,17]

The combined propulsion mission is very similar to the reference NTP profile. Departure and arrival orbits are the same. As in previous studies, two modes of combined propulsion were considered: the single burn and the multiburn. In the single burn case, the high thrust system is used for Earth departure, then jettisoned. NEP is used for the rest of the journey. The Mars arrival and departure orbit is 20077 km, as before. In the multiburn case, the high thrust stage is used for Earth escape, Mars capture, and Mars escape. A 250 km X 1 Sol Mars arrival and departure orbit are used. NEP is used for the Earth return maneuvers. The NTP stage is jettisoned after the Mars departure phase. Both high and low thrust outbound tankage are jettisoned after Mars arrival.

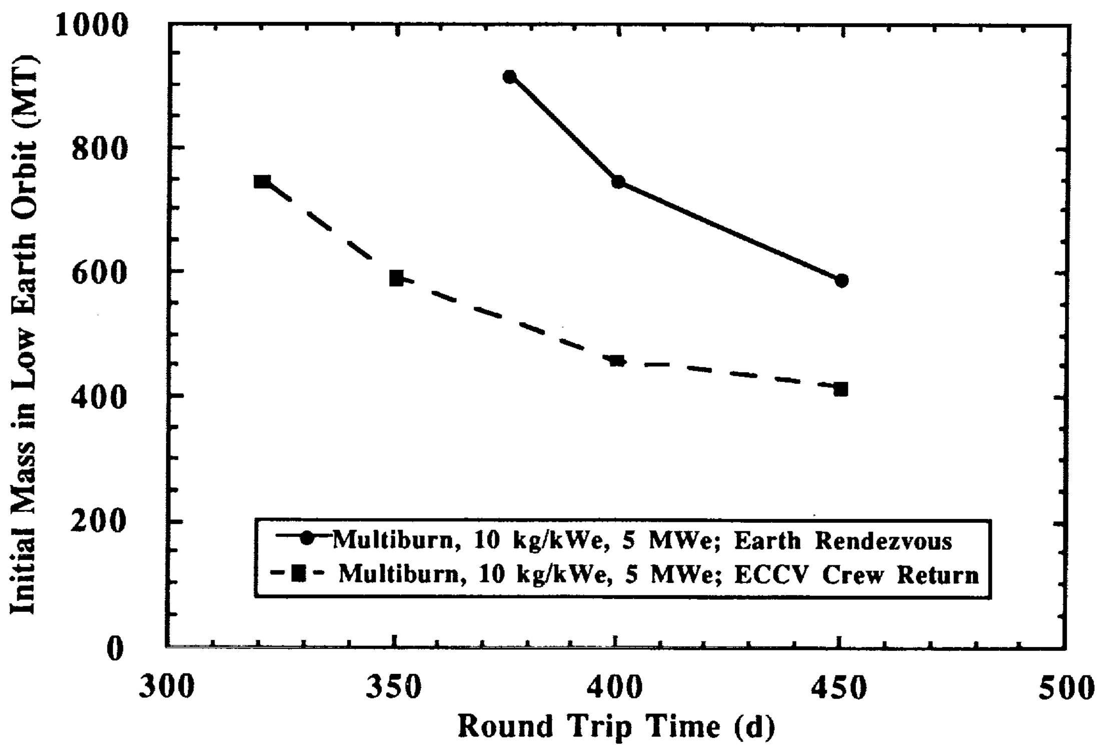

Figure 1. Effect of Earth Fly-by Return for 2018 All-Up Combined Propulsion Mission.

RESULTS

All-Up Combined Mission (2018)

The effects of using a Earth fly-by with ECCV return mode are shown in Figure 1. The results from the previous combined study, using full propulsive capture of the Mars transfer vehicle at Earth[6], are compared to a similar system using the fly-by return. The differences in the two studies are

1.) The previous, more massive mission used full propulsive capture at Earth; the recent results assume an Earth fly-by at 9.4 km/s.
2.) The Earth capture mission assumed a 900 second specific impulse, the fly-by assumes 925 s in keeping with baseline NTP performance projections.
3.) The Earth capture mission had two NTP stages, one 30 MT and the other 10 MT, jettisoned after Mars and Earth capture, respectively; the fly-by has a single, 20 MT stage that is jettisoned after Mars departure.

A mass savings of from 90 to 350 tons is possible for the same trip times by incorporating the fly-by maneuver; correspondingly, a 100 day reduction in trip time is possible for comparable initial mass. While some of the mass savings arise from the difference in staging scenarios and specific impulse, the majority of the trip time and mass savings arise from the reduced Earth return constraint, which allows more effective use of the NEP. This follows from similar results observed in the all-NEP studies.

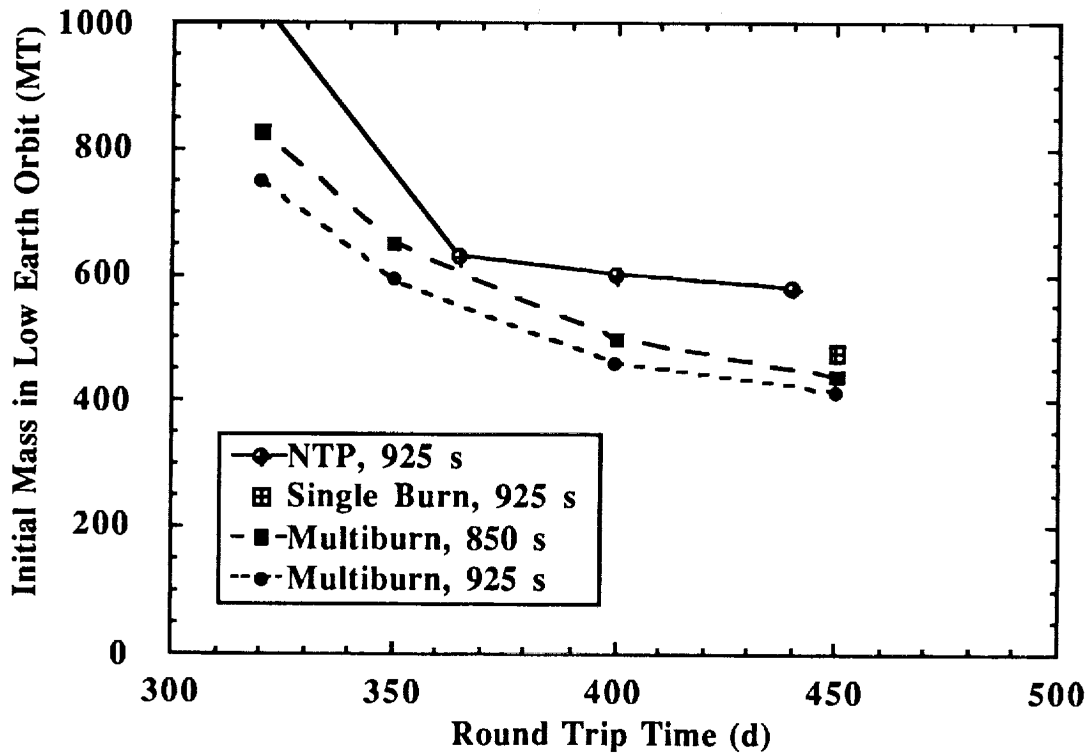

Figure 2. Combined Propulsion Mission Performance for the 2018 All-Up Mission.

The relative merits of the combined propulsion systems compared to the reference NTP system are shown in Figure 2. Three cases of combined propulsion are shown: a single burn 925 second, multiburn 925 second, and multiburn 850 second. The single burn option was found to be capable of a 450 day trip time, with a moderate mass savings of 100 tons over the 434 day NTP case. The multiburn cases produced similar reductions for trip times of 400 to 450 days, before almost equaling NTP performance in the vicinity of 1 year round trip time. Below one year, the combined propulsion systems increase relatively gradually in mass to a value of 750 to 825 MT at 320 days. The NTP mass climbs rapidly above 1000 MT, probably due to the lack of any Venus swing-by maneuver to ease the propulsion requirements, as was used in the slower trips. Thus, the combined propulsion systems use the capability of the NEP system to ameliorate the lack of gravity assists. This results in reduced variation in vehicle performance from opportunity to opportunity, and could allow wider launch windows than possible using gravity assists. The reduced variation with opportunity was also seen in the previous, all propulsive studies.

Split/Sprint Mission (2014)

Several alternatives have been assessed for this mission: All-NTP, All-NEP, and now, combined propulsion. Although this mission scenario includes both cargo and piloted vehicles, the analysis and comparison herein focusses upon the piloted vehicle only, since the cargo vehicle propulsion system is not likely to determine the propulsion system used for Mars mission. For reference, the NTP cargo vehicle initial mass was found to be 591.3 MT[11]; the NEP cargo vehicle was 392.5 MT[17]. Either of these numbers should be added to the piloted vehicle masses shown below to determine the total initial mass for the split mission. The two reference cases are the all NTP option, using NTP for both the piloted and cargo vehicles, and the all NEP option, using the 7.3 kg/kWe NEP system at power levels of 10 and 15 MWe. In the combined scenario, NTP stages operating at 925 and 850 seconds were considered for the multiburn cases. A multiburn case with a constrained outbound trip time of 150 days or less was also assessed for a 925 second NTP system. The combined NEP system was fixed at 5 MWe, 10 kg/kWe, 5000 seconds specific impulse for all combined mission analyses. The resulting piloted vehicle initial mass as a function of trip time is shown in Figure 3.

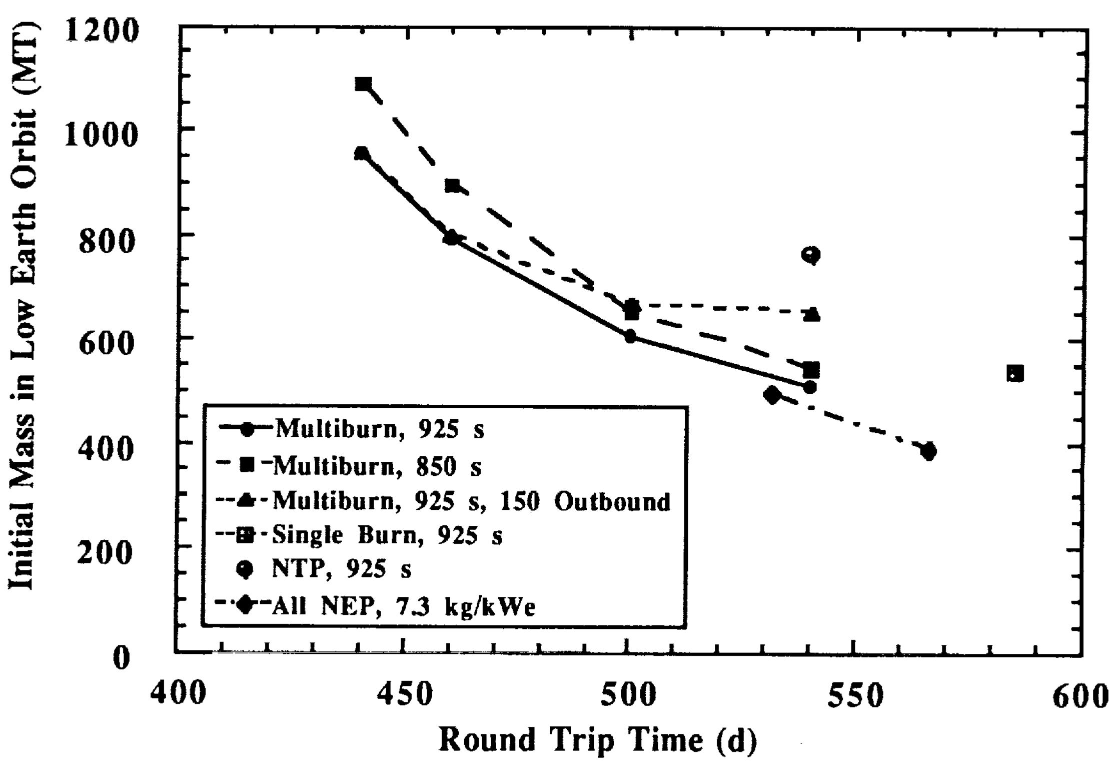

Figure 3. Combined Propulsion Mission Performance for the 2014 Split/Sprint Mission.

As in the 2018 mission study the use of a single high thrust stage at Earth departure does not provide significant mission benefit in terms of trip time reduction. A single high thrust burn is insufficient to allow a 5 MWe NEP system to meet the reference trip time of 540 days; however, a 585 day trip is possible, with an approximately 200 MT reduction in mass from the all-NEP system, and a 150 MT increase in mass over the 10 MWe all-NEP system. Both NTP and NEP systems can better this trip time.

The use of the multiburn combined propulsion scenario is seen to provide reductions in mass over the reference NTP system at the baseline trip time of 540 days. The reduction in mass for a 925 second, 10 kg/kWe, 5 MWe combined system at 540 days is approximately 200 MT from the all-NTP system; a comparable reduction is also available even for an 850 second NTP/NEP system using the multiburn approach. If the 150 day outbound trip time constraint is imposed, only a 100 MT reduction results for the 540 day mission. All of these systems are more massive than the reference 15 MWe NEP system for that trip time. However, the use of the combined systems is also seen to reduce trip time by as much as 80 days for initial masses comparable to the baseline NTP system mass while maintaining the 150 day outbound constraint. Such a trip time is also beyond the capability of the high power all-NEP systems, unless system specific mass is drastically reduced and power is raised to levels greater than 20 MWe[16]. It is interesting to note the relatively small dependence of the combined system mass upon the NTP specific impulse value. This is because the use of the NEP system keeps the high thrust ΔV values relatively low, minimizing the sensitivity. Similarly, the use of the high thrust systems to impart some velocity to the vehicle at departure or arrival reduces the mission sensitivity to NEP specific mass and power level.

CONCLUSIONS

By utilizing high and low thrust systems in those regimes in which they each excel, significant mission benefits can be obtained in terms of trip time and mass for piloted Mars missions. Implementing combined mission profiles that are similar to those developed for high thrust missions allows the full benefit of the concept to emerge. Specifically, planetary fly-bys and split/sprint mission modes have been shown to enable combined propulsion systems to reduce trip times using relatively near term system performance assumptions.

These results have been intended as scoping studies for the purpose of determining the potential advantages of combined propulsion. More detailed studies of trajectories, including more precise trajectory optimization, gravity loss effects, abort options, and conjunction missions are also required. In addition, the vehicle integration and design of a combined mission will involve compromises between the two extremes of high and low thrust system design. In particular, deployment and support of heat rejection radiators on board a high thrust/weight vehicle will pose a challenge, as will propellant tankage design and insulation. The presence of two nuclear reactors on board a single spacecraft, while providing some reassuring redundancy, also introduces payload and crew shielding requirements. These studies are beyond the scope of this analysis.

The impact of a combined propulsion system on technology development planning could be dramatic. The scoping studies reported herein have shown the benefits of utilizing both high and low thrust propulsion to ease the technology development risk of both. By using near term, less risky NTP and NEP systems in tandem, a space transportation system can be built in a timely fashion, with mission benefits beyond those attainable by a single system. The combined systems assessed above allow for 850 second, state of the art NTP systems, and 10 kg/kWe, 5 MWe systems scalable from existing reactor and thruster technology programs to be used to perform missions previously thought to require higher specific impulses (in the case of NTP) or power levels (in the case of NEP). In addition, the development of a 5 MWe NEP module is capable of providing cargo mission support for the split mission scenario. A less tangible but no less important consideration is the presence of two separate propulsion systems on a single spacecraft to provide additional redundancy for piloted missions.

REFERENCES

1.) STONE, J. R., SHAW, L. M., AND AUKERMAN, C. A., *Plans for the Development of*

Cryogenic Engines for Space Exploration, AIAA Paper No. 91 - 3438, 1991.

2.) GUNN, S. V., *Development of Nuclear Rocket Engine Technology*, AIAA Paper No. 89-2386, 1989.

3.) BOROWSKI, S. K., *Nuclear Propulsion - A Vital Technology for the Exploration of Mars and the Planets Beyond*, in Proceedings of the Case for Mars Conference III, 1988.

4.) RAGSDALE, R. G. AND WILLIS, E. A., *Gas-Core Rocket Reactors - A New Look*, AIAA Paper No. 71-641, 1971

5.) HICKMAN, J. M., ET. AL., *Systems Analysis of Mars Solar Electric Propulsion Vehicles*, AIAA Paper No. 90-3824, 1990.

6.) MONDT, J. F., *Overview of the SP-100 Program*, AIAA Paper No. 91-3585, 1991.

7.) HACK, K. J., GEORGE, J. A., RIEHL, J. P., AND GILLAND, J. H., *Evolutionary Use Of Nuclear Electric Propulsion*, AIAA Paper No. 90-3821, 1990.

8.) GILLAND, J. H., MYERS, R. M. AND PATTERSON, M. J., *Multimegawatt Electric Propulsion System Design Considerations*, AIAA Paper No. 90-2552, 1990.

9.) GILLAND, J. H., *Synergistic Use of High and Low Thrust Propulsion Systems for Piloted Missions to Mars*, AIAA Paper No. 91-2346, 1991.

10.) SYNTHESIS GROUP REPORT, *America at the Threshold, America's Space Exploration Initiative*, available from the Superintendent of Documents, U.S. Government Printing Office, Washington, D.C., 20402, June 1991.

11.) NASA LUNAR AND MARS EXPLORATION PROGRAM OFFICE, *Analysis of the Synthesis Group's Mars Exploration Architecture*, LMEPO Document XE-91-001, October, 1991.

12.) GILLAND, J. H., *Mission and System Optimization of Nuclear Electric Propulsion Vehicles for Lunar and Mars Missions*, IEPC Paper No. 91-038, AIDAA/AIAA/DGLR/JSASS 22nd International Electric Propulsion Conference, Viareggio, Italy, October 14-17, 1991. NASA CR 189058, 1991.

13.) BOROWSKI, S. K., *Nuclear Thermal Rocket Workshop Reference System - ROVER/NERVA*, Proceedings of NASA/DOE/DOD Nuclear Thermal Propulsion Workshop, Cleveland, OH, July 10-12, 1990. NASA CP-10079.

14.) GEORGE, J. A., *Multimegawatt Nuclear Power Systems for Nuclear Electric Propulsion*, AIAA Paper No. 91-3607, 1991.

15.) HACK, K. J., GEORGE, J. A. AND DUDZINSKI, L. A., *Nuclear Electric Propulsion Mission Performance for Fast Piloted Mars Missions*, AIAA Paper No. 91-3488, 1991.

16.) GEORGE, J. A., HACK, K. J., AND DUDZINSKI, L. A., *Fast Piloted Missions to Mars Using Nuclear Electric Propulsion*, in AIP Conference Proceedings No. 246: Ninth Symposium on Space Nuclear Power Systems, Albuquerque, NM, January 12 - 16, 1992, Vol. I, pp.389-401.

17.) DUDZINSKI, L. A., HACK, K. J., GEFERT, L. P., *Nuclear Electric Propulsion Benefits to Synthesis Missions*, in Proceedings of ANS Nuclear Technologies for Space Exploration Conference, August, 1992.

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*

2. REPORT DATE
November 1992

3. REPORT TYPE AND DATES COVERED
Final Contractor Report

4. TITLE AND SUBTITLE
Combined High and Low Thrust Propulsion for Fast Piloted Mars Missions

5. FUNDING NUMBERS
WU-539-72

6. AUTHOR(S)
James H. Gilland and Steven R. Oleson

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
Sverdrup Technology, Inc.
2001 Aerospace Parkway
Brook Park, Ohio 44142

8. PERFORMING ORGANIZATION REPORT NUMBER
E-7423

9. SPONSORING/MONITORING AGENCY NAMES(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135-3191

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA CR-190788

11. SUPPLEMENTARY NOTES
Project Manager, John S. Clark, (216) 433-7090.

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified - Unlimited
Subject Category 20

12b. DISTRIBUTION CODE

13. ABSTRACT ***(Maximum 200 words)***

The mission benefits of using both high thrust nuclear thermal propulsion (NTP) and low acceleration, high specific impulse nuclear electric propulsion (NEP) to reduce piloted trip times to Mars with reasonable initial mass are assessed. Recent updates in mission design, such as the Earth fly-by return, are assessed for their impact on previous studies. In addition, the Synthesis Commission split mission to Mars in 2014 is also assessed using combined propulsion. Results show an 80 to 100 day reduction in trip time over the reference NTP or NEP systems and missions, with comparable or reduced vehicle initial masses. The impacts of the mission and system analyses upon technology planning and design are discussed.

14. SUBJECT TERMS
Combined propulsion; Nuclear electric propulsion; Nuclear thermal propulsion; Electric propulsion; Mission analysis

15. NUMBER OF PAGES
12

16. PRICE CODE
A03

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	