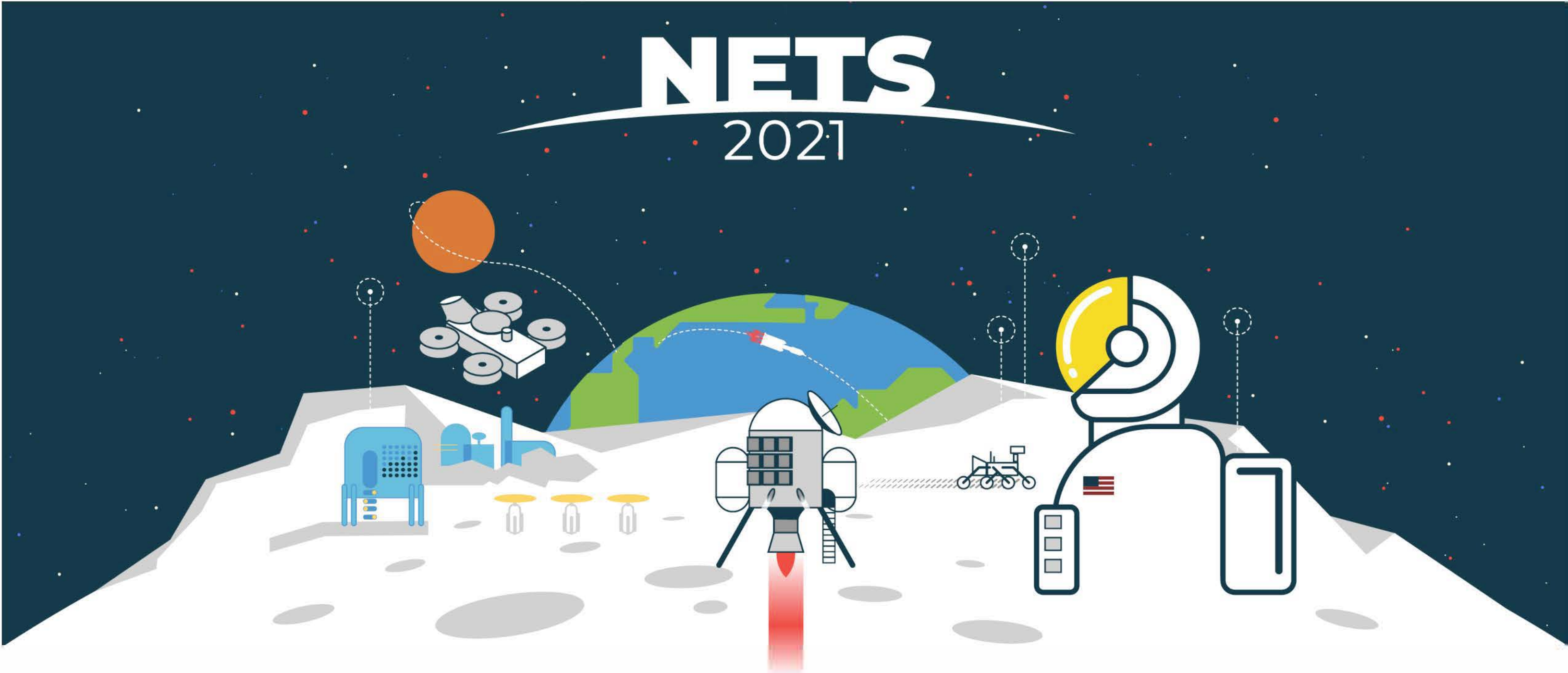

NUCLEAR and
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CONFERENCE PROCEEDINGS

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Track 3: Mission Concepts and Policy for Nuclear Space Systems

Technical Track Chair(s): Bhavya Lal & Susannah Howieson

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Author	Title
Kokan, Timothy	Introduction to a Human Mars Campaign Utilizing Nuclear Thermal Propulsion
Kumar, Soraj	Decoding Mission Design Problem for NTP Systems for Outer Planet Robotic Missions
Kuperman, Alan J.	Avoiding HEU In Space Reactors: An Emerging Consensus
Reynolds, Christopher B.	Modeling of NTP Engine Start-Up, Shutdown, and Cooldown and their Impact on ΔV
Smith, Clayton	The Case for a 50+ Year Radioisotope Power System

INTRODUCTION TO A HUMAN MARS CAMPAIGN UTILIZING NUCLEAR THERMAL PROPULSION

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Nuclear thermal propulsion (NTP) has been extensively researched as a potential main propulsion option for human Mars missions. NTP's combination of high thrust and high fuel efficiency makes it an ideal main propulsion candidate for these types of missions, providing architectural benefits including smaller transportation system masses, reduced trip times, increased abort capabilities, and the potential for transportation infrastructure reuse.

Since 2016, Aerojet Rocketdyne (AR) has been working with NASA and members of industry as part of the NASA Space Technology Mission Directorate. The initial goal of this project was to determine the feasibility and affordability of a low enriched uranium (LEU)-based NTP engine with solid cost and schedule confidence. Having shown feasibility and affordability, the current project focus is on maturing NTP fuel and reactor technology.

As part of this activity, AR has examined potential NTP vehicle configurations to support both short-stay and long-stay crewed Mars missions and potential vehicle synergy to support both mission types.

This paper presents an overview of an envisioned human Mars exploration campaign consisting of uncrewed demonstration flights, crewed short-stay missions, and crewed long-stay missions preparing for the eventual permanent human presence on the surface of Mars.

NOMENCLATURE

AR = Aerojet Rocketdyne
CFM = Cryogenic Fluid Management
CLV = Commercial Launch Vehicle
CONOPS = Concept of Operations
DSH = Deep Space Habitat
DSM = Deep Space Maneuver
EME = Earth-Mars-Earth
EMVE = Earth-Mars-Venus-Earth
EOI = Earth Orbital Insertion
EVA = Extra-Vehicular Activity
EVME = Earth-Venus-Mars-Earth
GCD = Game Changing Development
GRC = Glenn Research Center
Isp = Specific Impulse
LDHEO = Lunar Distance High Earth Orbit
LEU = Low Enriched Uranium
LH2 = Liquid Hydrogen
MAV = Mars Ascent Vehicle
MEO = Medium Earth Orbit
MOI = Mars Orbit Insertion
MSFC = Marshall Space Flight Center
MTV = Mars Transfer Vehicle
NASA = National Aeronautics and Space Administration
NRHO = Near Rectilinear Halo Orbit
NTP = Nuclear Thermal Propulsion
OMS = Orbital Maneuvering System
RCS = Reaction Control System
SLS = Space Launch System
SNP = Space Nuclear Propulsion
SPS = Surface Power System
SOI = Sphere of Influence
STMD = Space Technology Mission Directorate
TDM = Technology Demonstration Missions
TEI = Trans-Earth Injection
TMI = Trans-Mars Injection
ΔV = Delta-V, Velocity change

I. INTRODUCTION

Since 2016, AR has been working with NASA, the Department of Energy, and members of industry as part of the NASA Space Technology Mission Directorate (STMD)[1,2]. Initially under the Nuclear Thermal Propulsion Project within the Game Changing Development (GCD) Program, the goal was to determine the feasibility and affordability of a low enriched uranium (LEU)-based NTP engine with solid cost and schedule confidence.

Having shown feasibility and affordability, the project focus has evolved as part of the STMD Space Nuclear Propulsion (SNP) Project within the Technology Demonstration Missions (TDM) Program. The current focus of the SNP Project is in three major areas:

1. NTP fuel and reactor technology maturation;
2. Identification of nuclear electric propulsion (NEP) subsystem capability gaps and needs;
3. Advancement of critical cryogenic fluid management (CFM) technologies needed for SNP.

As part of this activity, AR has examined potential NTP vehicle configurations to support both short-stay and long-stay crewed Mars missions and potential vehicle synergy to support both mission types[3,4]. A natural follow-on to this work is the definition of a multi-mission human Mars campaign utilizing NTP. An overview of a potential campaign is provided in the following section.

II. HUMAN MARS CAMPAIGN OVERVIEW

Table I provides a sequence of missions for a potential human Mars campaign starting in the early 2030's with initial uncrewed flight demonstration missions, followed by a crewed short-stay Mars mission, and finally to crewed long-stay Mars missions in the 2040's preparing for the eventual permanent human presence on the surface of Mars.

The first uncrewed missions in the envisioned campaign would be used to test out the NTP system, vehicle systems, and deep space habitat on a dress rehearsal for follow-on crewed missions.

The first crewed mission to the Martian surface would use a short-stay opposition class trajectory and establish a minimal set of surface infrastructure to support a 30-day Mars surface stay.

Following the initial crewed surface mission, subsequent missions would transition to long-stay conjunction class trajectories. These trajectories take advantage of optimal Earth-Mars planetary alignments to allow for a simpler Mars Transfer Vehicle (MTV) and a longer surface stay time of >400 days. These long-stay missions would build on previous infrastructure, establishing a more permanent presence in the form of an outpost station. Following the first long-stay surface mission, both the tempo of the missions and the mass delivered to the surface of Mars would increase until a desired steady cadence of surface missions is reached.

TABLE I. Human Mars Campaign Mission Sequence

Mission Name	Year (Earth Departure)	Primary Mission Goal	Mission Details
#1a: Cislunar Uncrewed Flight Demonstration	2032	Test NTP crew vehicle in cislunar space	2-burn mission from MEO to LDHEO
#1b: Mars Uncrewed Flight Demonstration	2033	Test NTP crew vehicle on full Mars mission	Mars Opposition Class Mission – flyby or 50 day Mars SOI
#2: Mars Crewed Short Stay Mission – 30-Day Crew Surface Stay	2037	First humans on the surface of Mars	Mars Opposition Class Mission – 50 day Mars SOI, 30 day Mars surface
#3: Mars Crewed Long Stay Mission – 400+ Day Crew Surface Stay	2041	First human extended Mars surface stay	Mars Conjunction Class Mission – 400+ day Mars SOI and Mars surface
#4+: Mars Crewed Steady-State Long Stay Missions	2045+	Prepare for permanent human presence on the surface of Mars.	Series of Mars Conjunction Class Missions – 400+ day Mars SOI and Mars surface stays. Mission frequency is initially every other conjunction opportunity (52 months), transitions to every conjunction opportunity (26 months)

Figure 1 provides the required roundtrip mission in-space ΔV throughout the first 20 years of the envisioned campaign[5]. Lines for four different mission types are provided: opposition class Earth-Mars-Venus-Earth (EMVE), opposition class Earth-Venus-Mars-Earth (EVME), opposition class Earth-Mars-Earth (EME), and conjunction class EME.

The campaign missions enumerated in Table I are highlighted in Figure 1 as the optimal departure dates in order to minimize round-trip ΔV for each Earth-Mars mission opportunity selected. As can be seen in Figure 1, the initial short-stay opposition class missions require higher in-space ΔV. This higher required ΔV results in a larger, more complex MTV, but allows for shorter overall missions, benefiting crew health and mission reliability, and limiting the amount of surface infrastructure needed to support the crew while on the surface of Mars.

The following sections provide an overview of each of the missions within the envisioned campaign.

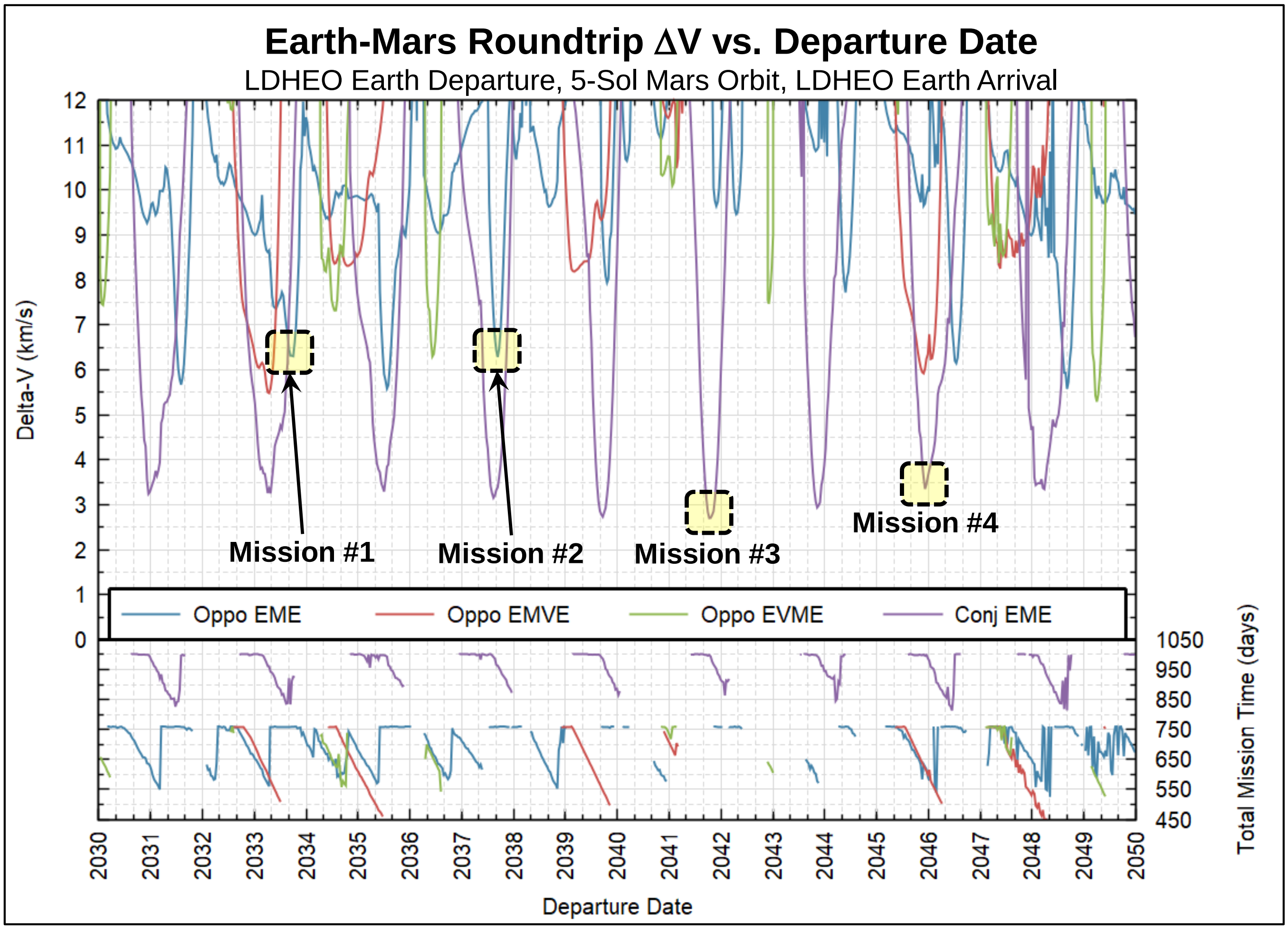


Fig. 1. Roundtrip Mission ΔV for Mars Conjunction and Opposition Missions from 2030-2050

II.A. Mission #1a: Cislunar Uncrewed Flight Demo

The first mission in the envisioned campaign starts with an uncrewed flight demonstration in cislunar space in 2032. The mission would start with the aggregation of the MTV in a medium Earth orbit (MEO) with a perigee of 2,000 km and an apogee of 6,200 km. This orbit is chosen to be above any high density orbital debris fields while also aligning launch vehicle payload mass and payload fairing volume capabilities.

After aggregation is completed in MEO, the MTV would perform a two-burn transfer to a lunar distance high Earth orbit (LDHEO). The MTV would loiter in LDHEO for a period of 3-6 months to allow for checkout of the vehicle performance and subsystem operation and to prepare for Mission #1b.

II.B. Mission #1b: Mars Uncrewed Flight Demo

The second phase of Mission #1 is a round-trip short-stay Mars opposition mission which begins in 2033 after the 3-6 month LDHEO checkout period is completed. The MTV travels to Mars, stays in Mars orbit for 50 days, and then returns back to Earth orbit. Upon return to Earth orbit, the MTV is inspected, refurbished as appropriate, and the propellant tanks refueled (LH2 for main propulsion and nitrogen tetroxide / hydrazine for reaction control system (RCS) propulsion) for Mission #2.

Missions #1a and #1b provide an uncrewed test of the MTV through the full in-space mission concept of operations (CONOPS) to be used in Mission #2.

II.C. Mission #2: Mars Crewed Short Stay Mission – 30-Day Crew Surface Stay

The second mission in the envisioned campaign is the first human mission to Mars. The MTV departs MEO with the first Mars crew in 2037 for a round-trip short-stay Mars opposition mission. Prior to the 2037 crew departure, a cargo mission in 2035 pre-deploys the required Mars surface assets to support the crew surface mission. These assets include a Mars Ascent Vehicle (MAV), Mars Surface Power System (SPS), and pressurized Mars surface transportation / habitat.

After the Mars surface assets are pre-deployed, the MTV with crew travels to Mars, descends to the Mars surface for a 30-day surface stay, returns back to the MTV in Mars orbit, and then returns back to Earth orbit. Upon return to Earth orbit, the crew rendezvous with a pre-deployed Orion crew vehicle and returns back to Earth.

The MTV is inspected, refurbished as appropriate, elements replaced as needed, and propellants refueled for Mission #3.

II.D. Mission #3: Mars Crewed Long Stay Mission – 400+ Day Crew Surface Stay

The third mission in the envisioned campaign is the first mission with an extended stay of humans on the surface of Mars. The CONOPS of Mission #3 are similar to those of Mission #2 with the exception that the mission is a long-stay Mars conjunction class mission with a crew stay on the surface of Mars of over 400 days. Due to the longer surface stay, additional Mars surface assets beyond another MAV, including a surface habitat, logistics module, and unpressurized surface transportation, are pre-deployed at the same Mars surface location as that used in Mission #2.

These additional Mars surface assets, along with those assets deployed in Mission #2, comprise the beginnings of a Mars surface outpost. These surface assets set the stage for the future expansion of the Mars surface infrastructure, and corresponding capabilities, in subsequent Mars missions.

II.E. Mission #4+: Mars Crewed Steady-State Long Stay Missions

The fourth mission in the envisioned campaign is the start of a series of long-stay Mars conjunction class missions preparing for the eventual permanent human presence on the surface of Mars. These missions, each with similar CONOPS to Mission #3, would initially land at the Mission #2 / #3 landing site to continue to build up the Mars surface outpost. Subsequent missions to additional surface locations to further explore various areas of interest are envisioned.

III. DETAILS OF FIRST CAMPAIGN MISSION: CISLUNAR AND MARS FLIGHT DEMONSTRATION MISSIONS

With an overview of the envisioned human Mars campaign provided in the previous section, details of the first mission in the envisioned campaign are provided here. The Mission #1 MTV is shown in Figure 2. It consists of a Core Stage, Inline Stage, Drop Tank Truss with seven Drop Tank Stages, and a Deep Space Habitat (DSH).

Figure 3 provides the CONOPS details for Mission #1. Aggregation of the MTV occurs in MEO and starts with the launch of the Drop Tank Truss element. The Drop Tank Truss element is the structural backbone for the Drop Tank Stages to attach to. It provides structural rigidity along with fluid, power, and data transfer between the other elements of the MTV. The Drop Tank Truss is launched in two segments on two commercial launch vehicle (CLV) launches. The Drop Tank Truss element segments leverage a service module to assist with rendezvous and docking with each other and with the next element launched: the Inline Stage.

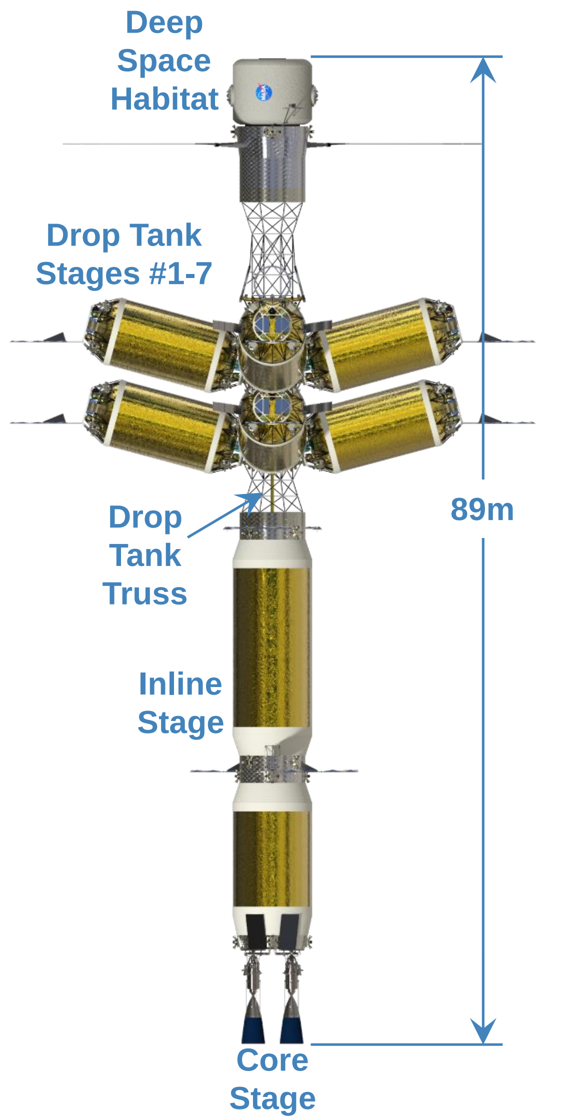


Fig. 2. Mars Transport Vehicle for Cislunar and Mars Flight Demonstration Missions

The Inline Stage is launched on a NASA Space Launch System (SLS) Block 2 launch vehicle to the same MEO orbit. It uses its on-board storable RCS propulsion to rendezvous and dock with the Drop Tank Truss. After mating is complete and mechanical, electrical, and fluid connections are checked out, the next element is launched: the Core Stage.

The Core Stage is also launched on a NASA SLS Block 2 launch vehicle. The Core Stage has two 25 klbf NTP engines. These engines have a nominal specific impulse (Isp) of 900 sec. For Mission #1, these engines operate at an Isp of 873 sec, leaving significant thermal margin in the reactor core allowing for future reuse. Similar to the Inline Stage, the Core Stage uses its on-board RCS system to rendezvous and dock with the rest of the MTV.

Once the center stack is mated and checked out, a series of seven Drop Tank Stages are launched on SpaceX Starship CLVs to the same MEO orbit. Each Drop Tank Stage uses its inboard RCS system to rendezvous and dock with the Drop Tank Truss.

With the Drop Tank Stages mated and their connections checked out, the MTV aggregation is complete. The MTV then performs the Mission #1a maneuvers which are two NTP main engine burns to raise the orbit from MEO to LDHEO. Each of these burns consumes enough liquid hydrogen (LH2) to empty the LH2 tanks of two Drop Tank Stages. Once those four Drop Tank Stage LH2 tanks are empty, the stages are jettisoned. This results in the MTV stack having three Drop Tank Stages remaining on the Drop Tank Truss after insertion in LDHEO.

Once in LDHEO, the MTV rendezvous and docks with the DSH which was launched, assembled, and checked out at the NASA Gateway in the lunar near rectilinear halo orbit (NRHO), and transferred down to LDHEO. The MTV, now with the DSH, loiters in LDHEO for a period of 3-6 months to allow for checkout of the vehicle performance and subsystem operation and to prepare for Mission #1b.

Mission #1b starts with the trans-Mars injection (TMI) maneuver to leave the Earth's SOI. After TMI, two additional Drop Tank Stages are jettisoned, leaving a single Drop Tank Stage remaining. The MTV then travels for 343 days before performing a deep space maneuver (DSM) burn to align the MTV's trajectory for intercept with Mars.

197 days after the DSM, the MTV performs a Mars orbit insertion (MOI) maneuver to insert into a 5-sol Mars orbit. The MTV stays in Mars orbit for 50 days. During that time, the MTV performs a series of maneuvers using its orbital maneuvering system (OMS) to reorient the orbit to prepare for Mars departure. While in Mars orbit, the MTV also has opportunities for remote sensing of Mars and its satellites, the potential deployment of robotic orbital or surface payloads, and for the examination of MTV performance and subsystem operation.

After 50 days in Mars orbit, the MTV performs the trans-Earth injection (TEI) maneuver and travels for 170 days back to Earth. As the MTV approaches Earth, it performs an Earth orbit insertion (EOI) maneuver to insert back in to LDHEO. The MTV then performs an OMS maneuver to lower its orbit to a loiter orbit in preparation for refueling and transfer back down to MEO for inspection, refurbishment as appropriate, and refueling for Mission #2 of the human Mars campaign.

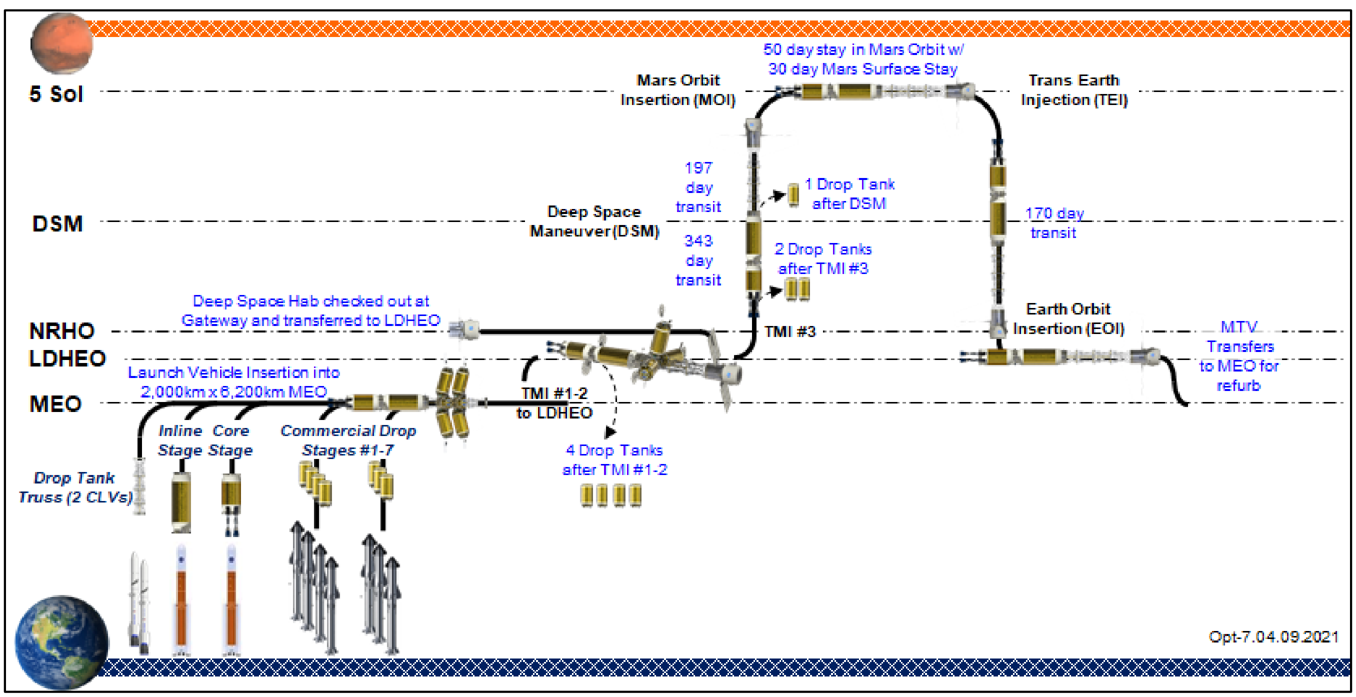


Fig. 3. Concept of Operations for Cislunar and Mars Flight Demonstration Missions

Figure 4 provides details of the MTV for Mission #1. The total round-trip heliocentric mission time for Mission #1 from Earth departure to arrival back at Earth is 760 days. The NTP engines are run at full power for 2.6 hours to complete the seven main engine burns. The MTV stack has a mass of 393 mT upon completion of aggregation in MEO, and a mass of 287 mT at Earth departure.

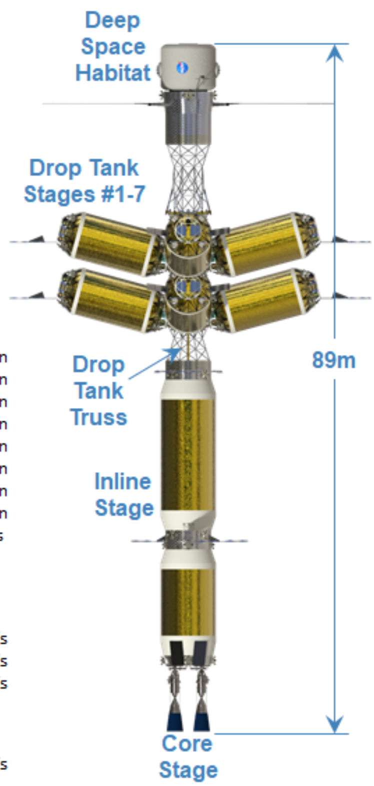
Mission #1 requires the 25 klbf NTP engines to operate at an Isp of 873 seconds to close the mission. These engines have a nominal specific impulse of 900 sec, leaving significant thermal margin in the reactor core allowing for future reuse.

Mission: 2033 Opposition

Mission Time				
Earth DSM	343	days		
DSM Mars	197	days		
Mars Stay	50	days		
Mars Earth	170	days		
Total	760	days		
Earth Sphere of Influence				
Departure Orbit	2,000 km x 6,200 km			
Arrival Orbit	LDHEO			
Mars Sphere of Influence				
Arrival / Departure Orbit	5 SOL			
NTP Primary Burns (8)*				
TMI #1 ΔV / Time	1,019	m/s	28.3	min
TMI #2 ΔV / Time	1,242	m/s	27.8	min
TMI #3 ΔV / Time	1,480	m/s	29.2	min
DSM #1 ΔV / Time	847	m/s	12.5	min
DSM #2 ΔV / Time	0	m/s	0.0	min
MOI ΔV / Time	1,940	m/s	22.6	min
TEI ΔV / Time	1,114	m/s	21.0	min
EOI ΔV / Time	1,021	m/s	15.9	min
Total Time			2.6	hrs
*ΔV does not include FPR				

Earth Sphere of Influence ΔV	RCS		OMS	
Rendezvous and Dock	25	m/s	0	m/s
Orbit Maintenance	90	m/s	0	m/s
LDHEO to Loiter Orbit	46	m/s	94	m/s

Mars Sphere of Influence ΔV	RCS		OMS	
Plane Changes	0	m/s	25	m/s



Vehicle Concept Characteristics

Deep Space Habitat Mass	45,000	kg, At Earth Departure
Drop Tank Truss Mass	6,834	kg
Inline (1 Total)		
Main Usable Propellant	45,748	kg of LH2
RCS Usable Propellant	5,264	kg of NTO/Hydrazine
Inert Mass	21,295	kg
Gross Mass	**72,307**	**kg**
Stage Length	21.6	m
Stage Outer Diameter	7.2	m
Drop Tanks (7 Total)		
Main Usable Propellant	23,284	kg of LH2
RCS Usable Propellant	271	kg of NTO/Hydrazine
Inert Mass	11,955	kg
Gross Mass	**35,510**	**kg**
Stage Length	15.3	m
Stage Outer Diameter	6.35	m
Core Stage		
Main Usable Propellant	30,637	kg of LH2
RCS Usable Propellant	5,200	kg of NTO/Hydrazine
Inert Mass	32,432	kg
Gross Mass	**68,269**	**kg**
Stage Length	21.7	m
Stage Outer Diameter	7.2	m
# of NTP Engines	2	
NTP Eng Thrust	25,000	lbf
NTP Eng Isp – Mission Req / Design Goal	873 / 900	sec
OMS Isp	450	sec
MTV Stack at TMI #1	393,009	kg (Does not include DSH)
MTV Stack at Earth Dep	287,211	kg

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Fig. 4. Mars Transport Vehicle Details for Cislunar and Mars Flight Demonstration Missions

IV. CONCLUSIONS AND FUTURE WORK

An overview of an envisioned human Mars campaign, consisting of uncrewed demonstration flights, crewed short-stay missions, and crewed long-stay missions preparing for the eventual permanent human presence on the surface of Mars, was provided. Details of the first campaign mission, an uncrewed cislunar and Mars flight demonstration mission was also provided. Future work will focus on the refinement of the campaign and detailed definition of the remaining crewed campaign missions.

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The authors would like to acknowledge the support of NASA Marshall Space Flight Center (MSFC) and Glenn Research Center (GRC), NASA STMD GCD and TDM Programs, and the Department of Energy engineers that continue working NTP for Mars crewed missions and other missions that can benefit from NTP.

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DECODING MISSION DESIGN PROBLEM FOR NTP SYSTEMS FOR OUTER PLANET ROBOTIC MISSIONS

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This paper discusses the current challenges of exploration of outer planets and proposes a Nuclear Thermal Propulsion (NTP) system for future deep space exploration missions. The mission design problem with respect to NTP system is presented where it is proposed that NTP powered missions need to integrate the requirements and constraints of mission objective, spacecraft design, NTP system design and launch vehicle limits into a self-consistent model. The paper presents a conceptual mission design to Jupiter based on the mission modeling techniques in the paper.

NOMENCLATURE

IMLEO – Initial Mass in Low Earth Orbit
JOI – Jupiter Orbit Insertion
JPL – Jet Propulsion Laboratory
LH_2 – Liquid Hydrogen
MIPS – Minimally-Intrusive Power generation System
NASA – National Aeronautics and Space Administration
NTP – Nuclear Thermal Propulsion
RTG – Radioisotope Thermoelectric Generators
TCM – Trajectory Correction Maneuver
TJI – Trans-Jupiter Injection
ΔV – change in velocity, km/s

I. INTRODUCTION

The robotic exploration of our solar system has expanded human knowledge on a diverse range of celestial bodies in our planetary system starting with its formation and evolution to search for life. The exploration of outer planets in particular have made fascinating discoveries of ocean worlds in gas giant and Ice giant systems. Some of the successful rendezvous missions over the decades to outer planets include Galileo (Jupiter), Cassini-Huygens (Saturn) and Juno (Jupiter) [1]. NASA is also developing future missions such as Europa Clipper (Europa) and Dragonfly (Titan) for the exploration of gas giant systems [2]. Table I mentions NASA's robotic missions to outer planets with details on trajectory and trip times.

Designing missions to outer planets is extremely challenging. Due to large distances of these planetary bodies, the ΔV required for such missions is very high. To date, chemical propulsion systems have been the go-to choice for deep space exploration missions. However, its low propellant efficiency has also been a challenge towards designing a dedicated mission to Ice giants without requiring a super heavy-lift launch vehicle [3]. Its limited ΔV capability and struggle against trip time and distance often necessitates the use of multiple gravity assist trajectories. The increased trip time due to gravity assist trajectories is directly proportional to the Phase- E cost of the mission lifecycle which can be a significant number for cost capped planetary missions.

TABLE I. Rendezvous robotic missions to outer planets.

Spacecraft	Destination	Trajectory	Trip time (yrs.)
Galileo	Jupiter	V-E-E-G-A	6.14
Juno	Jupiter	2+ dv-E-G-A	4.92
Cassini	Saturn	V-V-E-JG-A	6.71
Europa Clipper*[4]	Europa	M-E-G-A	5.5
Dragonfly* [5]	Titan	E-V-E-E-G-A	9.7

*Future mission currently in development.

I.A. Nuclear Thermal Propulsion

Nuclear Thermal Propulsion (NTP) can be the game changing technology for outer planet exploration missions. It's high specific impulse, over twice that of the best chemical rocket, combined with high thrust, enable missions that require high ΔV missions with short trip times.

The fundamental principle of NTP is remarkably simple when compared with the traditional chemical propulsion systems. In NTP system, the heat energy from fission reactor is transferred to the propellant which is then ejected through a de Laval nozzle. A schematic of a nuclear thermal propulsion engine is shown in Figure 1 below. The fuel is injected into the reactor core where it is heated to temperatures of about 2,700 K or above and then ejected via nozzle [6]. The temperature of the propellant heating is actually limited due to the structural integrity of the NTP engine. A small amount of propellant is also used to run the turbopumps which feed the propellant to the reactor core.

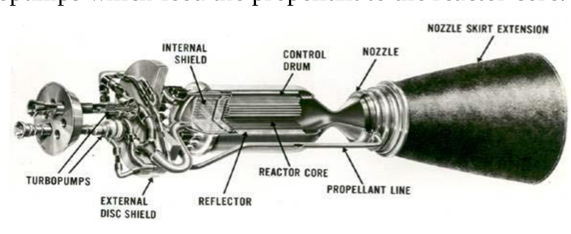

Fig. 1. Schematic of an NTP engine [7].

Another advantage of the NTP system is that it can be used for onboard electric power generation. A bi-modal nuclear engine for propulsive thrust and electric power for the control of the spacecraft would eliminate the requirement of Radioisotope Thermoelectric Generators (RTGs) [8]. Alternatives to bi-modal nuclear engine are also being explored to use NTP for electrical power generation such as Minimally-Intrusive Power generation system (MIPS) which can convert thermal energy from the reactor core at idle to an adequate amount of electricity to power the spacecraft, without compromising the reactor design and without impacting engine performance [9].

II. MISSION DESIGN FOR NTP SYSTEMS

Numerous studies have been conducted towards demonstrating the feasibility of NTP powered robotic missions for deep space exploration [10, 11]. Among many development challenges such as very high cost and long schedule of completing development, qualification, and production of these engines, the NTP systems for science missions have also not been aggressively considered in the past due to their large mass and inability to launch them on a single launch vehicle [12]. Requiring multiple launches not only increases the complexity of in-orbit assembly but also increases mission costs. The present analysis addresses the mass and launch vehicle challenges for NTP powered robotic missions. The graphical representation of the mission design problem for NTP systems is shown in Figure 2.

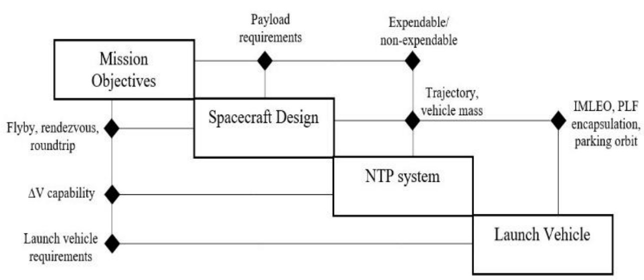

Fig. 2. Mission design problem for NTP powered missions.

The problem tackles multiple areas such as spacecraft and NTP system design based on mission objectives. The NTP system and spacecraft design parameters are also evaluated for launch vehicle constraints. This approach makes sure that the overall design is within the limits specified by each system. The problem statement begins with the science mission objectives which determines whether a mission will be a flyby, rendezvous or roundtrip. This information is used towards the spacecraft and NTP system design. Based on the payload requirements the spacecrafts subsystems are designed and evaluated to check if the overall design meets multiple parameters such as dry mass, power, and thermal etc. For the scope of this study, the spacecraft design will be only considered with respect to its total mass and dimensions. The NTP system configuration is then addressed which is based on the expendable or non-expendable nature of the mission along with ΔV requirements, engine thrust class (15klbf, 25klbf) and LH_2 propellant tank sizing. The spacecraft and NTP system configuration such as Initial Mass in Low Earth Orbit (IMLEO) and payload fairing encapsulation are evaluated based on launch vehicle limitations. Because this exercise is multidisciplinary in nature, the issues during the design of one system do not exist in isolation, but feed upon other systems as well. This problem is solved iteratively until multiple cross dependent parameters starting from spacecraft design to NTP systems and launch vehicle requirements are satisfied.

III. JUPITER RENDEZVOUS MISSION

We demonstrate an NTP powered Jupiter rendezvous mission based on the mission design problem presented in the previous section. The spacecraft design is based on expendable mission mode which consists of the spacecraft and NTP system. In the expendable mission mode, the NTP system is used only during the TJI and TCM and is then separated from the spacecraft. The spacecraft’s orbit insertion over Jupiter is conducted by the its onboard propulsion system. Therefore, by limiting the use of the NTP systems to TJI and TCM we simplify the cryogenic LH_2 storage problem significantly. NTP system parameters for this study have been referred from Refs. 13, 14 and 15 and are mentioned in Table II.

TABLE II. NTP system parameters.

Parameter	Values
Thrust (vac.)	15000 lbf
Specific impulse (vac.)	900 s
Nozzle area ratio	300:1
NTP engine T/W	2.65
LH_2 mass flow rate	7.56 kg/s

Estimating the spacecraft dry mass was the first step towards determining the configuration and total wet mass of spacecraft and NTP system. Referring to JPL studies on future mission concept studies to gas giant system (Ref. 16, 17), the dry mass of the spacecraft was estimated to be 2300kg. This required about 2050kg of chemical storable propellant for orbit insertion and trajectory correction maneuvers during the science mission operations. Based on the total wet mass estimates of the spacecraft which is 4350kg, the NTP system mass were calculated to be of about 21.36mT. This included NTP engine mass of 2560kg and LH_2 propellant mass if 12650kg. The spacecraft and NTP system mass breakdown are provided in the table III.

TABLE III. Spacecraft and NTP system mass breakdown.

Vehicle	Mass (kg)
NTP Engine	2,560
Tank dry mass	2,200
Spacecraft wet mass	4,350
LH_2 propellant (with 3% ullage volume)	12,650
Total 'wet' mass at launch	21,760

The total NTP injection stage length is 16.53m which includes 6.63m long NTP engine, 9.4m long LH_2 tank and interstage between NTP engine and LH_2 tank of about 0.5m long. The dimensions of the spacecraft is about 3.5m x 3.5m along with 0.5m long interstage between LH_2 tank and payload. This makes the total length of the spacecraft to be of about 20.53 m which meets the launch vehicle requirements with respect to IMLEO and payload fairing limits for ULA's Vulcan Heavy and Blue Origin's New Glenn [18, 19].

Fig. 3. Spacecraft and NTP system configuration

Trajectory design for the mission was divided into three sections. The first section involved Trans-Jupiter Injection (TJI) from its parking orbit of circular 1000km with 28.5 degrees inclination. During this phase NTP system provides the required ΔV of about 7km/sec for a direct Earth-Jupiter transfer orbit. The second phase of the trajectory involves heliocentric transfer during which the Trajectory Correction Maneuver (TCM) is performed. This correction is performed by the NTP system and after the TCM burn the NTP stage is separated from the spacecraft. The third and final phase of the trajectory involves Jupiter Orbit Insertion (JOI) which is performed by the spacecraft's onboard propulsion system.

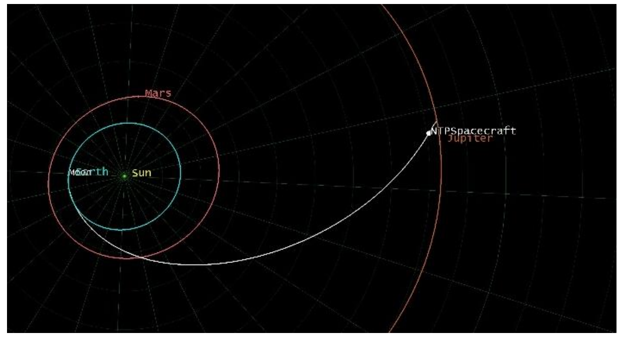

Fig. 4. E-J direct transfer trajectory

At the completion of JOI burn, the final captured orbit around the Jupiter has a period of 48.5 days with eccentricity of 0.96. The final spacecraft mass delivered to Jupiter is 3169 kg which includes about 869kg of fuel for orbit corrections during science phase of the mission. The converged trajectory shows that the trip time for the designed mission is 2.1 years using single high-class commercial launcher. This demonstrates that the NTP powered mission can reduce the trip time by a factor of two or more for similar class of spacecraft. Further, the direct transfer capability also increases the launch window for missions when compared with traditional chemical powered missions which have to be dependent on planetary alignments for trajectories requiring multiple planetary flybys.

IV. CONCLUSION

The paper addresses the mission design problem by demonstrating the capability of NTP system for Jupiter rendezvous mission. The integrated system design includes spacecraft and NTP system which should meet the requirements of launch vehicle and mission objectives. This approach also addresses the issue regarding system configuration where complete system needs be launched on a single launch vehicle to avoid any in-space assembly requirements. The full paper will include mission analysis for Ice giants and discussion on our future work towards implementing Model Based Engineering (MBE) and Model Based Systems Engineering (MBSE) for mission design.

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AVOIDING HEU IN SPACE REACTORS: AN EMERGING CONSENSUS

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Several years ago, it was assumed that prospective U.S. space power reactors would utilize fuel of highly enriched uranium (HEU) – despite such uranium being nuclear weapons-usable and therefore disfavored for civilian applications under longstanding U.S. and international nonproliferation policy. More recently, however, a U.S. government consensus has emerged opposing HEU use in space reactors, and instead advocating fuel of low-enriched uranium (LEU), which is not suitable for nuclear weapons under International Atomic Energy Agency guidelines. This paper first documents the emergence of the consensus against HEU space reactors and then recommends that the American Nuclear Society promote research and development of LEU-fueled space reactors, which may be the only politically plausible pathway for the United States to achieve nuclear power in space.

I. INTRODUCTION

Several years ago, it was assumed that prospective U.S. space power reactors would utilize fuel of highly enriched uranium (HEU) – despite such uranium being nuclear weapons-usable and therefore disfavored for civilian applications under longstanding U.S. and international nonproliferation policy. In January 2018, the U.S. government even tested a prototype of such a reactor using HEU fuel at a national laboratory.[1] More recently, however, a U.S. government consensus has emerged opposing HEU use in space reactors, and instead advocating fuel of low-enriched uranium (LEU), which is not suitable for nuclear weapons under International Atomic Energy Agency guidelines. This U.S. government consensus is remarkably broad, encompassing both the Executive and Legislative branches of government, both the House and Senate, and both the Republican and Democratic parties. The alternative of LEU-fueled space power reactors appears feasible, based on research conducted by the National Aeronautics and Space Administration (NASA) and the Department of Energy (DOE), and awaiting research and development (R&D) of some technical aspects. This paper first documents the emergence of the consensus against HEU space reactors and then recommends that the American Nuclear Society (ANS) promote research and development of LEU-fueled space reactors, which may be the only politically plausible pathway for the United States to achieve nuclear power in space.

II. BACKGROUND

The U.S. government consensus against HEU-fueled space power reactors is motivated by the desire of U.S. officials to sustain, rather than undermine, more than four decades of progress in the U.S.-led international nonproliferation policy of minimizing HEU outside of nuclear weapons.[2] The proliferation risks of HEU are well known.[3] Fifty kilograms of HEU is sufficient for a simple, gun-type nuclear weapon like the one dropped on Hiroshima, which would be straightforward even for terrorists to construct. Much less HEU would be sufficient for an implosion bomb having multi-kiloton yield, which states could accomplish and perhaps some terrorists.[4] Accordingly, the amount of HEU in even a small space power reactor, like the one tested in 2018, would be sufficient for one or more nuclear weapons. If the United States were to proceed with HEU space reactors, it could increase not only nuclear proliferation risks – due to other countries following the precedent – but also nuclear terrorism risks arising from the requisite terrestrial fuel cycles including in the United States.

The U.S. government initiated its HEU minimization policy in the 1970s.[5] Since then, the scope of the policy has grown to encompass many types of nuclear facilities: foreign research reactors, U.S. university and commercial research reactors licensed by the Nuclear Regulatory Commission (NRC), U.S. government research reactors operated by the Department of Energy (DOE), and foreign and domestic processing plants that produce medical isotopes. In addition, during the past six years, Congress has funded research and development of Navy LEU fuel in hopes of replacing HEU fuel for propulsion of submarines and aircraft carriers.[6] Most recently, in 2019, the U.S. Army mandated that proposed designs of its new Mobile Nuclear Power Plant (MNPP) must utilize LEU fuel.[7]

A guiding principle of the U.S. HEU minimization policy has been to avoid exceptions, on grounds that if any country were granted an exception for a facility, then other countries would demand exceptions too, potentially unravelling the policy. That is why, even though the U.S. government's original goal was to reduce foreign use of HEU, the implementation started by converting two domestic research reactors.[5] When the U.S. government momentarily violated this principle in the late 1980s, by proposing to build a new HEU-fueled research reactor, European countries reacted by threatening to stop converting their reactors from HEU to LEU – until the U.S. plan was abandoned in 1995.[5]

U.S. policy acknowledges that converting some facilities from HEU to LEU may require R&D, justifying a delay in but not an exemption from conversion. Among U.S. research reactors that had used HEU fuel, 17 already have been converted to LEU fuel, another is expected to close, and R&D is ongoing to achieve conversion of the final six. In countries supplied by the United States with enriched uranium, dozens of research reactors have been converted from HEU to LEU, and the final two are scheduled to be converted within a decade. In these countries, after the 1970s, only one additional HEU-fueled research reactor was constructed, and the U.S. government steadfastly refused to provide HEU due to the no-exception policy, which explains why the operator is now converting to LEU fuel.[8] For medical isotope production, the U.S. government likewise rejected foreign appeals for exceptions, so that only one major producer in the world still uses HEU targets, and it will fully convert to LEU targets by next year.[9] The Army offers no exception from its LEU requirement for MNPPs. The Navy must await further R&D before it can decide whether to convert to LEU propulsion reactors, but Congress has rejected appeals to exempt submarines from the LEU R&D requirement that also applies to aircraft carriers.[10] Thus, the U.S. government appears to have avoided exceptions from the HEU minimization policy in any domain – except obviously the U.S. nuclear weapons stockpile.

III. CONSENSUS ON LEU SPACE REACTORS

The spark for emergence of the U.S. government consensus against HEU space power reactors was the January 2018 test of the HEU-fueled prototype. U.S. nonproliferation officials quickly realized that proceeding with such a reactor could sabotage decades of hard-won international progress on HEU minimization. Even many advocates of space nuclear power came to oppose the HEU-fueled version because they realized the nonproliferation controversy might derail space nuclear power entirely.

Two legislative initiatives demonstrate that the Congressional opposition to HEU space reactors is bipartisan and bicameral. In June 2019, Rep. Bill Foster, a Democrat and the only physicist in the U.S. Congress, successfully added an amendment enacted as part of an appropriations bill. As he explained on the floor of the House of Representatives, the "Amendment directs NASA to work toward the development of a LEU space power reactor….The problem is that if all the spacefaring nations of the world start using large amounts of weapons-grade material in their space reactors, then it will be difficult to ensure that this Material would not be diverted to weapons programs in space and on Earth. If the U.S. develops a LEU space power reactor design, it is likely that this type of reactor design will be adopted as a de facto standard by other spacefaring nations, making Earth and space a safer place."[11]

In September 2020, the Republican-controlled Senate Commerce Committee reported to the full Senate a NASA Authorization bill including Section 506 on "Prioritization of Low-Enriched Uranium Technology." It provides as follows: "(a) *Sense of Congress*.--It is the sense of Congress that…HEU presents security and nuclear nonproliferation concerns…[T]he use of LEU in place of HEU has security, nonproliferation, and economic benefits, including for the national space program. (b) *Prioritization of Low-enriched Uranium Technology*.--The Administrator shall establish and prioritize, within the Space Technology Mission Directorate, a program for the research, testing, and development of a space surface power reactor design that uses low-enriched uranium fuel. (c) *Report on Nuclear Technology Prioritization*.--Not later than 120 days after the date of the enactment of this Act, the Administrator shall submit to the appropriate committees of Congress a report that-- (1) details the actions taken to implement subsection (b); and (2) identifies a plan and timeline under which such subsection will be implemented."[12]

The Executive Branch of the U.S. government also now favors LEU space reactors. In February 2020, NASA and DOE completed the final draft of a "trade study," comparing HEU versus LEU as fuel for space reactors. The report is not publicly available, but in June 2020 its findings were publicly characterized by the Deputy Chief Engineer of NASA's Space Technology Mission Directorate, at the ANS Annual Meeting. The study, he said, "Concluded that moderated HALEU-fueled reactors are competitive in mass with HEU-based designs."[13] (HALEU is "high assay" LEU, enriched above five percent but below 20 percent.) This undermined the main rationale of HEU proponents – that LEU would significantly increase the mass of a space power reactor and thus preclude or sharply increase the expense of launching it.[14]

Also in February 2020, NASA's Budget Estimate stated that its nuclear fission power project "will seek to identify design trades and collaborative opportunities with industry, and to the extent feasible take advantage of the interagency investment in a common fuel source for both nuclear power and propulsion systems."[15] Since NASA already had embraced LEU for propulsion reactors, this statement provided an economic-efficiency rationale for using LEU fuel also in space power reactors. In November 2020, the press reported that the Nuclear Technology Portfolio Lead in NASA's Space Technology Mission Directorate had concluded that, "A low enriched form of nuclear fuel will power the nuclear core" of future space power reactors.[16]

The White House first demonstrated concern over HEU-fueled space reactors in its August 2019 "National Security Presidential Memorandum on the Launch of Spacecraft Containing Space Nuclear Systems." The policy declared that, "Due to potential national security considerations associated with nuclear nonproliferation, Tier III [the most restrictive] shall also apply to launches of spacecraft containing nuclear fission systems and other devices with a potential for criticality when such systems utilize any nuclear fuel other than low-enriched uranium…The President's authorization shall be required for Federal Government launches in Tier III."[17]

The White House amplified its concern over HEU space reactors in its December 2020 "Space Policy Directive–6, the National Strategy for Space Nuclear Power and Propulsion." This directive effectively banned HEU-fueled reactors except in the absence of any alternative way of accomplishing the mission. It declares that, "The use of HEU in space nuclear power and propulsion systems should be limited to applications for which the mission would not be viable with other nuclear fuels or non-nuclear power sources."[18]

NASA is already implementing this new policy, according to the agency's March 2021 presentation to an NRC regulatory conference, stating that it "prefer[s] HALEU fission reactor solutions based on [the] March 2020 DOE study that showed masses comparable to HEU systems."[19] NASA also revealed that the "current government reference design calls for a segmented moderated HALEU reactor" for the fission surface power project. The government also continues to mandate LEU for nuclear thermal propulsion, as codified in a December 2020 statement of work: "The reactor shall use high assay low-enriched uranium (HALEU) fuel, or uranium fuel with lower levels of enrichment."[20]

IV. ACHIEVING SPACE NUCLEAR POWER

The NASA/DOE trade study of February 2020 documented that LEU reactors, by employing a moderator to thermalize neutrons, can avoid a large mass penalty in comparison to the baseline fast HEU reactor that excludes a moderator. The study also noted that LEU reactors "have greater complexity" and thus require additional R&D.[13] Such research, which DOE already is supporting, reportedly focuses on two potential moderators: zirconium hydride (ZrH) and yttrium hydride (YH). ZrH is the more mature technology, but YH could avoid hydrogen loss at higher temperature.

In light of the U.S. government consensus against HEU space reactors, pursuing R&D on LEU space reactors may be the only politically plausible pathway for NASA to achieve nuclear power in space. In addition, providing results of such research to other countries could reduce their perceived need to produce or handle HEU, thereby decreasing risks of diversion or theft for nuclear weapons and mitigating international security concerns and tensions.

Accordingly, the ANS and its Aerospace Nuclear Science and Technology Division could best facilitate space nuclear power by advocating R&D of LEU space power reactors. By contrast, if the ANS were to promote HEU-fueled space reactors, despite the U.S. government consensus against them, it might inadvertently undermine prospects for any space nuclear power. Simply put, in the context of space exploration, the pro-nuclear position is anti-HEU.

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MODELING OF NTP ENGINE START-UP, SHUTDOWN, AND COOLDOWN AND THEIR IMPACTS ON ΔV

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For the past 5 years, nuclear thermal propulsion (NTP) has been evaluated as a potential main propulsion option for crewed missions to Mars. The combination of high thrust and high efficiency (high Isp) make NTP ideally suited for these missions, where transfer times can be in excess of 300 days and vehicle masses can reach several hundred metric tons.

With increases in fidelity to engine and trajectory modeling, a desire to couple the NTP engine start-up/shutdown and cooldown transients with current finite-burn trajectory modeling arose. With engine start-up and shutdown duration on the order of 30 seconds each, and engine cooldown taking place over several hours post-shutdown, enough propellant is expended to provide useful impulse and impact several aspects of the end-to-end trajectory and vehicle sizing.

This paper details current NTP engine start-up/shutdown and cooldown modeling, and the impacts of this impulse on vehicle sizing and end-to-end trajectory modeling.

NOMENCLATURE

AR = Aerojet Rocketdyne
CFM = Cryogenic Fluid Management
DSH = Deep Space Habitat
GCD = Game Changing Development
HALEU = High Assay Low Enriched Uranium, <20% enriched with U235 isotope in Uranium fuel
Isp = Specific Impulse
LDHEO = Lunar Distant High Earth Orbit
LEU = Low Enriched Uranium
MEO = Medium Earth Orbit
NASA = National Aeronautics and Space Administration
NEP = Nuclear Electric Propulsion
NERVA = Nuclear Engine for Rocket Vehicle Applications
NRHO = Near Rectilinear Halo Orbit
NTP = Nuclear Thermal Propulsion
SLS = Space Launch System
SNP = Space Nuclear Propulsion
STMD = Space Technology Mission Directorate
TDM = Technology Demonstration Mission
USNC = Ultra Safe Nuclear Corporation
ΔV = Delta-V, Velocity change

I. INTRODUCTION

Since 2016, Aerojet Rocketdyne (AR) has been working with NASA, the Department of Energy, and members of industry as part of the NASA Space Technology Mission Directorate (STMD). Initially under the Nuclear Thermal Propulsion Project within Game Changing Development (GCD), the goal was to determine the feasibility and affordability of a low enriched uranium (LEU)-based NTP engine.

Having shown feasibility and affordability, the project focus has evolved as part of the STMD Space Nuclear Propulsion (SNP) Project within the Technology Demonstration Missions (TDM) Program. The current focus of SNP is in three major areas:

1. NTP fuel and reactor technology maturation
2. Identification of nuclear electric propulsion (NEP) subsystem capability gaps and needs
3. Advancement of critical cryogenic fluid management (CFM) technologies needed for SNP

Under the SNP project, AR has increased the fidelity of engine and trajectory modeling[1-3], leading to a more accurate representation of vehicle performance. The details of the engine and trajectory modeling updates, along with vehicle and mission impacts are discussed in the following sections.

II. NTP MARS ARCHITECTURE STATUS

The current NTP Mars architecture assumes an opposition class mission, which has the benefit of a shorter mission duration of less than 2 years compared to a conjunction mission. Figure 1 below shows a top-down view of the opposition mission trajectory for a 2039 opportunity. Compared to conjunction class missions, opposition class missions required 2-3 times higher ΔV, which result in larger and more complex transfer vehicles. The combination of high Isp and high thrust from NTP enables lighter and less complex vehicles when compared to conventional chemical propulsion.

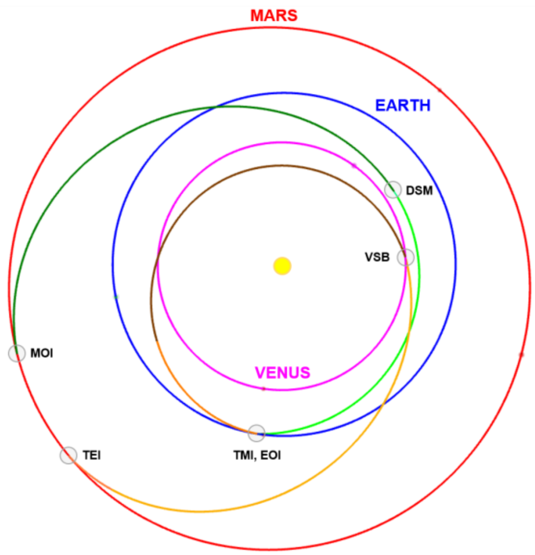

Figure 1 - 2039 Mars Opposition Mission Profile

Several vehicle configurations for Mars opposition missions were examined, trading aggregation orbit, launch vehicles, and number/type of elements[4]. The NTP vehicle shown in Figure 2 below is the culmination of that study, along with further, higher fidelity analyses. The primary propulsive elements include an NTP core stage, an inline stage, and two strap-on core stages, all launched with SLS Block 2. Twelve Starship launched drop tanks provide additional propellant and allow for element staging, which increases vehicle performance. The vehicle elements are launched to, and assembled in, a 2,000km x 6,200km Medium Earth Orbit (MEO), before transferring up to LDHEO to await crew arrival and Earth departure.

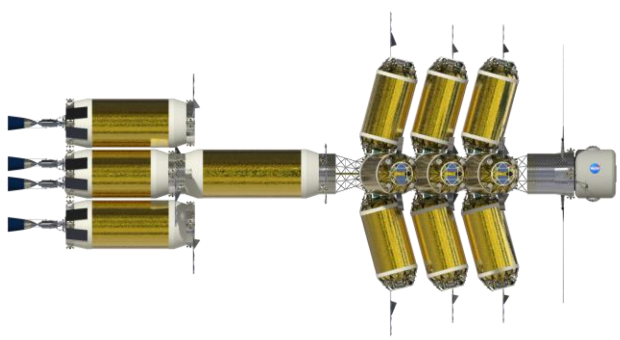

Figure 2 - Current NTP Opposition Vehicle

The gross mass of the NTP vehicle after assembly and before transfer to LDHEO is 705 mT. This gross mass consists of the entire vehicle stack shown in Figure 2 above, minus the Deep Space Habitat (DSH). The DSH aggregates in NRHO around the moon before transferring to LDHEO and rendezvousing with the NTP vehicle. The individual element masses are broken down in Table 1 below. Each engine provides 25,000 lbf of thrust and has a nominal Isp of 900 sec. For the 2039 opposition mission, the engines operate at an Isp of 880 sec, providing thermal margin in the reactor core allowing for future reuse.

Table 1 - NTP MTV Element Masses

Element	Mass	Quantity
NTP Core	61,922 kg	1
Strap-On Core	69,478 kg	2
Inline	73,349 kg	1
Drop Tank	35,724 kg	12

III. NTP ENGINE START-UP/SHUTDOWN & COOLDOWN MODELING

Contrary to traditional liquid rocket engines, which have start-up/shutdown times on the order of a few seconds, the current NTP engine requires 35 seconds for start-up and 30 seconds for shutdown. Post-shutdown, the continuous or pulsed flow cooldown period scales based on engine burn time, requiring 1 to 30 hours to reach the temperature handoff for radiative cooling. During each of these regimes, propellant flows through the engine at low rates, and will produce noticeable thrust and useful ΔV.

A. Start-Up/Shutdown

AR has been modeling both steady state and transient operation of the NTP system for a Mars mission full-scale 25,000-lbf thrust size. Transient NTP modeling has confirmed the closed-loop control approach that will permit a 35-second start ramp and a 30-second shutdown ramp. The NTP transient start holds the "start trajectory" to a nominal temperature rise (constant slope) to achieve the target reactor exit temperature near 4,860-degrees Rankine (2,700 Kelvin). The NTP engine operates off-design during the start and a variation in thrust and delivered specific impulse (Isp) is observed during the time it takes (e.g., 35 seconds) for the NTP engine to achieve steady-state operation. The 35-second start-up period is shown in Table 2 and Figure 3 below.

Table 2 – NTP Engine Start Ramp

Time (sec)	Thrust (lbf)	Power (MW)	Isp (sec)
0	25	0.5408	182.6
5	1,250	27.04	282.1
25	10,000	216.3	680.4
30	17,500	378.6	780.0
35	25,000	540.8	879.6

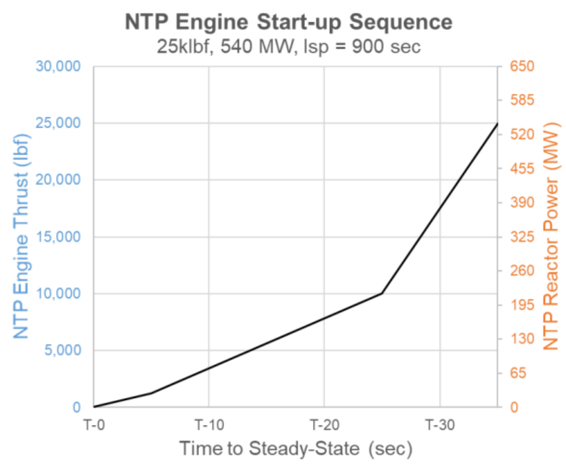

Figure 3 – NTP Engine Start-Up Sequence

Similarly, during the NTP shutdown ramp a variation in thrust and delivered specific impulse (Isp) occurs. The thrust and Isp data is provided to the NTP vehicle mission model and is used to calculate the quantity of hydrogen consumed during the start and shutdown times during each mission "burn." The thrust at shutdown variations are shown in Table 3 and Figure 4 below.

Table 3 – NTP Engine Shutdown Ramp

Time (sec)	Thrust (lbf)	Power (MW)	Isp (sec)
0	25,000	540.8	879.6
4	17,500	378.6	799.9
10	10,000	216.3	680.4
18	3,750	81.12	521.1
26	1,250	27.04	361.8
28	750	16.23	322.0
30	375	8.112	282.1

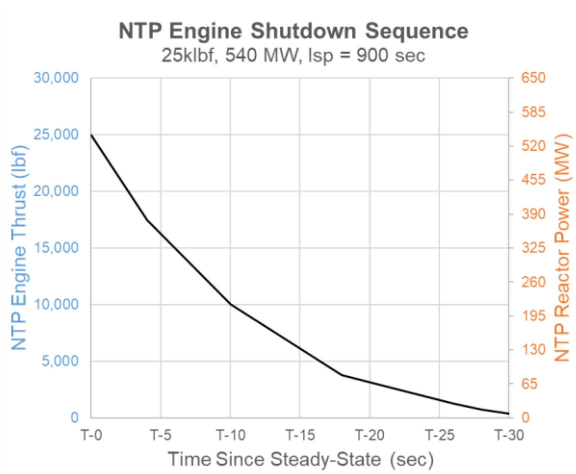

Figure 4 – NTP Engine Shutdown Sequence

B. Continuous & Pulsed Cooldown

AR, working with Ultra Safe Nuclear Corporation-Tech (USNC-Tech), has developed a transient cool-down model that has permitted analysis for both continuous and pulse cool-down flow post-shutdown. The hydrogen that is stored in the vehicle tankage and used for producing thrust is also supplied via a variable flow pump system for cooling down the reactor after each mission "burn." NTP reactor cool-down requirements are well understood and demonstrated during the Rover/NERVA (Nuclear Engine for Rocket Vehicle Applications) program[5-6].

After shutdown, the reactor remains hot and must be further cooled. This NTP engine cool-down can use continuous or pulse cooling flow through the reactor until the reactor reaches a minimum exit temperature that allows for radiating heat into space. This value is determined by the material in the core with the lowest margin to melt or lose significant structural strength. In a heavily moderated high-assay low enriched uranium (HALEU) core, it is usually the moderator material (e.g., Lithium Hydride)

Continuous cool-down can be an effective approach for post-shutdown cooling during testing. When considering a flight system it is more effective to pump the hydrogen to the reactor over the cool-down period, thus when using a pump, pulse cooling appears to be more feasible for flight systems. AR has analyzed both continuous and pulse cooldown with USNC-Tech. That analysis has shown that when optimizing the time and number of pulses for pulse cool-down, the hydrogen needed post-shutdown is very similar to the hydrogen consumed during a continuous cooldown. This paper shows some of the results from the continuous cool-down approach.

As the reactor temperature decreases, the cool-down flow rate decreases as well. The amount of cooling required varies with the length of each NTP mission burn. The cool-down duration and amount of propellant needed increases as the engine burn duration increases. Figure 5 below shows how the amount of hydrogen consumed will vary with each engine burn time.

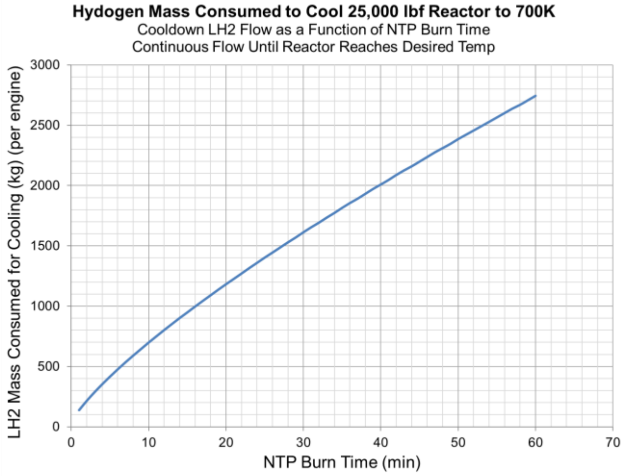

Figure 5 - NTP Cooldown Flow per Engine Burn Time

IV. END-TO-END TRAJECTORY MODELING

Previous mission trajectory analysis used a constant-thrust, constant-specific-impulse model that neglects the effects of start-up/shutdown and cool-down. This captured the effects of the finite burn (e.g., gravity and steering losses), but neglected the persistent, albeit low-level thrust produced by the cooling flow. In addition, this thrust occurs at lower specific impulse due to the decreasing exit temperature of the reactor.

A series of analyses using a piecewise linear representation of a full operating profile was used to determine the impact to each main engine firing. The initial and target conditions of each burn were obtained from the constant thrust/specific impulse end-to-end mission model, and each burn was modeled separately to determine the individual impacts. Then, a reserve was applied to each burn to account for increases in propellant usage. Figure 6 below shows a comparison between the two trajectory analysis techniques.

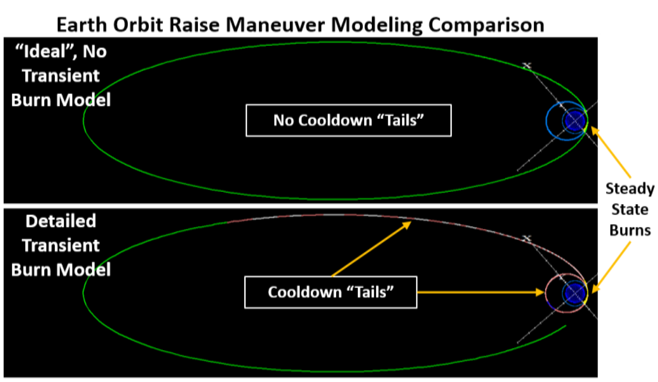

Figure 6 – Comparison of Modeling Methods

V. IMPACTS TO OVERALL ARCHITECTURE

Coupling the NTP engine start-up/shutdown and cooldown transient modeling with current end-to-end trajectory modeling has implications to vehicle sizing and performance. These implications include changes to how start-up/shutdown and cooldown propellant is allocated within element masses and to how the maneuver ΔV for each burn is calculated.

Alongside vehicle sizing, the overall mission and trajectory design is affected in key areas. The longer engine cooldown periods can have significant effects on highly elliptic orbits where small impulse maneuvers can have an amplified result. This section provides details on these effects.

Prior analysis assumed the propellant used for NTP engine start-up/shutdown and cooldown was reserved as a percentage of usable propellant, and remained on the vehicle as inert mass rather than being expelled with each burn. This method resulted in a higher vehicle burnout mass, while also not taking advantage of impulse that would be provided in real-world application.

Vehicle modeling was updated to take into account the performance benefits of the start-up, shutdown, and cooldown propellant. The ΔV provided from start-up, shutdown, and cooldown was calculated from the average Isp and required propellant mass during each mode. These ΔVs were then debited from the primary maneuver to calculate the steady state ΔV. Since the cooldown propellant requirement is a function of engine burn time, this algorithm iterates through until convergence. The average Isp and propellant mass per engine for start-up/shutdown and cooldown are shown in Table 4 below. The propellant masses for start-up and shutdown are constant between each burn, and only vary at the vehicle level with the number of engines used for each maneuver.

Table 4 – Start-Up and Shutdown Average Isp and Propellant Mass

	Average Isp	Propellant Mass
Start-Up	687.8 sec	191.7 kg
Shutdown	701.9 sec	158.7 kg
Cooldown	427.3 sec	Varies with Burn Time

For the opposition vehicle analysis, the primary performance metrics are the minimum required Isp and the required number of drop tanks to close the mission. Analysis was performed to determine the differences in these metrics between the two modeling methods. Table 5 below shows the performance using the two methods.

Table 5 – Comparison of Vehicle Modeling Methodologies

	Propellant Held as Inert Mass	Propellant Expelled as Usable Impulse	Percent Reduction
Required Isp	926.6 sec	879.6 sec	5.34%
Required # of Drop Tanks	16	12	33.3%

The required propellant for start-up/shutdown and cooldown is 8% of the total tanked usable propellant on the vehicle. The impulse gained through the start-up/shutdown and cooldown propellant accounts for almost 5% of the total mission ΔV. Using the propellant for start-up/shutdown and cooldown as usable impulse is not only more accurate from a vehicle sizing perspective, but also results in improvements to the primary performance metrics by a significant amount.

VI. CONCLUSIONS

Increases in fidelity to NTP engine start-up/shutdown and cooldown modeling in addition to end-to-end trajectory modeling allows for increases in vehicle performance through useable impulse. This paper details the engine modeling techniques and how these methods affect vehicle sizing and mission/trajectory design.

ACKNOWLEDGEMENTS

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THE CASE FOR A 50+ YEAR RADIOISOTOPE POWER SYSTEM

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The Johns Hopkins University Applied Physics Laboratory (JHU/APL) is leading the NASA funded Interstellar Probe study to explore the "Very Local" interstellar medium. To perform this exploration the mission will be required to last at least 50 years in regions of space where solar power is no longer practical. Additionally, several new studies for the National Academies' Planetary Science and Astrobiology Decadal Survey are planning missions lasting 20-35 years. The Decadal Survey is used to build consensus on priority of national science goals. These proposed missions are inconsistent with the NASA's current flight lifetime requirement of 14 years. Paramount to these proposed long-duration missions are questions about the longevity of such a mission. Evidence exists that space-borne Radioisotope Power Systems can indeed last a long time. LES-9, Voyager I, and Voyager II are over 40 years old, LES-8, Pioneer 10, and Pioneer 11 lasted 28, 30, and 22 years, respectively, and New Horizons is still active 15 years after launch. None of these missions was terminated because of an RTG failure.

This paper examines the historical record by way of statistical analyses, illustrates the theoretical performance through a long-duration mission, and discusses how reliability engineering and testing methods can be brought to bear to increase confidence in delivering sufficient power at end of mission.

I. INTRODUCTION

The Johns Hopkins University Applied Physics Laboratory (JHU/APL) is leading the NASA funded Interstellar Probe study to explore the "Very Local" interstellar medium. To perform this exploration the mission will be required to last at least 50 years in regions of space where solar power is no longer practical. Additionally, several new studies for the National Academies' Planetary Science and Astrobiology Decadal Survey are planning missions lasting 20-35 years. The Decadal Survey is used to build consensus on priority of national science goals. These proposed missions are inconsistent with the NASA's current Radioisotope Power Systems (RPS) life requirement of 14 years (flight). Paramount to these proposed long-duration missions are questions about the longevity of such a mission. Evidence exists that space-borne Radioisotope Power Systems can indeed last a long time. LES-9, Voyager I, and Voyager II are over 40 years old, LES-8, Pioneer 10, and Pioneer 11 lasted 28, 30, and 22 years, respectively, and New Horizons is still active 15 years after launch.

This paper explores the need for RPS designs that are intended to last much longer than the current requirement of 14 years (17 years after fueling) and explores the historical record for actual vs design lifetimes to show the feasibility of building long lasting RPS. We also exercise a current RTG performance model of the General-Purpose Heat Source RTG using the JPL Lifetime Performance Prediction (LPP) tool to make top-level inferences about power output at end-of-mission, and discusses how reliability engineering and testing methods can be brought to bear to increase confidence in delivering sufficient power at end-of-mission.

I.A. Background and Motivation

As part of the Interstellar Probe (ISP) study, we are exploring longevity issues across the spectrum to include aging of hardware (focusing on current electronics technology), materials, moving components (thruster valves), ground systems, workforce issues, and knowledge transfer between generations of scientists, engineers and managers. Chief among the equipment concerns is the RPS. For distances out into the interstellar medium where travel times of at least 50 years are needed, RPS are the only viable option currently available. This is inconsistent with the current published lifetime requirement of 14 years (17 years after fueling).

The ISP study team is not the only one thinking about long duration missions. Several whitepapers supporting the National Academies 2023-2032 Planetary Decadal Survey [1] have emphasized science in the outer solar system. The Outer Planets Assessment Group (OPAG) for example states "For the decade 2023-2032, OPAG endorses a new start for two directed missions: first, a mission to Neptune or Uranus (the ice giants) with atmospheric probe(s), and second, a life detection Ocean World mission".[1] Several PIs mention long mission times as part of their concept descriptions. Rymer

[1] All Planetary Decadal whitepapers referenced were accessed at: Planetary Science and Astrobiology Decadal Survey 2023-2032, The National Academies of Science, Engineering, Medicine. https://www.nationalacademies.org/our-work/planetary-science-and-astrobiology-decadal-survey-2023-2032.

describes a mission to Neptune and Triton needing a "12-16 year cruise phase to Neptune". Although not stated, the science collection portion of the mission could be 2-5 years.[2] Robbins discusses missions to revisit and orbit Pluto and as such notes that there are, "Significant time requirements to reach the system with a low enough capture velocity, and the power and related age issues that result." This would be a mission longer than New Horizons.[3] Cohen, notes in his discussion of a proposed Uranus mission the constraint that RPS lifetimes have, "With current technology (i.e., 14-year MMRTG flight design life), a typical baseline would be a <12-year cruise (potentially with a Centaur flyby) and a 2-year mission at Uranus with a system tour that enables surface mapping of the large satellites as well as spatial coverage of the planet & rings/small moons; this baseline could be significantly lengthened if the lifetime of future RPS were improved."[4] Neveu advocates for sample returns from Enceladus with various options with a Sun orbiter taking up to 34 years, a Saturn Orbiter taking 13-15 years, or landers taking more than 26 years.[5]

With growing demand to pursue longer lasting missions, we examined all U.S. missions that flew RPSs to see how long did they last? We also investigated the power output at end-of-life through current performance models and historical power curves extending them to 50 years.

II. HISTORICAL LIFE TIME DATA ANALYSIS

This section examines the historical record of RPS and provides a statistical analysis for probability of success and lifetime.

Table 2 is a listing of all RPS missions used in the dataset. The list is based on a NASA compilation[6] with the mission information provided from the SpaceTrak database[7]. For each table entry, mission name, type of RPS and quantity are provided. The dates listed are the mission launch date and the date of last contact showing the RPS as operational. For still active missions, no end date is given. Mission design and RPS design life are provided separately. The RPS failure column indicates an RPS failing prior to the RPS design life (see Section II.A for details on the two failures). Note, no nuclear heater units are included.

Of the 29 missions listed, 3 failed to become operational (Transit 5BN-3 and Nimbus B-1 failed to reach orbit while Apollo 13 ALSEP failed to reach the moon) and are removed from the analysis. This represents a total of 40 RPS units.

Some entries appear as mission data but since we are interested in how long the RPS equipment lasted, some of the end dates are dates of last contact and not mission end. For example, routine contact with Pioneer 10 ended in March 1997, but last documented contact was January 2007. Also the Triad mission ended in 1972, but signals from the spacecraft indicate that the RTG was still operational as of 2006.

Owing to the small sample sizes a traditional statistical analysis is augmented with a Bayesian analysis to show the uncertainty distributions in the results.

II.A. Probability of Success

RPS have an outstanding record for producing power for space missions. Of the 40 missions listed, all fulfilled their mission duration as designed. All but 2 units were working at the time when the mission ended. The reasons for missions ending are attributed to other systems or the spacecraft being retired after successful extended missions. Two units had significant anomalies and are considered to have failed for this paper, however, both anomalies presented after the initial mission objectives were met, but before the RPS design life. Although some ambiguity as to the nature of the failures exist, we consider them as failed to provide a worst-case view of the data. The two anomalies are Transit 4B and MSL/Curiosity.

After meeting all its mission objectives, the SNAP-3B system power intermittently dropped to zero for several days and then failed completely on June 1962, 7 months after launch. It is believed that either the RPS DC/DC converter failed, or that the thermoelectric converters in the power unit failed.[8]

On mission sol 456, MSL engineering operations staff observed an unexpected shift in bus balance voltage telemetry: the balance voltage shifted from its nominal ~11 volts to ~4 volts. The team has deduced that an internal low impedance short on the MMRTG as the only credible root cause of the anomaly. This short spontaneously cleared on sol 461 when the rover was asleep. The MSL team has learned to identify an internal MMRTG short and explicitly clear a persistent soft short, "invoking a battle short"[9]. The first time this procedure was used (sol 816) is considered the time of the anomaly for this paper. Again, this is a worst-case assumption but is considered since the method to clear was discovered by happenstance elsewhere in the instrument suite and not part of the power design. Over time, the frequency of shifts has increased.

With no mission ending failures, it would be tempting to declare 100% reliability of the systems. This is inaccurate as a predictor for these missions and especially for a mission of 50 years. We now incorporate the 2 RPS failures to present as a bounding case. Using Bayesian updating, probability of success metrics is computed, see Table 2.

Table 1. Probability of Success Metrics

	Units	Fails Prior to Sys Design Life	Mean	5th	95th
All	44	2	0.939	0.879	0.981
SNAP	28	1	0.944	0.877	0.988
RTG	16	1	0.922	0.829	0.982

* Prior: Beta Distribution with 5^{th}=0.75 and 95^{th}=0.99

II.B. Mission Design Life vs Actual Mission Duration

Computing how long the systems last is more difficult since all but 2 units were working at the time the missions ended. Figure 1 shows the RPS design life versus actual life. The dotted line is where design life equals mission duration. The markers represent the missions with green being active missions and red being the two anomalies. Note that the majority of mission are above the dotted line indicating the RPS are lasting longer than their design lives.

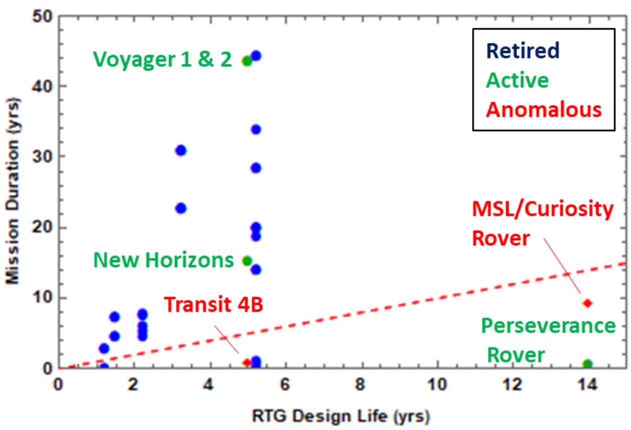

Figure 1. RPS systems last longer than their intended design lives

II.C. Lifetime Assessment

If one were to take the average mission duration (16.9 years) as the metric for how long these systems can last, that would not tell the correct story since all but 2 systems were operational at the end of the mission. A statistical analysis technique called a Survival Analysis is often used when some of the data is right censored (operation halted before a failure), but this struggles to determine the mean life. When the analysis is applied to the sub-populations of SNAP and RTG systems, the difficulty is magnified with only 1 failure for each. A Bayesian analysis is used to update a prior belief with the observations available to provide a distribution of lifetime.

The life model used is a Weibull distribution. It has 2 defining parameters, β which defines the shape indicating infant mortality or wearout, and η which is the "characteristic life". Both parameters are unknown and is what the Bayesian analysis will solve for. Our prior for each is defined as Lognormal distributions with 5^{th} and 95^{th} percentiles of:

β : between 0.5 and 5

η : between 10 and 75 years

Updating with all the data produces a distribution of time to failure (see Figure 3). The resulting mean lifetime is just over 100 years and a lower 5^{th} percentile around 50 years.

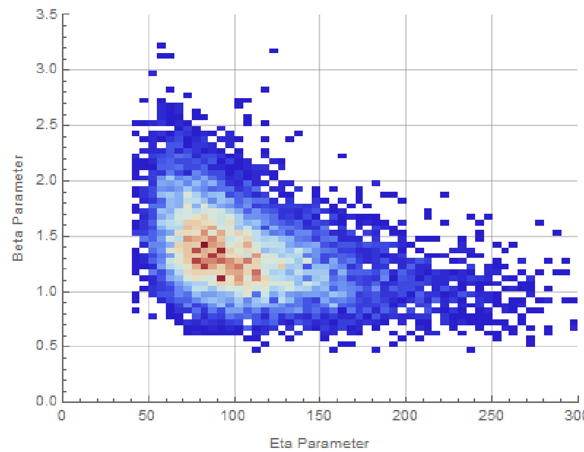

Figure 2. Bayesian assessment shows uncertainty of both Weibull distribution parameters

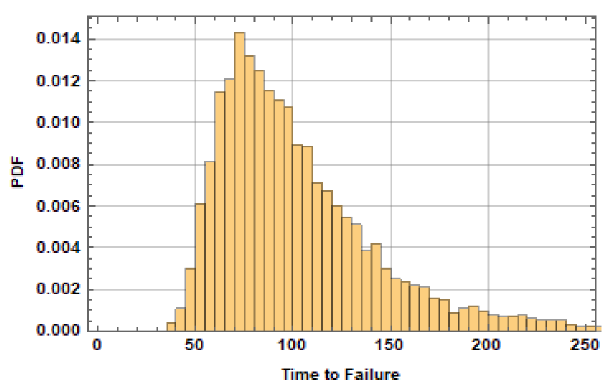

Figure 3. Bayesian assessment shows RPS time to failure distribution

III. RELIABILITY

While statistical assessments of past systems show a long life is possible, much work needs to be done to understand how the system degrades over time and how various time dependent failure mechanisms behave. A physics of failure and probabilistic approach is needed. Two activities are currently on-going supporting reliability analysis. The Risk Informed Life Testing (RILT) is a physics based assessment to model time dependent failure mechanisms.[10] The RPS office is currently constructing a reliability framework to apply to all RPS technologies to inform the probability of delivering advertised power at EOL.

IV. TIME DEPENDANT PERFORMANCE ANALYSIS

Life Performance Prediction (LPP) is a capability developed at NASA's Jet Propulsion Laboratory (JPL) used to model thermoelectric conversion physics in Radioisotope Thermoelectric Generators (RTGs) for a variety of missions and storage environments over long durations. Having the ability to accurately model and predict performance is critical in mission and operations planning in order to optimize the science return during mission activities. LPP accepts several types of inputs to perform predictions in order to achieve the level of accuracy these missions require. Features unique to each RTG such as thermoelectric and insulation material properties as a function of time and temperature coupled with mission characteristics such as fin root temperature and load voltage allow LPP to output results of interests such as power, internal resistance, and interface temperatures. Experimentally collected data that is of importance to the long-term performance predictions for RTGs includes degradation effects of key interfaces and insulation (i.e. electrical and thermal contact resistance of hot side interfaces and changes to insulation thermal conductivity). By relying on large amounts of relevant test data, LPP can accurately match measured results gathered from flagship missions such as the MMRTG on Curiosity and Perseverance.

LPP was used to examine the power output over time and estimate the point at which the unit would no longer produce power. Figure 4 shows a 16 GPHS unit operating for 73 years until no power is produced with the following inputs: 224 W beginning of mission power, hot and cold junction temperatures of 830°C and 296°C, and a constant 30V bus.

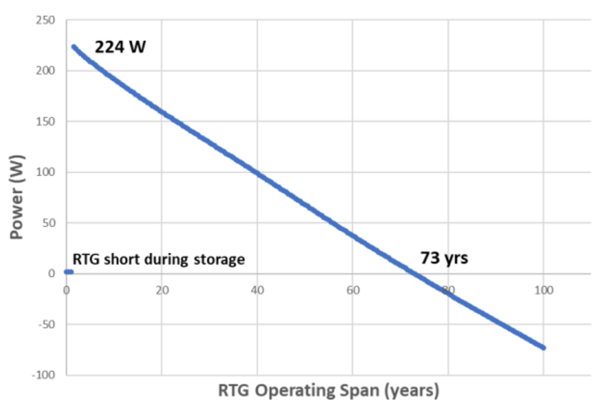

Figure 4. GPHS-RTG power prediction for ISP through 100 years with 16 GPHS

A similar analysis with a 18 GPHS units shows power being produced out to 85 years.

V. CONCLUSIONS

RPS design have progressed to keep pace with the demands of space missions. The life achieved from these systems continues to out-perform their stated design life regardless of era. While the statistical analyses of life and power performance show that multi-decadal missions are possible, caution must be used with these results. The old adage about extrapolating beyond the data set applies. Further, limitations on the material degradation and potential chemical reactions have not been fully examined here. However, the results do show promise of extended life lasting several decades.

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Table 2. RPS Unit Listing

Mission Name (No. of Units)	Power Source	Launch Date	RPS End Date	Mission Design Life	RPS Design Life*	RPS Anomaly	
Transit 4A	SNAP-3B	6/29/1961	7/1/1962	0.05	5.00		
Transit 4B	SNAP-3B	11/15/1961	8/2/1962	0.05	5.00	X	RTG failure in June
Transit 5BN-1	SNAP-9A	9/28/1963	12/22/1963	0.05	5.00		
Transit 5BN-2	SNAP-9A	12/5/1963	11/1/1964	0.05	5.00		
Transit 5BN-3	SNAP-9A	4/21/1964	4/21/1964	0.05	5.00	n/a	Launch failure
SNAPSHOT	SNAP-10A	4/3/1965	5/16/1965	1.00	1.00		SNAP lasted 43 days
Nimbus B-1 (2)	SNAP-19	5/18/68				n/a	Launch failure
Nimbus III (2)	SNAP-19	4/14/1969	1/22/1972	2.00	1.00		SNAP salvaged from NIMBUS B
Apollo 12 ALSEP	SNAP-27	11/14/1969	7/1/1977	2.00	2.00		
Apollo 13 ALSEP	SNAP-27	4/11/1969				n/a	Device did not reach the moon
Apollo 14 ALSEP	SNAP-27	1/31/1970	7/1/1977	2.00	2.00		
Apollo 15 ALSEP	SNAP-27	7/26/1971	7/1/1977	2.00	2.00		
Pioneer 10 (4)	SNAP-19	3/3/1972	1/23/2003	7.00	3.00		
Apollo 16 ALSEP	SNAP-27	4/16/1972	7/1/1977	2.00	2.00		
Triad 1	TRANSIT-RTG	9/2/1972	7/1/2006	1.00	5.00		NRC indicates RTG still operational as of 2006.
Apollo 17 ALSEP	SNAP-27	12/7/1972	7/1/1977	2.00	2.00		
Pioneer 11 (4)	SNAP-19	4/6/1973	11/24/1995	7.00	3.00		
Viking 1 lander (2)	SNAP-19	8/20/1975	11/13/1982	1.25	1.25		RPS design for 1 year travel + 90 days ops
Viking 2 lander (2)	SNAP-19	9/9/1975	4/12/1980	1.25	1.25		RPS design for 1 year travel + 90 days ops
LES 8 (2)	MHW-RTG	3/15/1976	7/1/2004	3.00	5.00		
LES 9 (2)	MHW-RTG	3/15/1976	5/20/2020	3.00	5.00		
Voyager 2	MHW-RTG	8/20/1977		4.50	5.00		
Voyager 1	MHW-RTG	9/5/1977		4.50	5.00		
Galileo (2)	GPHS-RTG	10/18/1989	9/21/2003	8.00	5.00		
Ulysses	GPHS-RTG	10/6/1990	6/30/2009	5.00	5.00		
Cassini (3)	GPHS-RTG	10/15/1997	9/15/2017	11.00	5.00		
New Horizons	GPHS-RTG	1/19/2006		15.00	5.00		
MSL/Curiosity Rover	MMRTG	11/26/2011	12/8/2014	2.50	14.00	X	Battleshort first used on sol 816
Perseverance Rover	MMRTG	7/30/2020		2.50	14.00		

*RPS design life information found in Vining, C. B., and D. M. Rowe. "CRC Handbook of Thermoelectrics." Ed., DM Rowe, CRC Press, Inc., Florida (1995): pages 520-534.