AIAA Propulsion and Energy Forum
August 24-28, 2020, VIRTUAL EVENT
AIAA Propulsion and Energy 2020 Forum

10.2514/6.2020-3850

Mars Opposition Missions Using Nuclear Thermal Propulsion

Christopher B. Reynolds[1], Claude R. Joyner II[2], Timothy Kokan[3], Daniel J.H. Levack[4], and Brian J. Muzek[5]

[1]Aerojet Rocketdyne, Huntsville, Alabama 35806, USA
[2]Aerojet Rocketdyne, West Palm Beach, Florida 33410, USA
[3]Aerojet Rocketdyne, Huntsville, Alabama 35806, USA
[4]Aerojet Rocketdyne, Canoga Park, California 91309, USA
[5]Aerojet Rocketdyne, Huntsville, Alabama 35806, USA
(256) 971-2642; christopher.reynolds@rocket.com

Abstract

Since 2016, Aerojet Rocketdyne (AR) has been working with NASA and other industry partners to improve the design and increase the feasibility of Low Enriched Uranium (LEU) NTP engine systems that may provide programmatic and cost benefits over older Highly Enriched Uranium (HEU) designs. From the project's inception, mission analysis has primarily focused on Mars conjunction class missions, on the order of 900 days total mission duration and 600 days Mars surface stay time. However, within the past year, that focus has shifted to shorter duration Mars opposition class missions. Opposition class missions have the benefit of shorter total mission duration (less than 2 years), but result in Mars surface stay times on the order of one month. Generally, these missions also require two to three times the ΔV of their conjunction counterparts.

In order to satisfy the requirements of a Mars opposition mission, a study was performed to determine the optimal NTP Mars Transfer Vehicle (MTV) architecture, while also taking advantage of previous work completed for Mars conjunction missions. Trades were performed on NTP main engine Isp, number of NTP engines, NTP MTV assembly/aggregation orbit, Mars aggregation orbit, MTV stage/drop tank diameter, and stage/drop tank launch vehicle. From these trades, several NTP MTV configurations were developed, each sized to fulfill the requirement of closing a Mars opposition mission in the mid to late 2030s. Each configuration uses a combination of SLS delivered NTP stages and either SLS or commercially delivered drop tanks. By utilizing the drop tank approach, the mass ratio of the NTP MTV can be maximized over the entire Mars opposition mission. The propellant mass fraction of the entire vehicle stack can be increased at the start of the mission, while also reducing the dry mass that is carried throughout the mission.

This paper selects four configurations for examination, and shows the rationale behind several of the architectural decisions made, and the design maturity taken throughout the study. Engine parameter sensitivities are also presented for two configurations, showing the effect of engine Isp and engine mass on number of commercial drop tanks. Additionally, ground rules and assumptions for Mars cargo delivery are also detailed. Similar to the baseline Mars conjunction architecture, cargo delivery for a Mars opposition mission can be performed with a NTP cargo vehicle that is derived from the NTP MTV crew core. Two vehicle options are presented that are derived from the NTP crew core, with payload numbers provided for three cargo mission options.

[1] Engineer, Mission Architecture & Analysis / Systems Engineering, 950 Explorer Blvd NW, AIAA Member
[2] Fellow, Advanced Propulsion System Architecture Development / Systems Engineering, 17900 Beeline Hwy / M/S 712-67, AIAA Fellow
[3] Principal, Mission Architecture & Analysis / Systems Engineering, 950 Explorer Blvd NW, AIAA Member
[4] Program Manager, Advanced Launch Vehicle Propulsion / Program Management, P.O. Box 7922 / MS RFB19, AIAA Member
[5] Senior Engineer, Mission Architecture & Analysis / Systems Engineering, 950 Explorer Blvd NW, AIAA Member

I. Nomenclature

AR	=	Aerojet Rocketdyne
AR&D	=	Automated Rendezvous and Docking
BVP	=	Boundary Value Problem
CONOPS	=	Concept of Operations
DAC	=	Design Analysis Cycle
DSM	=	Deep Space Maneuver
EMVE	=	Earth-Mars-Venus-Earth (Opposition trajectory type)
EOI	=	Earth Orbit Insertion
EVME	=	Earth-Venus-Mars-Earth (Opposition trajectory type)
GCD	=	Game Changing Development
HEU	=	Highly Enriched Uranium
I_{sp}	=	Specific impulse, a measure of efficiency (thrust per mass of fuel burned)
LDHEO	=	Lunar Distant High Earth Orbit
LEO	=	Low Earth Orbit
LEU	=	Low Enriched Uranium
MEO	=	Medium Earth Orbit
MOI	=	Mars Orbit Insertion
MTV	=	Mars Transfer Vehicle
NASA	=	National Aeronautics and Space Administration
NERVA	=	Nuclear Engine for Rocket Vehicle Application
NRHO	=	Near Rectilinear Halo Orbit
NTP	=	Nuclear Thermal Propulsion
NTR	=	Nuclear Thermal Rocket
OMS	=	Orbital Maneuvering System
SLS	=	Space Launch System
SOI	=	Sphere of Influence
STMD	=	Space Technology Mission Directorate
TEI	=	Trans-Earth Injection
TLI	=	Trans-Lunar Injection
TMI	=	Trans-Mars Injection
TVI	=	Trans-Venus Injection
VSB	=	Venus Swing-By
WSB	=	Weak Stability Boundary
ΔV	=	Velocity change, m/s

II. Introduction/Background

Over the past 60 years there have been several efforts to study and define the optimal architecture for human exploration of Mars [1-3]. Solutions have ranged from large scale bi-propellant chemical transportation systems to highly-efficient nuclear electric propulsion vehicles, and everything in between. A key observation when examining these previous studies is that many of them have highlighted advanced propulsion systems as an enabling technology for near term human exploration, specifically, nuclear thermal propulsion or NTP. This is evident when looking at the NERVA program, ran jointly by NASA and the Atomic Energy Commission, with a goal of providing a nuclear power upper stage for the Saturn V [4]. In addition to this nuclear powered upper stage, nuclear thermal propulsion was already seen as a potential technology for Mars exploration. Over the past four years, Aerojet Rocketdyne (AR) has been working with NASA and other industry partners to access the feasibility of using NTP for human exploration of Mars.

Initial studies looked to define the architecture for a Mars conjunction mission using NTP and many of the elements envisioned for NASA's lunar exploration program [5]. As the vehicle and engine evolved, opposition class missions were examined at a top level for feasibility, and starting in late 2019, opposition class missions became the focus of effort and a study was began at AR to examine several vehicle options and configurations that could enable these more difficult, but worthwile, missions. The primary focus of this paper is to detail the intitial conclusions and vehicle configurations from the NTP opposition class missions study.

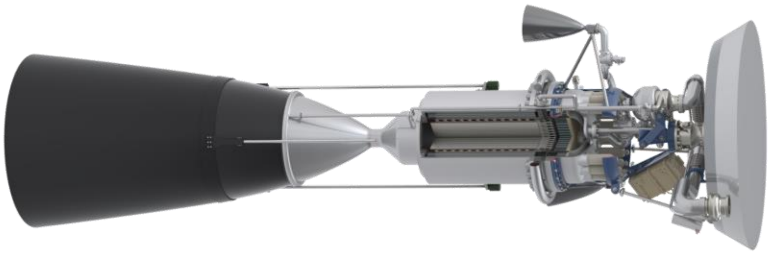


Figure 1. Modern 25,000 lbf NTP Engine Cutaway

A Nuclear Thermal Rocket (NTR), as shown in Figure 1, provides thrust by heating a propellant (in this case hydrogen) that is passed through a nuclear fission reactor. As the propellant exits the reactor it is expanded through a nozzle generating significant momentum that propels the vehicle. By using hydrogen as the working fluid, NTP has much higher exhaust velocities, due to the low molecular weight, and therefore higher specific impulse than conventional chemical rockets - though not as high as electric propulsion. NTP provides advantages over electric propulsion due to the high thrust of the engines, which translates into reaching interplanetary velocities much more quickly. When the high specific impulse is coupled with the high thrust of NTP, high performance mission architectures can be designed.

III. Baseline Conjunction Class Mission

The current conjunction architecture has been in development since 2016 as part of a NASA Space Technology Mission Directorate (STMD) Game Changing Development (GCD) program. Starting with the mission trades and conceptual design of the NTP vehicle and engine, the architecture has evolved to become a robust and extensible solution for the human exploration of Mars. The baseline mission begins with the aggregation and assembly of the NTP vehicle in NRHO, using a SLS Block 2 launch vehicle to throw the individual elements on a TLI trajectory. With the vehicle assembled, the NTP stack transfers to LDHEO and awaits the arrival of the crew. Each mission assumes a conjunction class trajectory. After Trans Mars Injection (TMI), the NTP vehicle with crew takes as little as 160 days to get to Mars, significantly reducing crew transit times versus other propulsion options.

To return to Earth, the NTP vehicle performs both Trans Earth Injection (TEI) and Earth Orbit Insertion (EOI) to insert into LDHEO, where the crew is met by Orion and returns to the surface. The NTP vehicle then performs a maneuver to return to NRHO via a long-duration WSB trajectory. A representative CONOPS for a Mars conjunction class mission in 2033 is shown in Figure 2.

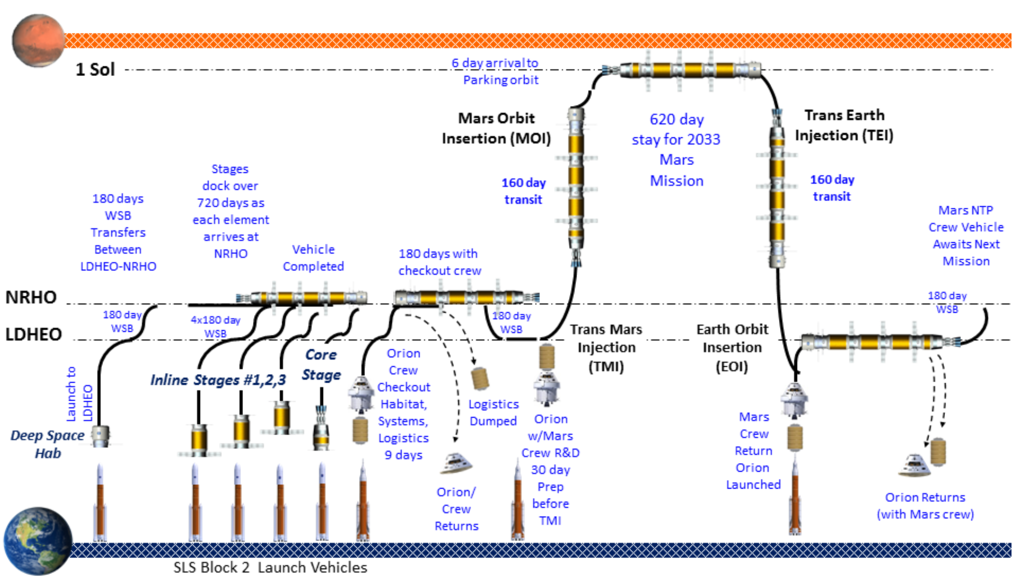


Figure 2. 2033 Mission Bat Chart

In the years leading up to a manned Mars mission, crew surface assets are prepositioned using several cargo launches. For these cargo missions, a derivative of the NTP crew core can be used. This NTP cargo vehicle launches off SLS Block 2 vehicle and rendezvous and docks with the Mars cargo lander in HEO. The vehicle then performs TMI to put the stack on a minimum energy trajectory to Mars. After Mars Orbit Insertion (MOI), the payload descends to the surface and the NTP cargo vehicle preforms a disposal burn to place itself into a heliocentric orbit. Further definition of the NTP cargo vehicle is available in the sections below.

A. Crew Vehicle

The baseline NTP conjunction crew vehicle consists of a core stage with three NTP engines each producing 25,000 lbf of thrust, and three common inline stages. Both elements are mass limited, sized for the SLS Block 2 TLI payload capability of 45 mT [6]. By constraining the launch mass of each of the elements, as the dry mass of the vehicle changes, the gross mass of the vehicle remains the same. With this approach, changes to vehicle dry mass or to payload mass only change the inbound and outbound transfer times based on the impulse (ΔV) capability of the NTP vehicle. The baseline NTP Mars Crew Vehicle is shown in Figure 3.

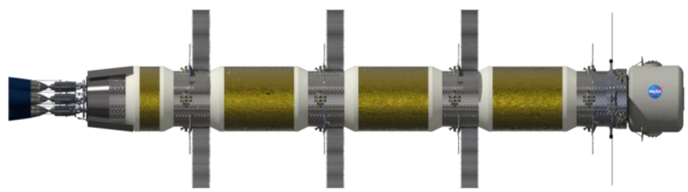


Figure 3. Mars Conjunction Baseline NTP Vehicle

Over time, the NTP conjunction vehicle has gone through several Design Analysis Cycles (DACs) that have increased the modeling fidelity of subsystem mass estimates, architecture analysis, and NTP engine definition [5]. Due to its fidelity, the NTP conjunction vehicle provides an ideal jumping-off point for the initial design of the elements for an opposition class NTP vehicle. As shown in later sections, the same NTP engine and NTP vehicle elements are used to develop a vehicle capable of completing a Mars opposition mission, with modified elements used for NTP vehicle and architecture optimization. Figure 4 below shows details of the Mars conjunction NTP vehicle.

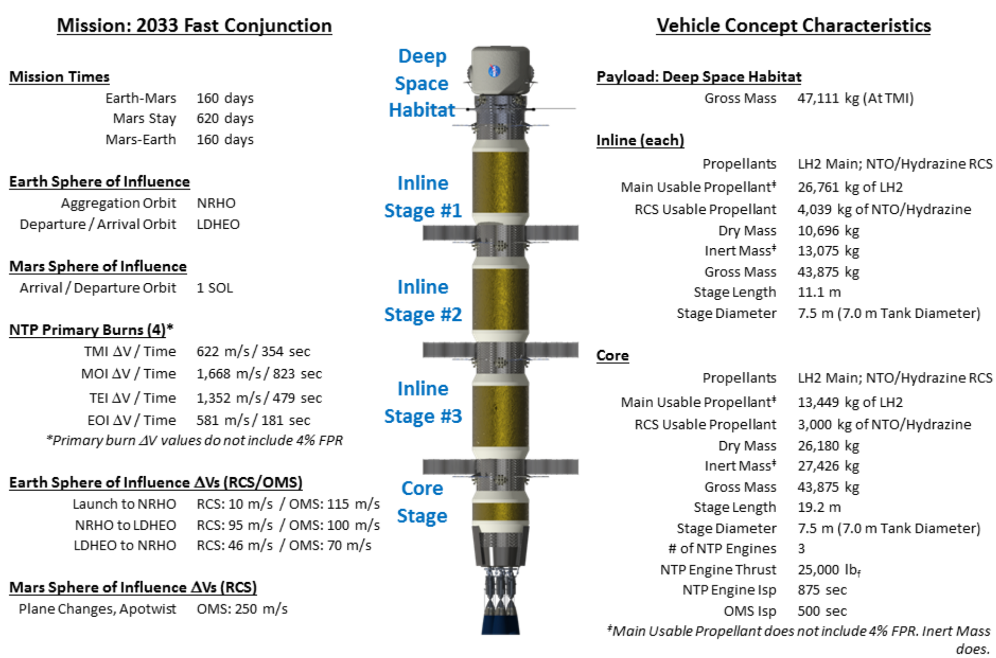


Figure 4. Mars NTP Conjunction Baseline Vehicle Baseball Card

B. Cargo Vehicle

The baseline NTP cargo vehicle for conjunction missions is a modification of the NTP crew core (Figure 5 below) using one NTP engine instead of three and including a larger propellant tank. While the elements for the crew vehicle are mass limited to the SLS Block 2 TLI payload capability, the cargo vehicle is sized to be delivered to the same orbit as a 54 mT Mars cargo lander, resulting in a NTP cargo vehicle sized to be 54 mT. This eliminates any major orbital maneuvering required for payload alignment, leaving only the AR&D maneuver to be done with RCS.

Figure 5. Mars NTP Cargo Vehicle Commonality with Crew Vehicle (Conjunction Mission)

For Mars opposition missions, the same design philosophy is followed for the NTP cargo vehicle, deriving the vehicle from the NTP crew core. For configurations that use a larger diameter NTP MTV (i.e. 9.1m tank diameter), the NTP Cargo Core can be volume limited in the larger SLS 10m payload fairing, resulting in a 90 mT NTP Cargo Core. This allows extra cargo mass to be delivered to Mars or reduces the number of launches necessary to field the required Mars surface assets. Further detail on the NTP Cargo Core and missions is provided in later sections.

C. Trajectory Considerations

Figure 6 below shows ΔV requirements and transfer times for Mars conjunction and opposition missions, with opportunities starting in 2030 and extending to 2050. The purple curve represents conjunction missions, with the blue, green, and red curves representing various opposition class missions. The dashed black lines represent baseline vehicle performance used as reference for this study.

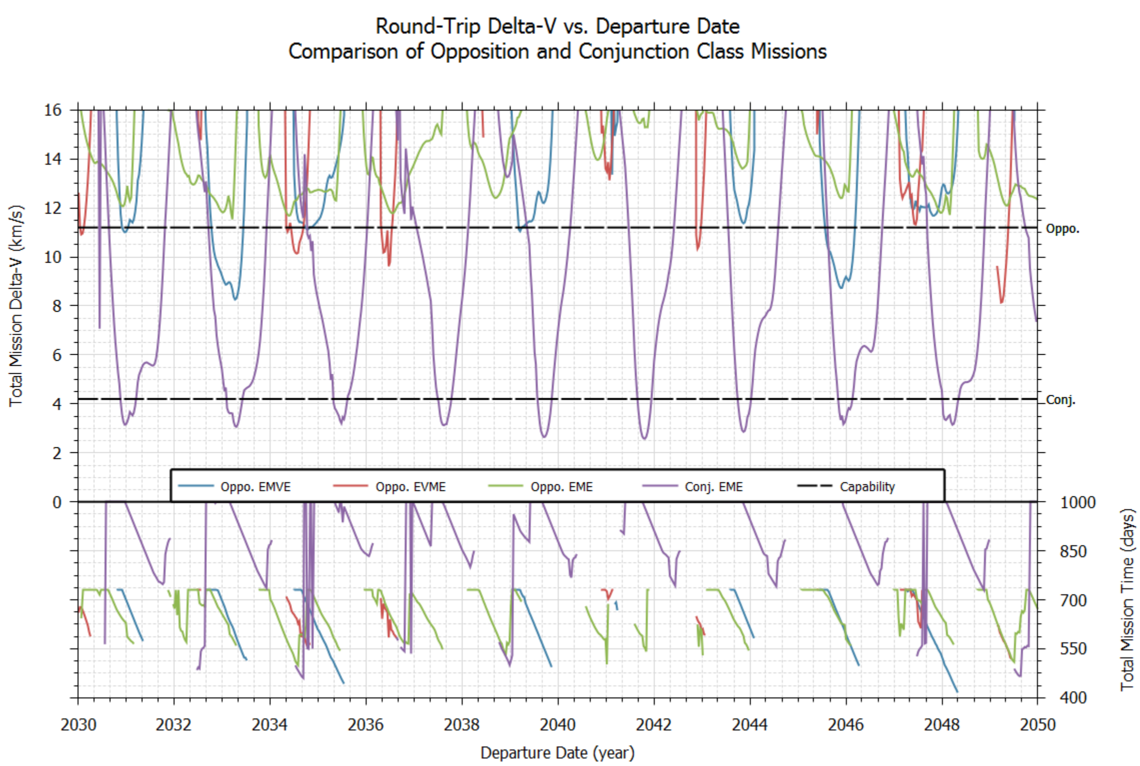


Figure 6. Mars Conjunction DV Requirements and Mission Times

IV. Opposition Study Ground Rules & Assumptions

Ground rules and assumptions were defined at the onset to bound and direct the study. Previous Mars opposition studies were examined to see what constraints were made, and how those constraints affected and influenced the results [1-3]. As a result of the literature review, Mars missions were constrained to surface landings in either 2035 or 2039. This would allow time to develop the necessary technologies, preposition any surface assets, and allot time for missions to occur alongside NASA's envisioned lunar exploration campaign.

A. Mission Ground Rules

Table 1 shows the primary architecture ground rules and assumptions used for each of the NTP vehicle configurations detailed in the sections below. These assumptions were used to generate the patched conics and end-to-end trajectory models that are inputs for the vehicle sizing models. Vehicle level assumptions are provided in a later section.

Table 1. Study Ground Rules and Assumptions

Architecture Level Assumptions	
Mars Mission Opportunity	2035, 2039
Surface Stay Time	> 50 days
Crewed Time Away From Earth	< 2 years
Launch Vehicles Assumed	SLS Block 2, New Glenn, SpaceX Starship
Earth Aggregation/Departure Orbits	LEO, LDHEO, NRHO
Mars Aggregation Orbits	5-Sol

The trajectory/mission analysis developed for this study is examined in a previous paper starting from initial assumptions and approximation and progressing to the development of a detailed end-to-end trajectory model [7]. The details provided below on the trajectory model is specific to the 2035 opposition opportunity, but was updated to include the 2039 opportunity details when the primary modeling was complete.

B. Patched Conics Trajectory

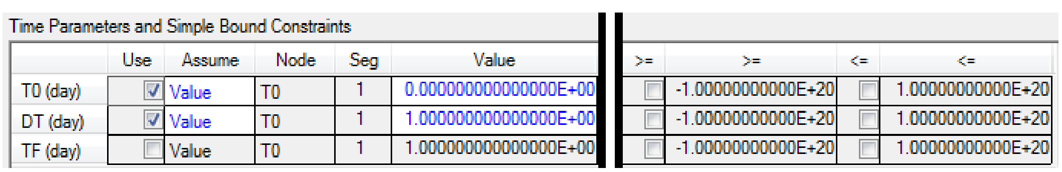
As with the previous Conjunction mission analysis, trajectory analysis was performed with NASA's Copernicus orbital mechanics and trajectory code [8]. The basic heliocentric analysis is constructed of five Copernicus segments, which depend on the type of Opposition mission flown. Each of the interplanetary segments is constructed with a Lambert's BVP solution matching the target body position and velocity after the segment delta-time. The summary of these segments is given in Table 2, and a graphic representation is shown in Figure 8.

Table 2. Summary of Copernicus Segments

Segment No.	EMVE	EVME
1	Trans-Mars Injection (TMI) to Deep Space Maneuver (DSM)	Trans-Venus Injection (TVI) to Venus Swing-by (VSB)
2	Deep Space Maneuver (DSM) to Mars Orbit Insertion (MOI)	Venus Swing-by (VSB) to Mars Orbit Insertion (MOI)
3	Mars SOI Stay	Mars SOI Stay
4	Trans-Venus Injection (TVI) to Venus Swing-by (VSB)	Trans-Earth Injection (TEI) to Deep Space Maneuver (DSM)
5	Venus Swing-by (VSB) to Earth Orbit Insertion (EOI)	Deep Space Maneuver (DSM) to Earth Orbit Insertion (EOI)

Segment time specification in Copernicus allows the specification of any two of initial time (T0), delta time (DT), or final time (TF), as shown in Figure 7. In order to perform a sweep across Earth departure date, the initial time value of segment 1 is fixed. Instead of specifying the initial time of all four segments, however, it is prudent to specify segment delta-times for all four segments. In addition to being internally consistent, efficient transfers are generally found within a range of transfer times, and using the bound constraints reduces the computing effort wasted optimizing where the delta-times are extremely long or where the transfer time becomes negative. In addition, the constraint of minimum Mars SOI stay time is made a bound constraint, which is handled more efficiently by the optimizer than a general linear or nonlinear constraint. This reduces the number of free variables by one (initial time is fixed) and the number of nonlinear constraints by one (Mars SOI time is a bound constraint). Reducing the number of free variables and nonlinear constraints generally helps the optimizer to find a solution more quickly.

The inclusion of a Deep Space Maneuver (DSM) complicates the analysis somewhat. In order to take full advantage of the Lambert solver, the position at which the DSM occurs is specified in heliocentric spherical co-ordinates. The distance from the Sun is constrained by a simple bound constraint to be between the orbits of Venus and Mars, and the solar ecliptic declination (angular distance from the ecliptic) is constrained via simple bound constraint to be ±30° from the ecliptic. The right ascension is allowed to vary as necessary around the celestial sphere.


Time Parameters and Simple Bound Constraints

	Use	Assume	Node	Seg	Value	>=	>=	<=	<=
T0 (day)	☑	Value	T0	1	0.000000000000000E+00	☐	-1.00000000000E+20	☐	1.00000000000E+20
DT (day)	☑	Value	T0	1	1.000000000000000E+00	☐	-1.00000000000E+20	☐	1.00000000000E+20
TF (day)	☐	Value	T0	1	1.000000000000000E+00	☐	-1.00000000000E+20	☐	1.00000000000E+20

Figure 7. Copernicus Time Specification Menu. Some options are removed for clarity.

The Δv values calculated by Copernicus for this trajectory are heliocentric in nature; that is, they assume that the vehicle starts in a solar orbit with the position and velocity of the planet they are orbiting. The impulsive Δv function in Copernicus provides a capability for performing a "zero-SOI" swing-by maneuver, which accounts for the patched-conics approach at the Venus swing-by. However, no such capability is built into Copernicus for planetary departure and capture. The solution to this is to take advantage of Copernicus's plug-in feature. Because the visa-vis equations are explicit, Copernicus's parser plug-in capability is leveraged [9].

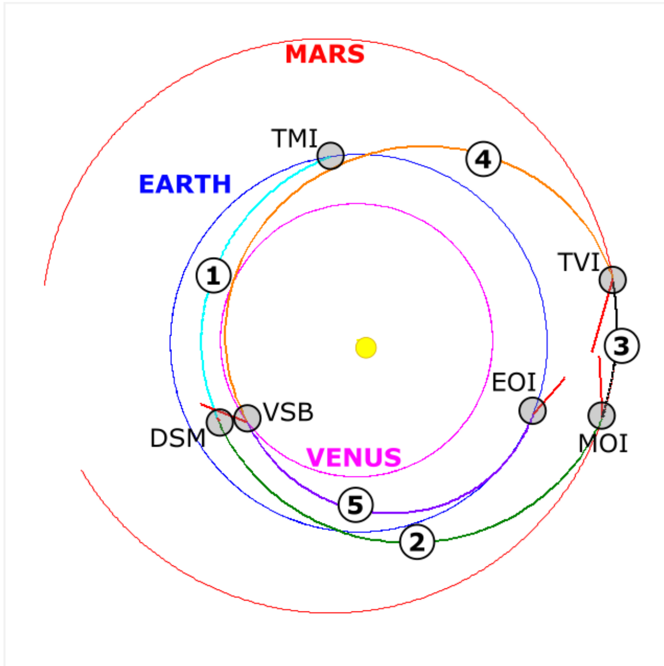


Figure 8. Overview of patched-conics EMVE mission analysis. Segment endpoints are circled and segment numbers are shown.

The parser plug-in feature allows the user to extract specific variables from any segment in a Copernicus deck and use them in an auxiliary calculation. Any values so calculated can be subsequently used as part of an objective function or non-linear constraint, or fed back into another segment's initial conditions. In this instance, the Mars- and Earth-relative Cartesian states (three components each of position and velocity) are read into the plug-in. The planet-relative velocity at intercept or departure is equivalent to the planetary SOI hyperbolic excess velocity. By calculating the magnitude of the planet-relative velocity, the Δv at planetary departure or capture can be adjusted into the planetary SOI. An illustration of the maneuvers used is given in Figure 8. The Copernicus program summary of these variables is shown in Figure 9 below.

Edit	Variable	Type	Assume	Node	Seg	Value
	mu (km**3/s**2)		Value			3.986000000000000E+05
	rp (km)		Value			6.778000000000000E+03
	ra (km)		Value			3.231620000000000E+05
	ISP (s)		Value			3.200000000000000E+02
	m0 (kg)	Mass [m]	Inherit	T0--	1	1.000000000000000E+03
	g0 (m/s**2)		Value			9.806649999999999E+00
	Rx (km)	State [Rx]	Inherit	T0+	1	0.000000000000000E+00
	Ry (km)	State [Ry]	Inherit	T0+	1	0.000000000000000E+00
	Rz (km)	State [Rz]	Inherit	T0+	1	0.000000000000000E+00
	Vx (km/s)	State [Vx]	Inherit	T0+	1	2.660961927023077E+00
	Vy (km/s)	State [Vy]	Inherit	T0+	1	3.501152683218006E+00
	Vz (km/s)	State [Vz]	Inherit	T0+	1	1.790070870613166E+00

(a)

Edit	Variable	Type	Push	Value
	v_inf (km/s)			4.747961900656827E+00
	v_ph (km/s)			1.183887486566938E+01
	v_pe (km/s)			1.073310854275719E+01
	dV_pc (km/s)			1.105766322912189E+00

(b)

Figure 9. (a) Input and (b) Output parameters for the parser plug-in. Relative velocity is marked with the green box, SOI-adjusted Δv is marked with red.

The sum total of all six propulsive maneuvers SOI-adjusted Δv is used as the objective function for optimization. To perform the sweep of departure dates, the CopPy Python library provided with the Copernicus executable is used. CopPy allows the user to script Copernicus runs and to alter the input decks programmatically from within a Python 2.7 or 3.X language user-written script. Because the Lambert's BVP solver in Copernicus operates directly on the input deck impulsive Δv parameters, these can also be extracted from the input file after it has been processed by Copernicus. After extraction, the magnitude and date of each maneuver are written to a comma-separated values (.csv) file for post-processing and inspection.

C. End-to-End Trajectory

Copernicus is also utilized for the complete n-body analysis of the end-to-end mission stack. However, because of the level of detail and computation time required for such a study, the end-to-end trajectory is only performed after an architecture and mission have been selected. The trajectory as modeled includes 31 segments, 112 control variables, 11 inequality constraints, and 58 equality constraints. Despite the seeming intractability of the problem, it is further separated into "groups" to ease understanding and computation.

The other major difference between the patched-conics model and the n-body model is the separate treatment of large main-propulsion burns and orbital maneuvering system (OMS) burns. Main propulsion is used for large Δv maneuvers, namely planetary capture and departure and the deep space maneuver if it is large enough. The OMS system is used for small adjustments, such as deep space targeting maneuvers, the Venus swing-by, and orbital adjustment maneuvers in Mars SOI. The key propulsion values are specified in zero-duration segments (see Table 3), and all propulsive maneuvers inherit from these two segments. This ensures consistency across the input deck and allows faster modification should these parameters change.

Table 3. Propulsion Characteristics for Finite Burns

	Main Propulsion	OMS
Thrust-to-Weight	0.057	0.0002
Specific Impulse (s)	900	500

The initial guess for each mission group is drawn from the results of the patched-conics model. To do this, an initial time and state is specified near each SOI transition in terms of hyperbolic orbit parameters. The Copernicus state input dialog for these parameters is shown in Figure 10. The three components of the hyperbolic excess velocity from the patch-conics approximation can be entered directly into the first three elements of the state specification. B-plane angle (BTheta), periapsis radius (Rp) and true anomaly (Ta) are allowed to float between specified bounds to match a desired planetary SOI state.

	Use	Assume	Node	Seg	Value
VoutX (km/s)	☑	Value	T0-	5	-2.237899003548695E+00
VoutY (km/s)	☑	Value	T0-	5	-1.263090627273493E+00
VoutZ (km/s)	☑	Value	T0-	5	2.684165044163018E+00
BTheta (deg)	☑	Value	T0-	5	-5.313022790329670E+01
Rp (km)	☑	Value	T0-	5	6.883069682296880E+03
Ta (deg)	☑	Value	T0-	5	6.197699688126208E+01

Figure 10. Copernicus State Specification for Hyperbolic Elements.

A specific parking orbit is desired at both Earth and Mars SOI. In order to assure that this orbit is achieved, it too is specified explicitly in a state dialog. The escape orbit and parking orbit are then matched via a finite burn (see Figure 11). On the hyperbolic trajectory, one parameter is free to orient the orbit for arrival/departure (B-plane angle). For the elliptic parking orbit, the apogee and perigee altitude are specified and three variables are free to orient the orbit parallel to the departure orbit: longitude of the ascending node, inclination, and argument of periapsis. The optimizer is free to use these parameters to align the orbits for minimum losses.

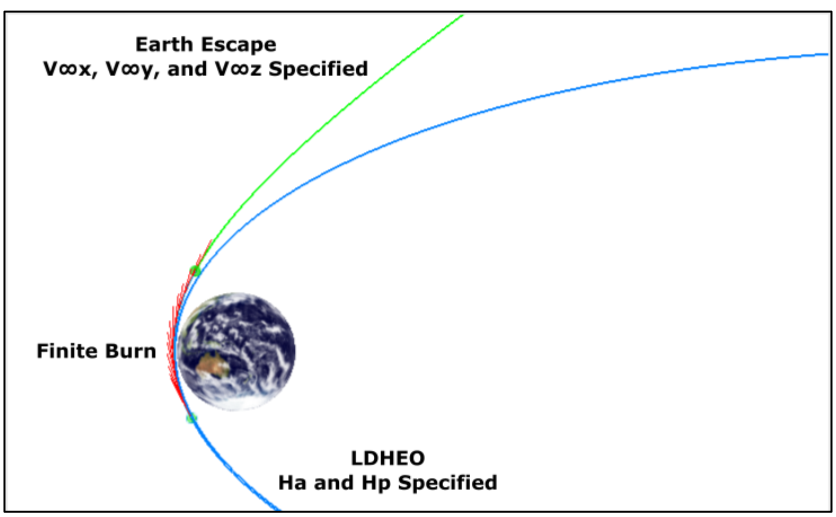


Figure 11. Example of Earth/Sun SOI Patch via Hyperbolic Orbit Specification

At Mars, the situation is similar. However, because the departure parking orbit and arrival parking orbit may not be aligned, there must be some mechanism to get from one to the other. Three OMS burns are placed between the inbound and outbound parking orbits along with inter-burn coasts to space them out. Because both parking orbits are the same shape (same apoapsis and periapsis radius), conceptually, these burns must provide an inclination change, an apse line shift, or both. Because of the high eccentricity of the parking orbit (see Figure 12) these non-tangential maneuvers are best accomplished near apoapsis where the orbital velocity is low. In addition, a portion of these maneuvers can be provided during the arrival and departure burns by using a slightly non-tangential burn that does not occur at periapsis. In any case, the states of the hyperbolic arrival and departure orbits and elliptical parking orbits are specified and finite burns are inserted between to patch them together. This allows the optimizer to orient the orbits and burns as necessary to minimize ΔV.

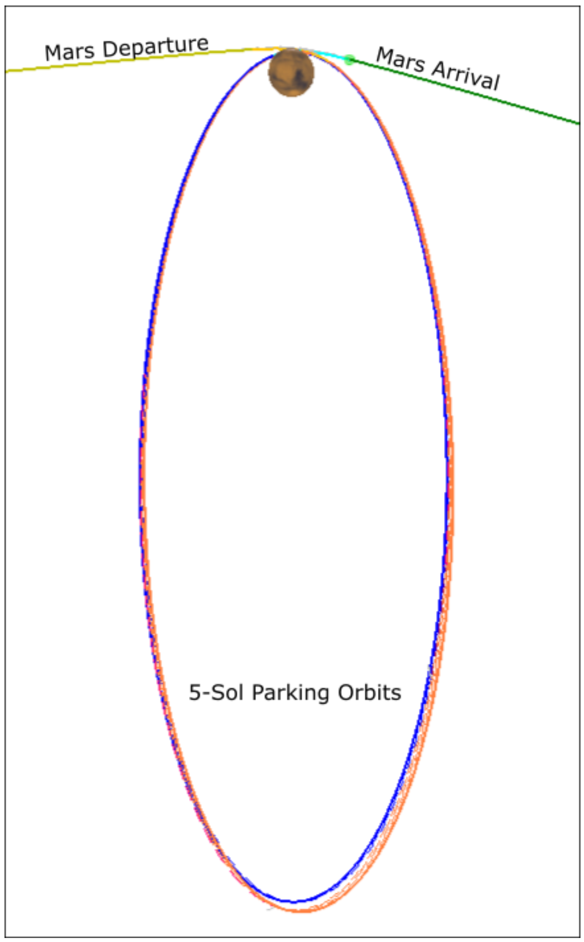


Figure 12. Mars Orbit Operations

Because there are two different propulsion systems acting in this model, straight Δv is no longer an appropriate objective function. Instead, the mass ratio of the vehicle is tracked through the entire mission, and the mass after Earth arrival is used as the objective function and maximized. To make a direct comparison between the patched-conics and n-body results, an effective Δv from the n-body model can be calculated by assuming the main propulsion specific impulse and calculating an effective main propulsion Δv via Equation 1.

$$\Delta v_{eff} = g_0 \cdot I_{sp,main} \cdot \ln\left(\frac{m_0}{m_f}\right) \quad (1)$$

D. NTP MTV Stage Assumptions

Along with architecture level ground rules and assumptions, several vehicle and propulsion system level assumptions were made to simplify the analysis. This was primarily done to create a level playing field by which each NTP opposition vehicle configuration could be compared. The primary vehicle and propulsion level assumptions are shown in Table 4.

Table 4. Vehicle and Propulsion Ground Rules and Assumptions

Vehicle Level Assumptions	
Flight Performance Reserves	5%
Payload Attach Fitting Mass Allocation	2.5% of Launch Mass
Mass Growth Allowance + Mass Margin	30%
NTP Elements Available	Core Stage, Strap-On, Inline, Drop Tank
Deep Space Habitat Mass at TMI	40,000 kg – 45,000 kg
Drop Tank Truss Structure Mass	3000 kg
Propulsion System Assumptions	
NTP Engine Thrust	25,000 kg
NTP Engine Isp	~900 sec
NTP Engine Mass	5,500 kg
OMS Thrust	600 lbf/thruster (1,200 lbf/engine)
OMS Isp	500 sec
RCS Isp	333 sec

As a result of the assumptions made in the study, the NTP engine specific impulse was set at 900 sec, pushing for each vehicle to close at that currently attainable performance. If higher specific impulse is obtained for the engine, each vehicle configuration would translate that higher Isp directly to vehicle performance margin. For each opposition vehicle configuration, a truss structure is assumed for drop tank attachment. A mass of 3,000 kg was assumed for the truss structure based on previous mass estimation efforts. A detailed structural analysis of the drop tank truss structure is ongoing for specific vehicle configurations.

V. NTP Mars Opposition Vehicle Configurations

Over the course of the study, several vehicle configurations were examined using a variety of launch vehicles and element designs, with the goal of reducing the gross mass of the vehicle while also staying within the architecture and vehicle level GR&As as detailed in the sections above. Figure 13 below contains details on each of the configurations examined and shows the design path taken to arrive at the most recent vehicle architecture. Each vehicle configuration is sized to complete a Mars opposition mission with overall ΔVs in the range of 8.4-11.2 km/s.

Case	Aggreg. Orbit	Gross Mass	No. of Elements	No. of Drop Tanks	No. of NTPs / Cumulative Burn Time	No. of Launches	SLS PLF Used
Reference Conjunction Vehicle	NRHO	203 mT	Core: 1 Inline: 3	None	3 Engines/ 0.6 hours	SLS: 4 CLV: 0	8.4 m LONG
1	NRHO	440 mT	Core: 1 Inline: 3	6 (SLS)	3 Engines/ 1.4 hours	SLS: 10 CLV: 0	8.4 m LONG
2	LEO	844 mT	Core: 1 Inline: 2	18 (New Glenn)	4 Engines/ 2.7 hours	SLS: 3 CLV: 18	8.4 m LONG
3	MEO	476 mT	Core: 1 Inline: 2	5 (Starship)	4 Engines/ 2.0 hours	SLS: 3 CLV: 5	8.4 m LONG
4	LEO	649 mT	Core: 1 Inline: 2	4 (SLS)	4 Engines/ 3.1 hours	SLS: 7 CLV: 0	10 m LONG
5	LEO	921 mT	Core: 1 Inline: 2	18 (New Glenn)	4 Engines/ 2.9 hours	SLS: 3 CLV: 18	10 m LONG
6	MEO	465 mT	Core: 3 Inline: 1	4 (Starship)	4 Engines/ 2.0 hours	SLS: 4 CLV: 4	8.4 m LONG
7	LEO	795 mT	Core: 3 Inline: 1	16 (New Glenn)	4 Engines/ 3.7 hours	SLS: 4 CLV: 16	8.4 m LONG
8	LEO	723 mT	Core: 3 Inline: 1	12 (New Glenn)	4 Engines/ 3.4 hours	SLS: 4 CLV: 12	Nose Cone PLF
9	LEO	717 mT	Core: 3 Inline: 1	11 (New Glenn)	4 Engines/ 3.4 hours	SLS: 4 CLV: 11	Nose Cone PLF

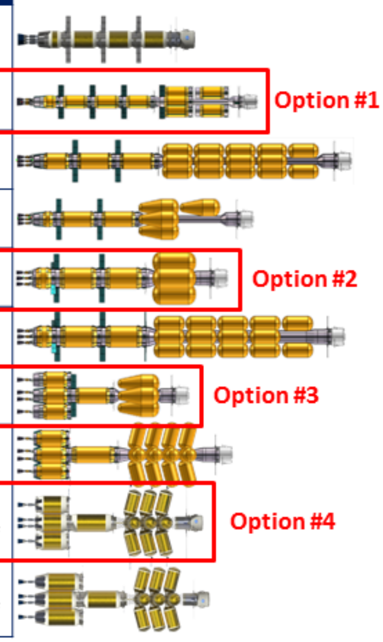


Figure 13. NTP Opposition Vehicle Configurations

Four opposition vehicle options (highlighted in Figure 13 above) were chosen for further detail, looking at the launch/ mission CONOPS and detailed vehicle mass estimates. These options were chosen to span the trade space of the study, and to also demonstrate the design maturation throughout the study.

A. NTP MTV Option #1

At the onset of the study, a vehicle configuration was examined that used the same propulsive elements (core and inline stage) as the NTP conjunction vehicle, while also following similar architecture assumptions (i.e. aggregating at the moon in NRHO). This was done do determine the number of NTP elements needed to complete the opposition mission, while limiting new development of stages not already designed for conjunction missions. Due to the increased ΔV required for the opposition missions, vehicle staging was implemented, reducing the vehicle gross mass required to achieve those higher ΔVs. Additional inline stages were used as truss-mounted drop tanks, allowing for staging at three locations along the mission (TMI, DSM, and MOI). Figure 14 below shows the launch and mission CONOPS up to MOI for Option #1.

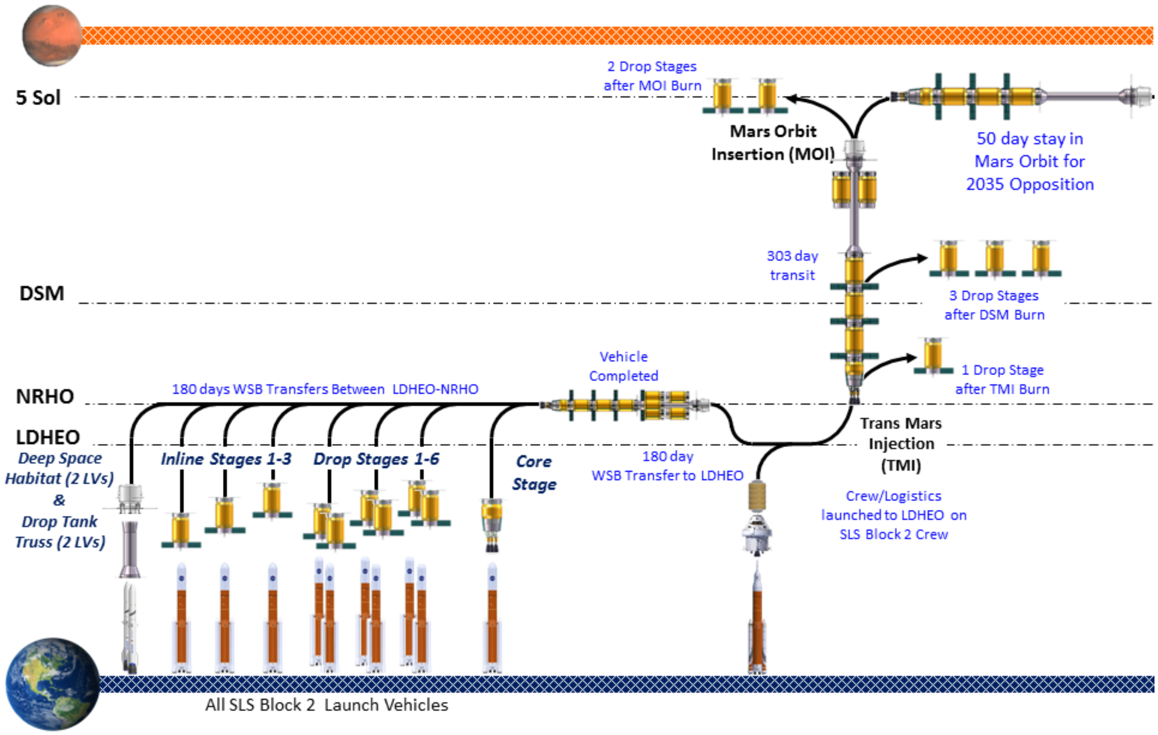


Figure 14. Option #1 Bat Chart

The vehicle configuration for Option #1 uses a core stage, three inline stages, and six inline-based drop tanks. With this vehicle using the same propulsive individual elements as the reference NTP conjunction vehicle, each element is sized to the TLI throw capability of the SLS Block 2 (45 mT). After vehicle assembly in NRHO, the entire stack (including the deep space habitat (DSH) transfers to LDHEO to await arrival of the crew. During TMI, only enough propellant to drain one full drop stage is used, resulting in only that stage being discarded. After the DSM and MOI, three drop stages and two drop stages are removed respectively. After MOI, a vehicle stack consisting of the core stage, three inline stages, the drop tank truss structure, and the DSH is located in a 5-sol Mars orbit. The return trip to Earth (including the Venus fly-by) can be completed with this residual NTP vehicle.

The general characteristics of the vehicle and its individual elements are shown below in Figure 15. The overall vehicle gross mass at TMI is 440 mT, much greater than that of the reference conjunction vehicle. This increase in gross mass is directly attributed to the higher required ΔV for the 2035 Mars opposition opportunity (~2X the total ΔV of a conjunction mission).

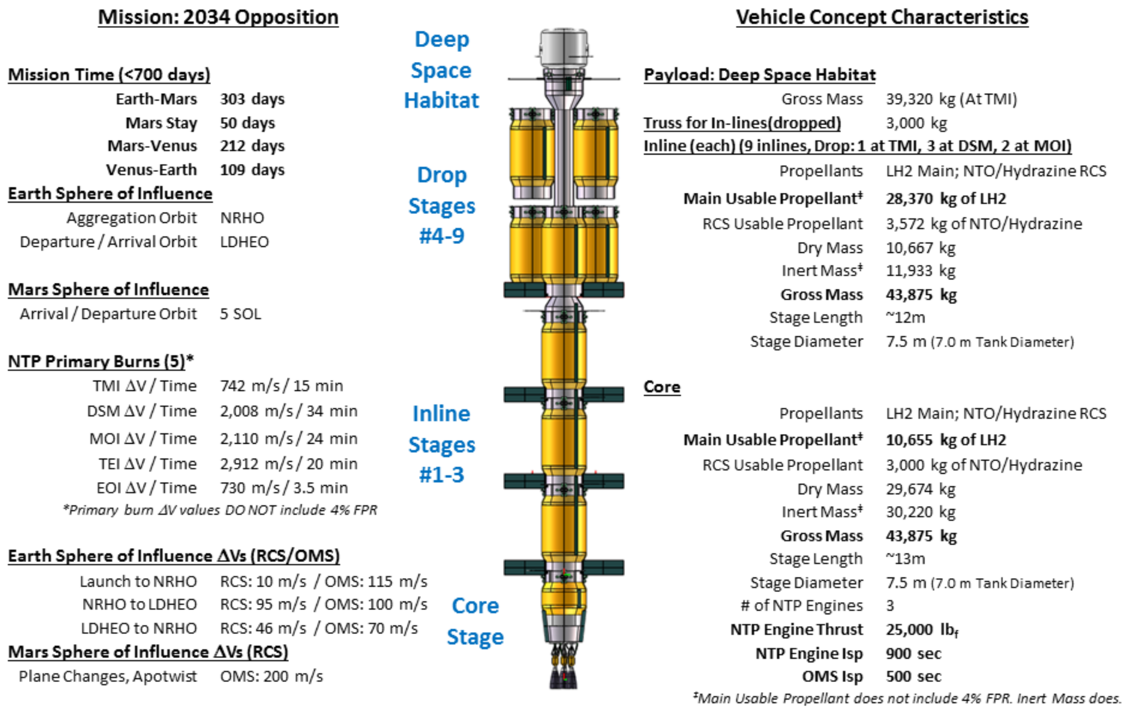


Figure 15. Option #1 Baseball Card

B. NTP MTV Option #2

At lower required ΔVs, the combination of lower gross mass stages and higher aggregation orbits proves fruitful, as the relationship between launch capability and payload fairing volume allows for mass limited stages. When the required mission ΔV more than doubles, changes at the architecture and vehicle level are required to reduce number of launches, and to become more mass efficient at the stage level. The next three NTP opposition vehicle configurations assemble and aggregate in LEO or MEO rather than NRHO like Option #1 or the reference conjunction vehicle. This is done to increase the mass/volume of each NTP element while also allowing for more of the total mission ΔV to be performed by the more efficient propulsion.

For Option #2, the primary goal was to use propulsive elements launched entirely on a SLS Block 2 launch vehicle with 10m Long payload fairing. This was done to be able to launch the largest (from both a mass and volume perspective) elements possible into LEO, increasing the overall propellant mass fraction of the vehicle, thus reducing the vehicle gross mass at TMI. Post-launch, each element is inserted into a 1,300 km LEO for assembly into the complete vehicle stack. This vehicle configuration consists of the core stage, two inline stages, and four SLS launched drop tanks. The aggregation/assembly and mission CONOPS are shown in Figure 16.

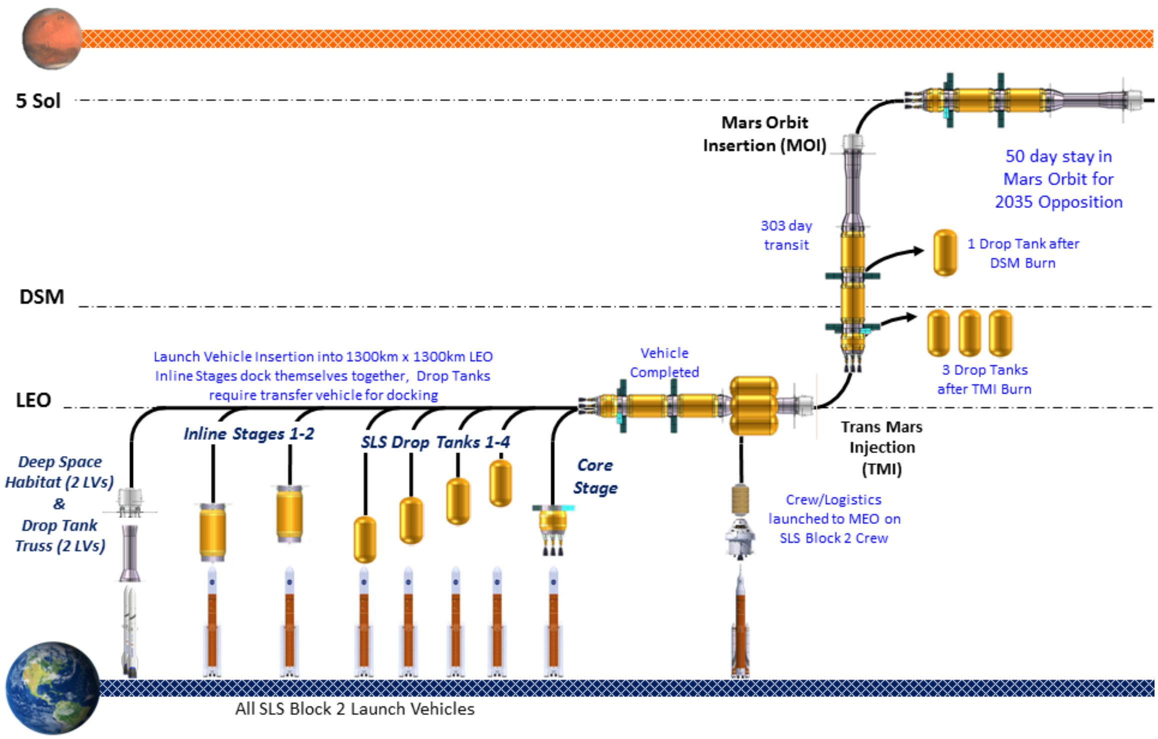


Figure 16. Option #2 Bat Chart

After vehicle assembly in LEO, the crew arrives in Orion and the mission begins. In order to minimize gravity losses, the TMI maneuver is split into two burns of equal burn time, while also adding a fourth NTP engine to the core stage. After TMI, sufficient propellant has been expended to remove three drop tanks. The final drop tank is disposed of after the DSM, leaving the remaining propellant in the core stage and inline stages to complete the rest of the mission.

Figure 17 below shows the mass estimates, ΔVs, burn times, and general statistics for the vehicle and each individual element. The overall gross mass at TMI is 649 mT, higher than that of the Option #1 opposition vehicle, but much lower than some of the other options examined under the study. This is due to the higher gross mass and larger volume of the individual elements. Each element is sized to the lift capability of the SLS Block 2 when inserting into a 1,300 km LEO, which results in launch masses of 93.6 mT (87.5 mT gross mass) for each of the elements, with element volumes utilizing the entire length of the 10m Long payload fairing.

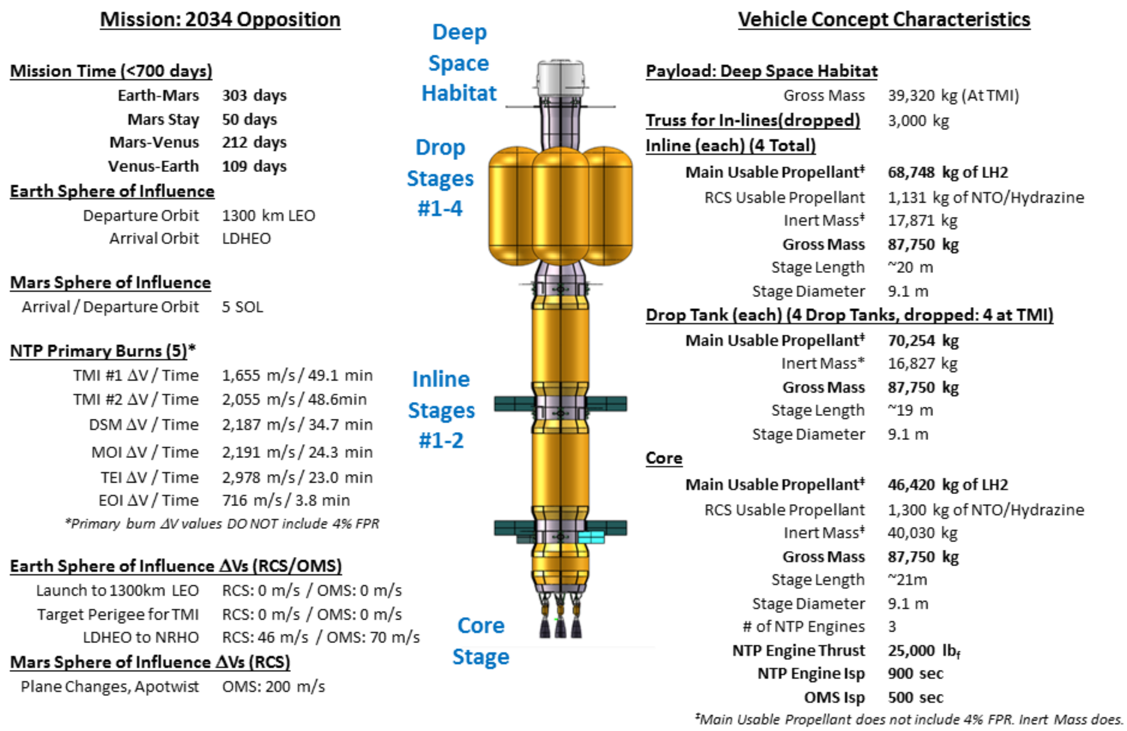


Figure 17. Option #2 Baseball Card

C. NTP MTV Option #3

Option #3 was the first configuration examined that utilized strap-on core stages. These stages are single engine core variants of the main NTP core stage and provide additional thrust and propellant for the vehicle while in Earth's gravity well. Having the additional thrust of the strap-on core stages allows for one NTP engine to be removed from the main NTP core stage, leaving two engines on the main vehicle stack. The thrust provided by these two engines is sufficient to limit gravity losses throughout the rest of the mission, while increasing the propellant available on the core stage. This approach of having strap-on core stages (also called side mounted boosters) has been explored in past Mars architecture studies [10-11] and is a unique way of adding performance while keeping the overall vehicle mass down. Additionally, the strap-on core stages have commonality with the envisioned NTP cargo vehicle for prepositioned cargo delivery to Mars. This could allow for element reuse, either reusing old NTP cargo vehicles as the strap-on core stages, or reserving propellant in the strap-on core stages to return to Earth orbit to be refueled for additional missions (crewed or cargo).

The Option #3 vehicle is comprised of a NTP core stage, two strap-on core stages, a single inline stage, and four drop tanks. For this configuration, the drop tanks are launched on the SpaceX Starship launch system, and are sized to take advantage of the payload capability and volume provided. The other propulsive elements are launched on the SLS Block 2 launch vehicle with the 8.4m Long payload fairing. The stages are mass and volume limited, maximizing the LH_2 volume within each tank while allowing SLS to throw each stage as high as possible. This vehicle sizing method results in an MEO aggregation and assembly orbit of 2,000km x 13,000km. The drop tank gross masses were determined using a launch vehicle model anchored to data provided in the Starship payload user's guide [12].

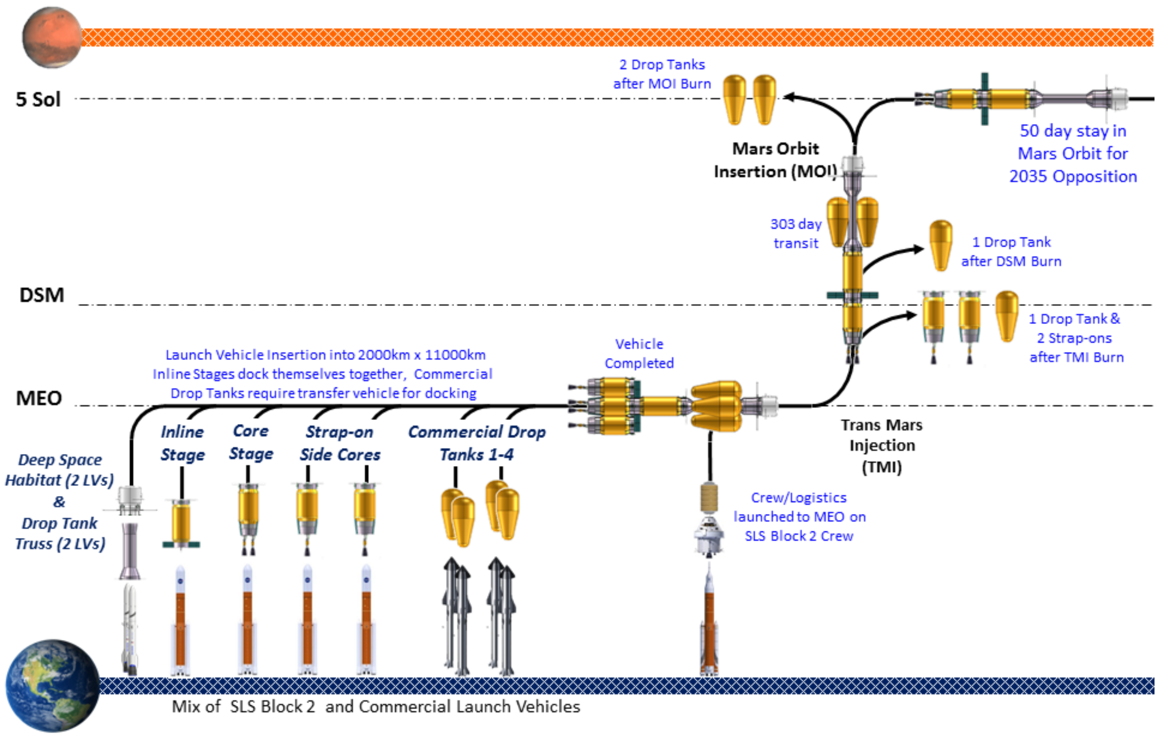


Figure 18. Option #3 Bat Chart

The aggregation/assembly and mission CONOPS for Option #3 are shown in Figure 18. After launching each of elements into orbit, and assembling the vehicle, the crew would arrive in Orion to MEO and would transfer to the deep space habitat. From there, a two-burn TMI would occur, utilizing propellant from the strap-on core stage and one of the drop tanks. The strap-on core stages and a single drop tank would be staged after the burn, leaving the main stage vehicle with three drop tanks, an inline stage, and the primary NTP core stage. After the DSM and MOI burns, a single drop tank and two drop tanks would be staged respectively, leaving the NTP vehicle with a single inline and core stage to perform the remainder of the mission.

Figure 19 below shows the general statistics of Option #3 at the vehicle and individual element level. The gross mass of the vehicle at TMI is 465 mT, showing that the addition of the strap on stages can provide a mass efficient solution even with the aggregation departure orbit starting deep within Earth's gravity well. The SLS launched NTP elements are sized to 62 mT (the mass and volume limit for the MEO aggregation orbit), and the Starship launched drop tanks have a launch mass of 52 mT.

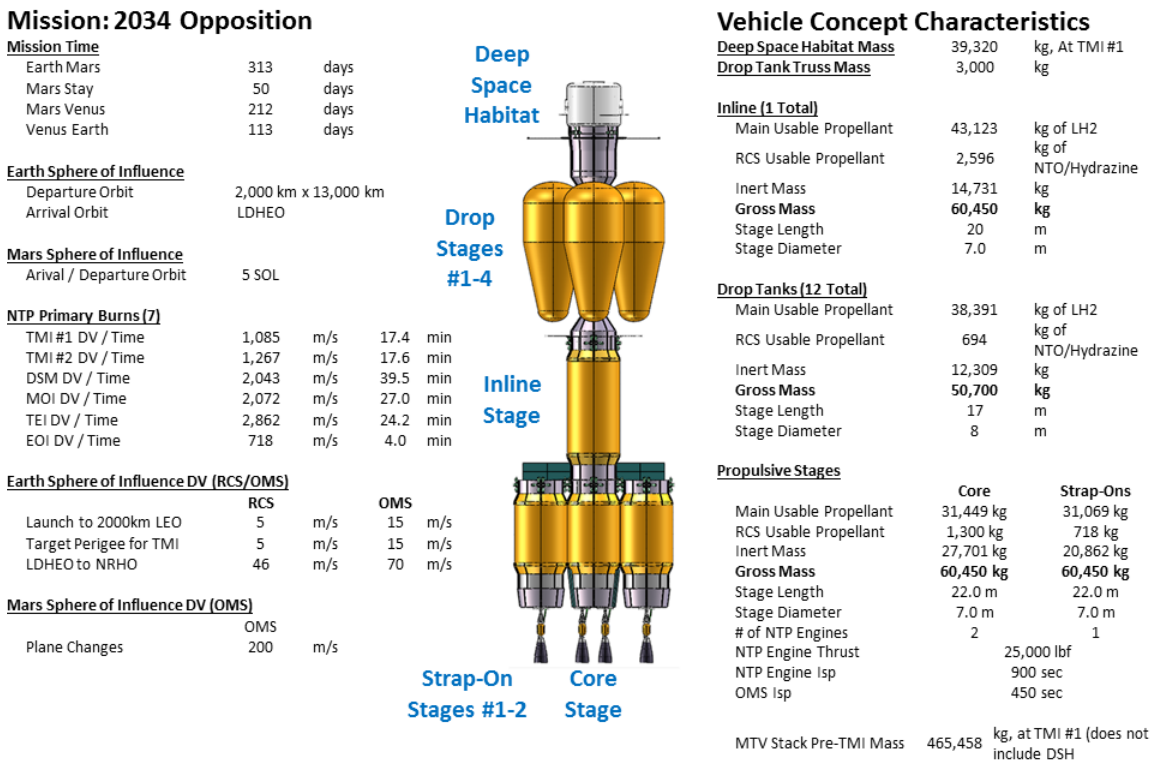


Figure 19. Option #3 Baseball Card

D. NTP MTV Option #4

The lessons learned throughout the Mars opposition study resulted in the Option #4 vehicle configuration. Like Option #3, Option #4 uses strap-on core stages to increase the vehicle thrust and propellant load while in Earth's gravity well. This vehicle also uses drop tanks launched on the New Glenn launch vehicle, with each tank canted 15 degrees off perpendicular to allow for more efficient packaging. The primary difference between Option #4 and the previous options examined concerns the outer diameter of the primary propulsive elements (main core stage, strap-on core stages, inline stage). In order to maximize the propellant volume on the core stages and inline stage, the outer diameter of the tanks was increased from 7.0m to 8.4m, allowing the stages to sit flush on top of SLS with a small nose cap to protect the stage during launch. This diameter is the same as the SLS core stage, EUS LH_2 tank, and the standard 8.4m Long payload fairing, thus not changing the outer mold line of the primary vehicle structure. Figure 20 illustrates the concept of attaching a nose cap to the top of each stage during launch.

Figure 20. NTP Option #4 Elements w/ Nose Cap

The Option #4 vehicle is comprised of a NTP core stage, two strap-on core stages, a single inline stage, and 12 New Glenn launched drop tanks (shown in Figure 21 below). The individual elements are inserted into a 1,300 km LEO by the launch vehicle for assembly into the complete NTP vehicle. To ensure that the outer mold line of the SLS launch vehicle remains unchanged, the gross mass of the core and inline stages were adjusted to maintain the same overall launch vehicle length (core stage, interstage, EUS, payload fairing). Also, to increase the performance at the overall vehicle level, the vehicle was assembled without the deep space habitat, which aggregates separately in NRHO. At the start of the mission, a two burn maneuver raises the aggregation orbit to LDHEO (2,000 km x 316,784 km); utilizing the propellant from six drop tanks and the two strap-on core stages. These spent stages are discarded from the vehicle, leaving only the six remaining drop tanks attached to the inline and NTP core stage.

Figure 21. Option #4 Drop Tank in New Glenn PLF

At this point the deep space habitat would transfer to LDHEO and the crew would arrive in Orion. The remaining drop tanks would be used to perform both the TMI and DSM burns (2 for TMI and 4 for DSM). Specific to Option #4, the opposition opportunity was changed from 2035 to 2039. This was primarily done to enable each vehicle to capture most opposition opportunities within the 15 year Earth-Mars synodic cycle. A benefit of this configuration when compared to the other options that utilize the strap-on core stages is that the strap-on core stages are discarded before Earth departure, and thus can be more easily recovered for reuse. Figure 22 below shows the aggregation and mission CONOPS for Option #4.

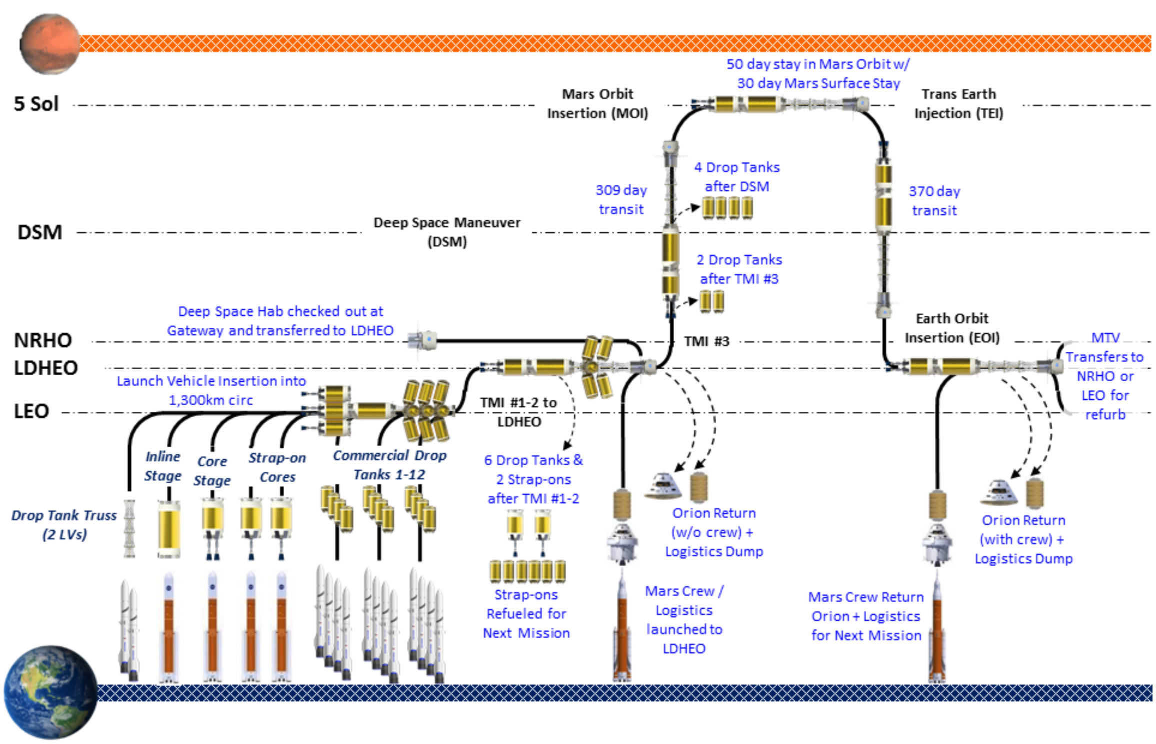
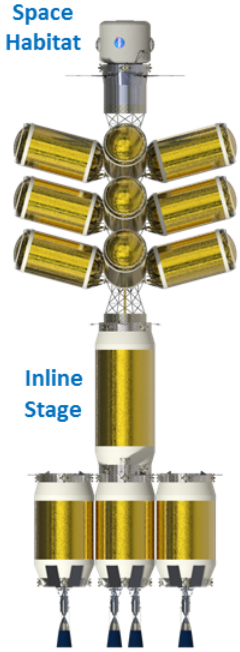


Figure 22. Option #4 Bat Chart

Figure 23 contains the mass estimates, ΔVs, mission times, and other general characteristics for the Option #4 vehicle configuration and individual elements. The overall gross mass at TMI is 723 mT. This vehicle is heavier than some of the other options examined, but shows robustness in both vehicle and mission design. Due to the sizing philosophy of each of the primary NTP elements, the masses are different in order to maintain the same overall SLS Block 2 vehicle length. The NTP core stage, strap-on core stage, inline stage launch masses are 78.5 mT, 76 mT, and 93.6 mT respectively. The launch mass of the New Glenn launched drop tanks is 35 mT, based on internal vehicle models anchored to public data [13]. Also, there was a desire to limit the number of drop tank launches with Option #4 to the current number of 12. In order to maintain 12 drop tanks, as the element masses were updated, the specific impulse of the NTP engine was raised or lowered. Shown below, an Isp of 904 seconds for the NTP engine is required to close the 2039 Mars opposition mission with 12 total drop tanks.

Mission: 2039 Opposition

Mission Time				
Earth Mars	309	days		
Mars Stay	50	days		
Mars Venus	211	days		
Venus Earth	159	days		
Earth Sphere of Influence				
Departure Orbit	1,300 km LEO			
Arrival Orbit	LDHEO			
Mars Sphere of Influence				
Arival / Departure Orbit	5 SOL			
NTP Primary Burns (7)				
TMI #1 DV / Time	1,291	m/s	32.6	min
TMI #2 DV / Time	1,632	m/s	33.1	min
TMI #3 DV / Time	853	m/s	25.1	min
DSM DV / Time	3,332	m/s	72.6	min
MOI DV / Time	1,429	m/s	17.3	min
TEI DV / Time	1,908	m/s	18.5	min
EOI DV / Time	989	m/s	7.3	min
Earth Sphere of Influence DV (RCS/OMS)				
	RCS		OMS	
Launch to 2000km LEO	0	m/s	0	m/s
Target Perigee for TMI	0	m/s	0	m/s
LDHEO to NRHO	46	m/s	73	m/s
Mars Sphere of Influence DV (OMS)				
	OMS			
Plane Changes	50	m/s		



Vehicle Concept Characteristics

Deep Space Habitat Mass	45,000	kg, at TMI #1
Drop Tank Truss Mass	3000	kg
Inline (1 Total)		
Main Usable Propellant	61,646	kg of LH2
RCS Usable Propellant	5,782	kg of NTP/Hydrazine
Inert Mass	23,832	kg
Gross Mass	**91,260**	**kg**
Stage Length	21.4	m
Stage Diameter	8.4	m
Drop Tanks (12 Total)		
Main Usable Propellant	21,538	kg of LH2
RCS Usable Propellant	260	kg of NTP/Hydrazine
Inert Mass	12,327	kg
Gross Mass	**34,125**	**kg**
Stage Length	14.4	m
Stage Diameter	6.35	m

Propulsive Stages	Core	Strap-Ons
Main Usable Propellant	42,694 kg	42,695 kg
RCS Usable Propellant	700 kg	4,417 kg
Inert Mass	33,144 kg	27,027 kg
Gross Mass	**76,538 kg**	**74,139 kg**
Stage Length	22.0 m	22.0 m
Stage Diameter	8.4 m	8.4 m
# of NTP Engines	2	1
NTP Engine Thrust	25,000 lbf	
NTP Engine Isp	904 sec	
OMS Isp	450 sec	
MTV Stack Pre-TMI Mass	723,042	kg, at TMI #1 (does not include DSH

Figure 23. Option #4 Baseball Card

E. Engine Isp and Mass Sensitivities

Sensitivities to NTP engine mass and Isp were performed to understand how each architecture would be affected by increasing or decreasing performance. The variable traded against for each assessment is the number of drop tanks required to close the mission, as that parameter encompasses both vehicle complexity/mass and total ΔV capability. The target NTP Isp for each architecture is 900 sec, with an extended goal of 1000 sec. NTP Isp is traded along a range, from 875-1000 sec in order to see how the increase (or decrease) in Isp changes the capability of each architecture. The NTP engine mass trades ranged from 3,500-6,500kg for each engine. The trades shown in the sections below are for options #2 and #4 detailed above.

1. NTP Engine Isp

Figure 24 shows the NTP Isp sensitivity for vehicle option #2. Over the range of specific impulse traded, the number of drop tanks ranged from 5 drop tanks at the lower end of the range, down to 3 drop tanks at the high end of the range. With this vehicle configuration having large SLS launch drop tanks, a lower sensitivity to changes in the vehicle and architecture are to be expected.

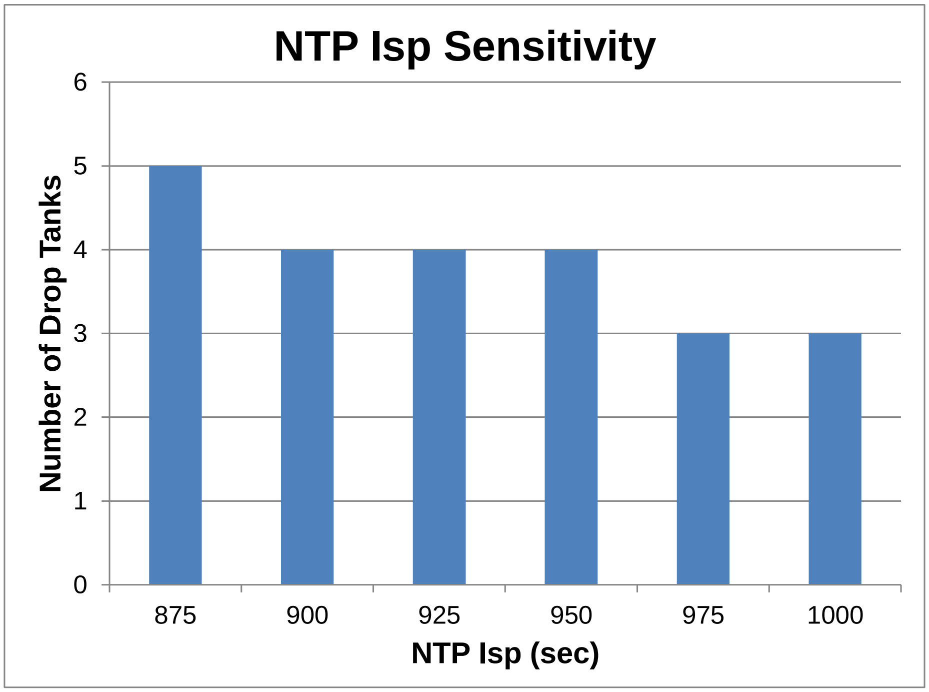


Figure 24. NTP Isp vs. Number of Drop Tanks for Option #2

For the NTP vehicle option #4, the sensitivity to Isp is much greater due to the lower gross mass of the drop tanks (Figure 25). The number of drop tanks range from 14 at the lower end of the range, to 7 drop tanks at the high end. As a check against the baseline vehicle configuration, option #4 has 12 drop tanks in total. If the number of drop tanks remained constant, the increase in specific impulse to 1000 sec would result in a 10% increase in total mission ΔV. This margin could be used to decrease transfer time, increase payload, widen the abort capability, or to provide margin over departure windows.

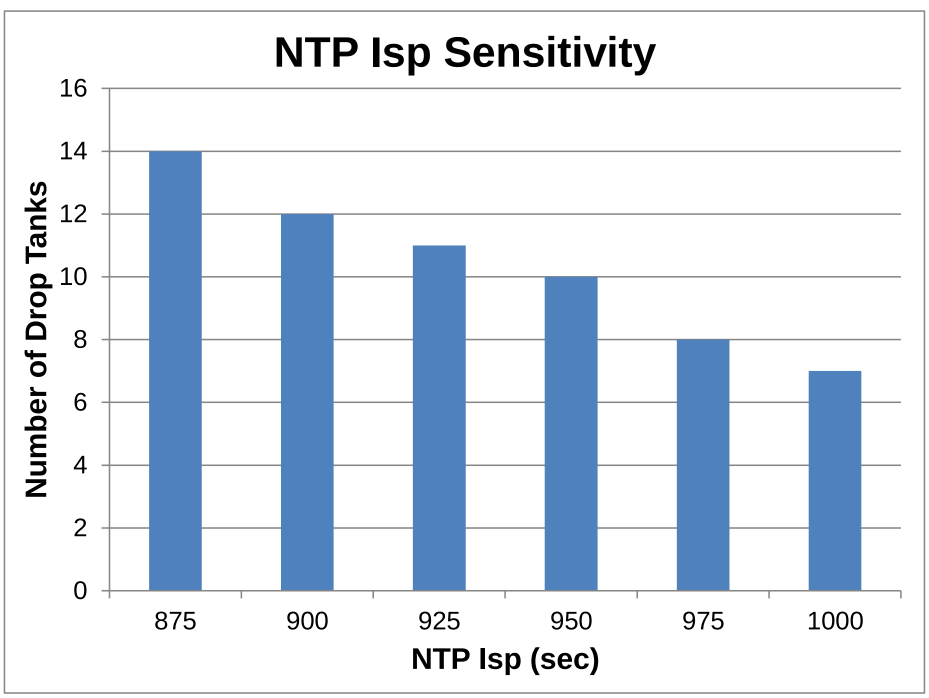


Figure 25. NTP Isp vs. Number of Drop Tanks for Option #4

2. NTP Engine Mass

Figure 26 shows the NTP engine mass sensitivity for option #2. The baseline option #2 vehicle configuration has ample performance margin to allow for the engine mass to increase (from 5500 kg to 6500 kg) while not requiring an additional drop tank. On the lower end of the range, the decrease in engine mass to 3500 kg does not provide enough additional performance to reduce the number of drop tanks to 3. This insensitivity to engine gross mass is primarily due to the large launch mass of the drop tanks (93.6 mT) which provide close to 900 m/s of ΔV for each drop tank on the vehicle stack.

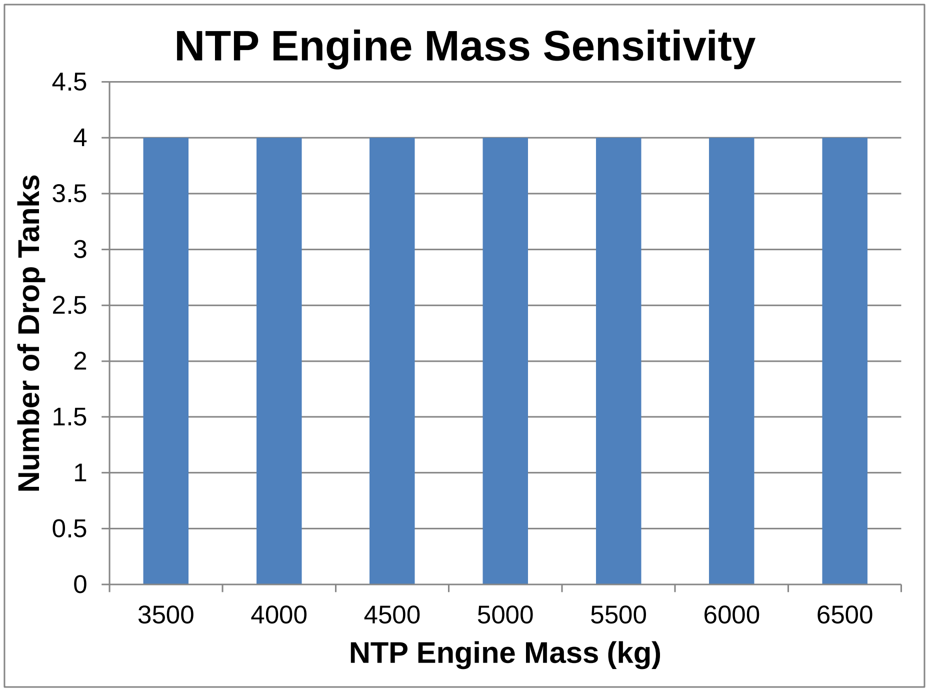


Figure 26. NTP Engine Mass vs. ΔV Capability for Configuration #2

Vehicle option #4 has a more pronounced sensitivity to changes in NTP engine mass due to its lower gross mass drop tanks (Figure 27). At the low end of the engine mass range, option #4 would require 11 drop tanks to complete the opposition opportunity, but at the high end of the range the vehicle would require 15 drop tanks.

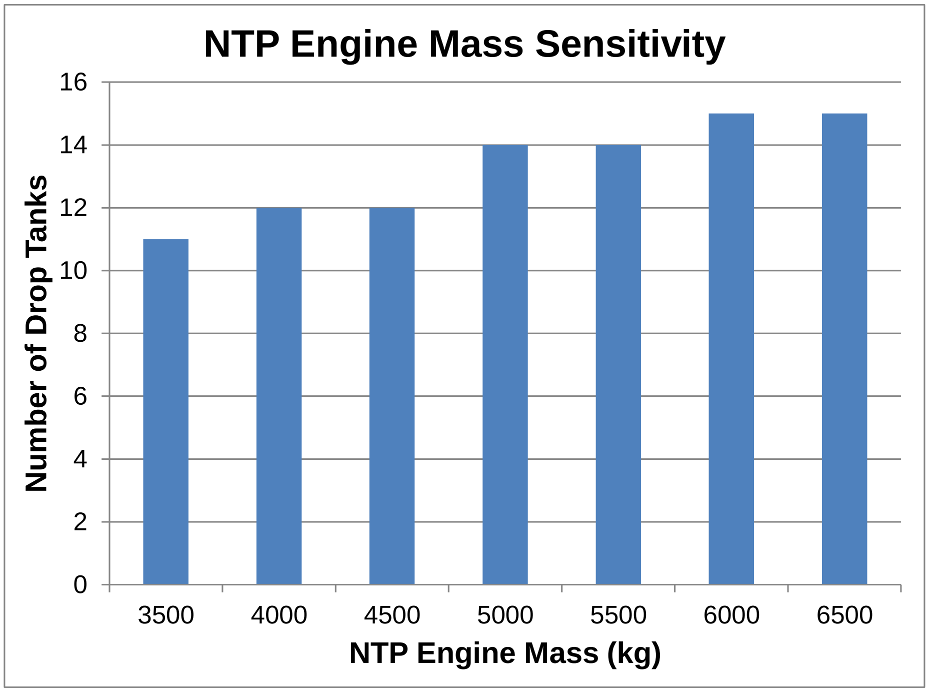


Figure 27. NTP Engine Mass vs. Number of Drop Tanks for Configuration #4

VI. Mars Opposition Cargo Missions

Cargo and resupply missions will be required in support of crewed exploration of Mars in order provide crews on the surface with habitation, logistics, and descent/ascent vehicles. As with any Mars architecture, there are several options for cargo delivery using both SEP, NTP, and chemical propulsion. For this study, NTP was baselined for cargo delivery using a derivative of the primary NTP core, with only a single NTP engine rather than multiple engines. There are multiple parallels between the conjunction mission cargo analysis and the opposition cargo analysis detailed below.

A. Cargo Ground Rules & Assumptions

For the cargo portion of the study, some top level assumptions were made to simplify the analysis so that focus could be put on capability of a few options rather than examining several cases at a high level. The assumptions taken are shown below.

- Landers aggregate in a 2,000 km x 40,000 km orbit
- 3 x 54 mT Landers required for a crewed Mars exploration mission
- Landers insert into a 5-sol Martian orbit (250 km x 112,000 km)
- Options examined for both propulsive capture and aero capture of the landers

Three primary missions were examined for Mars cargo using NTP. These missions use the NTP cargo stage for TMI and MOI, TMI alone, and TMI plus a maneuver to return the NTP cargo stage to Earth. The bat charts below (Figures 28-30) show the mission CONOPS for each of these options.

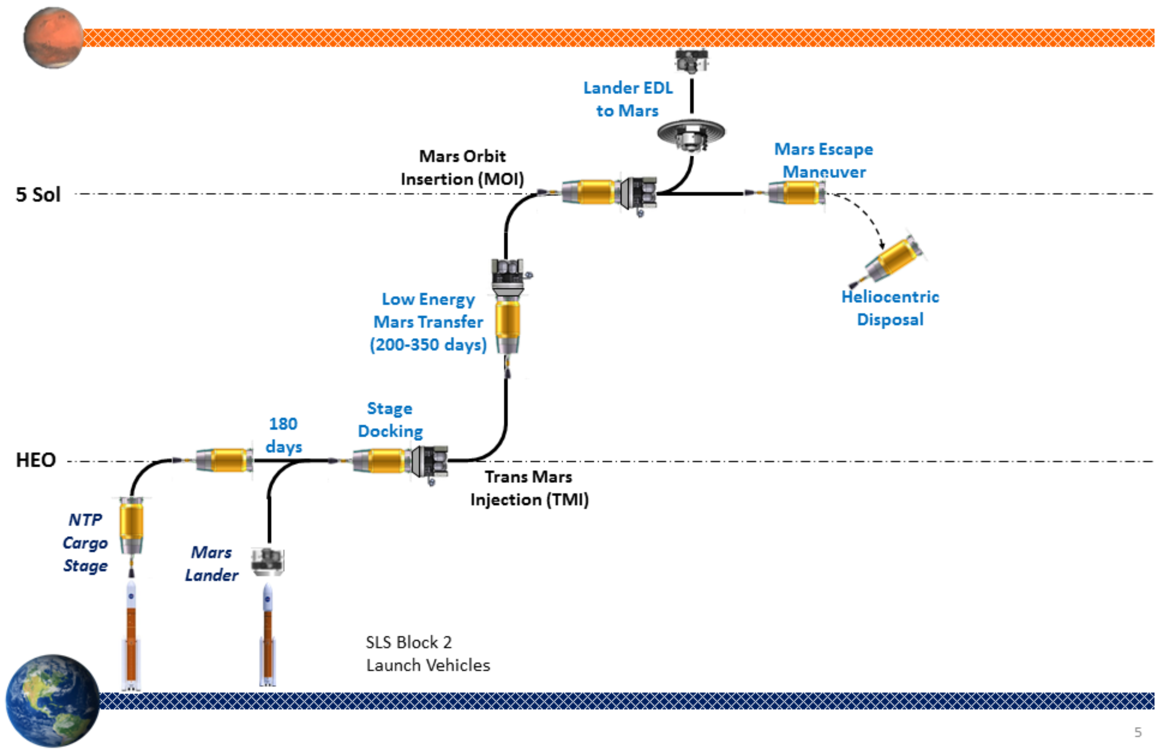


Figure 28. Cargo Mission #1: TMI + MOI

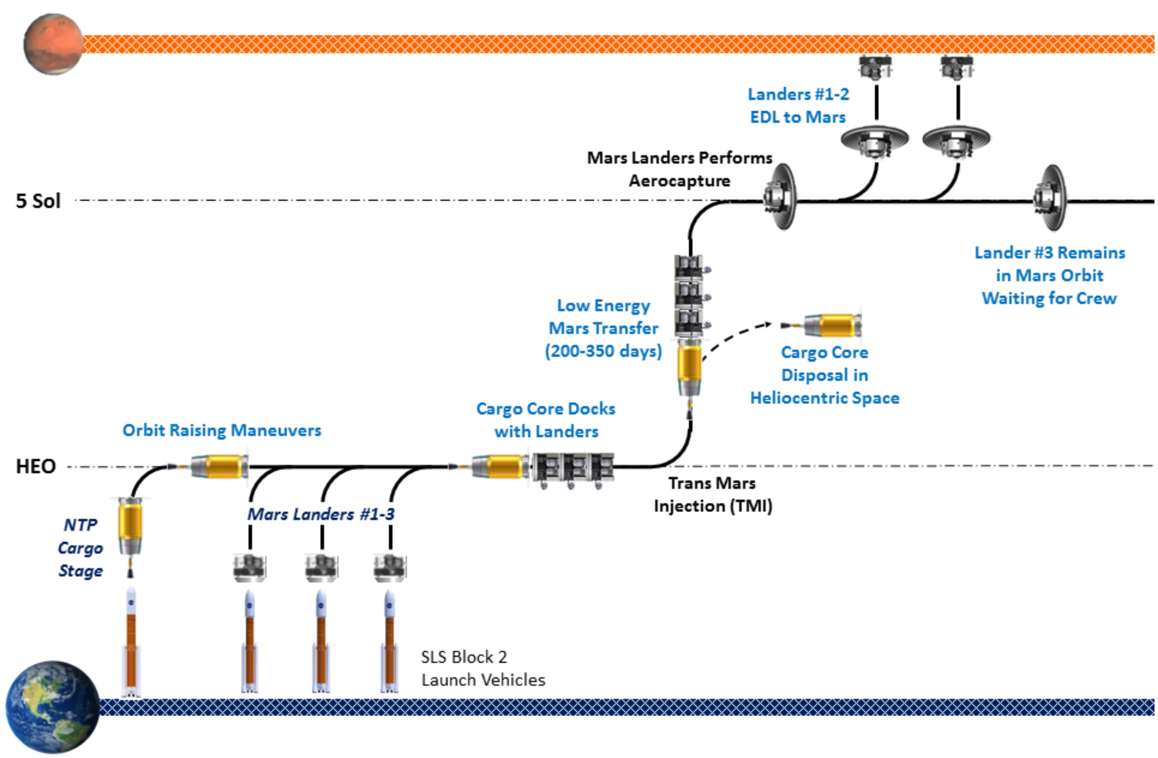


Figure 29. Cargo Mission #2: TMI Only

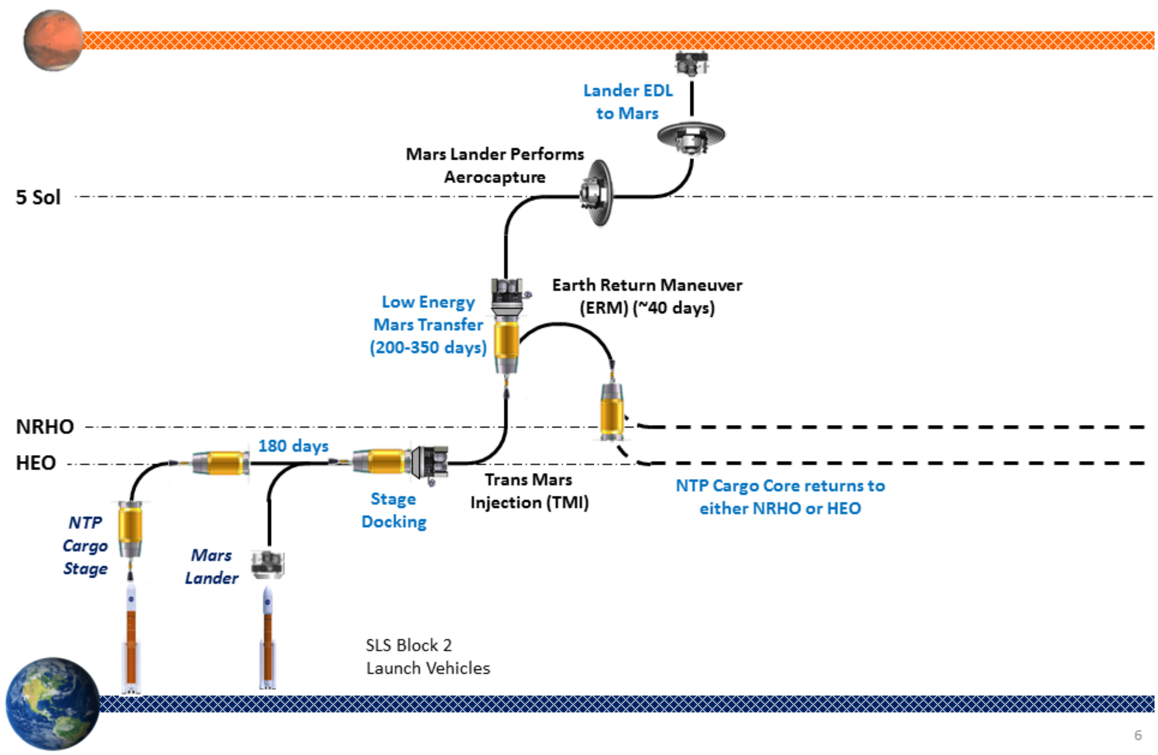


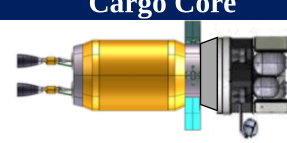
Figure 30. Cargo Mission #3: TMI + Earth Return Maneuver (ERM)

Notable for cargo mission #3 (Figure 30) is that the returned NTP cargo stages can possibly be used as the strap-on cores in the NTP crewed vehicle configurations that utilize them. Ideally, this use of NTP cargo cores would be interchangeable, where strap-on cores can also be used as NTP cargo cores after flying on a crewed mission.

B. Cargo Core Mission Options

For the cargo portion of the study, two vehicles were designed based on the options examined for the NTP opposition crewed vehicle. Each NTP cargo vehicle is based on the primary NTP core, using a single NTP engine rather than multiple engines. The diameters of the stages are also taken from several of the NTP crewed vehicle options, using both 7m and 9.1m. Details for both options are provided in the Table 5.

Table 5. Mars Cargo Vehicle Options

	NTP Cargo Vehicle #1 54 mT / 7m Cargo Core	NTP Cargo Vehicle #2 590 mT / 9.1m Cargo Core
		
LV Drop Off Orbit	400 km x 40,000 km	1,300 km LEO
Aggregation Orbit	2,000 km x 40,000 km	2,000 km x 40,000 km
Payload Mass Range	55 mT – 127 mT	76 mT – 193 mT

Mars Surface Payload Requirement	3 x 54 mT Landers	3 x 54 mT Landers
No. of SLS Launched for Cargo Mission	5 SLS	4 SLS
Total ΔV Available w/ Single Lander	2,557 m/s	3,189 m/s

Details on capabilities for each NTP cargo vehicle across the three cargo missions are shown in Table 6. The payload masses delivered to Mars vary greatly depending on the mission option chosen, with the largest payload being close to 200 mT delivered to a Mars 5-sol orbit. This case aligns with the larger 90 mT NTP cargo vehicle and the TMI only mission, where the landers aero capture at Mars. With these payload values, several landers could be delivered to Mars using a single NTP cargo core, saving launches and creating schedule margin.

The lightest payload delivered to Mars corresponds to the smaller 54 mT NTP cargo vehicle and the TMI + MOI mission option. In this case, the NTP cargo core can deliver 55 mT and perform the propulsive insertion into a 5-sol Martian orbit. This option would require a larger number of launches to proposition three Mars landers, but would reduce the mission complexity by not having to aero capture such a large payload upon Mars arrival.

Table 6. NTP Cargo Vehicle Payloads

	NTP Cargo Vehicle #1	NTP Cargo Vehicle #2
TMI + MOI	55 mT	76 mT
TMI Only	82 mT	120 mT
TMI + Earth Return Maneuver (ERM)	127 mT	193 mT

A significant capability can be found when opting for the TMI + ERM mission, where the both NTP cargo core options can deliver greater than one lander, and still return to Earth orbit for later reuse.

While cargo is an integral part of a crewed Mars exploration campaign, this analysis is primarily at a high level, showing capabilities to deliver large masses or provide reusability for further missions. Additional work would be needed to determine accurate mass estimates for the lander systems, as well as number of landers required to support a short-term stay at Mars.

VII. Conclusions & Future Work

Additional effort has been put forward on reducing the risk of each of the vehicle configurations detailed above, while also fleshing out the study ground rules. Further work is ongoing to define a detailed mission campaign across several Mars opposition opportunities, looking at launch vehicle availability and other areas. Other analyses are also in work, going backwards and using the current opposition NTP elements to derive a conjunction class vehicle concept using NTP. While conjunction class missions were the focus of earlier effort, showing commonality with the current NTP opposition elements would prove beneficial for future architectural analyses.

Also, for presentation in upcoming papers, AR has begun work on a comparison of transportation systems for the human exploration of Mars. Current analysis is looking at high power NEP vehicles, hybrid NEP/Chemical systems, and some of the NTP options presented in this paper. A rendering of one of the early NEP concepts is shown in Figure 31. The primary goal of incorporating these other advanced propulsion concepts into the Mars opposition study is to better understand the trade space, and to determine the best overall propulsion system for a crewed Mars opposition mission.

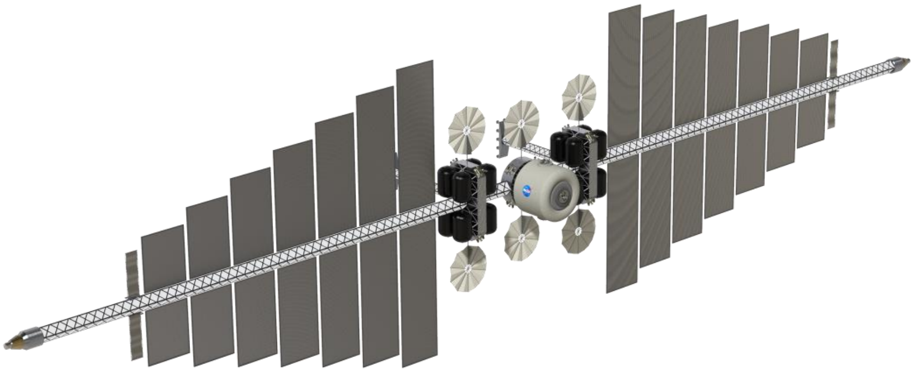


Figure 31. AR High Power NEP Transportation Vehicle Concept

As detailed in this paper, there are several options for using NTP for Mars opposition missions, each of which is enabled with attainable and currently in development NTP engines. Each of the crewed vehicle options detailed in this paper close for a Mars opposition mission, with later configurations showing the design maturity taken throughout this study. These more mature configurations, like the Option #4 detailed above, show a robust vehicle capable of reducing the number of launches and covering a wide window of opportunities, while also providing an advanced propulsion system that translates to multiple architectures and missions.

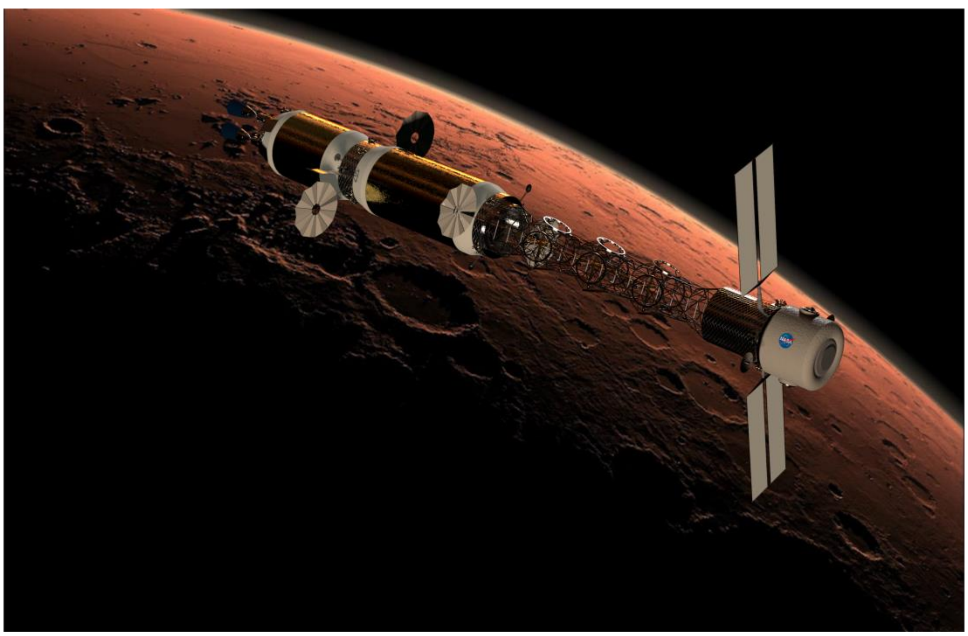


Figure 32. NTP Transfer Vehicle in Mars Orbit

References

[1] Drake, B. G., "Human Exploration of Mars Design Reference Architecture 5.0", NASA/SP-2009-556, 2009.
[2] Braun, W. V., "Manned Mars Landing Presentation to the Space Task Group", NASA, August 4, 1969.
[3] Borowski, S. K., McCurdy, D. R., Packard, T. W., "Modular Growth NTP Space Transportation System for Future Human Lander, NEA and Mars Exploration Missions", AIAA Space 2012 Conference & Exposition, Pasadena, California, AIAA 2012-5144, 2012.
[4] *NERVA Engine Reference Data*, Aerojet Nuclear Systems Company, NERVA Program, Contract SNP-1, September 1970.
[5] Levack, D. J. H., Horton, J. F., Jennings, T. R., Joyner II, C. R., Kokan, T., Mandel, J. L., Muzek, B. J., Reynolds, C. B., and Widman, F. W., "Evolution of Low Enriched Uranium Nuclear Thermal Propulsion Vehicle and Engine Design", AIAA Propulsion and Energy Forum and Exposition 2019, Indianapolis, Indiana, AIAA 2019-3943, 2019.
[6] "Space Launch System (SLS) Mission Planner's Guide", NASA, 20190000736, 2019.
[7] Muzek, B. J., Horton, J. F., Joyner II, C. R., "Mission Analysis for Mars Opposition Missions 2033 to 2048", AAS Guidance, Navigation, and Control Conference, Breckenridge, Colorado, AAS 20-065, 2020.
[8] Williams, J., Senent, J. S., Ocampo, C., Mathur, R., and Davis, E. C., "Overview and Software Architecture of the Copernicus Trajectory Design and Optimization System", 4th International Conference on Astrodynamics Tools and Techniques, Madrid, Spain, 2010.
[9] Williams, J., *Copernicus Version 4.6 User Guide,* NASA Johnson Space Center, pg. 380-1.
[10] "Integrated Manned Interplanetary Spacecraft Concept Definition", Boeing, 1968.
[11] "Space Transfer Concepts and Analysis for Exploration Missions", Boeing, 1990.
[12] *Starship Users Guide,* Revision 1.0, SpaceX, March 2020.
[13] *New Glenn Payload User's Guide*, Revision C, Blue Origin, October 2018.